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Performance Assessment of Ejector Augmented Pulsed Detonation Rockets

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Abstract

There is increasingly compelling numerical and experimental data in the present times to justify the pulsed detonation engine concept as a viable alternative high performance propulsion system. As in the case of steady ejector based rockets, the pulsed detonation rockets (PDRs) may also be operated in an ejector mode, to attain a significant augmentation of thrust. Analysis and computations of this form of thrust augmentation are few and scarcely available in the open literature. This work attempts to utilize traditional analytical techniques as well as current CFD tools to assess the performance of ejector augmented PDRs. PDRs may be used effectively in this mode at low altitude and Mach number part of a flight trajectory for future SSTO or similar missions.

Introduction

The ejector augmented PDR provides the means to enhance thrust and specific impulse at low speeds beyond that provided by conventional rockets, by adding momentum to an entrained air flow. Steady state devices based on the ejector rocket are relatively simple to analyze. The analysis of pulsating flows is significantly different from that of steady flows. Here, a combined analytical and computational strategy is adopted for this purpose.

The operation of an ejector augmented rocket is described schematically in Fig. [1]. A rocket engine is embedded in an incoming airstream. Rocket exhaust is referred to as the primary flow, while the air inflow is the secondary flow. By entraining secondary flow and adding momentum to it at appropriate conditions of mass flow rate, temperature and pressure, it is possible to enhance the performance of the pure rocket system significantly. Further, in a device such as a scramjet, the ejector offers a viable alternative for improving performance at low speeds.

The principles and thrust augmentation mechanisms of ejector-augmented rockets have been described by Alperin and Wu [1]. Much is known by way of the aerodynamic features of the primary and secondary flows in an ejector. Simplified inviscid and incompressible arguments have been used (e.g. Heiser [9]) to model the ejector process at low speeds with good comparison with test data. Detailed effects of compressibility, dealing in particular with the upstream effects of back pressure on the secondary flow were formally presented by Fabri and Siestrunck [8] in a landmark paper. Their work identified flow regimes in the compressible operation of an ejector and provided convenient parameters with which the performance may be quantified. They provided a basis by which multi-dimensional features in the ejector flow may be accounted for. (Subsequent work, e.g. Emanuel, [7] simplified these arguments and provided a quasi-one-dimensional analysis of the ejector process for certain situations.) Heiser and Pratt [10], in their text on hypersonic propulsion have provided a modular set of equations to evaluate ejector performance, making adjustments for various types of losses.

A problem which greatly strains the operation of the ejector concept is one of mixing of the primary and secondary streams. If this mixing is incomplete, thrust augmentation suffers and various loss mechanisms emerge. It is necessary to achieve this mixing in as short a distance as possible in order to minimize the

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size and weight penalty caused by the engine. Hypermixing concepts based on producing vortical structures from an array of primary rocket exhaust plumes have been devised (e.g., Bevilaqua [2]) and are considered feasible in various forms.

A pulsating primary flow (as in a pulsejet engine,) enhances mixing with the secondary flow. This is the result of unsteady vortical features (particularly at high peak pressures,) which excite modes of flow interaction that are not available in a steady state ejector mixing duct. Resonant entrainment of secondary flows depending on geometry has also been observed at low speeds (Parikh and Moffat [14].) An experimental and analytical investigation into this process was performed at Hiller Aircraft Corp. [12] in the 1960s.

In the present time, the development of a pulsed detonation engine (PDE) has received global attention due to the promise of high cycle efficiency and specific impulse that it holds. Some thoughts and numerical simulations of momentum transfer between PDRs and air inflow have been developed in Ref. [4]. Commercial interest has emerged in these concepts (e.g., [3]). The high values of post-detonation pressures, and the high frequencies being considered for the PDEs (about 200 Hz) make the ejector PDE concept an extremely attractive candidate for future missions.

Two forms of practical implementation of the ejector-PDR are of interest in the present study. These are sketched in Figs. [2] and [3]. The first is a top-wall mounted PDR, with minimal interference with the inflow. This is particularly beneficial at supersonic secondary flow speeds, when strong internal shocks may be avoided. The second is a strut mounted PDR configuration, in which struts are placed inside and across a scramjet air intake consisting of rocket engines. This form allows for better mixing between primary and secondary flow streams due to the vorticity and pressure gradients caused by the internal shock system.

