9th AUSTRALASIAN
FLUID MECHANICS CONFERENCE,
AUCKLAND, 8-12 DECEMBER 1986

Investigation of a Flow Generated by Control Tabs in a Rocket Nozzle Exhaust

C. M GOURLAY[1], J. M. SIMMONS[1] and R. F. MEYER[2]

[1]Department of Mechanical Engineering, University of Queensland, Australia.
[2]Faculty of Engineering, University of Auckland, New Zealand.

ABSRACT

A shock tunnel has been used to study the flow of air, argon and carbon dioxide past tabs obstructing a rocket nozzle exhaust. The tabs are a means of thrust vector control. Schlieren photographs show a Mach reflection pattern with a Mach disc in the region enclosed by the tabs. Pressure measurements indicate that the formation of a Mach disc enhances the forces on the tabs. A preliminary heat transfer study is reported, and the factors which influence the available test time are discussed.

INTRODUCTION

Many schemes have been studied for thrust vector control of rockets, for example, gimballed motors, secondary injection into the nozzle and insertion of vanes into the exhaust flow. The usual aim is to obtain control of the direction of the thrust vector. The work reported here is a continuation of the study by Simmons et al. (1986) of a scheme in which three tabs are inserted into the rocket nozzle exhaust (Figure 1), thereby enabling not only a change of the direction of the thrust but also a reduction of its magnitude. Thrust magnitude modulation has potential application for the control of solid propellant rockets.

A sketch of the Mach relection shock wave pattern observed by Simmons et al. (1986) for symmetric tab insertion is shown in Figure 2. If the Mach disc (normal shock) is sufficiently far upstream, the reflected shock waves strike the tabs and the high pressure downstream of the Mach disc raises the pressure on part of the tabs. The resultant effective reduction of the amplitude of rocket thrust can be expected to be greater than that with tab configurations for which a Mach disc does not form. The Mach disc location (L_s in Figure 2) is taken as the main characteristic of the shock wave pattern.

A mathematical model of the flow field is required in order to predict the magnitude of the axial and radial components of the forces on the tabs for the purpose of designing a control system. A simple mathematical model was proposed by Simmons et al. (1986) for symmetric tab insertion but it was necessary to calibrate it against their experimental data. The aim of the study reported here was to extend the experimental data in three ways. Firstly, for the case of symmetric tab insertion, various test gases were used to give a range of gas properties typical of that found in rocket exhausts. Secondly, pressure distributions over the tabs were measured to complement the shock wave patterns obtained with schlieren techniques, and thirdly, several configurations with asymmetric tab insertion were studied with a view to later extension of the mathematical model.

Heat transfer to the tabs in a rocket exhaust will be high and problems of structural integrity might occur. The first steps in the development of techniques for measuring heat transfer are reported.

EXPERIMENTAL TECHNIQUES

Experiments were conducted in the small free piston shock tunnel TQ in the Department of Mechanical Engineering at the University of Queensland (Figure 3). The nozzle produced a Mach number of 4.07 for air, 5.66 for argon and 3.38 for carbon dioxide. It had an exit diameter of 70 mm with a semi-angle of divergence of 7 degrees. Full details of the model are in Simmons et al. (1985,1986) and Gourlay and Simmons (1985). Three tapered tabs were inserted, initially symmetrically, around the cicumference of the nozzle exhaust flow with the geometry defined in Figure 2. The configuration of the symmetric cases was changed by moving all tabs radially, thereby varying D. Asymmetric configurations were obtained by displacing one tab from a symmetric configuration.

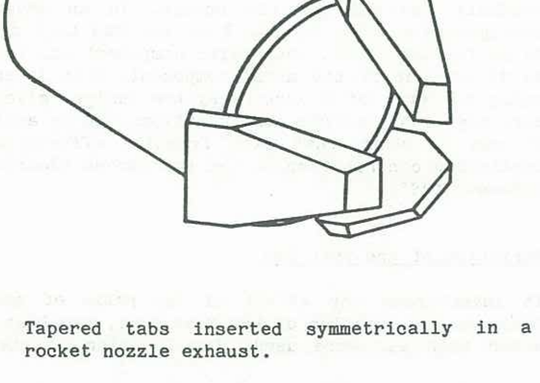

Fig 1: Tapered tabs inserted symmetrically in a rocket nozzle exhaust.

