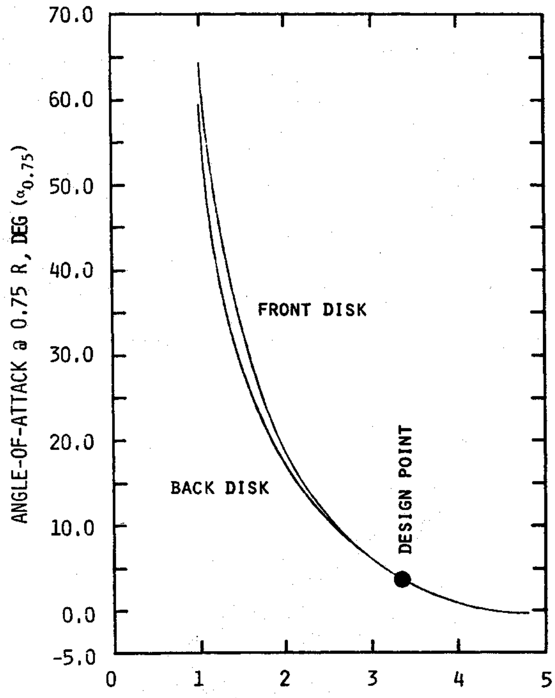

Fig. 3 Angle of attack of front/aft propeller(s) at 75% radial location as function of advance ratio.

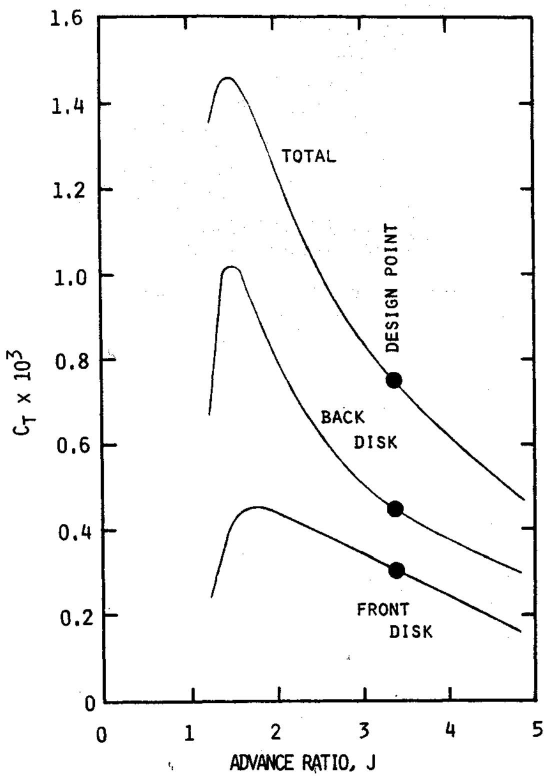

Fig. 4 Thrust coefficients of front/aft propeller(s) as function of advance ratio for off-design mode.

Summary

Using the method of Playle et al., as given by Davidson, the off-design performance of counter-rotating propeller configurations has been investigated. As in Playle's original study for the fixed-pitch condition, the off-design analysis for a variable-pitch or constant-speed condition of the counter-rotating propeller yielded a relatively flat propeller efficiency curve for advance ratios of 1.5–5. This type of performance has again indicated the full extent of the performance advantages of counter-rotating propeller configurations when analyzed for off-design conditions.

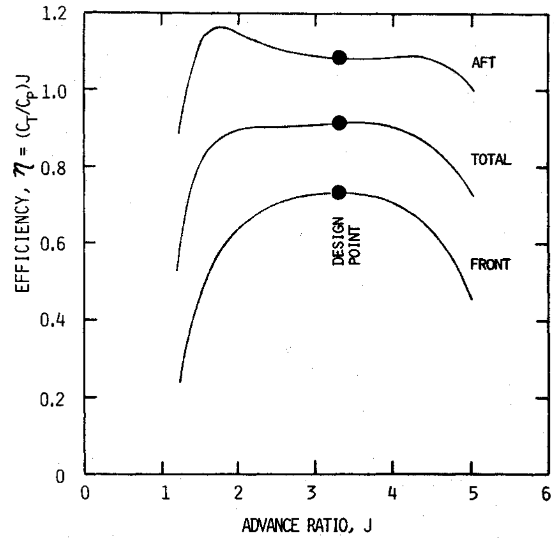

Fig. 5 Propeller efficiency of counter-rotating propeller configuration in off-design mode.

