Engineering Notes

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Rocket Nozzle Expansion Ratio Analysis for Dual-Fuel Earth-to-Orbit Vehicles

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Introduction

FULLY reusable Earth-to-orbit vehicles will eventually be needed to augment or replace existing vehicles such as the Space Shuttle. Previous studies[1] have identified two candidate vehicle systems. One is a single-stage vehicle, and the other has two stages and uses parallel burn. Both systems use hydrocarbon engines in parallel with hydrogen engines at liftoff. The hydrocarbon engines are shut off before reaching orbit, and the hydrogen engines are used by themselves to complete the ascent to orbit. In a previous study,[1] the hydrogen engine characteristics were based on the Space Shuttle main engine (SSME) with a two-position nozzle. The expansion ratios for the nozzle were 40 and 150.

The purpose of this Note is to report the results of a recent study to analyze the effects of expansion ratios. Recent SSME data and estimates of the characteristics of the SSME with modified expansion ratios are used. Both single-stage and two-stage systems are examined.

Engine Characteristics

The engine characteristics used in this study are based on the SSME at full power level (109% of the normal power level), which produces a vacuum thrust of 2.28 MN at a mixture ratio of 6.0. Characteristics provided by the manufacturer are used to derive the analytic equations presented in the Appendix. These equations, rather than values from the curves provided by the manufacturer, are used to ensure that the engine characteristics are a smooth function of the expansion ratio. The specific impulse and engine mass are shown as a function of expansion ratio in Fig. 1. The data points shown indicate that the specific impulse equation fits the provided data. The engine mass data provided is linear, and the equation represents the curve provided as accurately as the data could be read. The mass data shown is for a fixed powerhead, with constant chamber pressure, mixture ratio, throat area, and propellant flow. Only the nozzle is changed.

The engine characteristics provided by the manufacturer are based on regenerative cooling for the nozzle extension. This approach is required because the heat transfer analysis performed by the manufacturer indicates that a radiation-cooled nozzle would reach temperatures greater than those acceptable for current materials. If a radiation-cooled nozzle could be used, the nozzle mass could be reduced significantly, and the results of this study would be changed in favor of higher expansion ratios.

Analyses and Vehicle Characteristics

The analyses and vehicle characteristics for this study are the same as those for a previous study,[1] with one minor difference: the liftoff thrust-to-weight ratio for the two-stage system was increased from 1.3 to 1.5. Continuing analyses[2] indicate an advantage for the higher value. For the single-stage vehicle, the hydrocarbon engine provides 0.8 of the total thrust, and the hydrocarbon-engine propellant is about 0.66 of the total propellant mass. For the orbiter of the two-stage vehicle, the hydrocarbon engine provides 0.8 of the total thrust, and the hydrocarbon-engine propellant is about 0.71 of the total propellant mass. These thrust and propellant fractions were found to be optimum in Ref. 1 and are not varied in this study. The analyses start with the calculation of a trajectory optimized to provide the maximum burnout mass for the given gross mass for each expansion ratio considered. The specific impulse and exit area of the hydrogen engines vary with expansion ratio and, therefore, affect the trajectory. This approach results in small changes in the time at which the hydrocarbon engines are shut off. The results of the trajectory calculation are then used to size the vehicles to the required payload of 13.6 Mg. The trajectory analysis provides the correct propellant ratios regardless of the vehicle size and does not need to be repeated after the sizing.

In all cases, the hydrogen engines used two-position nozzles. In this study, the lower expansion ratio was varied, while the higher expansion ratio was held constant at 120. Then, the lower expansion ratio was held constant at 40 or 60 while the higher expansion ratio was varied.

Results

The results are presented in the form of vehicle system dry and gross mass required to place a 13.6 Mg payload in orbit as a function of the rocket nozzle expansion ratio. Both the high and low ends of the expansion ratio are varied. Previous studies[3] indicate that the life-cycle costs are minimized by minimizing dry mass, and so the selections are made to minimize dry mass. Minimizing the gross mass is not as important, but the trends can help in understanding the effects of expansion ratio.