Present analysis does not depend critically upon the exact configuration used. In a quasi-one-dimensional sense, mixing is modeled by assuming a value of an associated efficiency, and shock losses may be treated in terms of entropy increase. Some CFD analysis has been also pursued which avoids such assumptions.

The following assumptions are used in quasi-1D performance predictions made in this work:

(i) All effects are one-dimensional. Emanuel [7] presents a quasi-one dimensional analysis of compressible ejector flows and shows that qualitative and important global quantitative data matches the more elaborate Fabri [8] theory without detailed modeling of two dimensional non-isentropic mixing processes. This is assumed to be sufficient for a first approximation in this work.

(ii) Constant total pressure in the combustor

(iii) Assumed fraction of primary flow total temperature regained from an arbitrary afterburning process.

(iv) Perfect matching of primary and secondary flow static pressure at the entrance to the ejector.

(v) Most importantly, the perfect gas ejector analysis is valid in a cycle averaged sense. This is easily shown to be true in the incompressible case. For the case of a compressible gas undergoing pulsed cyclic processes, the results are expected to be valid at lower Mach numbers. For compressible flows, this analysis gives the performance of an equivalent steady state ejector with the same total pressure and temperature ratios between the primary and secondary flows. This number can be scaled using empirical relations obtained from perfect gas CFD studies in the earlier part of this work.

Engine Characterization

It is required to specify the volume of individual PDR tubes which provide a required level of thrust at a given operating frequency. When some preliminary estimate for the thrust required and the specific impulse expected are available, the fuel mass flow rate $\dot{m}_f$ can be deduced. With this number and the cycle time of the PDRs (N in number) and the density of the propulsive mixture, the volume of each PDR can be sized according to:

$$\Omega_p = \frac{\dot{m}_f \Delta t}{N\rho_p}$$

The primary PDR nozzle exhaust area A_p may be determined from mean mass flow rate considerations (primary mass flow rate, or fuel mass flow rate), using an exponential model for density and velocity variations in a detonation wave profile. With due manipulation, the mean mass flow rate leaving a set of PDRs may be evaluated to be:

$$\overline{\dot{m}}_p = \frac{NA_p U_{\max}}{\Delta t}\left[\frac{\rho_{\min} + \rho_{\max}}{2\lambda}\right]$$

where $u(t) = U_{\max} e^{-\lambda t}$,

$$\rho(t) = \rho_{\min} + (\rho_{\max} - \rho_{\min})e^{-\lambda t}$$

In the above, λ is the time constant for the exponential variations of velocity and density behind a detonation wave. Typically the value of λ is different for different flow variables. However, as a simplistic first-cut approximation, it has been assumed to be identical. It is also assumed in this analysis that $\lambda\Delta t >> 1$, (where Δt is the time period of each PDR cycle.) Typical values of $\lambda\Delta t \approx 10$ are well within the scope of the present analysis. For a cycle frequency of 200 Hz, this yields a value of $\lambda = 2000$.

The ratio of the primary to secondary mass flow rate is determined in the quasi-one-dimensional cycle averaged ejector mode analysis. This number must be in accordance with the estimate for "proper" afterburning of incoming air such that the net increase in specific energy due to afterburning is maximized. An alternate method of taking this into consideration is to include it as a loss mechanism and assume that the total temperature of the mixed primary and secondary flows reaches a smaller fraction of the initial total temperature of the primary flow. The cross sectional area of the secondary flow in order for stoichiometric afterburning to be achieved, is:

$$A_s = \frac{8N\Omega_p \rho_p}{\eta_{ab} u_s \rho_s \Delta t}$$

Total pressure in the detonation chamber exhaust is computed using data from the NASA-CEA code. This number is required in the ideal cycle analysis based on Heiser and Pratt [10]. First, the total temperature is obtained using the energy equation:

$$T_t = \frac{C_p T + \frac{u^2}{2}}{C_p}$$

in which the specific heat of the detonation products for a H2/O2 PDR is 16.279 kJ/kgK, and a post-detonation temperature of 3682 K with velocity of 1543 m/s are used. A total temperature of 3755 K from these relations can then be used to compute the stagnation pressure from the relation:

$$\frac{p_t}{p} = \left(\frac{T_t}{T}\right)^{\gamma/(\gamma-1)}$$

where the value of γ is taken to be 1.1287 from the CEA analysis. Using a post-detonation static pressure of 19.045 bar, this results in a stagnation pressure of 20.378 bar.

