ANALYSIS OF INCREASED COMPRESSION FACTOR ON EJECTOR-ROCKET PERFORMANCE

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Abstract

The present paper examines the benefits of increasing the compression ratio within the ejector section of a simple ejector/rocket. Relying on the ejector effect alone, the obtainable thrust is compared to that of both a hydrogen and kerosene fuelled first stage rocket engine typical of those used for space launch. At static sea level conditions, it is found that increasing the compression ratio is more beneficial than increasing the entrainment ratio in terms of thrust augmentation, while the benefit is reduced at higher flight altitude and speed. The hypothesis of using exit area constriction as a means of increasing ejector performance is also examined.

1 Introduction

One way of incorporating the benefits of airbreathing into rocket based launch vehicles is through the use of an ejector system. This idea is central to the development of Rocket Based Combined Cycle (RBCC) engines, where it is the ejector effect which is primarily responsible for any increased performance over traditional rocket systems during the initial phases of launch. A typical RBCC engine would transition between a) ejector, b) ramjet, c) scramjet, and d) pure rocket modes. However, even in the absence of high speed propulsive modes which include the combustion of atmospheric oxygen (i.e., ramjet and/or scramjet operation), a RBCC with an ejector mode has the potential to improve performance through the entrainment and compression of atmospheric air even at low speeds, acting to increase the total massflow through the engine and increase the thrust produced.

The thrust augmenting potential of ejectors has been studied as far back as 1949 with the work of Von Kármán [20]. This type of theoretical treatment is often focused on accurately estimating the air inflow conditions under the various ejector operating regimes (see Fabri and Siestrunck[13]), as the ratio of entrained air to rocket exhaust massflows, also referred to as the entrainment ratio (α), is a key operating parameter. For the case where the exit pressure of the ejector is sufficiently low, the entrained air is capable of choking rapidly within a short distance from the ejector entrance. Under these conditions, numerous theoretical methods have been developed to assess the resulting flowfield. Dutton and Carroll[8, 9] examine ejector configurations in which, in addition to the entrainment ratio, other relevant parameters such as the compression ratio (the ratio of the total pressure at the ejector exit plane to that of the entrained air, π_m) are maximized. They also examine the limitations imposed on this operating regime due to the influence of the second possible operating regime, that of choked exit flow (due, for example, to heat release within the ejector section).

These two ejector parameters, α and π_m, define a given ejector's performance. Since $\alpha = 0$ represents a pure rocket, a minimum value for this parameter is required for an ejector/rocket engine to be considered “airbreathing”. Given the ejector's main function as a jet pump, the maximization of the compression ratio is also a key performance indicator, even in applications where the ejector is used in non-propulsive systems (Emanuel[10]).

To improve the accuracy of the theoretical treatment for lower entrainment ratios while also providing a means of determining the flow details within the ejector section itself (not simply at the inflow and outflow planes), Chow and Addy [3] develop a method where one can approximate the viscous effects within the shear layer between the air and rocket streams. This allows for the estimation of the massflow passing through the mixing layer to be added to the entrained air massflow calculated based on the assumption of no mixing up to the choke point. They also show that the theory provides a reasonable estimation of the wall pressure within the ejector as compared to experimental results, while Chow and Yeh[4] show that this theory can be extended to parabolic ejector sections (in addition to a constant area ejector). At higher entrained air massflow rates, Peters et al.[18] develop a method that accounts for the effects of turbulence within the mixing layer as well as allowing for equilibrium chemical reactions to occur within the ejector section (as would be expected when the core rocket is operating under fuel rich conditions). The inclusion of chemical reactions and hence heat release during the mixing process forces the ejector into the second main operating regime, where the entrainment ratio is no longer dominated by upstream choking but rather choking further downstream within the ejector. This mode of operation is also examined by Masuya et al.[16] where chemical equilibrium calculations are added to a one dimensional flow model using an experimentally determined wall pressure distribution.