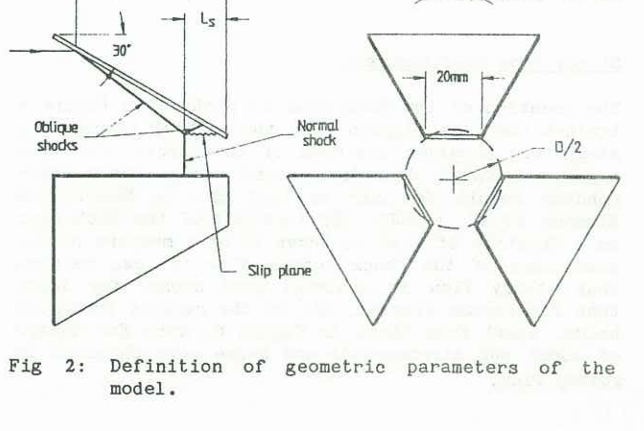

Fig 2: Definition of geometric parameters of the model.

A dual-pass colour schlieren system was used for flow visualization studies. A pulsed stroboscope with 10 microseconds duration of flash was triggered in single-shot mode from the rise in output of a stagnation pressure transducer immediately upstream of the nozzle of the shock tunnel (Figure 3). The stagnation temperature in the test section was calculated from measurements of the primary shock speed which in turn was obtained from two pressure transducers in the shock tube. The test section Mach number was calculated with a one-dimensional analysis from the area ratio of the nozzle and the dependence of the specific heat on temperature.

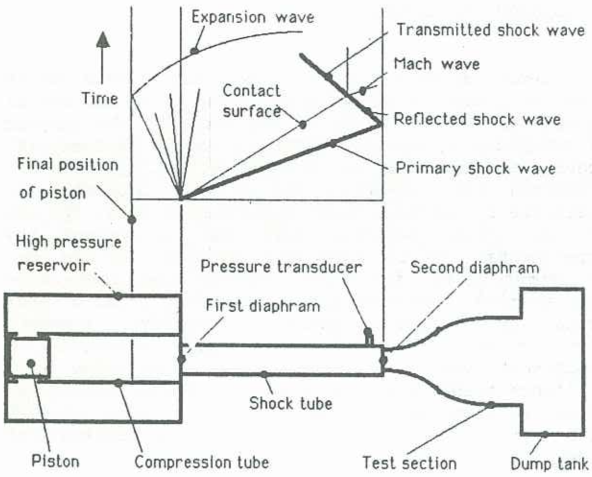

Fig 3: Schematic of shock tunnel with wave diagram.

The pressure distributions on one tab were obtained with six piezo-electric pressure transducers mounted in two configurations (Figure 4). Two techniques were evaluated for the measurement of heat transfer to the tabs, namely thin film resistance gauges and calorimeter thermocouple gauges (Figure 4).

EXPERIMENTAL RESULTS

Shock Wave Patterns

Figure 5 shows a representative schlieren photograph of a symmetric configuration with the top tab viewed edge-on. Flow is from left to right and impinges on the tabs, generating a Mach reflection pattern. The shock wave pattern was found to be insensitive to variation of the test section stagnation conditions. For example, with argon test gas, separate increases in stagnation temperature from 4064 K to 7364 K and stagnation pressure from 2.06 MPa to 2.76 MPa produced no measurable change in the location of the Mach disc for a given tab configuration. Typical values of stagnation pressure and temperature are : 1.6 MPa and 1500 K for air, 2.76 MPa and 6150 K for argon, and 1.74 MPa and 1890 K for carbon dioxide.

