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Testing the External Burning Propulsion Concept

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Introduction

THEORETICAL analysis[1] has indicated that external burning propulsion using the exhaust from a solid propellant as the fuel may offer performance superior to a conventional rocket in certain applications. In this propulsion concept, shown schematically in Fig. 1, the exhaust from a fuel-rich solid propellant is injected transversely through the vehicle boundary layer into the supersonic airstream surrounding the separation bubble at the vehicle base. There mixing and supersonic combustion occur, and the resulting pressure rise is transmitted to the vehicle base through the subsonic portion of the near wake. The analysis of Strahle[1] indicates that base pressures greater than the freestream static pressure are achievable.

Because of the encouraging analytical results, the authors undertook the experimental evaluation of this external burning propulsion concept. The purpose of this Note is to describe the testing techniques employed and to indicate testing considerations which are unique either to the propulsion concept or to the testing configuration.

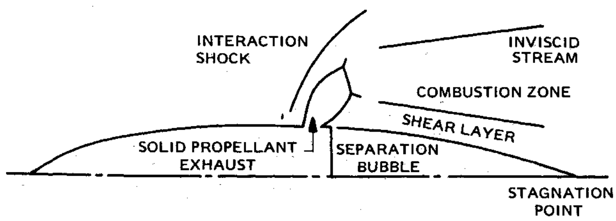

Fig. 1 Schematic diagram of the external burning propulsion concept.

Test Configuration

The ideal test configuration for this propulsion concept is probably a circular annular nozzle in which the base of the centerbody serves as the model of the vehicle base. However, as described below, the ratio of the nozzle diameter to the centerbody diameter should be as large as possible to eliminate interference effects. Thus, for the model sizes of interest, this configuration in full scale requires air flow rates not generally available except in the largest aerodynamic test facilities. Reduced scale models would probably be acceptable for a purely aerodynamic study, where interaction distances scale with model radius. However, for external burning, the combustion zone length is a second characteristic length, and the ratio of this length to the model radius strongly influences performance. If chemical reaction rates, as opposed to mixing rates, determine the length of the combustion zone, then the above ratio will not scale with model radius, and the results from reduced scale models could be misleading. Accordingly, a semiannular half model was selected to allow the maximum width of the annular air flow passage without reducing the scale of the model.

A diagram of the half-model configuration is shown in Fig. 2. During a test, air flows from the air supply through the nozzle, around the model section, and into the enclosed freejet test section. The fuel, i.e., the fuelrich exhaust from the solid propellant rocket motor mounted external to the test section, is injected into the airstream from the model just downstream of the point where the air enters the test section. The fuel mixes and burns with the air flow in the test section, and the combined flow then passes from the test section into the variable area diffuser. From the diffuser, the air exhausts to the atmosphere. For tests conducted at atmospheric freestream pressure, the freejet enclosure and diffuser are not needed, and the air/fuel mixture exhausts directly to the atmosphere.

The advantages of the half-model configuration over the full model are numerous. First, the required air flow rate is halved. Second, the fuel supply can be mounted external to the model and test section, simplifying the design. Third, the measurement of pressure distributions in the wake downstream from the model is easily accomplished using pressure ports mounted on the base plate extension (see Fig. 2). These pressure distributions have greatly aided the interpretation of test results. Finally, the extraction of temperature and pressure data from the model is simplified.

A possible disadvantage of the half model configuration is that the boundary layer on the base plate extension might provide a communication path between the subsonic near wake and pressure fields further downstream from the base. If this were the case, we would expect to see differences between the base pressure and downstream pressure distribution measured with a half model and those measured with a full model. In Fig. 3, the base pressure and downstream pressure distribution measured with the half model configuration are compared to data obtained with a full model by Reid and Hastings.[2] Except at $x/r = 4$, the agreement between the two sets of data is very good, providing no indication that the boundary layer on the plate affects the test results.