Ideal Cycle Analysis

The equation set presented by Heiser [9] is used with cycle averaged quantities to determine the performance of PDR-ejectors. Both compressible and incompressible cases have been studied, to see the range of applicability of simpler models. Perfect gas CFD codes have been run to estimate the effect of the unsteadiness in the PDR flow over performance. In all of these simulations, the following properties have been used (with the number convention from Heiser and Pratt):

$$\frac{P_{tp}}{P_0} = 20, \qquad \frac{T_{tp}}{T_0} = 12.5 \qquad \text{at takeoff}$$

$$\frac{A}{A_p^*} = 7.7, \qquad \gamma = 1.4 \qquad \frac{A_{10}}{A_0} = 0.8 \qquad \frac{p_{10}}{p_0} = 1.$$

Net thrust has been seen to be augmented by a factor of 1.7 times when the secondary flow Mach number is zero, and gradually lose its effect as the Mach number increases beyond 1.5. As a simple model, a linear variation of thrust augmentation from 1.7 to 1 between secondary Mach numbers of 0 and 2 has been assumed in this analysis to correct for the unsteadiness. Further, PDR specific impulse is augmented by partial filling of the detonation chambers. This has been considered in this analysis. Fill fractions going from 33% to 100% during the range of operation of the ejector have been considered in order to match the take-off specific impulse from SSTO missions of interest.

Pulsed Detonation Engine Simulations

Numerical modeling of pulsed detonation rockets has been performed using the NASA codes BURN2 and VULCAN. A seven species – seven reaction model of Hydrogen-Air chemistry has been selected. It has been observed that more complex models do not yield significantly different predictions of the flow quantities of interest. These codes were modified to perform multicycle studies, such that a time averaged performance data could be obtained. Fig. [4] shows a sample plot of thrust and specific impulse from a multi-cycle airbreathing pulsed detonation engine, in which Hydrogen and air are assumed to be perfectly mixed at the stoichiometric proportion. As has been observed in the literature (Kailasanath [11]), the outflow boundary conditions in quasi-1D CFD studies determine to a large extent the final values of specific impulse obtained. Characteristic boundary conditions with a relaxation gap for pressure seems to provide reliable values of specific impulse. Detonation pressure rise and wave speeds used in these computations compare favorably with Ref.[11], and the Gordon-McBride (Ref.[13]) CEA code from NASA-Glen.

Multicycle Thrust and Specific Impulse

While a multicycle reacting gas calculation of ejector rockets is possible, present codes have been seen to be computationally intensive. In order to resolve detonation waves and properly model their diffusion into a secondary flow stream, a mesh resolution of within a few millimeters is essential. This results in prohibitive CPU time requirements for realistic ejector-PDR simulations, where the linear dimensions of the engine are about a meter, and the need to correctly model mixing processes limits the time step used. As a solution to this crisis, a simplified Euler equation solver (with no chemical reactions) has been used, feeding detonation wave profiles as inflow conditions.

A stream thrust analysis, such as that developed in Ref. [10] has been used to evaluate parametrically the thrust and specific impulse of the ejector PDR device. Cycle averages were based upon exponential variations of flow quantities at the exhaust of PDR tubes, such as those developed in Ref. [5]. Details of the mathematical model will be presented in the final paper.

Results

Two dimensional simulations of reacting flow with detonation and turbulent mixing were performed using the NASA-Langley code VULCAN. Due to the large CPU time taken for the execution of such calculations, very few simulations were performed with this model. A sample mixing region at the exhaust of a PDR, of a flow that is yet and mixing and burning is shown in Fig. [5].

Ideal cycle analysis based upon cycle averaged flow quantities is presented in Figs. [6] and [7]. Fig. [6] shows the incompressible flow estimate of thrust augmentation, while Fig. [7] includes the effects of compressibility and non-isentropic mixing. The incompressible flow estimate represents at low speeds the maximum attainable thrust augmentation. At supersonic speeds, the possibility of afterburning increases the potential thrust that may be generated.