Steady Flow Establishment

The location of the Mach disc is plotted in Figure 6 against the time elapsed from the initial increase in stagnation pressure upstream of the nozzle, for both argon and carbon dioxide as test gases. The corresponding result for air as test gas is taken from Simmons et al. (1986). The location of the Mach disc as a function of time is taken to be a measure of the steadiness of the shock tunnel flow. It can be seen that steady flow is achieved much sooner for argon than for carbon dioxide. All of the results presented below, apart from those in Figure 6, were for delays of about 800 microseconds and hence were obtained in steady flow.

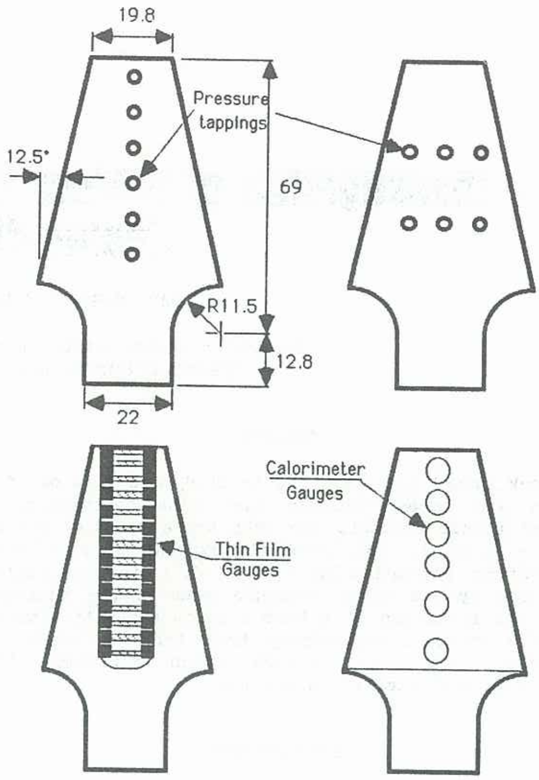

Fig 4: Location of pressure transducers and heat transfer gauges.

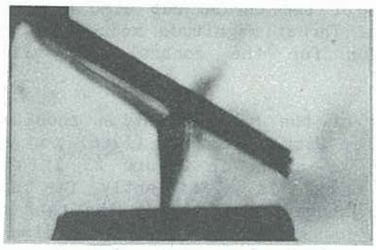

Fig 5: Schlieren photograph of Mach reflection pattern for D = 30.3 mm with air at Mach 4.1.

Tab Pressure

A representative pressure distribution over a tab is shown in Figure 7. It is for the symmetric configuration corresponding to Figure 5. The lack of symmetry across the tab is as yet unexplained. Integration of the pressure distributions on the three tabs gives components of force along (axial) and normal to (radial) the axis of the nozzle. In an asymmetric configuration with D = 30.3 mm for two tabs and D = 40 mm for the other, the radial component was found to be 10 percent of the axial component. This is encouraging in terms of a control system design, given that more highly asymmetric configurations can be achieved. It can be shown that skin friction effects make a negligible contribution to the tab forces (Gourlay and Simmons, 1985).

Variation of the Test Gas

To investigate the effect of the ratio of specific heats on the location of the Mach disc, two test gases other than air were used. The location of the Mach disc as a function of the separation of the tabs, for a symmetric case, is given for the two gases and for air in Figure 8. The result for air is taken from Simmons et al. (1986). The ability of a mathematical model to predict these trends can be studied by varying the ratio of the specific heats.

Heat Transfer

Heat transfer measurements have been obtained by two methods (see Figure 4). With thin film resistance gauges a problem was encountered in the harsh environment of the shock tunnel. The gauges were actually removed by the hot flow and by the impingement of metal diaphram particles. This was attributed to the fact that the gauges were mounted on a compression surface rather than an expansion surface where such gauges have been used with success. The stagnation enthalpy of the flow was reduced to subject the gauges to lower temperatures. This met with some success. However, some of the measured temperature rise might have been due to erosion of the gauges. Results obtained by this method for a single tab and for a system of three tabs are given in Figure 9. They are taken from Meyer (1985).