Design Considerations

Proper design of the test configuration is critical. First, if there is a static pressure mismatch between the supersonic freestream and the ambient pressure surrounding the nozzle, either expansion waves or a shock wave will originate at the lip of the outer nozzle wall and run toward the wake. If these waves intersect the wake too near to the model base, they will influence the measured base pressure. This problem is illustrated in

Presented as Paper 73-1226 at the AIAA/SAE 9th Propulsion Conference, Las Vegas, Nev., November 5–7, 1973; submitted November 16, 1973; revision received February 28, 1974. This work was supported in part by the Naval Ordnance Systems Command, Naval Ordnance Station, Indian Head, under Contracts N00174-71-C-0139, N00174-72-C-0105, and N00174-73-C-0240; and in part by the Air Force Rocket Propulsion Laboratory under Contract F04611-72-C-0044.

Index categories: Airbreathing Engine Testing; Airbreathing Propulsion, Subsonic and Supersonic; Combustion in Gases.

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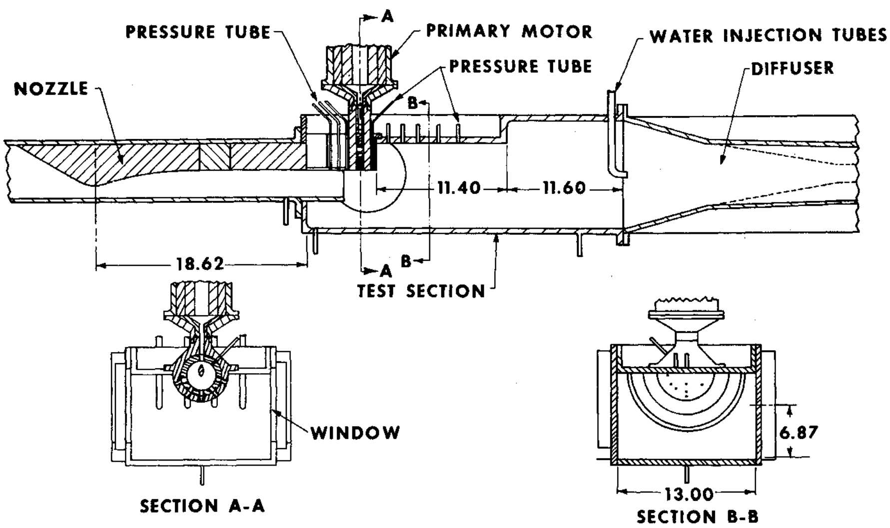

Fig. 2 Half-model test configuration.

Fig. 4, where the measured base pressure ratio is plotted vs the ratio of the pressure measured in the freejet enclosure to the freestream static pressure.

The test model should be designed so that expansion or compression waves resulting from a pressure mismatch will intersect the wake too far downstream to affect the base pressure. Reid and Hastings[2] determined that for a Mach 2 flow a pressure mismatch will not affect the base pressure if the leading waves intersect the wake centerline farther than five model radii from the base. However, this criterion can lead to a large annular nozzle gap width and excessively large air flow rates.

An alternate solution is to maintain a precise match between the freestream pressure and the ambient pressure. This is easily accomplished when the freestream pressure is one atmosphere, and the fuel/air mixture is exhausted directly to the surroundings. However, for tests conducted at subatmospheric freestream pressure, a freejet enclosure and diffuser are required, and the ambient pressure is strongly influenced by the fuel flow rate and the intensity of combustion. Then, active control of the ambient test section pressure is required.

There are two obvious ways to control this pressure. First, a cooling water spray can be injected at the entrance to the diffuser and the ambient pressure controlled by controlling the cooling water flow rate. Second, the diffuser throat area can be varied during the test. The second alternative was chosen here, but for large fuel flow rates, a cooling water spray was also used to reduce the range of throat area variation required. The mechanism by which the test section pressure is controlled by varying the diffuser throat area is described thoroughly in Ref. 3.