The bulk of the CFD runs were made on an internally written Euler equation solver at HyPerComp, Inc. This code is second order accurate in space and time, and imposes characteristic boundary conditions and propagates exiting waves into a plenum chamber, thereby avoiding boundary effects on high strength unsteady shock phenomena. Secondary flow Mach numbers of 0.15 and 1.5 have been studied and the results are presented in Figs. [8] – [13]. The code also performs an exponential fit to the thrust profile that is generated, thereby locating a cycle-averaged asymptotic thrust and specific impulse generated by the ejector PDR. Pressure, temperature profiles are provided to the primary flow as an inflow boundary condition. The purge part of the PDR cycle is not modeled in this study, and represents a possible loss mechanism. Sample pressure profiles in subsonic and supersonic secondary flow ejector PDRs are shown in Fig. [15] and [16]. The effect of the leading shock waves in pumping the secondary flow momentum is clearly visible.

A sample mission was studied with the assistance of mission data from Lockheed Martin Tactical Aircraft Systems in Fort Worth, TX. This consisted of a vehicle weighing about 900,000 lb (GTOW), delivering a 40,000 lb payload to a low earth orbit. Simplified and unclassified mission data was used in obtaining preliminary sizing and performance expected of the ejector-PDRs. A PDE based multimode engine was conceptualized in this study, comprising of four stages from takeoff to orbit. These stages are analogous to the more conventional scramjet based combined cycle. A future paper [16] will present the overall vehicle concept with all the stages.

The ejector PDR is the first stage in the proposed single flow path multimode engine concept. The specific impulse estimate from present analysis is shown in Fig. [14] for the cases of (a) a continuous flow ejector, (b) a PDR ejector with the same cycle averaged total pressure, and (c) a PDR ejector with variable cowl geometry. Losses pertaining to nonideal PDR operation, improper mixing, have been accounted for by assuming nondimensional efficiencies pertaining to these processes. A more comprehensive study would provide greatly enhanced understanding of these processes.

Conclusion

Semi-numerical tools to compute the performance of ejector augmented pulsed detonation rocket are under development. In the event that the usage of pulsed detonation rockets becomes prevalent in future times, there is bound to be interest in using them in an ejector augmented format. Work pertaining to efficient afterburning mechanisms for ejector PDRs and novel ejector geometries with pulsed mixing enhancement devices is being currently pursued. An experimental program to validate some of the trends observed in this paper, is under progress in the University of Texas at Arlington.

Acknowledgement

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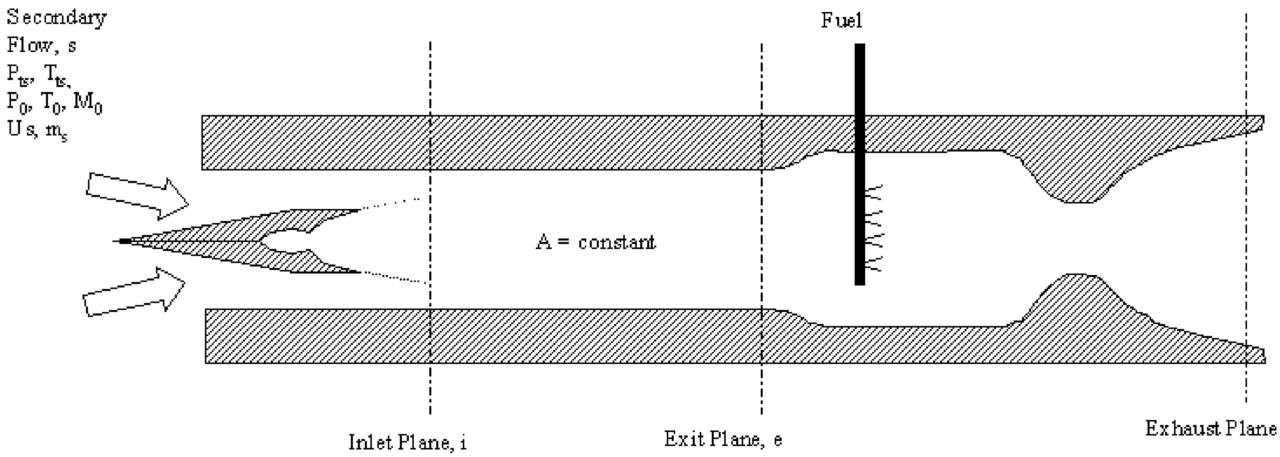

