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INTERACTIONS BETWEEN THE EXTERNAL FLOW AND ROCKET EXHAUST NOZZLE

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ABSTRACT

The problem areas of nozzle separation, base burning, and jet pluming associated with the rocket exhaust and external flow interactions are reviewed from an aerodynamic viewpoint, assuming that the operating conditions correspond to missile or space vehicle launch and boost. The study indicates that the separation characteristics of nozzles having area ratios up to 25:1 will not be significantly influenced by the external flow. It is also shown that interaction of the external flow and rocket exhaust gas can result in combustible mixtures in the base region. One example of base burning is cited. The large effect of the exhaust jet plume on afterbody pressure distributions is discussed for high altitude operation. Finally, design compromises are suggested to alleviate the aforementioned problems.

INTRODUCTION

Exhaust nozzles have received considerable attention over the years, e.g., references 1 and 2. However, most of these studies are limited to the internal flow characteristics of the exhaust nozzle and thus neglect the strong interactions of the external flow on the exhaust nozzle performance or the effect of the exhaust flow on the external flow.

A limited amount of work on the interactions of the exhaust jet and external flow for the ramjet and turbojet application has been reported (ref. 3) but little information, reference 4, exists for the rocket configuration. A partial explanation for this state of affairs is related to

the complicated flow interactions which occur in the base region as a fixed afterbody and exhaust nozzle combination operates over a speed range of from takeoff to hypersonic conditions and a nozzle pressure ratio of 40 to possibly 10,000. Such operating requirements involve the problem areas of nozzle separation characteristics at less than design pressure ratio, the base burning due to entrainment of unburned fuel in the base region, and flow separation on the external body due to jet pluming. Since the interactions are so closely related to the hardware geometry no general quantitative answer is possible; however, it will be the purpose of this report to discuss the considerations involved in the selection of the various design compromises.

DISCUSSION

Nozzle Separation

The proper perspective for discussing the problem areas cited previously can be obtained from a review of the calculated characteristics of a series of fixed geometry convergent-divergent nozzles operating over a wide range of pressure ratios. Figure 1 presents the variation with nozzle pressure ratio of the thrust to ideal thrust ratio for 15° conical nozzles having area ratios of 8 and 25:1 as well as an isentropic nozzle of area ratio 25:1. These calculations are based on air flowing through the nozzles and discharging to quiescent air.

At the low pressure ratios the 8:1 nozzle has the highest thrust ratio since it is operating near its design pressure ratio of 100. As the pressure ratio is increased to 1000 the thrust ratio of the 8:1 nozzle decreases due to underexpansion and significantly higher thrust ratios can be obtained with 25:1 area ratio nozzles having design pressure ratios on the order of 500. Unfortunately severe penalties are incurred at the low pressure ratios with the 25:1 nozzles due to overexpansion of the flow. The problem then is to design a nozzle, preferably fixed geometry, which has the thrust advantage of the 8:1 nozzle at low pressure ratios and the advantages of the 25:1 nozzle at high pressure ratios. Satisfactory solution of this problem and application to a satellite booster stage missile would increase the second stage payload 25 percent.

Since the thrust penalties shown in figure 1 at less than design pressure ratio were calculated by assuming some over expansion of the nozzle flow, it may be argued that these penalties are excessive. Thus determination of nozzle separation characteristics is very important and has been the subject of many investigations since Prandtl's work in 1907. A bibliography of this work would be too long to discuss here, but two examples might be references 5 and 6. Unfortunately with all of this quiescent data it is not possible to predict nozzle separation characteristics with mathematical or engineering certainty. Additional uncertainties are introduced when the effect of external flow are considered.

The complications due to the external flow are illustrated by the sketch of the local flow conditions shown in figure 2. Qualitatively the base region represents the zone of adjustment or interaction between the external flow and the exhaust nozzle flow. It is here that two streams having different boundary layer conditions, velocities, temperatures, and pressures mix and interact with each other. Knowledge of the base pressure is of course a prerequisite to estimating whether the nozzle flow will overexpand or separate, since the base pressure is the effective discharge or back pressure for the nozzle. For example, at low pressure ratios, the

flow in the nozzle can overexpand until the wall pressure (p_1) becomes less than the base pressure (p_2 or p_b). This continues until a critical value of base to nozzle pressure is reached (p_2/p_1) (ref. 7), at which condition the flow breaks down and separates from the nozzle wall. This of course is equivalent to a change in local flow direction and results in the oblique shock shown and across which occurs the pressure rise (p_2/p_1). The expansion of the external flow and separation of nozzle flow occur in such a manner that the flow conditions are satisfied in the wake region where the two flows are mixed to pressure (p_3) by means of the trailing shock. Thus the determination of the base pressure requires the simultaneous solution of the external and internal flow conditions as is so ably demonstrated in reference 8. The base pressure, in turn, represents the nozzle back pressure and as such is related to the nozzle separation characteristics.