More recently, the flowfield within ejectors has been examined numerically, especially as it relates to the incorporation of the ejector within the larger class of RBCC engines. Matesanz et al.[17], using an explicit, finite element based algorithm and a $k\varepsilon$ turbulence model, examine a simple axisymmetric ejector with a single rocket placed along the axis. West et al.[21] and Ruf[19] examine a similar configuration but with the addition of downstream hydrogen injection (as would occur in an RBCC engine operating in a diffusion and afterburning mode, see Daines and Segal[5]) using a pressure based reacting flow solver (seven species, nine reaction, H_2/O_2) and an extended $k\varepsilon$ turbulence model. More complicated rocket arrangements have also been studied, where Daines and Merkle[7] compare a single central rocket along the axis to an annular rocket located so as to evenly divide the entrained air massflow, while Daines and Bulman[6] study the effect of dynamically switching the angle of the rocket exhaust entering the ejector section (both solve the Favre averaged Navier-Stokes equations with the $k\varepsilon$ turbulence model of Chen and Kim). In each case, the entrainment ratio and the level of thrust augmentation is found to increase over the simple single central rocket configuration.

However, even as a stand alone propulsive device, Alperin and Wu [1] show that when compared to a pure rocket, a constant section ejector followed by a propulsive nozzle is capable of producing increased levels of thrust. Consistent with the nature of ejector operation, they also note that for best performance a high degree of compression within the ejector is critical. Therefore, this paper examines the benefits of increasing the second major performance parameter of an ejector, the compression ratio, on the thrust produced by an ejector/rocket engine. This is done using a theoretical framework in which one can solve for the ejector inflow conditions under the third main operating regime, that in which the back pressure within the ejector duct is sufficient to influence the entrained air inflow conditions. Similar to the work of Alperin and Wu[1], this also includes the second operating regime where the exit flow is choked (as this corresponds to a unique back pressure producing sonic mixed flow at station 2 in Fig. 1). Once the ejector inflow conditions are obtained, the effect of increasing the compression ratio is evaluated on the thrust augmentation of the ejector/rocket over the thrust of the same rocket in isolation. The use of a constricting area ejector section is also proposed as a means of obtaining this increase in the compression ratio following the hypotheses suggested by Makaron and Fedyayev[15] and Escher[11], and a limited analysis of the potential implications of this configuration is performed.

2 Ejector Theory

There are two main modes of operation for an ejector/rocket engine, differentiated by the behaviour of the incoming air. During the initial stages of launch when the flight Mach number is subsonic, the incoming air is entrained into the ejector duct by the action of the rocket exhaust. This pumping action acts to increase the total massflow through the ejector duct and the engine is said to be operating in *ejector rocket* or simply *ejector* mode. However, as the flight Mach number is increased to values beyond Mach one, the air inflow is generally determined by external conditions such as flight Mach number and inlet shock structure (unless at some downstream point within the ejector the conditions are sufficient to unstart the inlet). In this case, the engine is considered to be operating in *air augmented rocket* or *ram rocket* mode.

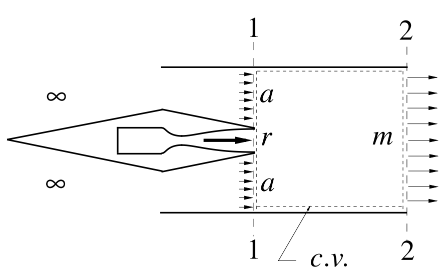


Fig. 1 Ejector section of an ejector/rocket or RBCC engine

For the cases in which the engine is operating in ejector mode, the critical performance parameter is the incoming air Mach number (M_a), as it is this "airbreathing" aspect of the ejector that offers improved performance over pure rocket systems. To determine this value it is often convenient to assume an aerodynamic choking condition, where due to the expansion of the high pressure rocket exhaust into the entrained airstream the subsonic incoming air encounters a converging area streamtube and hence accelerates towards Mach one, sometimes referred to as a Fabri choke (Aoki et al. [2], Kanda and Kudo [14], Etele [12]). However, this presupposes two conditions: (a) the static pressure in the rocket exhaust is sufficiently higher than that in the entrained airstream at the ejector inlet so that the rocket exhaust expands into the airstream and thus creates a converging air streamtube and (b) the conditions downstream of the ejector are such that they do not influence the conditions at the inlet. This can lead to a restriction in the variety of scenarios that can be examined, especially those where conditions further downstream of the ejector can have a significant impact on the entrained airflow (i.e., heat addition). Under these circumstances, a more general model is required.