Figure 1: Schematic for Ejector mode analysis

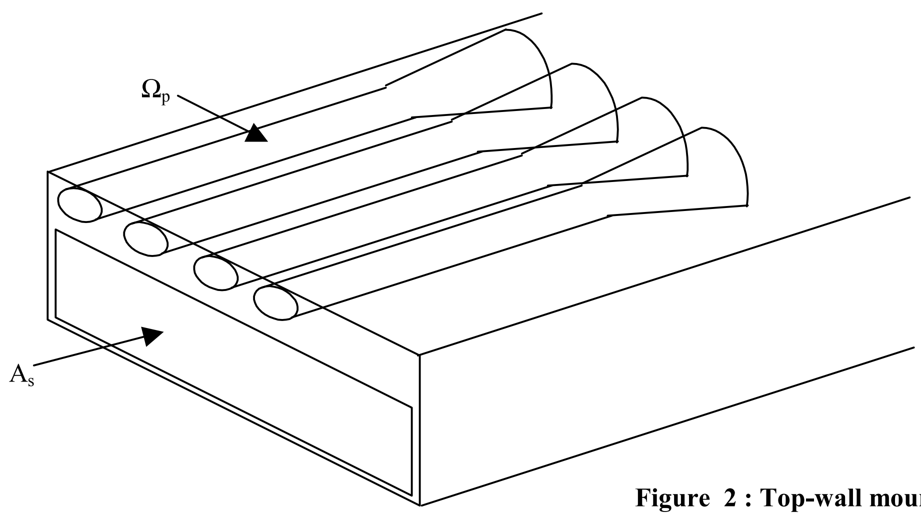

Figure 2 : Top-wall mounted PDRs

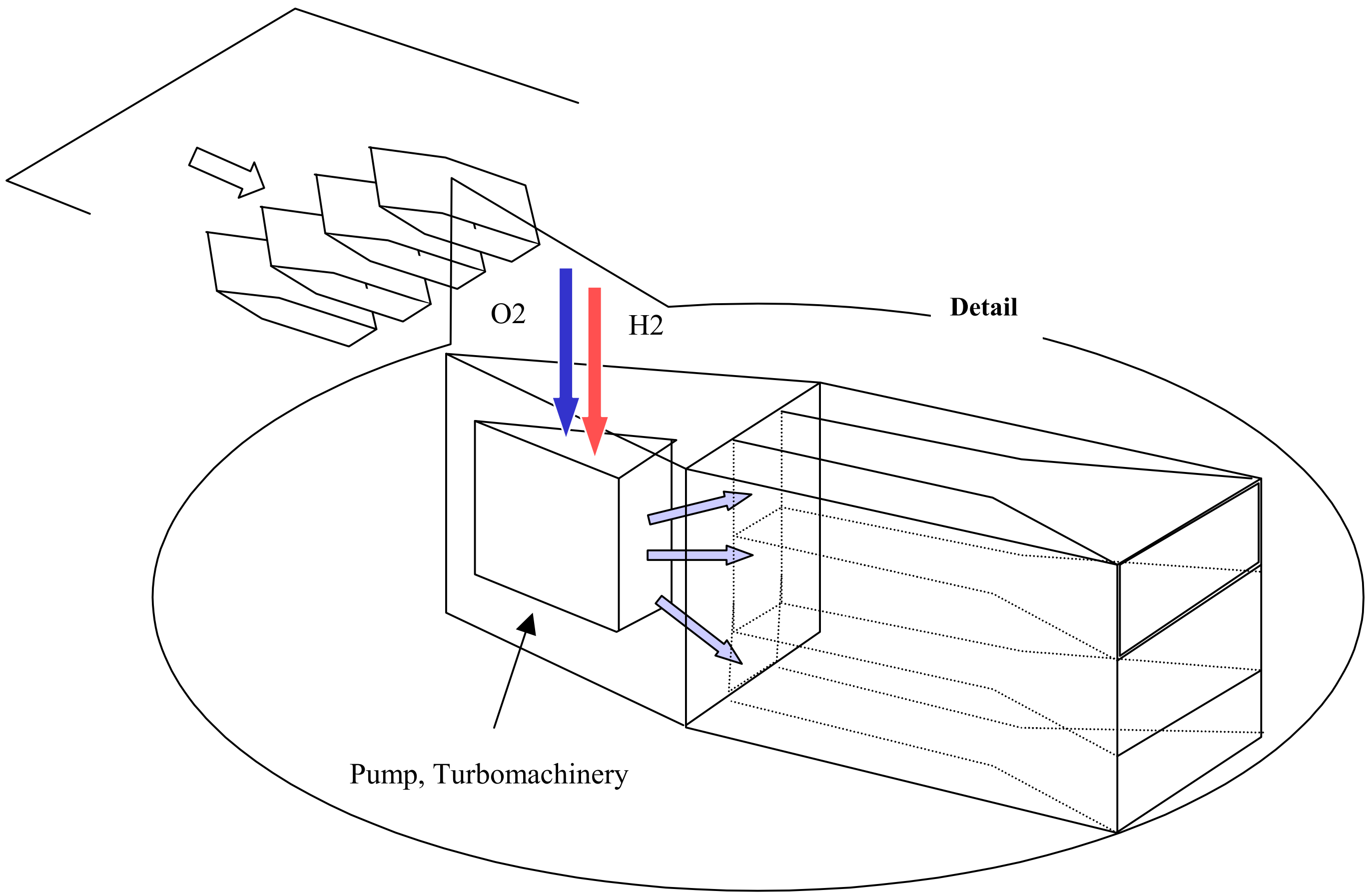

Figure 3: Strut Embedded PDRs

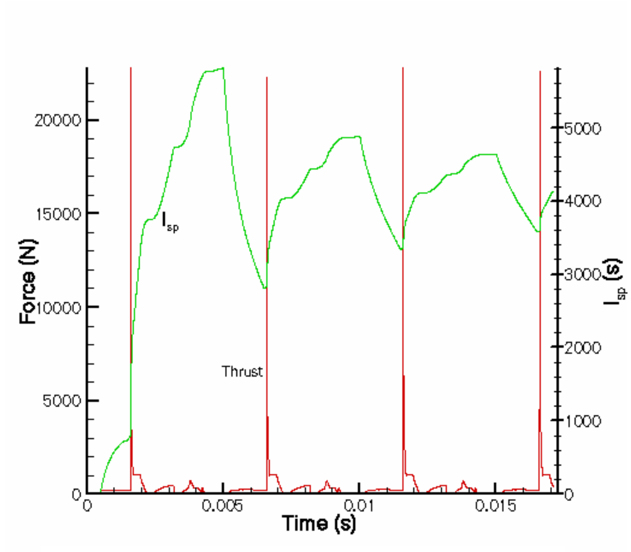