The base pressure variation for a representative configuration having a nozzle area ratio of 25:1 and body to exit diameter ratio of 2.0 has been calculated by the method of reference 8. The results are summarized in figure 3 as the variation with Mach number of the nozzle pressure ratio referenced to base pressure (p_b) and stream pressure (p_o). As would be expected the base pressure results in a significant shift of the nozzle operating condition at moderate supersonic speeds. Whether this has a significant effect on the nozzle thrust at less than design conditions depends on the nozzle separation characteristics. Unfortunately there is much disagreement in the literature (refs. 1, 2, 5, and 6) on nozzle separation pressure ratios so a range of separation pressure ratios having a mean value of 0.4 design is shown in figure 3. The important point is that this corresponds to operation in the speed range where base pressure is about equal to stream pressure and thus no large base pressure effects can be anticipated. Furthermore it is questionable that any separation and thus reduction in thrust loss in the subsonic speed range can be assumed for nozzle area ratios of 25:1 or less. This is particularly true if contoured rather than conical nozzles are used, since unpublished data indicate that overexpansion to about 0.2 of design can be obtained with contoured nozzles. Thus high thrust losses due to overexpansion in high area ratio convergent-divergent nozzles is indicated.

A logical question at this point might be: "What techniques can be used to maintain high thrust over the operating pressure ratio range." Recognizing that the thrust penalty is due to overexpansion of the flow in the nozzle one approach would be to force separation by means of secondary air injection along the nozzle wall. For best performance this means a range of secondary weight flows must be provided, such as occurs in the airplane ejector nozzle. Figure 4 schematically illustrates the application of the airplane ejector principle (ref. 9) to the rocket exit problem. In principle the nozzle would be aerodynamically self adjusting, i.e., at low speeds and altitudes large quantities of secondary flow would pass through the ejector (fig. 4(a)). This secondary airflow in turn prevents the overexpansion of the rocket exhaust and thrustwise the performance corresponding to an 8:1 nozzle is obtained. As higher speeds and altitudes are reached the rocket exhaust pressure increases relative to the ambient pressure and the secondary flow automatically decreases (fig. 4(b)). This results in thrust performance corresponding to a 25:1 area ratio nozzle during the later portions of the flight.

Another way of looking at the nozzle overexpansion problem is that the nozzle wall shelters or isolates the internal flow from the ambient pressure thus permitting overexpansion. This nozzle wall can be eliminated, as far as isolating the internal and external flows is concerned, by using the plug type nozzle shown in figure 5. Admittedly the plug nozzle like the convergent-divergent nozzle can be influenced by base pressure, however overexpansion to one-fifth of the design pressure ratio at take-off is impossible since the outer boundary of the nozzle exhaust flow is an unrestrained and adjustable stream line, i.e., the expansion occurs externally. This fact has been demonstrated by a number of quiescent air investigations, such as reference 10. Assuming the same base pressure conditions as were calculated previously for figure 3 the performance of the plug and convergent-divergent nozzles are compared in figure 5 as the variation of nozzle thrust ratio with flight Mach number. It is obvious from this figure that a 25:1 area ratio plug nozzle has the inherent potential of operating with negligible thrust losses over the operating pressure ratio range or stated another way it combines the thrust advantages of the 8:1 nozzle at low pressure ratios with the advantages of the 25:1 area ratio at the high pressure ratios.

Base Burning

The mixing or interaction of external flow and exhaust nozzle flow in the base region introduces another problem, namely base burning. The problem is illustrated qualitatively in the hypothetical flow conditions illustrated in figure 6. It has been accepted for many years that the so-called dead air space at the base includes a trapped vortex and references 8 and 11 utilized this concept in predicting base pressures. The flow model shown assumes that there is a continuous interchange of air across the outer and inner separating stream lines due to mixing and that a vortex pattern as indicated is established. Potentially this increases the base temperature above any radiative heat considerations in two ways: first, hot exhaust gas is circulated into the base region and secondly the fuel-rich rocket motor exhaust gas is mixed with air from the external flow to provide a possible combustible mixture. The combination has in some cases resulted in very high temperatures and failures of hardware located in the base region due to overheating. The amount of combustibles introduced into the base region will not only be a function of the previously discussed parameters which affect base pressure such as base heighth and relative velocities of external and internal flow but, in addition, the oxidant-fuel ratio of the rocket motor. Therefore to illustrate how base heating can occur it will be arbitrarily assumed that 3 or 6 percent of the mixture in the base consists of exhaust gas from the rocket motor. From these assumptions and a simple heat balance consideration the base temperatures indicated in figure 7 were calculated as a function of oxidant-fuel ratio for JP-LOX rocket motor. Temperatures in the 300° to 500° F range are estimated simply due to mixing the exhaust gas with the external air. However, when it is assumed that the mixture is ignited in the base and all of the heat available in the exhaust gas is released, temperatures of 1000° F to more than 2000° F are estimated for the usual oxidant-fuel operating range of 2.0 to 2.5. This argument is not advanced as a rigorous study but merely to indicate the significance of the problem; however, burning does occur when the base serves as a flameholder and when hot sparks or some other ignition source exists for a particular configuration. Figure 8(a) shows a JP-4 - LOX rocket engine operating with

supersonic external flow with no evidence of burning in the base. In fact thermocouple measurements indicated temperatures only slightly higher than stream values. In figure 8(b) the oxidant-fuel ratio has been decreased. The hazy region indicates base burning, and this was confirmed by a recorded temperature of about 1000° F.