If the stagnation conditions entering the ejector, the ejector inlet geometry, and the rocket exhaust Mach number entering the ejector are all known quantities (T_a^o, T_r^o, p_a^o, p_r^o, A_a, A_r, and M_r), in addition to the gas composition of both the rocket and air streams (γ_a, γ_r, C_{p_a}, and C_{p_r}), then one can define two equations in which there remain only three unknowns. Two of these unknowns are Mach numbers, the first being the Mach number of the entrained airstream (M_a) while the second is that of the mixed flow (M_m). The third unknown is the mixed flow pressure (p_m), which reflects the fact that information downstream in the engine can influence the properties at the ejector inlet (unlike the case of a choked entrained airstream where the necessity of prescribing a downstream parameter is eliminated through the assumption of $M_a = 1$ shortly after entering the ejector).

Since M_m appears as a squared term in both equations, the result will be two possible solutions for a given massflow ratio $\alpha = \dot{m}_a/\dot{m}_r$. However, the additional constraint of a specific mixed flow pressure will uniquely determine which of the two Mach number solutions is obtained (high static pressure = subsonic mixed flow, low static pressure = supersonic mixed flow).

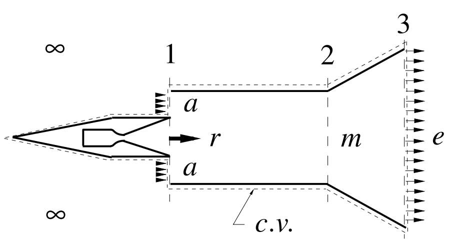


Fig. 2 Control volume surrounding entire ejector/rocket engine

To calculate the thrust of an ejector/rocket as shown in Fig. 2 one can write,

$$\mathcal{T} = \underbrace{A_e(p_e - p_\infty) - A_a(p_a - p_\infty)}_{\mathcal{T}_p} + \underbrace{\dot{m}_j\left[(\alpha+1)u_e - \alpha u_a\right]}_{\mathcal{T}_m} \quad (1)$$

where both pressure forces and a change in momentum contribute to the propulsive force of the engine. Specifying the mixed flow static pressure yields a solution for M_a as previously described, while setting the exit plane pressure (p_e) yields the flow Mach number leaving the engine through,

$$M_e^2 = \frac{2}{\gamma_m - 1}\left[\left(\frac{\pi_m p_a^o}{p_e}\right)^{\frac{\gamma_m - 1}{\gamma_m}} - 1\right] \quad (2)$$

where here the compression ratio has been introduced, which represents the ratio of mixed flow to entrained air total pressures,

$$\pi_m = \frac{p_m^o}{p_a^o} \quad (3)$$

With Eq. 1 one can calculate the total thrust of a rocket/ejector by specifying only two variables, the static pressure at both the end of the ejector section and the engine exit plane (p_m and p_e respectively). It should be noted that in specifying both p_m and p_e we are implying that the nozzle shape is a variable, since the ejector exit area, which is the same as the nozzle entrance area (station 2, Fig. 2), is a set parameter. Under the assumption of isentropic flow through the nozzle (Eq. 2) there will be a unique expansion ratio required to produce the given exit pressure specified. Therefore, for a given p_m, as we alter p_e we are altering the expansion ratio between stations 2 and 3 in Fig. 2. In fact, for cases where p_m is high and $M_m < 1$, a low exit pressure (p_e) requires the flow through the nozzle to go from subsonic to supersonic speeds thereby implying a converging/diverging nozzle shape from station 2 to 3.