Figure 4: Sample multicycle airbreathing PDE performance

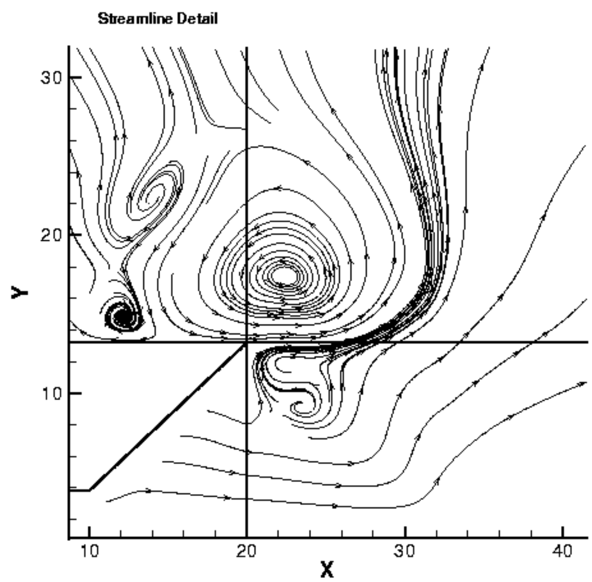

Figure 5: Unsteady ejector – secondary flow mixing

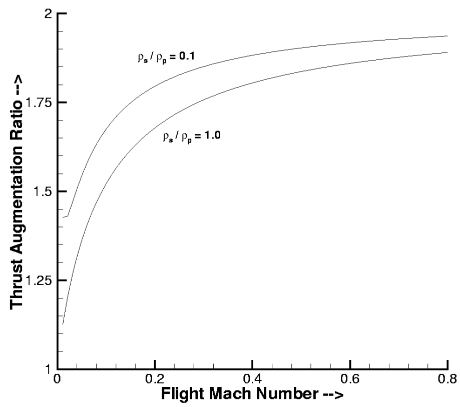

Figure 6: Ideal Cycle Averaged incompressible flow estimate of ejector thrust augmentation