Acknowledgment

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Flow Generated by Ramp Tabs in a Rocket Nozzle Exhaust

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Introduction

MANY schemes have been studied for thrust vector control of rockets. The usual aim is to obtain control of the direction of the thrust vector while minimizing unwanted reduction of its magnitude. Reported here is a study of a scheme in which three ramp tabs can be partially inserted into the nozzle exhaust at large angles to the flow, thereby

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generating not only a change in the direction of the thrust of the rocket, but also a significant reduction of its magnitude. Thrust magnitude modulation has potential application to the control of solid propellant rockets. A pictorial view of a configuration with tapered ramp tabs is shown in Fig. 1. The three tabs can be inserted either symmetrically, to modulate the magnitude of the thrust, or asymmetrically, to generate a transverse component of force on the rocket. A simple mathematical model of the flowfield has been developed for symmetric tab insertion. Its formulation has been guided by a series of flow visualization studies.

Experiments

Experiments with air as the test gas were conducted in a small free piston shock tunnel with a Mach 4.1 nozzle. Details are in Ref. 1 and 2. Three tapered tabs were inserted symmetrically around the circumference of the nozzle exhaust flow, with geometry defined in Fig. 2. The configuration of the model was changed by moving the tabs radially, thereby varying the exit diameter D of the imaginary truncated cone on which the tabs lie. Static pressures were measured along the centerline of one tab. A dual-pass color schlieren system was used for the flow visualization studies. A pulsed stroboscope tube with 10-μs duration of flash was triggered in a single-shot mode from the rise in stagnation pressure immediately upstream of the nozzle of the shock tunnel. Photographs were taken 820 μs after the initial increase in the stagnation pressure, thereby using a period of at least 400 μs of steady Mach number between the flow starting transient and the arrival of the test/driver gas interface. The shock wave pattern was found to be insensitive to small variations in the stagnation conditions[1,2] (1.6 MPa and 1490 K).

Figure 3 shows two representative schlieren photographs, with the top tab viewed edge on. Air flows from left to right

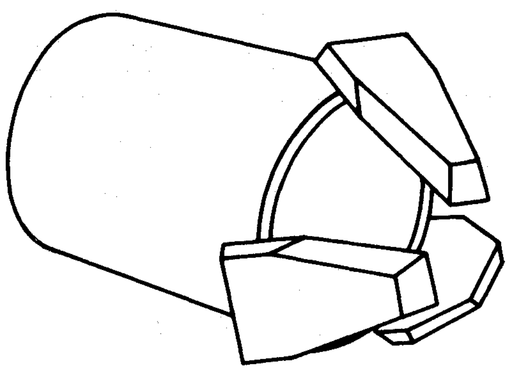

Fig. 1 Tapered ramp tabs inserted symmetrically in a rocket nozzle exhaust.

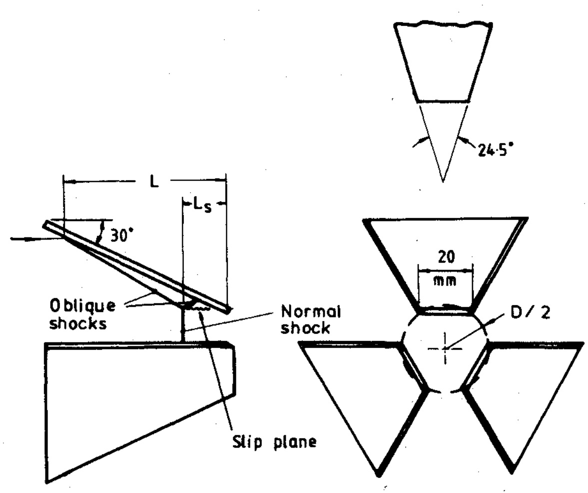

Fig. 2 Definition of geometric parameters of the model.