The thrust of a pure rocket can also be found from Eq. 1 by setting $A_a = \alpha = 0$ (i.e. a pure rocket will have no air entrainment area and hence no massflow of air), where it is also assumed that the frontal area of the rocket equals the nozzle exit area. The increase in thrust using an ejector/rocket as compared to a rocket alone can then be evaluated using the thrust augmentation ratio,

$$\Phi = \frac{\mathcal{T}_{ejector}}{\mathcal{T}_{rocket}} \quad (4)$$

3 Results

The term "airbreathing" implies a massflow of air passing through the engine, thus one of the key requirements of the ejector section is to increase this massflow as much as possible. The lower the value of α, the more the ejector/rocket system resembles a pure rocket, and hence the less of an airbreathing engine one obtains. Therefore, the first criteria an efficient ejector must possess is a minimum value of α, which in this paper will be taken as between 0.75 and 1.00. A second criteria also exists in that, in addition to simply entraining air, the ejector must also act on this air in a manner that increases the potential thrust obtainable from the flow leaving the ejector section.

In cases where the ejector is a component within a larger Rocket Based Combined Cycle (RBCC) engine, this requires that the flow exiting the ejector be as close to fully mixed as possible. In this type of engine fuel is often added in an afterburning type application so as to utilise the oxygen contained within the entrained air (a configuration referred to as diffusion and afterburning). While uniform burning and heat release require mixed flow entering the afterburning section, performance can also be increased by increasing the total pressure of the flow leaving the ejector. Higher values of compression ratio indicate that the transfer of energy from the rocket exhaust stream to the mixed flow has been accomplished with fewer losses. Even in cases where the ejector/rocket is used on its own without any further fuel addition (i.e., in an application similar to that shown in Fig. 2), increasing the compression ratio can have a significant impact on engine performance.

To assess this impact two different rocket configurations are examined, representative of both the Delta IV and Atlas E/F first stage launch engines which are hydrogen and kerosene fuelled respectively. The rocket exhaust temperatures in Table 1 are based on equilibrium combustion calculations for the various fuels at the total pressures and equivalence ratios listed. In the case of thrust and specific impulse, both the values at standard sea level conditions and those under vacuum conditions are calculated to illustrate the importance of $\mathcal{T}_p$, where the only difference between the two conditions is the drag force resulting from the net pressure difference between the surrounding atmosphere and the nozzle exit pressure (an effect which is absent under vacuum conditions, see Eq. 1).

Two operating conditions are examined. The beginning of a launch cycle is represented by a static sea level condition where the ejector effect would be strongest. A condition just prior to the point at which the vehicle reaches Mach one is chosen as a representative location where past which (i.e. higher and faster), the ejector effect would begin to diminish significantly and the engine would operate more as a ram rocket with the air inflow conditions being determined by the flight conditions as opposed to the ejector effect itself. Both conditions are listed in Table 2, along with the major ejector parameters resulting from the combination of rocket/flight conditions. It should be noted that the ejector outer radius is set so that $\sigma = A_r/(A_r + A_a) = 0.1$ in each case, independent of the Mach number (and hence A_r) required to match the static pressures between the rocket and air streams at the ejector inlet.

Table 1 Rocket configurations

Variable	Delta IV [(1)]	Atlas E/F [(1)]
Fuel	Hydrogen	Kerosene
ϕ	1.33	1.49
p_r^o [MPa]	9.73 (96 atm)	4.86 (48 atm)
T_r^o [K]	3634	3668
Area Ratio	21.5	25.3
$\mathcal{T}$ [kN] (vacuum)	2,948 (3,344)	276 (386)
I_{sp} [s] (vacuum)	382 (433)	240 (336)