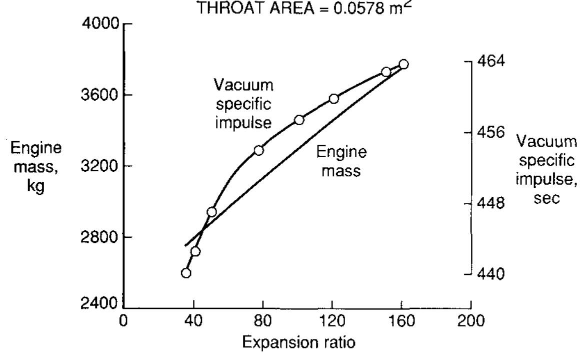

Fig. 1 Effect of expansion ratio on engine mass and specific impulse.

Received Oct. 24, 1986; revision received Nov. 28, 1988.

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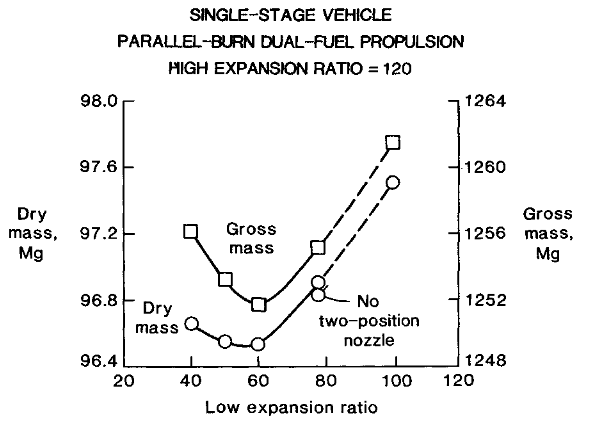

Fig. 2 Effect of low expansion ratio on dry and gross mass for single-stage vehicles.

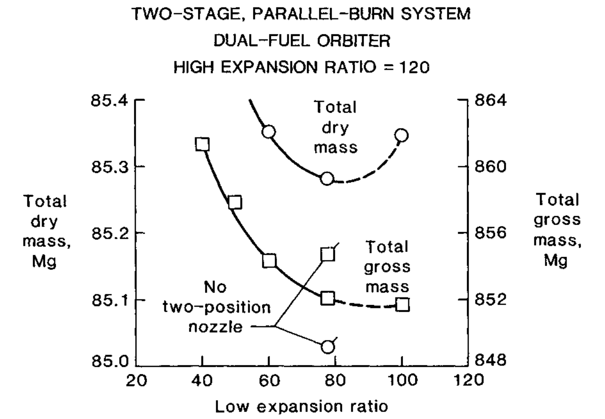

Fig. 4 Effect of low expansion ratio on dry and gross mass for two-stage vehicle systems.

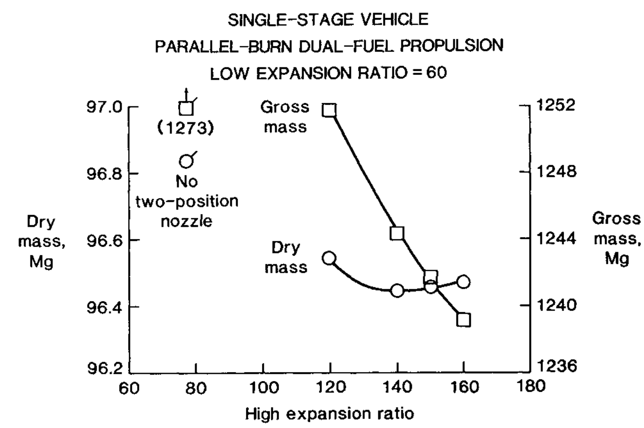

Fig. 3 Effect of high expansion ratio on dry and gross mass for single-stage vehicles.

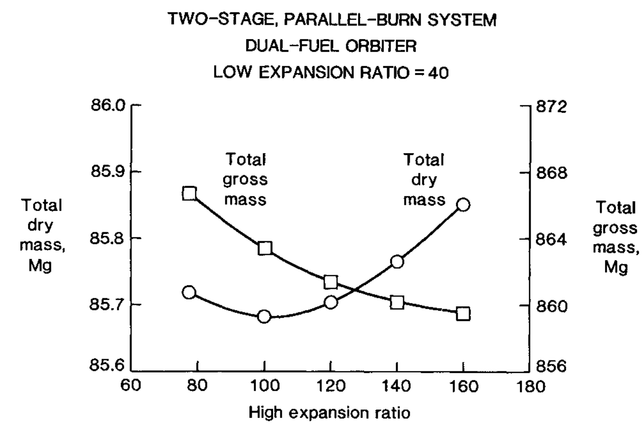

Fig. 5 Effect of high expansion ratio on dry and gross mass for two-stage vehicle systems.