One obvious means of counteracting the base heating problem is to provide some sort of insulation. However, this will result in a weight penalty and is not considered to be an adequate solution. Figure 9 indicates some alternate techniques which could be considered for convergent-divergent nozzle installations. Part (a) is a flush base configuration and corresponds to the model shown in the previous figure. Since the entrainment of the unburned fuel in the base is a function of the mixing length, retracting the body base relative to the nozzle exit (fig. 9(b)) must reduce the mixing length and also the entrainment of unburned fuel. Adding base bleed as indicated in figure 9(c) might have three advantages, namely the local velocity would be too high for combustion to seat, the fuel-air mixture would be too lean to ignite and finally, the source of ignition could be chilled and eliminated. Boattailing and retracting the base as indicated in part (d) eliminates the mixing zone and the base area; however, such extreme steps would also subject the rocket nozzle to external stream loads. This, of course, would affect the motor structure and for a gimballed motor would increase the actuating force requirements.

Two other techniques could have been added to figure 9. They are the plug and ejector type nozzles which were suggested previously to maintain high thrust at less than design pressure ratio. These nozzles have little or no base area or mixing length for exhaust gas entrainment. Accordingly, in addition to the off-design thrust advantage, the ejector or plug nozzles have the further advantage of alleviating the base burning problem.

Jet Pluming

So far, the discussion has considered problems associated with the early portion of the vehicle's flight. As the flight progresses and higher altitudes are attained the problem of the rocket exhaust flow interfering with the external flow due to jet pluming is encountered. This of course will be a function of the motor characteristics and vehicle flight plan. For purposes of discussion figure 10 presents the assumed variation of the vehicle altitude and nozzle pressure ratio (H_c/p_o) for a chamber pressure of 600 pounds per square inch absolute. With this as a starting point the jet pluming characteristics can be determined by using reference 12. The results of these calculations are shown in figure 11 as the variation of initial jet angle, θ_i, with nozzle pressure ratio. The estimated ratio of jet plume diameter to rocket motor throat diameter is also presented. At 200,000 feet altitude the nozzle pressure ratio (H_c/p_o) is more than 100,000. This results in a jet turning angle of about 75° and a final jet diameter which is 70 times the rocket motor throat diameter. Or stated another way the jet diameter is about ten times the body diameter for a representative vehicle. As indicated by the inserted sketch, the jet plume may act as a forward facing step to the air flowing along the external surface. This, of course, will cause flow separation on the body and change the aerodynamic loading. Depending on the center of pressure - center of gravity relationship of the vehicle this can result in an increase or decrease in vehicle stability. The extent of the

separation and the magnitude of the pressure in the separated region is of course a prerequisite for analysis of this problem. A semianalytical approach could be used based on work such as that presented in reference 13; however, the major portion of these data apply to two-dimensional separation due to forward facing steps and its application to the three dimensional case is open to question. Furthermore, the estimated pressure rise across the oblique shock due to separation varies from 4 to 16 depending on the reference data used. Accordingly, exploratory experimental programs have been initiated and an example of the results is presented in figure 12 **at a low angle of attack.** The model is a strut supported blunted cone cylinder model utilizing high pressure nitrogen to simulate the exhaust nozzle discharge. Due to flow and pressure limitations in the supply system it was necessary to install a plug in the exit of the model to obtain the desired nozzle pressure ratio simulation. The resulting sonic discharge velocity does not provide exact simulation but is believed to be adequate. At a pressure ratio of about 250 (fig. 12(a)) there is a slight increase in pressure on the body near the exit and a very weak oblique shock is formed on the leeward side. When the pressure ratio is increased to about 6000 (fig. 12(b)) the schlieren photo indicates that separation extends forward to the model shoulder and significant changes in pressure occur on the body. For example on the bottom or windward side, the pressure near the base indicates a pressure rise of about three. More importantly this pressure rise is not constant in the separated region but decreases rapidly with distance from the exit, indicating that simplified analytical approaches may not be adequate.

The previous discussion has shown that the initial angle of the exhaust jet plume and thus the extent of the separation is dependent on the nozzle pressure ratio (H_c/p). For a given chamber pressure and nozzle area ratio the controlling parameter becomes the ratio of the static pressure at the nozzle exit to the local external static pressure. This immediately suggests two techniques for reducing the jet pluming, namely increase the local external pressure or decrease the nozzle exit pressure. The local external pressure can be increased by adding an expanding conical section or flare to the afterbody. The pressure rise will of course be a function of the flare angle. The nozzle exit pressure can be decreased by increasing the nozzle area ratio, however this will result in high thrust losses at low pressure ratios if a convergent-divergent type nozzle is used. On the other hand, the ejector or plug type nozzle discussed previously can satisfy the additional requirements of reduced static pressure at the nozzle exit.

SUMMARY

The problem areas of nozzle separation, base burning, and jet pluming as influenced by the rocket exhaust and external flow interactions have been reviewed from an aerodynamic viewpoint for operating conditions corresponding to missile or space vehicle launch and boost. It was demonstrated that:

1. For the flight path assumed, nozzle separation characteristics are not significantly influenced by external stream effects for nozzle area ratios up to 25:1.

2. The interaction of the external and internal flows can result in the entrainment of enough unburned rocket fuel in the base to form combustible

mixtures. For example, in one case this mixture did ignite and raise the base temperature to about 1000° F.

3. Significant changes in the afterbody pressure distribution were caused by jet pluming and shock-boundary layer interactions at higher flight altitudes.

It is also indicated, from aerodynamic considerations, that the use of plug or ejector type nozzles having effectively high area ratios (25:1) may alleviate not only the thrust penalty at less than design pressure ratios due to the lack of nozzle separation but also the unfavorable aspects of base burning and jet pluming.