(1) Rocket parameters approximating the first stage of the launch system

Table 2 Ejector operating parameters

Variable	Delta IV	Atlas E/F	Atlas E/F
Altitude [m]	Sea Level	Sea Level	6,300
M_∞	0	0	0.8
p_a^o [kPa]	101.33	101.33	69.04
T_a^o [K]	288	288	279
$\sigma = A_r/A_m$	0.10	0.10	0.10
$\theta = h_a^o/h_r^o$	0.021	0.038	0.037
$\zeta = p_r^o/p_a^o$	96	48	70

With the variables in Tables 1 and 2, all the required information to solve the ejector flowfield is known except for the mixed flow pressure and the exit pressure. The mixed flow pressure is used to control the entrained air massflow rate, and thus closes the solution for a particular ejector condition. However, for a given p_m, it is desirable to have the rocket exhaust conditions (M_r, p_r) configured to allow for the maximum massflow of entrained air, which occurs when the rocket exhaust Mach number is adjusted so that the static pressure ratio between the two streams entering the ejector is unity (i.e., there is no expansion of the rocket exhaust into the airstream). For the launch systems examined here, this requires raising the rocket exit pressure and thus reducing the Mach number by changing the area ratio as compared to the values listed in Table 1. Keeping the remaining parameters constant results in a throttling effect for each case, where for the Delta IV configuration, the vacuum thrust and specific impulse decrease by approximately 3%. However, since a lower rocket exhaust Mach number creates both an increased static pressure over the nozzle exit plane in addition to a reduced exit plane area, both of these factors contribute to a reduction in the pressure drag exerted on the rocket and the net effect at sea level conditions is an *increase* in the total thrust by 2%. Therefore, in comparing ejector configurations to the rocket alone at static sea level conditions, if the throttled rocket thrust is used to evaluate Φ, then the results can be considered conservative with respect to the actual launch system (i.e., without alteration to achieve a specific rocket exhaust pressure).

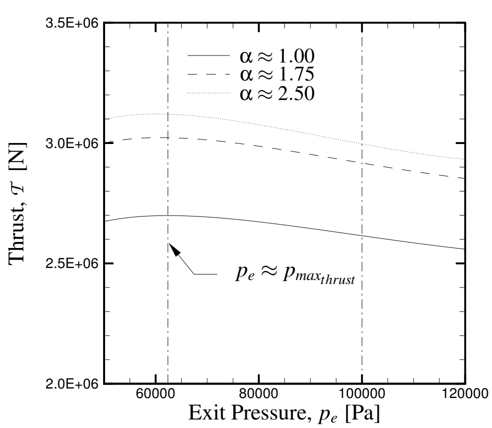


Fig. 3 Variation of ejector thrust with exit pressure (and hence nozzle geometry) for a matched static pressure condition between rocket and air streams at ejector inlet

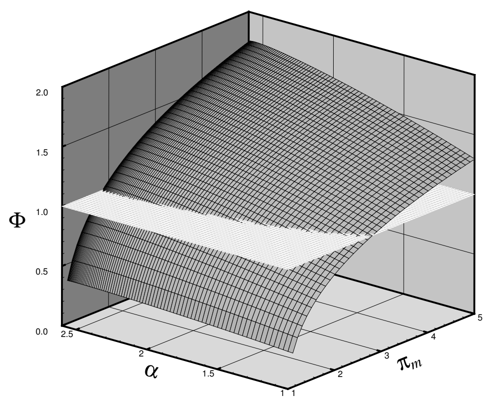
Using the throttled rocket conditions ($p_r/p_a \approx 1$), the effect of varying the exit pressure at the engine exit plane (station 3, Fig. 2) on the total thrust is shown in Fig. 3. As can be seen, there exists an optimum exit pressure in terms of producing the greatest thrust over the range of massflow ratios considered, however, at approximately 65 kPa this is well below the ambient pressure at sea level. If one sets the exit pressure to 100 kPa (i.e., the flow is approximately expanded to the local ambient pressure at sea level), although this does not coincide with the optimum exit pressure, the net effect is to decrease the overall thrust augmentation as shown in Fig. 5, thereby adding a conservative effect.