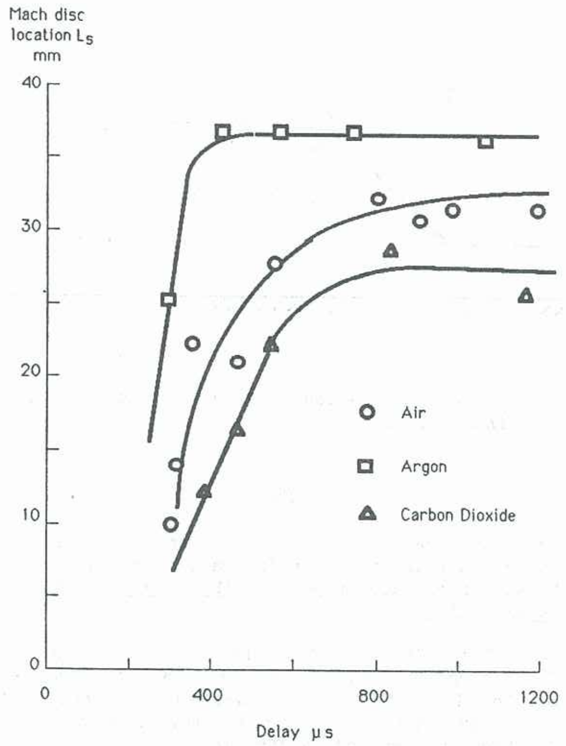

Fig 6: Mach disc location versus delay time for three test gases.

The thin-film gauge results are compared with theory in the report by Meyer (1985) and the predictions are reproduced here in Figure 9. Meyer assumed a flat plate laminar boundary layer to predict the heat transfer to a single tab in the efflux of a conical nozzle. He used Eckert's reference enthalpy method for the heat transfered to the tab and he calculated the pressure from an axi-symmetric characteristics analysis of the nozzle flow impinging on the tab, assuming locally plane flow and obtaining an average oblique shock angle from experiment. The predictions correlate well with experiments using a single tab at the upstream end. For the case of three tabs with the Mach disc there are high heat transfer rates near the downstream end of the tabs because of the high pressures.

Because of the problems with thin film gauges another

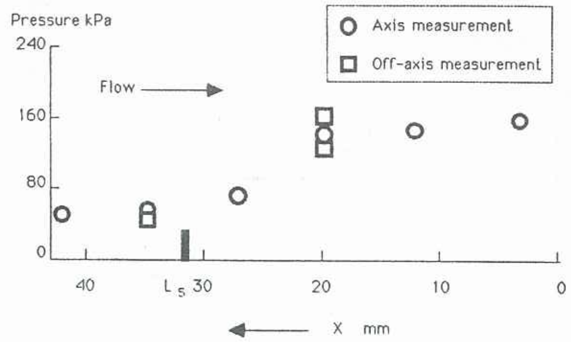

Fig 7: Static pressures on and off axis of symmetry of tab for conditions corresponding to Figure 5.

approach was taken; the use of calorimeter thermocouple gauges. These gauges were not eroded by the flow but their mechanical integrity was not high due to the fine scale of construction necessitated by the short duration flows. Constantin wire of 0.025 mm diameter was ultrasonically welded to the back of a sheet of 0.05 mm brass shim mounted on a specially modified tab (Figure 4). Two problems were encountered. The glue became hot and the shim separated from the tab substructure. Also, the wires became detached from the shim.

The theory of heat transfer measurement using calorimeter gauges is given in Schultz and Jones (1973). The one result obtained by this method gave an almost linear temperature-time response, as expected from the theory. This is encouraging but more development of the technique is needed for this application.