APPENDIX - SYMBOLS

A	area
C_F	thrust coefficient
C_{Fi}	thrust coefficient at perfect expansion
D	diameter
H	total pressure
M	Mach number
θ_i	initial angle of jet plume
p	static pressure
γ	ratio of specific heats

Subscripts:

b	base
c	rocket chamber
e	exit
j	jet
o	free stream
t	throat
1	initial condition or ahead of shock
2	conditions after shock

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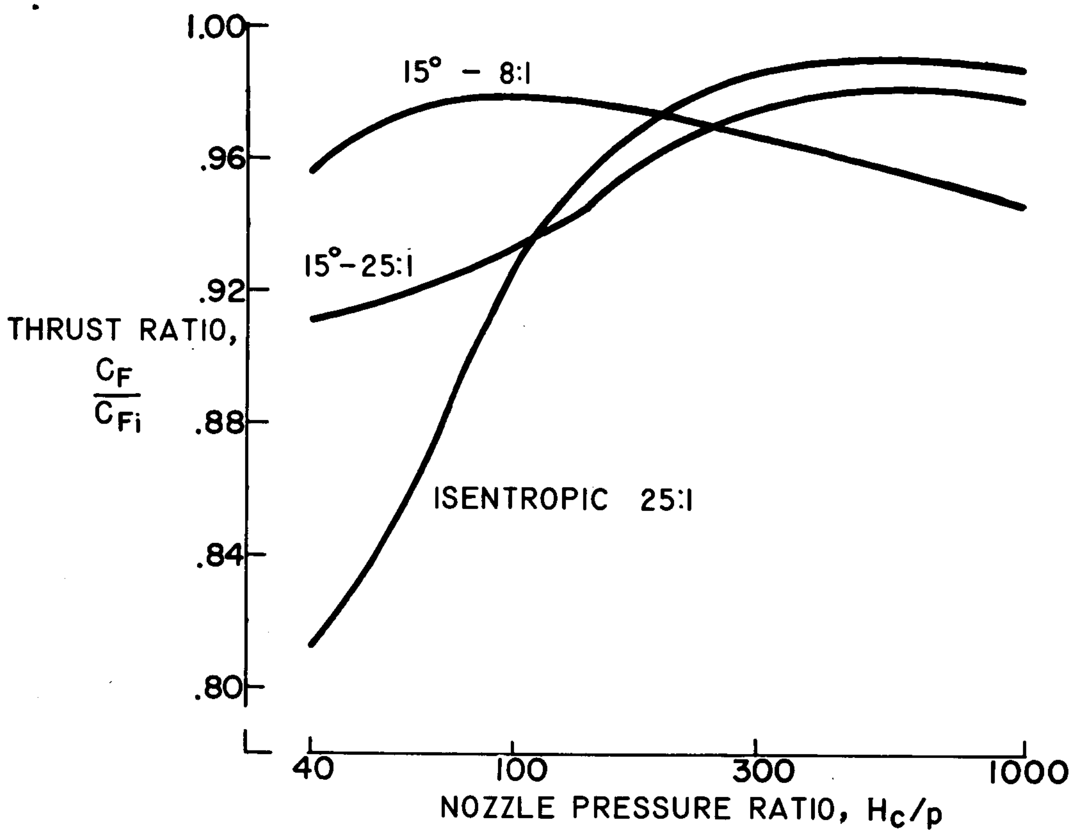


FIGURE 1 - Nozzle thrust characteristics.