Results for varying the low expansion ratio of a single-stage vehicle are shown in Fig. 2. These results indicate that both gross mass and dry mass are minimized for an expansion ratio of 60. The low expansion ratio does not change the engine mass when the higher expansion ratio is not changed; therefore, the best low expansion ratio is the one that results in the least propellant consumption. This provides the lowest dry mass and the lowest gross mass. The results for the optimization of the high expansion ratio, shown in Fig. 3, are quite different. Increasing the high expansion ratio increases the vehicle dry mass directly, in addition to the indirect impact from sizing the vehicle to the modified propellant consumption. The dry mass reaches a minimum at an expansion ratio of 140, where the effect of nozzle dry mass balances the effect of propellant consumption. The gross mass, on the other hand, continues to decrease with expansion ratio. The minimum gross mass would be at an expansion ratio greater than 160. The results for an engine with a fixed nozzle with an expansion ratio of 77.5 like the SSME are also shown in Figs. 2 and 3. The dry mass is about 0.3% higher than with the two-position nozzle, and the gross mass is off the scale at 1273 Mg. The impact of the high expansion ratio on the propellant consumption considerably outweighs the impact of the mass of the mechanism to extend the nozzle for the single-stage vehicle.

The corresponding results for two-stage vehicle systems are shown in Figs. 4 and 5. The totals of the orbiter and booster dry and gross masses are given. The results for the low expansion ratio are qualitatively similar to those for the single-stage vehicle, except that the minimum dry mass occurs at an expansion ratio of 77.5. The curve is dashed for expansion ratios greater than 77.5 because such nozzles may not be possible because of flow separation at sea level. The results for the high expansion ratio indicate a minimum dry mass at an expansion ratio of 100. Results are also shown for a fixed nozzle. Surprisingly, the dry mass with the fixed nozzle is considerably lower than for the best combination of two expansion ratios, and the gross mass is only slightly greater than with two expansion ratios. This result is quite different from the corresponding result for the single-stage vehicle. Two factors seem to lead to this different result. First, there is only a small difference between the low and high expansion ratios for minimum dry mass. Second, the sensitivity of vehicle gross mass to the high expansion ratio is less. This can be seen by comparing the gross mass curves of Figs. 3 and 5. This reduced sensitivity is an expected result of staging.

In order to understand the impact of the mass of the mechanism that extends the nozzle, a calculation was made with the mechanism mass included but with both expansion ratios at 77.5. The results, compared to the case with no mechanism mass, show that the propulsion system mass of the orbiter increases 151 kg, the orbiter dry mass increases 230 kg, and the booster dry mass increases 59 kg. About half of the increase is the mass of the mechanism itself, and the remainder is the result of resizing. Reducing the mass estimated for the extension mechanism could lead to a two-position nozzle being optimum, but the reduction would have to be by a factor of two or more. Actually, the finding that a fixed nozzle is better is a pleasing result because the two-position nozzle would require some development expense.

Conclusions

The results of this study indicate that for hydrogen engines of dual-fuel vehicles, the expansion ratios for the minimum vehicle system dry mass are different for single-stage and two-stage systems. For single-stage systems, expansion ratios of 60 and 140 in a two-position nozzle provide a minimum dry mass.

The minimum dry mass is about 0.3% less than with a single-position nozzle. For two-stage systems, a single expansion ratio of 77.5 provides a lower dry mass than any two-position nozzle.

The nozzle characteristics are based on regenerative cooling of the nozzle extension. If a radiation-cooled nozzle could be used, the nozzle would have a lower mass, and higher expansion ratios would provide minimum dry weight.

Appendix

The equation for vacuum specific impluse, I_{sp}, is

$$I_{sp} = a(e/60)^b$$

where e is the expansion ratio, a is 450.5 s, and b is 0.004375 for expansion ratios less than 60 and 0.02945 for expansion ratios greater than 60. This equation represents the data as accurately as could be read from the provided curves. It applies only to the case considered in which the engine powerhead is held constant and only the nozzle expansion ratio is varied.