With the engine exit pressure chosen, as well as the base rocket configuration against which the ejector/rocket is to be compared, it is possible to evaluate the effect of increasing the compression ratio within the ejector section on the overall engine performance. Figure 4 shows these results, using both a subsonic and supersonic mixed flow solution at the mixed flow plane. Although these surfaces represent the complete range of possible solutions, only those configurations which yield $\Phi > 1$ actually increase the thrust produced over the same rocket used in isolation. For both cases it is seen that at a given massflow ratio α, increasing the compression ratio within the ejector section increases the overall thrust augmentation. A subsonic mixed flow produces the largest region of solutions above the $\Phi = 1$ plane, where not only is this region reduced in size for the supersonic solution, but the minimum value of π_m required to simply match the stand alone rocket thrust is dramatically increased over that required for a subsonic mixed flow at the same value of α.

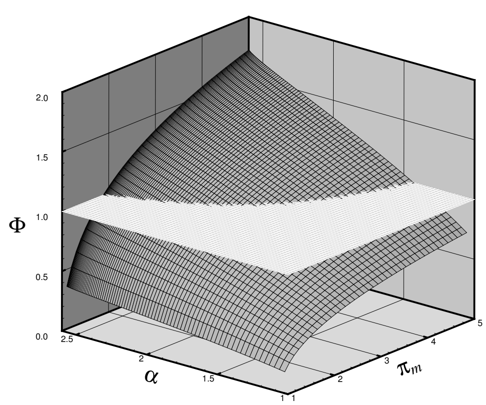
Figure 4 also shows that increasing the entrainment ratio increases thrust augmentation. In fact, at a particular compression ratio (i.e., that which would be obtained in a straight ejector section), there exists a minimum entrainment ratio that must be achieved in order to increase the engine thrust (again, this minimum value is higher for the supersonic mixed flow solution). Therefore it is instructive to consider how these variables are, and can be, manipulated. In the case of the entrainment ratio it is the mixed flow pressure which is varying. As shown in Fig. 5, as one increases p_m from very low values (low α), the entrainment ratio varies along the supersonic branch of the curves eventually reaching a maximum near 2.6. As this pressure continues to increase, the entrainment ratio begins to decrease again following the subsonic branch down to values below unity. Therefore, if one chooses to vary α in an effort to increase the thrust augmentation, an upper limit on α will exist coinciding with the condition at which the flow chokes leaving the ejector (i.e., the point separating the subsonic and supersonic branches in Fig. 5).

This is true irrespective of the choice of the other operating conditions as illustrated by the results for the ejector based on the Atlas E/F first stage rocket engine (Fig. 6). At static sea level conditions, the reduced combustion pressure of this rocket configuration acts to decrease the total pressure ratio, ζ, by approximately 50%, while the change in gas properties due to the difference in fuels results in an increase in θ of approximately 81% (see Table 2). However, as can be seen, there still exists a maximum value of α at approximately 3, and similar to the previous results, the subsonic mixed flow surface has the largest region of solutions which result in an increase in thrust.

In terms of increasing the compression ratio, one possible method could be the use of a constricting area ejector section. Unlike manipulating the massflow ratio, constricting the ejector area has no theoretical limit, although a practical limit exists in that one cannot reduce the exit



(a) Subsonic mixed flow at station 2 (Figs. 1 and 2)



(b) Supersonic mixed flow at station 2 (Figs. 1 and 2)

Fig. 4 Thrust augmentation for an ejector based on a Delta IV launch rocket (static sea level conditions, p_e = 100 kPa)

