Technical Notes

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Thrust Improvement with Ablative Insert Nozzle Extension

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Introduction

THE Marshall Space Flight Center (MSFC) is investigating the conceptual feasibility of increasing the thrust performance of the Space Shuttle Main Engine (SSME) by using a conical nozzle extension fitted with an ablative insert in order to achieve a low-cost, near-term gain in payload. The ablating insert would provide a controlled increase in nozzle expansion ratio during launch and early climbout (first 30-60 s) so as to reduce thrust loss from nozzle over-expansion in the lower atmosphere. In order for the scheme to be successful the receding surface of the insert must remain aerodynamically smooth to avoid flow disturbances that could cause flow separation, the recession must be circumferentially uniform over the surface of the insert in order to avoid side forces due to nozzle asymmetry, and the rate of recession must be matched to the climbout rate in order to control nozzle over-expansion and its attendant thrust losses.

This Note summarizes a study performed by the Jet Propulsion Laboratory (JPL) in the areas of: 1) defining the near-wall flow environment in the extended nozzle/insert region, 2) selecting potential insert materials, 3) conceptualizing an extension/insert geometrical configuration, and 4) identifying future experimental efforts necessary to verify the feasibility of the concept.

The concept also may be applicable to propulsion systems for vehicle launch from orbital trajectories near planets with dense, high-pressure atmospheres. Of course, specific propulsion requirements would have to be evaluated for such maneuvers.

Concept

The present SSME engine operates at a nominal chamber pressure and temperature of 20.24 MPa and 3892 K, respectively. The nozzle has a throat radius (R_T) of 13.09 cm and an area expansion ratio (ϵ) of 77.5:1. At the nominal oxidizer-to-fuel ratio (O/F) of 6.0:1 (LO_2/LH_2) the engine produces a vacuum specific impulse of 4443 N-s/kg. The nozzle exit pressure is approximately 37.9 kPa which is adequate according to the Summerfield criteria ($P_{exit}/P_{amb} > 0.3$) to provide full nozzle flow at the launch pad.

It is desired to seek a performance improvement by increasing the nozzle expansion ratio in coordination with the Shuttle climbout in order to avoid nozzle-flow separation. An approximately 1.37 m long conical extension, expanding the area ratio to approximately 100:1, could be incorporated within the available space in the gimbal envelope, with only minor adjustments to present engine rigging. The placement of an ablative insert within the conical extension to form a cylindrical port having approximately the present 77.5:1 nozzle exit diameter would provide control of flow separation for launch and climbout during the first 30-60 s of flight.

Figure 1 schematically depicts the extension/insert geometry as outlined above at four stages of insert regression: 1) full insert prior to SSME ignition, 2) ⅔ of the insert remaining, 3) ⅓ of the insert remaining, and 4) insert fully consumed leaving the conical extension only.

Either passive or active ablative materials are conceptually useful in the scheme. The effects of either ablative mode on thrust performance for the overall Shuttle launch and climbout was not quantitatively evaluated, however, those effects