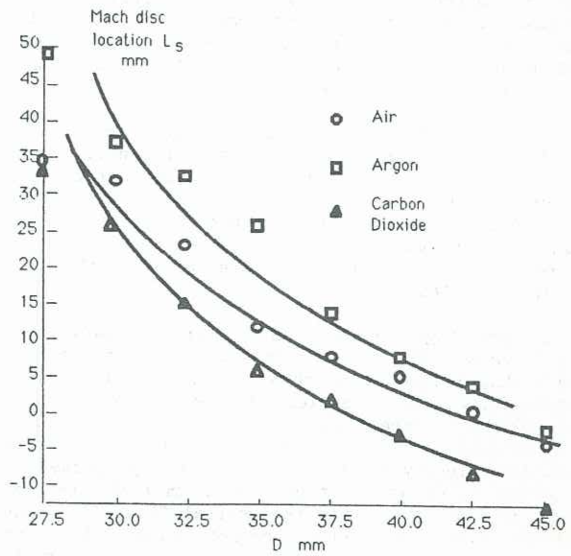

Fig 8: Location of Mach disc versus amount of symmetric tab insertion for three gases.

TEST TIME DURATION

The free-piston shock tunnel used in these studies produces high stagnation pressures and temperatures, but only for a short time. The theoretical test time is limited by one of the following phenomena:

(A) The reflected expansion wave generated by the bursting of the diaphram reflects from the piston and arrives at the interface of the driver and test gases (Figure 3).

(B) The flow of test gas is exhausted on the arrival of the interface at the test section.

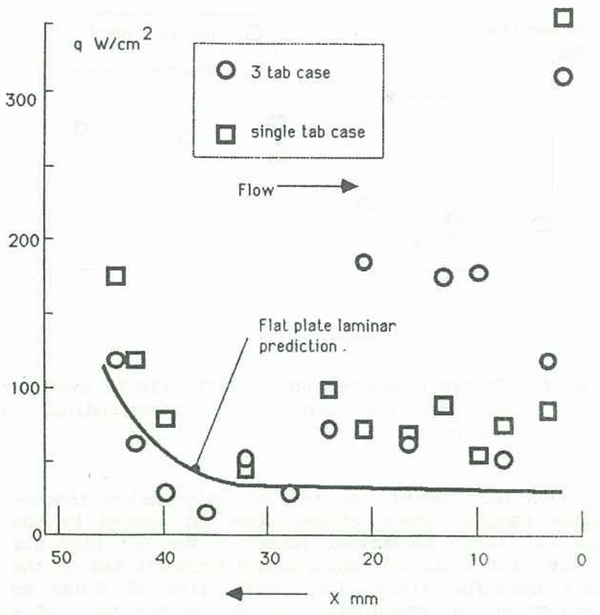

Fig 9: Measured heat transfer rates for conditions corresponding to Figure 5.

(C) If there is a difference in density across the interface then a 'non-tailored' condition exists. This means that the shock wave reflected upstream from the end of the shock tube will reflect back from the interface either as a shock wave or an expansion, as well as being partly transmitted. If a shock is reflected back towards the nozzle the interface is accelerated towards the end of the tube and the test time is shortened. If an expansion is reflected the pressure drops in the stagnation region, also truncating the test time. If 'tailored' conditions exist then only a Mach wave is reflected back towards the nozzle from the interface.

(D) The boundary layer between the primary shock wave and the interface acts as an aerodynamic sink absorbing mass from this region. The contact surface accelerates and the shock wave decelerates, causing their relative separation distance to decrease. The amount of test gas available to be expanded through the nozzle is then reduced, together with the test time (Mirels 1963).

(E) If the stagnation pressure in the boundary layer is too low to allow it to negotiate the pressure rise through the reflected shock wave, the boundary layer separates and a bifurcation of the reflected shock wave results (Figure 10). The flow which is processed by the two oblique shock waves in the foot has a different velocity to that of the flow processed by the normal shock wave. It can be shown that the foot flow is moving faster than the central flow. After the reflected shock wave passes through the interface the flow being processed by the shock wave system is driver gas flow which has a much lower stagnation temperature than the test gas. Since the foot flow is travelling faster than the central flow, driver gas can arrive at the end of the shock tube before the interface. This will reduce the temperature of the test gas flow and truncate the test time.