The second major design problem is perhaps unique to testing the external burning propulsion concept. Strong shock waves will originate upstream from the injection ports as a result of the transverse fuel injection into the supersonic airstream. These waves will be reflected from the freejet boundary back toward the wake as expansion waves. If the reflected waves intersect the wake too near to the base, they too can affect the measured base pressure. This problem is illustrated in Fig. 5. The leading expansion wave was computed to intersect the wake at the same axial location where a maximum was observed in the downstream pressure distribution. Most likely, these expansion waves limited the measured base pressure rise.

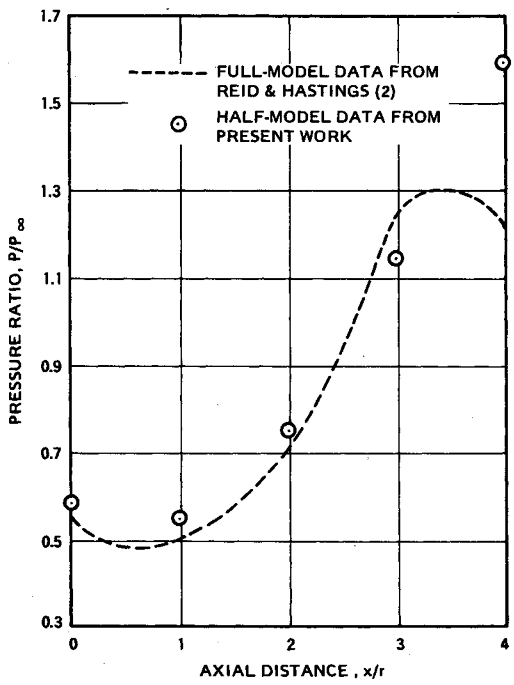

Fig. 3 Comparison of downstream pressure distributions for half and full models.

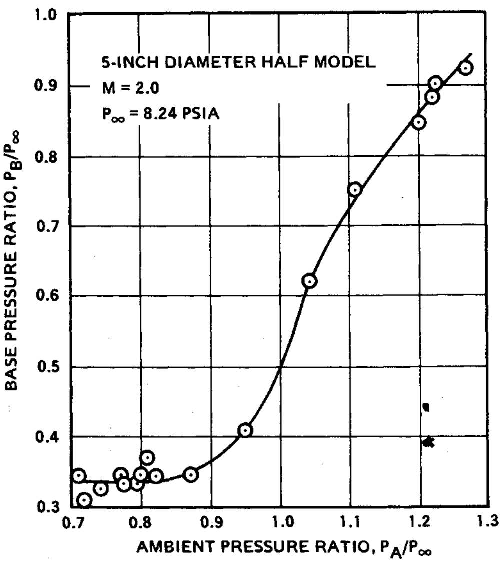

Fig. 4 Base pressure ratio as a function of ambient pressure ratio.

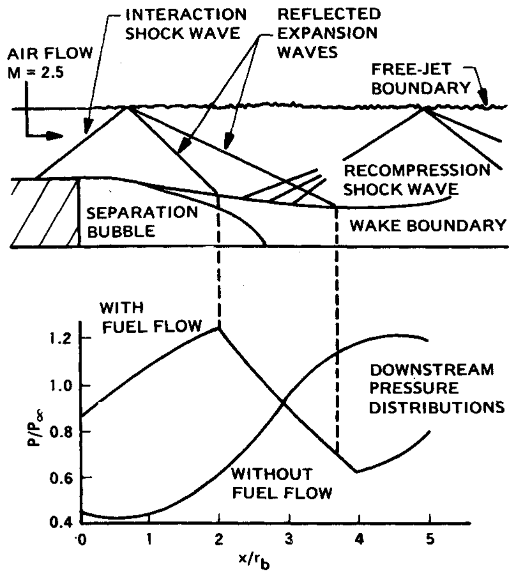

Fig. 5 Interaction shock wave reflection and relation to downstream pressure distributions.