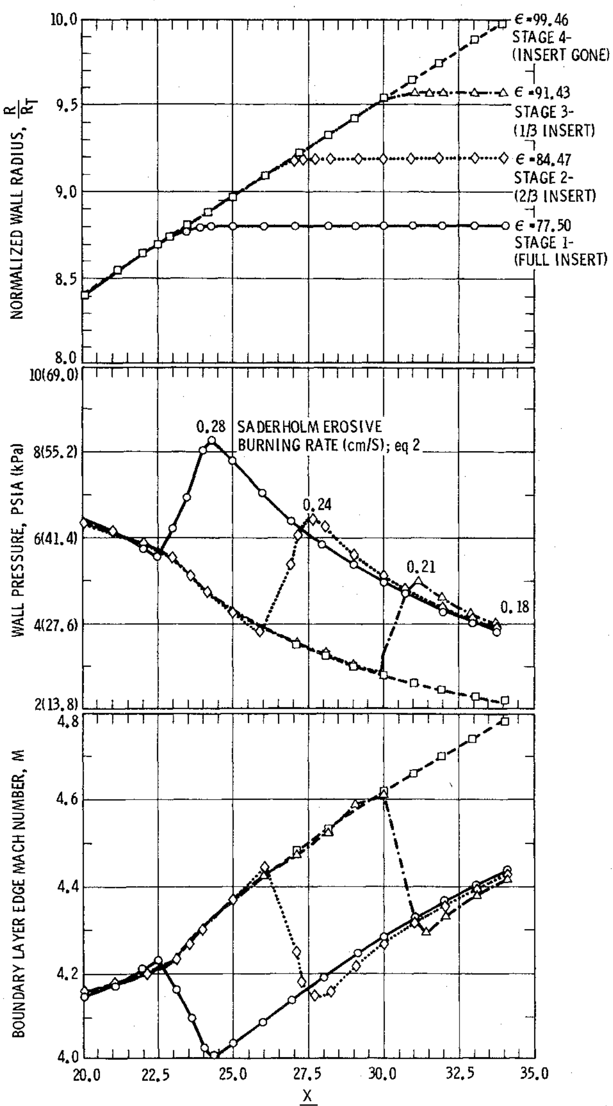

Fig. 1 Nozzle contour depicting four stages of insert regression and wall pressure and boundary-layer edge Mach number vs nondimensionalized axial distance.

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are expected to be relatively small compared to the effect of the area ratio increase for the SSME burn into high altitudes. Experimental evaluation of each ablative mode is necessary to identify beneficial differences in control of the ablation rate, receding surface characteristics, and insert structural integrity.

Analysis

In order to estimate the boundary-layer conditions along the surface of the conceptual SSME nozzle extension/insert, calculations were made by MSFC using the two-dimensional kinetics/boundary-layer module (TDK/BLM) computer codes. The calculations were made for the nominal engine operating conditions, outlined in the preceding section, and for a constant 1111 K wall temperature for the extension/insert. A nonablative wall was used for each of the four insert-regression stages depicted in Fig. 1.

The existing SSME nozzle exit is at $X/R_t = 23.52$ ($\epsilon = 77.50$) where X is axial distance from the throat. The extension/insert exit is at X/R_t of 34.98 ($\epsilon = 99.46$). As shown in Fig. 1, a generous fillet radius of curvature (~86.4 cm) was fitted to smooth the transition from the existing SSME nozzle to the extension/insert.

Boundary-layer edge conditions along the nozzle were calculated from the TDK code (pressure, temperature, density, velocity, and Mach number). Also, local Reynolds number, displacement and momentum thickness, friction coefficient, Stanton number, and thrust decrement were calculated. Predicted wall static pressures and boundary layer edge Mach numbers are shown in Fig. 1.

Although the TDK program does not calculate shock wave properties, the method of characteristics procedure apparently predicts the presence of an oblique compression wave (shock) system, as evidenced by a rapid 19 kPa rise in wall pressure near the junction of the wall with the cylindrical insert (see Fig. 1, stage 1). This wave system evidently progresses steadily downstream as the insert recedes. Note that the wave system is not present once the insert is gone (stage 4). The existence of this shock wave could be deleterious from two considerations: 1) the possibility of local shock-induced separation of the boundary layer, causing circulatory flow in the separation bubble which might increase the local regression rate of the insert, and 2) the possibility of sufficient separation so as to cause the extension not to produce full flow at the exit.

A criterion for the critical pressure rise for boundary-layer separation by shock waves is discussed in Ref. 1. The pressure rise ΔP_s(kPa) to cause turbulent boundary layer separation is

$$\Delta P_s = 2.25 \rho_\infty V_\infty{}^2 / (Rey_\infty)^{0.2} \tag{1}$$

where ρ_∞ is density (g/cm^3), V_∞ is velocity (m/s), and Rey_∞ is Reynolds number at the edge of the boundary layer. For appropriate values of ρ_∞, V_∞, and Rey_∞ from the TDK calculations, ΔP_s is 70.9 kPa for stage 1, 51.2 kPa for stage 2, and 45.1 kPa for stage 3. This suggests that the 19 kPa pressure rise across the present shock is not sufficient to cause local separation. The empirical relation given above for ΔP_s was obtained for an air flow with an upstream Mach number of 3.0.[2] At the higher Mach number of this study the estimated ΔP_s would be somewhat less.

The existence and impact of the shock system discussed above would need to be verified experimentally using techniques such as described in Refs. 3-5.