(F) The interface can become unstable after it has been processed by the reflected shock wave pattern, if bifurcation exists, since there is a shear layer emanating from the point where the oblique and normal shocks meet.

Because shock wave/boundary layer interaction (E) normally presents the worst limitation on the test time, an analysis was performed. Davies and Wilson (1969) proposed this mechanism to explain experimentally observed reductions in stagnation temperature at the end of a shock tube. By using a similar approach to Davies and Wilson and assuming tailored shock tube operation, the maximum length of test time was calculated to be 960 microseconds. This includes the time it would take the contaminated flow upstream of the nozzle to be expanded through the nozzle and to arrive at the test section. The conclusion is that the schlieren photographs were taken in the steady regime without contamination of the test gas by cold driver gas.

CONCLUSION

Experimental data has been generated for evaluation of a mathematical model of the control forces produced by tabs in a rocket nozzle exhaust. The existence of a Mach disc in the region enclosed by the tabs has been shown to enhance the magnitude of the pressures that are generated on the surface of the control tabs, and hence the drag forces. The Mach disc has also been shown to produce higher heat transfer rates to the tabs than would occur if it were absent. The heat transfer to the tabs requires further study. A shock tunnel permits convenient changes of test gas, but careful attention must be paid to the requirements for steady test flow.

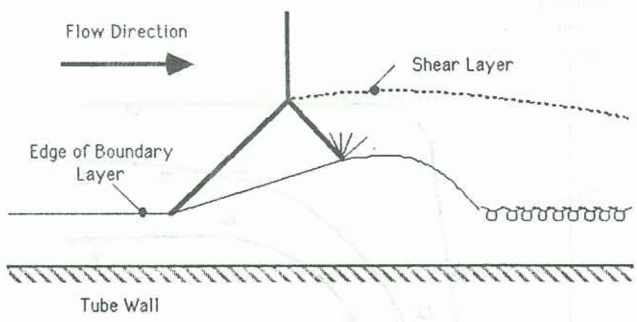

Fig 10: Shock bifurcation due to boundary layer interaction.

REFERENCES

Davies, L; Wilson, J L (1969): Influence of Reflected Shock and Boundary-Layer Interaction on Shock-Tube Flows. Shock Tube Symposium, The Physics of Fluids Supplement I, 37-43.

Gourlay, C M; Simmons, J M (1985): Measurement and Prediction of Pressures on Ramp Tabs in a Rocket Nozzle Exhaust. Report No. 8/85. Dept. of Mechanical Engineering, University of Queensland, Brisbane, Queensland, Australia.

Meyer, R F; (1985): Heat Transfer and Pressure Measurements on Rocket Tabs used to Control Rocket Nozzle Flow. Report No. 6/85. Dept. of Mechanical Engineering, University of Queensland, Brisbane, Queensland, Australia.

Mirels, H (1963): Shock Tube Test Time Limitation Due to Turbulent-Wall Boundary Layer. AIAA Journal Vol. 2, No. 1, 84-93.

Schultz, D L; Jones, T V (1973): Heat Transfer Measurements in Short-Duration Hypersonic Facilities. AGARDograph No. 165.

Simmons, J M; Gourlay, C M; Leslie, B A (1985): The Flow Field Generated by Inclined Tabs Obstructing a Rocket Nozzle Exhaust. Report No. 3/85. Dept. of Mechanical Engineering, University of Queensland, Brisbane, Queensland, Australia.

Simmons, J M; Gourlay, C M; Leslie, B A (1986): The Flow Field Generated by Ramp Tabs in a Rocket Nozzle Exhaust. AIAA Paper 86-0282.