and impinges on the tabs, generating a Mach reflection pattern, i.e., an interaction of oblique and reflected shock waves with a nearly normal shock wave or Mach disk. A sketch of the shock wave pattern is shown in Fig. 2. The location L_s of the Mach disk with changes in D is apparent. If the Mach disk is sufficiently far upstream, the reflected oblique shock waves strike the tabs and the high pressure downstream of the Mach disk raises the pressure on part of the tabs. The resultant reduction of the amplitude of rocket thrust can be expected to be greater than that with tab configurations for which a Mach disk does not form. Figure 4 shows the movement of the Mach disk with changes in the amount of symmetric insertion of three tabs. The amount of insertion is defined by D (Fig. 2).

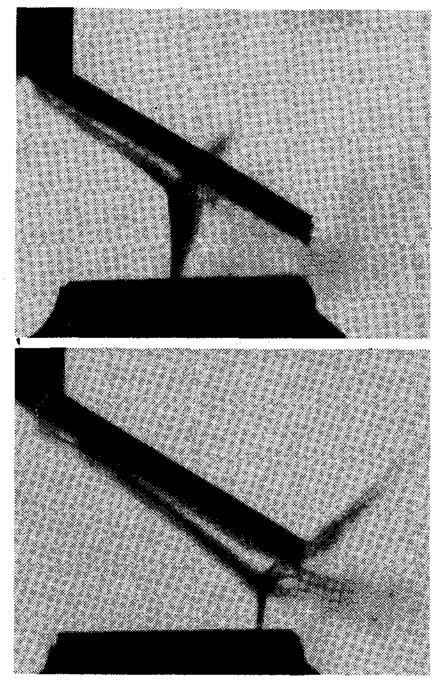

Fig. 3 Schlieren photographs of Mach reflection pattern: top, D=30.3 mm; bottom, D=37.5 mm.

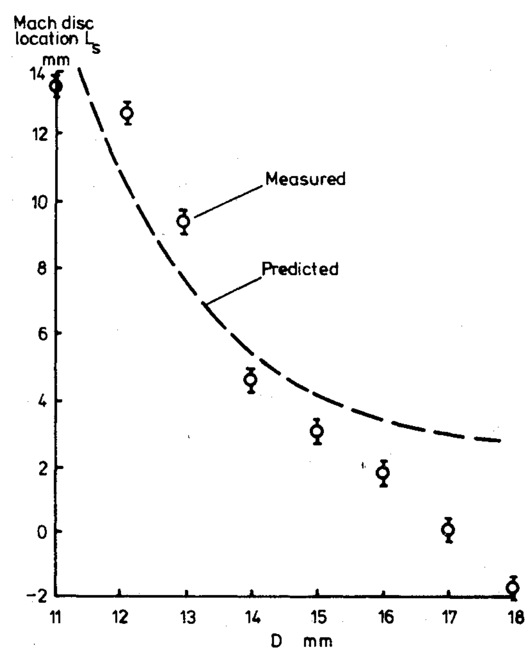

Fig. 4 Dependence of location L_s of Mach disk on amount of tab insertion (defined by D).

This set of measurements was used for the calibration of the mathematical model.

Mathematical Modeling

Analysis of the complicated three-dimensional flowfield with a Navier-Stokes code is likely to be very difficult, with extensive parameter variation being prohibitively time consuming. In order to identify promising configurations, a relatively simple mathematical model has been developed. A detailed description is available.[1,2]

The three tapered tabs are regarded as lying on the surface of an imaginary converging duct (Fig. 2). Flow enters the duct at its large end and then takes one of two paths. Some passes through the wall of the duct between the tabs and the remainder leaves the duct through its small end. The basis, then, of the modeling of this three-dimensional flow is the concept of one-dimensional steady compressible flow through a converging duct with a porous wall. The finite open areas for outflow between the tabs are spread around the duct as a continuous distribution of porosity. This approach is justified by a series of shock tunnel tests[1] with sets of 3, 6, and 12 tabs, with each set providing the same ratio of open to closed area around the duct. The reduction from 12 to 6 tabs had an insignificant effect on the location of the Mach disk. With 3 tabs, the Mach disk was located 7% closer to the downstream end of the tabs than it was with 12 tabs.