Assuming it is correctly calculated by TDK, the exit wall pressure of 27.6 kPa for the extended nozzle with full insert (Fig. 1) compares to 30.3 kPa obtained from Summerfield's criteria for full flow. Thus, the extended nozzle geometry might not produce full flow for about the first 900 m of climbout.

An estimate of the maximum average ablation rate required for the 15.3 cm thick insert (exit end) in order to avoid premature overexpansion after lift off is obtained by first noting from Fig. 1 that the exit pressure for the bare conical extension is approximately 15.2 kPa. From Summerfield's criteria, the approximate allowable ambient pressure is 50.3 kPa. The altitude corresponding to this pressure is approximately 5791 m and the time to achieve this altitude is approximately 44 s. Thus, the maximum average ablation rate for the entire thickness of the material is 15.3 cm/44 s = 0.35 cm/s.

With the insert fully consumed (stage 4), a calculated vacuum specific impulse of 4495 N-s/kg represents a potential gain of about 1% over the 4443 N-s/kg presently delivered. This could increase the Shuttle payload by approximately 2631 kg.

Potential Insert Materials

Passive

The primary candidate for the passive ablation mode is a low-density ablative insulation originally evolved at JPL under a NASA technology development program as an internal case insulation for advanced upper-stage motors. This material is a microballon insulation composed of approximately 18% phenolic microballons, 75% polymeric binder, and 7% curing agent by weight. The attributes of this material for this concept include (not necessarily in order of importance): 1) very light weight (density 0.64 g/cm^3); 2) low-cost, available constituents; 3) ease of application to nozzle extension; and 4) potentially dependable regression rate.

The insulation would be cast in place, cured, and machined to the desired contour. Development of a suitable bonding agent is anticipated to be straight-forward.

The microballon insulation would provide a high temperature gradient across its thickness and is expected to regress uniformly and predictably. However, regression rates would have to be determined experimentally since this material has never been used in a high-velocity region. Very little solids residue would remain to coat the nozzle as the flow would tend to scrub them away.

Based on the extension/insert dimensions (Fig. 1) and the density of the microballon insulation, the initial mass of the microballon insulation is 509 kg per SSME engine or 1527 kg total for the Shuttle. This represents only 0.09% of the 1.71 x 10^6 kg of liquid and solid propellant used by the Shuttle, and thus is indicative of the relatively minor effect of the insert consumption mode on overall thrust performance, other than through the area-ratio control that the insert allows.

Active

The primary candidates for the active (solid propellant) ablation mode are from a series of propellants previously investigated by JPL for use in the separation motors for the Shuttle Solid Rocket Boosters. Three of the composite propellant formulations from that series are shown in Table 1 which also shows burning rate and density.

The three formulations were selected on the basis of relatively low ballistic burning rates. An estimate of the erosive burning rate r (m/s) was obtained by application of the

Table 1 Candidate solid propellant insert materials

Ingredient/Property	Propellant SSM-1	SSM-2	SSM-8
Ingredient (wt%)			
NH_4ClO_4	79.00	85.50	78.00
Al_2O_3	—	—	0.50
Fe_2O_3	—	—	0.50
C (graphite)	0.50	0.50	0.50
HTPB binder	20.5	14.00	20.50
Property			
Burning rate, cm/s (@ 10.3 MPa)	1.50	1.45	1.52
Density, g/cm^3	1.58	1.69	1.59

empirical Saderholm Erosive Burning Model[6]:

$$r = 4.45 \times 10^{-7}(MP)^{0.71} \tag{2}$$

where M = local Mach number and P = local static pressure in Pa. These estimates ranged from 0.18 to 0.28 cm/s depending on the surface location as shown in Fig. 1, and are of the same order of magnitude as the 0.35 cm/s maximum average ablation rate required for the extension/insert concept. It is noted, however, that the correlation was obtained only up to high subsonic M and with P only down to 4 MPa, which are far from the conditions in the nozzle being considered ($4.0 < M < 4.4$ and $27.6 < P < 56.5$ kPa); therefore erosive burning rate data for the present application would have to be determined experimentally.

Based on an average density of 1.63 g/cm^3 for the solid propellant and the volume of the full insert, the mass of the propellant is 1306 kg per SSME engine or 3918 kg total for the Shuttle. Again, this represents only 0.23% of the total mass of liquid and solid propellant aboard the Shuttle.

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