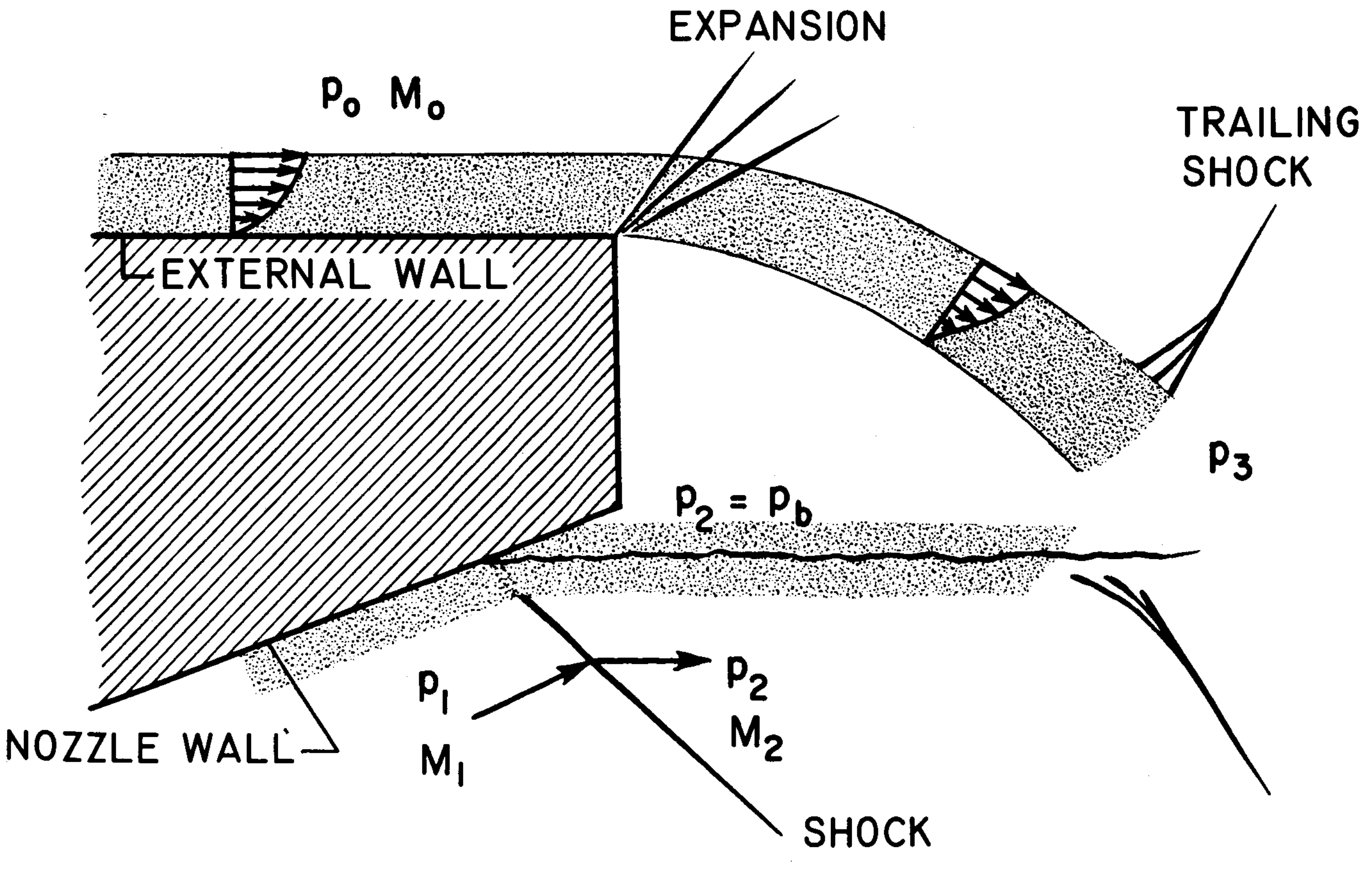


FIGURE 2 - Flow model of base region.

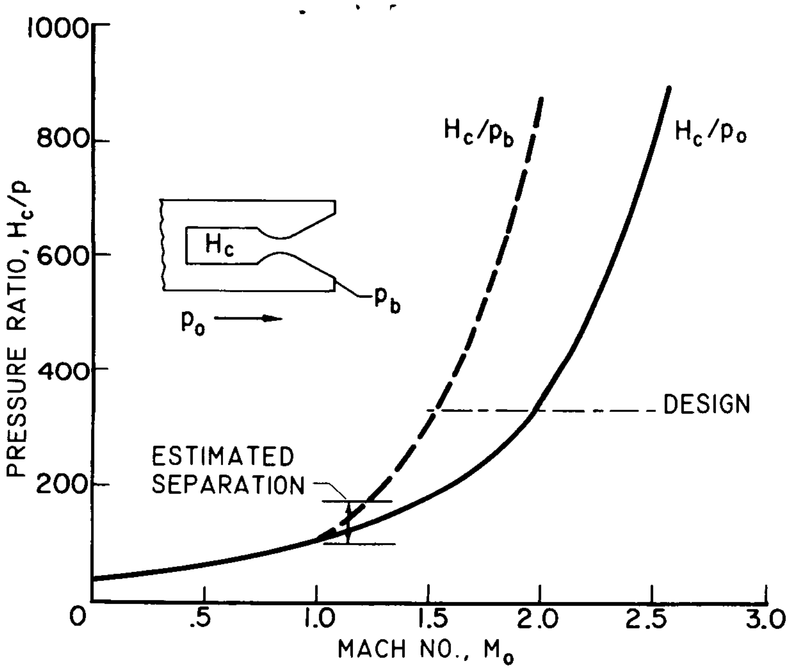


FIGURE 3 - Effect of external flow on pressure ratio

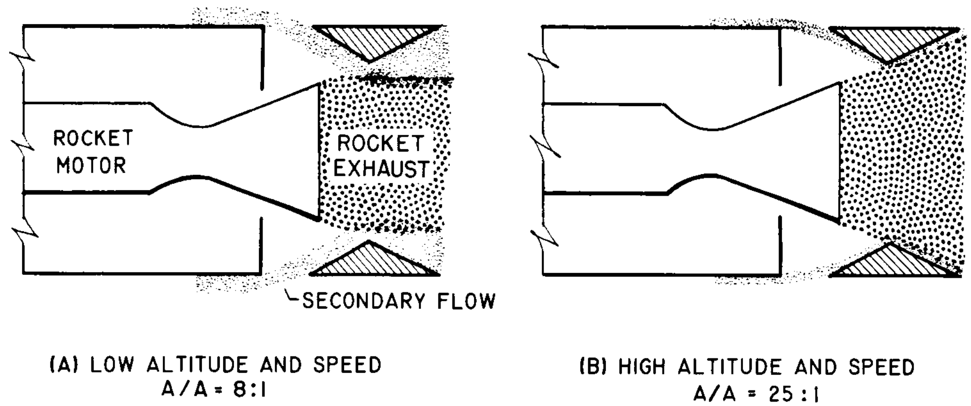


FIGURE 4 - Rocket ejector.