There is no simple solution to this problem other than increasing the nozzle gap width so the leading reflection intersects the wake centerline further than five model radii downstream from the model base. This criterion is easily stated but not easily applied, for prediction of the position and the angle of the leading expansion wave is difficult. Both these parameters are influenced by the fuel flow rate, the nature of the fuel, and the intensity of combustion. Both the position and the angle of the shock wave influence the path of the leading reflected expansion wave.

Test results were satisfactory when an annular nozzle gap of 2.5 in. was used with a 2.5 in. radius model and a Mach 2.0 freestream. However, the gap width cannot be scaled down with model radius because the injection shock wave is curved. As the gap width is reduced proportionally to model radius, the distance between the base and the point of intersection of the shock wave with the freejet boundary decreases more than proportionally to the radius, and the angle of the shock wave at the intersection point increases. Both these effects cause the leading reflected expansion wave to intersect the wake closer to the model base

Conclusions

The external burning propulsion concept can be adequately tested in a small test facility having limited air flow capabilities. The use of a half model appears adequate, but the configuration design requires care to ensure that flow disturbances resulting from fuel injection or a pressure mismatch do not adversely affect the experimental results.

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Viscous Effects in Massively Ablating Planetary Entry Body Flowfields

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Introduction

The large surface ablation rates caused by severe radiation-dominated re-entry heating can significantly alter the boundary layer and hence the heat-transfer and aerodynamic characteristics of planetary entry vehicles, requiring a sound theoretical understanding of strong ablation-blowing effects. An important but heretofore-untreated fluid mechanical aspect of this problem is the nonsimilar flow development *downstream* of the stagnation point, particularly with regard to the thickening of the mixing layer between the ablation and freestream gases, its influence on the pressure distribution, and its possible impingement on the surface with high local heating which terminates the strong blowing regime. As a first step toward providing a complete theory, the present Note describes an approximate engineering treatment of this problem. Complete details are given in Ref. 1.

Outline of Analysis

The problem considered is flow around a massively ablating two-dimensional or axisymmetric hypersonic blunt-nosed body at zero angle of attack (Fig. 1). To simplify the analysis, the radiation terms are neglected and the flow is assumed to be a nonreacting homogeneous perfect gas mixture with a Prandtl number of unity and a constant density-viscosity ($\rho\mu$) product. Furthermore, the Reynolds number is presumed sufficiently large and the blowing rates moderate enough to permit the use of boundary-layer theory and the neglect of transverse curvature effects by using the Mangler approximation $r \simeq r_w(x)$. Although the laminar flow case is treated here, this three-layer model approach can also be applied to turbulent flows. Another important approximation is neglect of the explicit pressure gradient effects on the solution to the shear layer equations in an appropriately transformed coordinate system. This is known to yield reasonably good engineering solutions for pressure distribution and heat transfer around highly cooled hypersonic blunted bodies including mass transfer.[2] Finally, the local inviscid flowfield is approximated by a Newtonian relationship between the inviscid pressure and the local effective body shape defined by

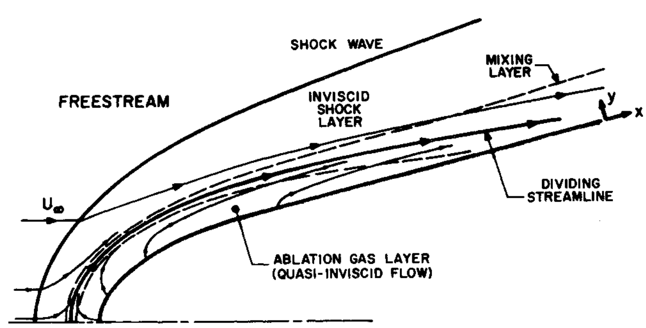

Fig. 1 Massive blowing flow configuration.

Presented as Paper 73-716 at the AIAA 8th Thermophysics Conference, Palm Springs, Calif., July 16–18, 1973; submitted October 10, 1973; revision received April 3, 1974. This work was partially supported by NASA under Contract NAS-10648-18.

Index category: Boundary Layers and Convective Heat Transfer—Laminar.

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