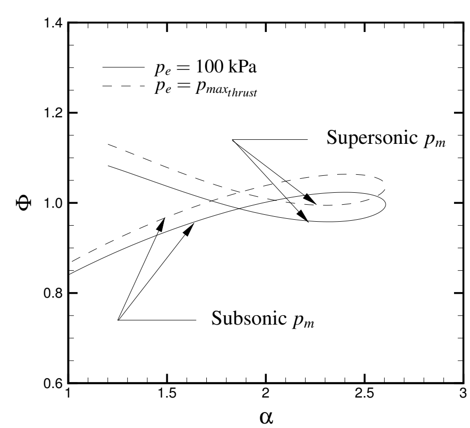


Fig. 5 Ejector thrust augmentation with varying entrainment ratios (as obtained by varying the ejector exit pressure p_m)

area beyond the point at which one can no longer pass the massflow of rocket exhaust entering the ejector. At this point α would be zero and hence an even more useful limit would be the area at which α is reduced to some arbitrary minimum value (i.e., 0.75-1.00). Using this minimum as a limit, there would then exist a maximum amount of constriction that could be applied, and hence a maximum increase in the compression ratio that could be reasonably assumed.

The use of a reduced ejector exit area implies a reduced α compared to a similarly configured straight ejector, since the rocket conditions are fixed leaving only the entrained air massflow rate to vary. However, as shown in Figs. 4 and 6, reducing α reduces the thrust augmentation, which acts to offset some of the benefit of increasing the compression ratio. To assess which of these two parameters, α or π_m, would be more beneficial to increase, the results shown in Fig. 6 are reduced to a two dimensional $\alpha - \pi_m$ plane in Fig. 7. The solid line represents the case of a straight ejector, where for a given α a fully mixed flow would produce the particular compression ratio indicated (the left branch represents a subsonic mixed flow solution, the right a supersonic mixed flow). The two broken lines represent the intersection of a given Φ plane with the solution surface in Fig. 6, where both the $\Phi = 1.0$ and $\Phi = 1.1$ traces are shown.

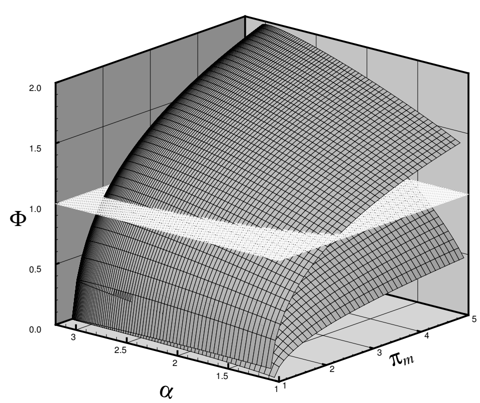


Fig. 6 Thrust augmentation for an ejector based on an Atlas E/F launch rocket (static sea level conditions, $p_e = 100$ kPa). Upper surface = subsonic, lower surface = supersonic

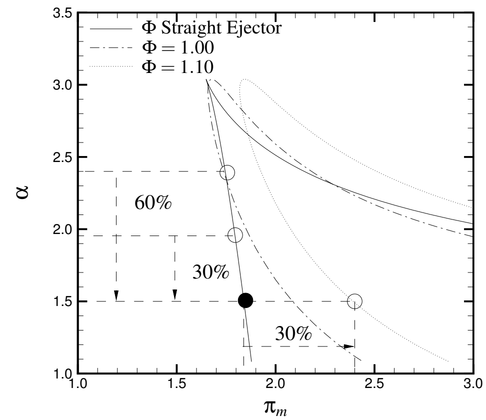


Fig. 7 Effect of changing α and π_m on thrust augmentation (Atlas E/F, static sea level conditions, $p_e = 100$ kPa)