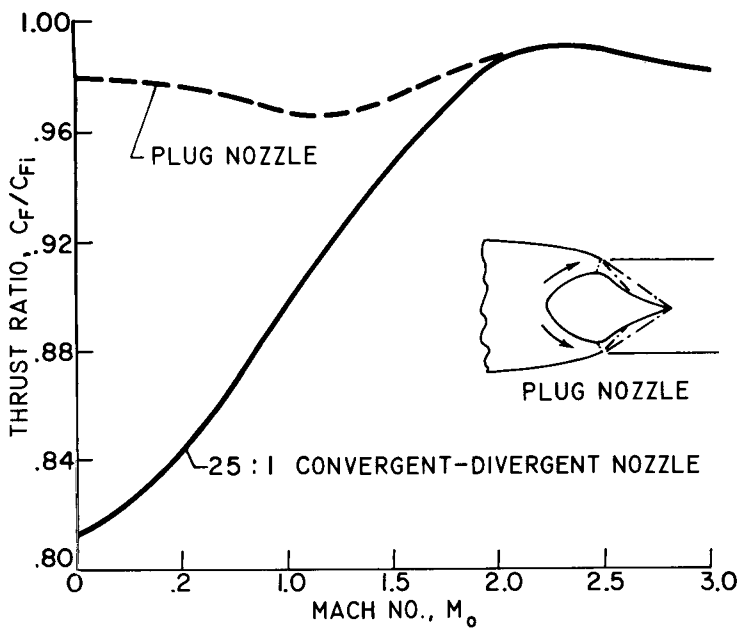


FIGURE 5 - Plug nozzle characteristics.

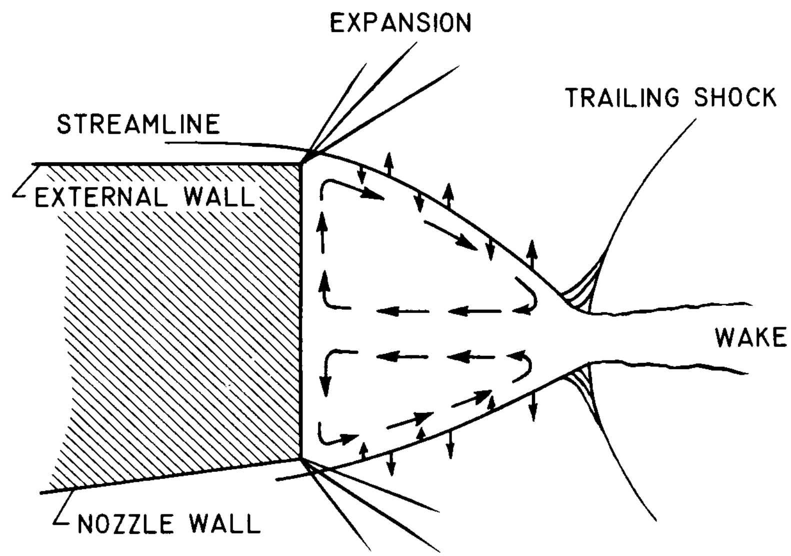


FIGURE 6 - Base vortex flow.