The engine mass m is first calculated in English units because the information supplied by the manufacturer was provided in English units. The first step is to calculate the mass from

$$m = ce + d$$

where c is 17.76 lb and d is 6768 lb. This equation fits the linear curve provided by the manufacturer. It gives the mass for an engine with a single expansion ratio and a vacuum thrust of 600,000 lb. This mass is then scaled to match the SSME mass of 3123 kg by multiplying by 3123 kg and dividing by m for $e = 77.5$. The mass at the thrust level required for the sized vehicle is found by multiplying by the required vacuum thrust and dividing by the SSME thrust of 2293.1 kN. The mass calculated in this manner is correct for a single-position nozzle. For a two-position nozzle, the same mass is correct if the higher expansion ratio is used and a mechanism mass is added.

For a two-position nozzle, the mechanism mass r is calculated from

$$r = f(e^{0.5} - 1)p$$

where f is 0.0455 s and p is the propellant flow rate, which can be calculated by the thrust divided by the specific impulse and, for r in mass units, by the acceleration of gravity. This equation was not derived from the recent data provided for this study. For the SSME at full power level and for an expansion ratio of 150, the mechanism mass is given by this equation to be 264 kg.

Acknowledgment

The author wishes to express appreciation to Mr. Frank Kirby of the Rocketdyne Division, Rockwell International Corporation, for providing the data used to derive the engine specific impulse and mass equations without which this study could not have been accomplished.

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Shock Wave Asymmetry of Sphere Cones at Angles of Attack

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Introduction

As was pointed out several years ago by Greenberg and Traugott,[1] in the case of flow over a blunted cone at an angle of attack, conditions exist where, depending on the Mach number and the cone angle, the shock layer (the region of flow confined between the shock wave and the body) on the windward side will be thicker than that on the leeward side. They further point out that it was Gonor[2] who postulated the existence of two types of flow for a circular cone at angle of attack in supersonic/hypersonic flow: a "normal" or "regular" regime in which the leeward shock layer is thicker than the windward shock layer and a "reversed" flow regime in which the opposite is true. The existence of these types of flow regimes was stated to be dependent on the Mach number of the flow and the effective semicone angle (see Fig. 1). Figure 1 is reproduced from Ref. 1. It must be pointed out in this context, however, that Gonor's[2] paper seems to contain some errors, viz. that the theoretical expression for the effective semicone angle θ^* does not reproduce the theoretical curves for the larger semicone angles θ_0 as shown in his Fig. 1 at $M = 4$. This fact has also been alluded to by the "Reviewer's Comment" at the end of that paper. Be that as it may, existence of "regular" and "reversed" regimes seem to be confirmed by Greenberg and Traugott.[1] In the experimental situation, θ^* may be defined as $\theta_0 \pm \alpha$ where α is the angle of attack of the cone.

Greenberg and Traugott further point out that this phenomenon is also true for spherically blunted cones, and they present a photograph in support of this conclusion. This short Note shows further evidence in support of their conclusion and adds the observation that this flow feature becomes quite prominent in hypersonic, high enthalpy streams.

Experiments and Discussion of Results

Tests on a spherically blunted cone of semiangle 13 deg (θ_0) at various angles of attack α were conducted in the Australian National University free-piston driven shock tunnel T3[3] with nitrogen as the test gas. The flow was generated by a conical nozzle with exit and throat diameters of 203 and 12.7 mm, respectively. The freestream was dissociated nitrogen with speeds (U_∞) varying between 6 and 8 km/s and a Mach number M_∞ of approximately 6. The corresponding reservior enthalpies h_0 ranged from 30–52 MJ/kg, respectively. Since the freestream was dissociated nitrogen, freestream flow conditions at the exit plane of the nozzle were calculated using a computer program based on the method of Lordi et al.[4] for nonequilibrium gas expansions.

Self-luminosity photographs, taken with a fast electromechanical shutter (speed ≈ 800 cm/s) especially developed for these experiments,[5] were used to study shock waves around the body. Some of these photographs are shown in Fig. 2. A considerable effort was spent to ensure that the observed shock shapes are caused by test gas only and not by helium

Received Aug. 3; revision received Oct. 24, 1988.

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