If one starts with a straight ejector at an entrainment ratio of 1.5 as indicated by the solid circle in Fig. 7, one can see that the resulting thrust is actually less than that of the stand alone rocket (i.e., it lies to the left of the $\Phi = 1.0$ curve). However, if one constricts the ejector area so as to achieve a 30% increase in the compression ratio, it is possible to produce a 10% increase in thrust as shown by the open circle on the $\Phi = 1.1$ curve. If the constricted ejector is operating at maximum massflow conditions (i.e., the flow is choked), this implies that the equivalent straight ejector is operating at less than maximum entrained air capacity and could in fact yield a massflow ratio in excess of 1.5. Depending on the relationship between the degree of constriction and the associated decrease in entrained air massflow, if one assumes the constriction required to achieve a 30% increase in the compression ratio results in a 30% reduction in α, restoring this massflow of air to the straight ejector still yields a net decrease in thrust as compared to the rocket alone (i.e., the point still lies to the left of the $\Phi = 1.0$ curve). In fact, as previously discussed there exists a minimum entrainment ratio required for the straight ejector to match the thrust of the equivalent stand alone rocket, which in this case is approximately $\alpha = 2.4$ (the intersection of the solid line and the $\Phi = 1.0$ curve). As shown in Fig. 7, even if the ejector constriction required to obtain a 30% increase in π_m produced a 60% decrease in the massflow ratio, one would still obtain a net result better than the straight ejector. In this case the straight ejector would just barely produce thrust equivalent to the rocket alone, while the constricted ejector with 60% less entrained air would yield a 10% increase in thrust.

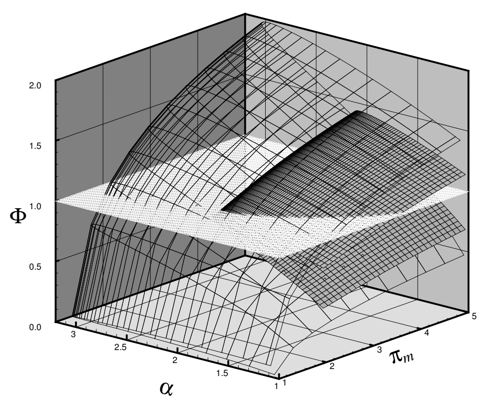


Fig. 8 Thrust augmentation for an ejector based on an Atlas E/F launch rocket (Altitude = 6,300 m, M_∞ = 0.8, p_e = 100 kPa)

Going back to Fig. 6, the size of the solution surface above the $\Phi = 1.0$ plane indicates there are numerous possible ejector configurations that would yield a thrust augmentation greater than unity at the beginning of a launch trajectory. Figure 8 compares this result to the solution surface at an altitude of 6,300 m and a freestream Mach number of 0.8. In this case none of the rocket parameters have been altered, however, the decrease in the airstream total pressure results in an increase in the total pressure ratio (ζ) of approximately 46%. This in turn affects the resulting static pressure ratio between the entrained air and rocket exhaust streams entering the ejector, where now $p_r/p_a \approx 1.5$ creating some expansion of the rocket exhaust into the entrained airstream. This effect, coupled with the fact that a decrease in p_a^o will decrease $\dot{m}_a$, yields a decrease in the maximum possible value of α as shown by the shift to the right of the shaded surface in Fig. 8. Since much of the solution region above the $\Phi = 1.0$ plane at sea level is located near the maximum massflow ratio, a reduction of this value reduces the range of solutions that will yield a thrust augmentation greater than unity. At this point the benefits of incorporating the ejector into a larger RBCC engine become more apparent, as the introduction of additional fuel can begin to be used in a ramjet type application as the freestream Mach number increases beyond sonic conditions.

4 Conclusions

Using a theoretical framework, the compression ratio of an ejector is examined from the point of view of improving the performance of an ejector/rocket engine. It is shown that at static sea level conditions, increasing the compression ratio can increase the overall thrust as compared to rocket engines typical of space launch systems. It is also shown that it is more beneficial in terms of increased thrust to increase the compression ratio rather than the entrainment ratio within an ejector. Ejector effectiveness as a function of flight altitude and speed is also examined, where at an altitude of 6,300 m and M_∞ = 0.8, it is shown that the ejector effect on its own is reaching the limits of its ability to improve thrust. Although the use of a constricting area ejector is suggested as a method of obtaining increased levels of compression, further study is required to validate such a configuration.

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