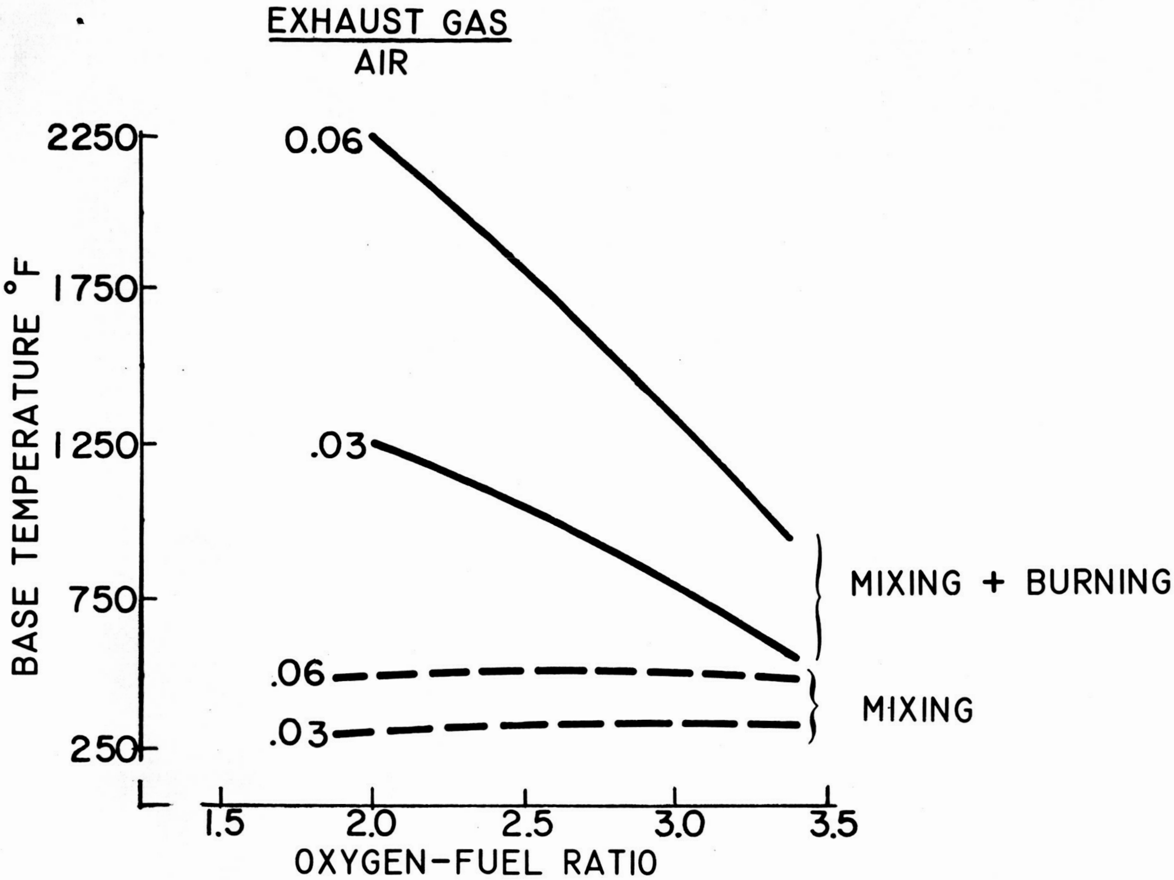
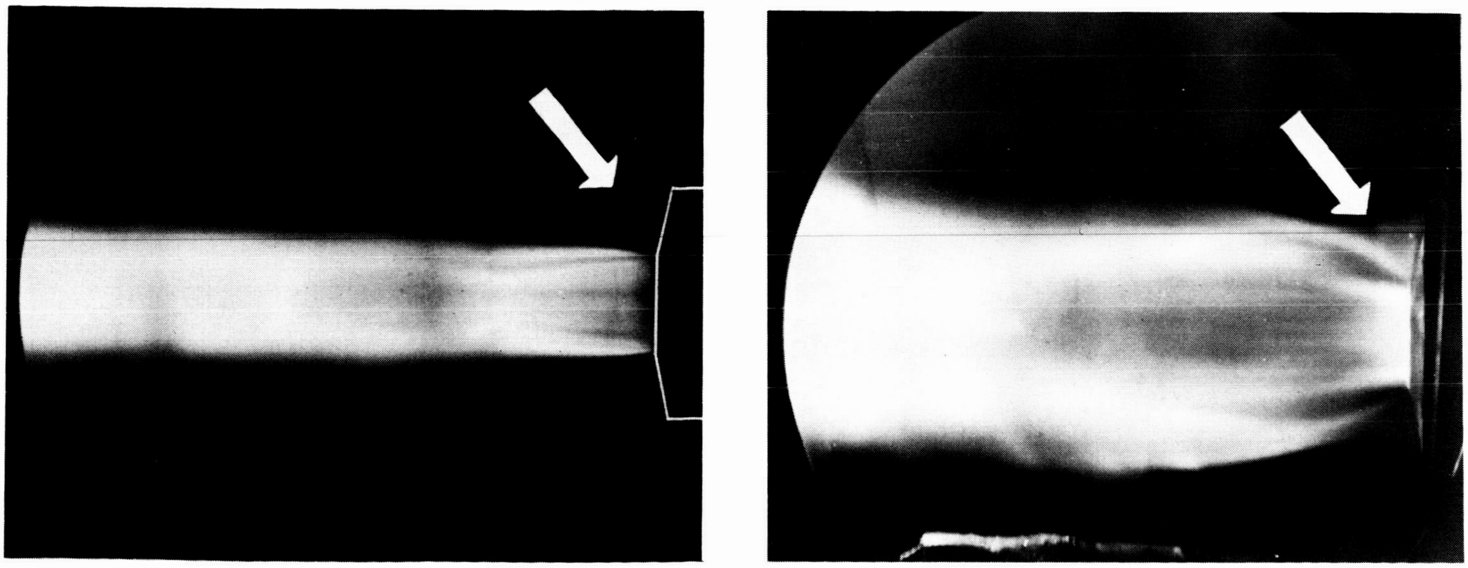


FIGURE 7 - Estimated base temperature.



(A) NO BURNING

(B) BURNING

FIGURE 8 - Base region - rocket burning. E-537

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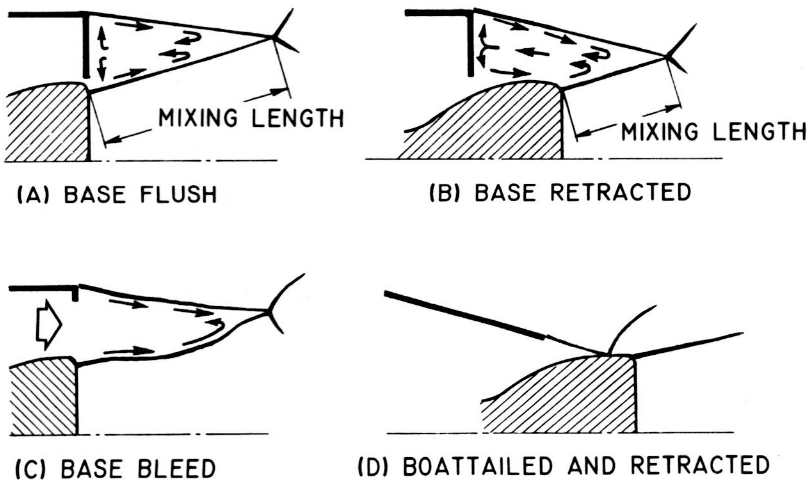


FIGURE 9 - Base configurations.