Figure 5 depicts a simple, physical model of steady flow through a porous axisymmetric duct with outlet diameter D and semi-angle α. The length L of the duct depends on α and the amount of insertion of the tabs in the rocket nozzle exhaust. The approaching supersonic flow from the nozzle is assumed to be parallel to the axis. The static pressure at the nozzle exit is assumed to act on the outside surface of the duct. A Mach disk (normal shock wave) stands at a distance $x=L_s$ upstream of the duct outlet, thereby dividing the duct flow into a supersonic and a subsonic region. Oblique shock waves are not considered at this stage, but they are included when pressures on the tabs are determined. In effect, this implies that the flow immediately upstream of the Mach disk is not affected by the tabs and has the nozzle exit conditions.

The mathematical model is essentially a one-dimensional mass balance between the flow entering the subsonic region of the duct across the Mach disk and the flow leaving this region through the downstream end of the duct and through the porous wall. The balance is achieved only when the normal shock is in the correct position. Modeling of the subsonic region requires knowledge of conditions on all of its boundaries. The porous wall does not present difficulties because typical conditions are such that the flow through the wall is always choked. However, the downstream boundary condition is difficult to formulate because the subsonic flow continues downstream of the duct exit plane. The subsonic flow region in Fig. 5 can be modeled if the boundary condition on the exit plane of the duct is assumed to be the static pressure behind the Mach disk. This implies that the central stream tube in Fig. 5 neither expands nor contracts. There is a balance between the convergence of the imaginary porous duct and the expansion-like influence of the outflow through the porous wall. This assumption is justified by the pressure measurements on the tabs[3] and order of magnitude arguments,[1] which show that the dominant processes by which the high pressure flow behind the Mach disk adjusts to the much lower ambient pressure must occur downstream of the duct exit, and hence, outside the domain of the model.

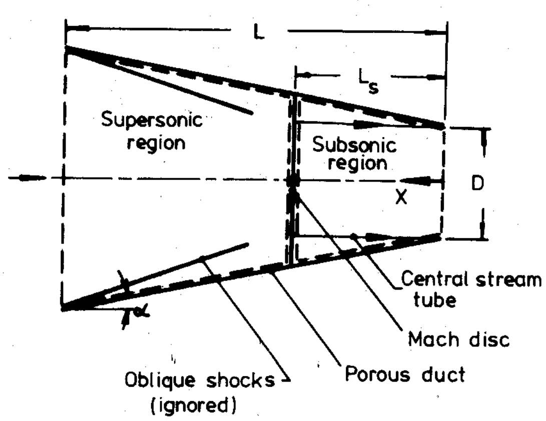

Fig. 5 Simple model of flow through porous duct.

The three component flows in the mass balance pass across areas that must be specified, namely, the areas of the Mach disk, the duct exit, and the porosity of the wall. Complicated three-dimensional effects, such as curved shock waves along the tabs and orifice flow complexities, make the effective areas less than the geometric areas. This difficulty only can be handled in a one-dimensional model by multiplying each of the three geometric areas by an empirical constant. The curve in Fig. 4 shows the ability of the mathematical model to be calibrated against the shock tunnel measurements by suitable choice of the three empirical constants. The values of the constants applied to the areas of the Mach disk, the outflow through the porous wall in the subsonic region, and the duct exit are 0.85, 0.70, and 0.73, respectively. The predictions and the measurements differ most for large duct exit diameters D. This is not serious because the Mach disk is then so far downstream that the reflected shock waves clear the ends of the tabs, leaving the tab pressures dependent only on the incident oblique shock waves.

At this point, the limit to the one-dimensional model is reached. The simple model enables a prediction of the location of the Mach disk, and hence, a reasonable approximation to the location of the point at which the Mach disk, the incident, and the reflected oblique shock waves meet (Fig. 2). It predicts a pressure in the subsonic region, which is used to determine both the choked outflow through the porous wall and the flow across the exit of the duct. Pressures on the tabs are determined from two-dimensional theory for straight oblique shock waves. The ability of the model, with the above values of the empirical constants, to predict tab pressures is shown by the good agreement with pressure measurements in Ref. 3.

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