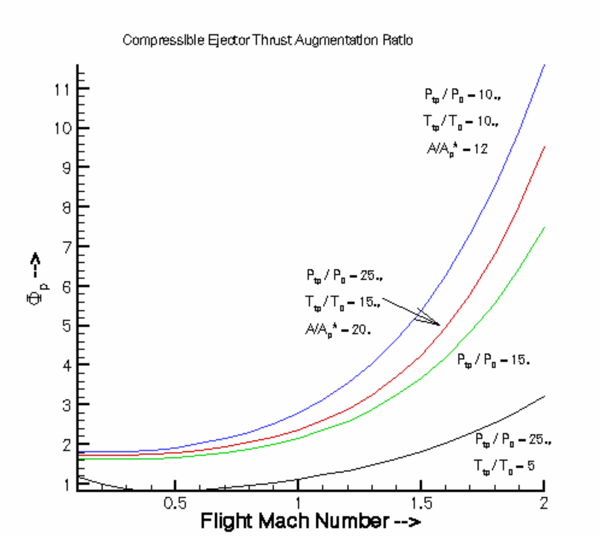

Figure 7: Ideal cycle compressible flow estimate of ejector system thrust augmentation

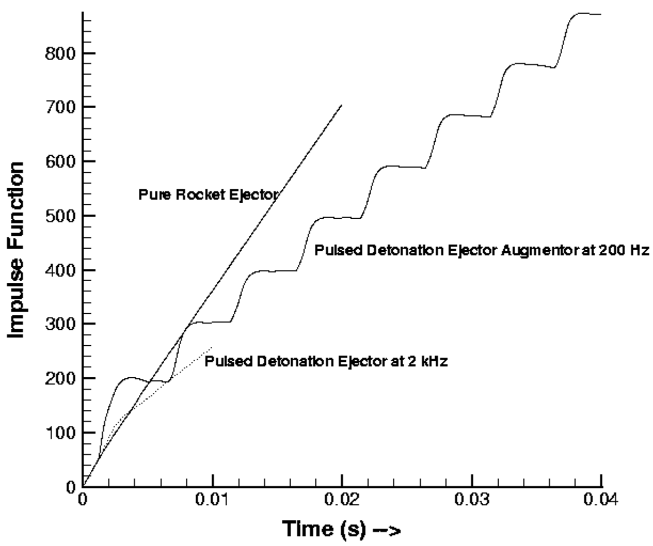

Figure 8: Impulse Variation for Ms = 0.15

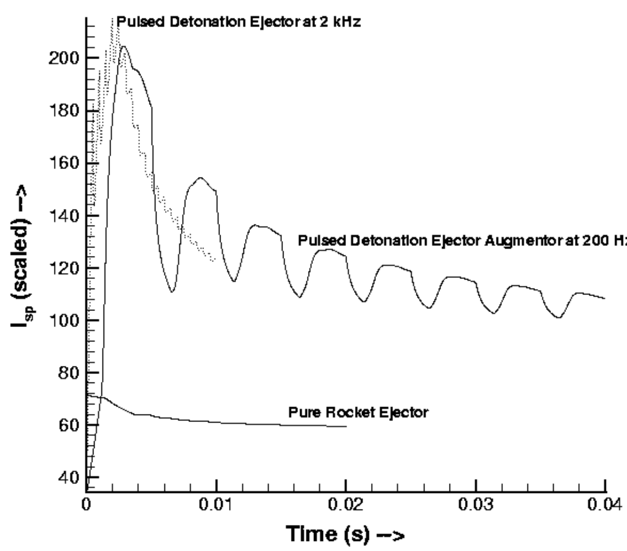

Figure 9: "Isp" comparisons for Ms = 0.15

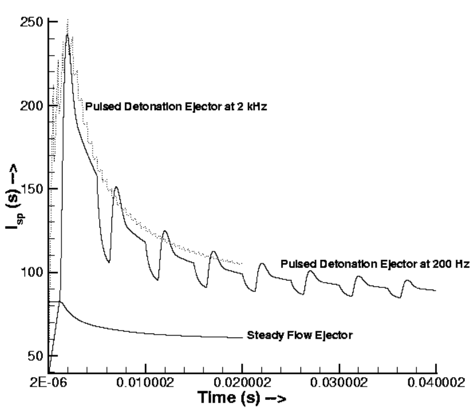

Figure 10: "Isp" comparison for Ms = 1.5