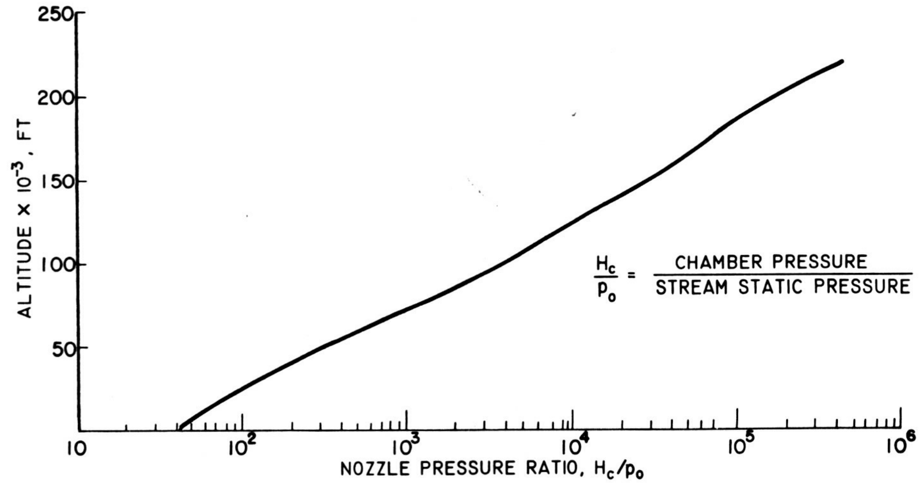


FIGURE 10 - Flight path.

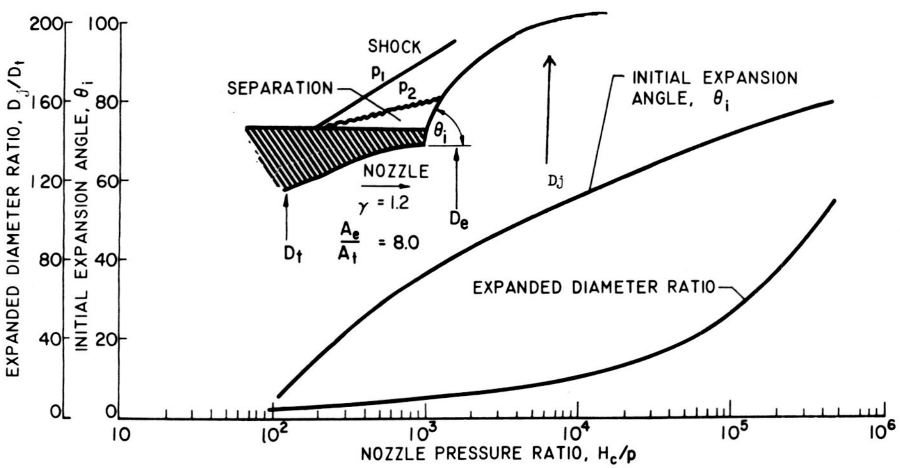
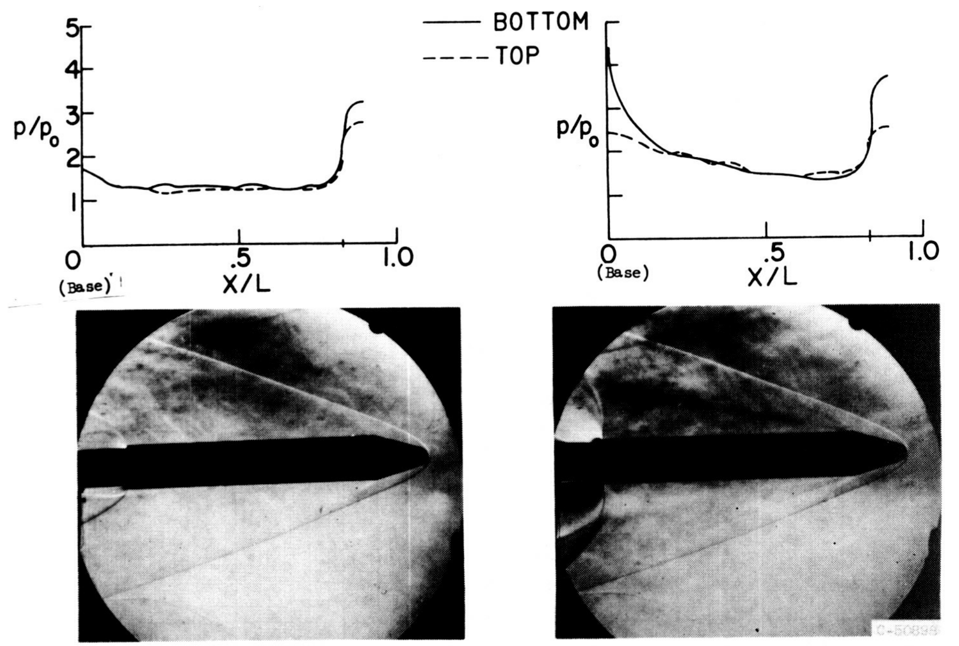


FIGURE 11 - Exhaust jet characteristics.



(a) $\frac{Hc}{P_o} = 250$

(b) $Hc/p_o = 6000$

FIGURE 12 - Exhaust jet pluming.

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