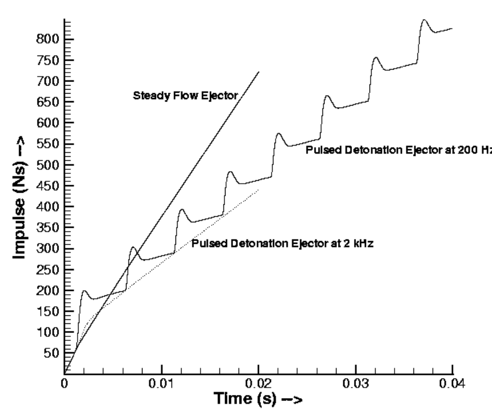

Figure 11: Impulse variation for Ms = 1.5

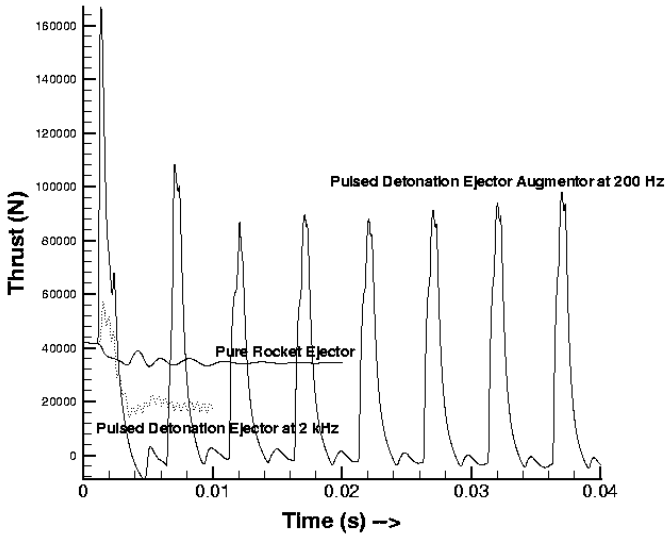

Figure 12: Multicycle Thrust Variation

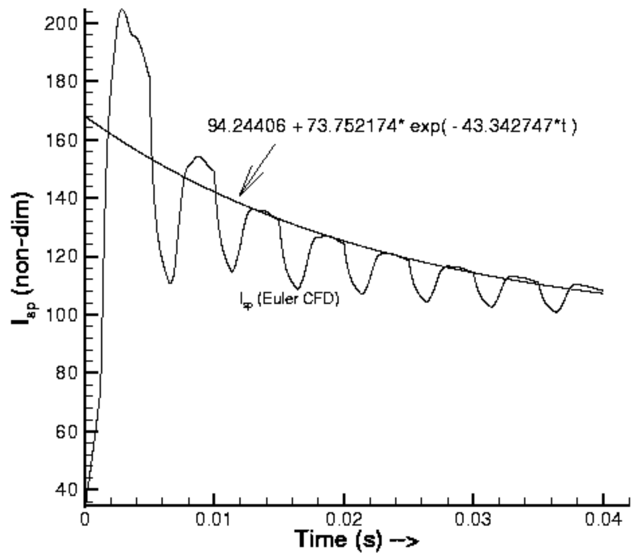

Figure 13: Exponential curve fit for Isp

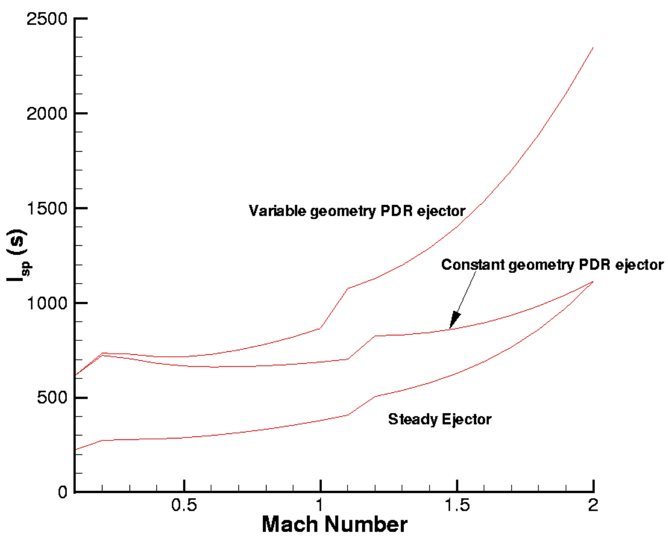

Figure 14: Sample SSTO trajectory specific impulse

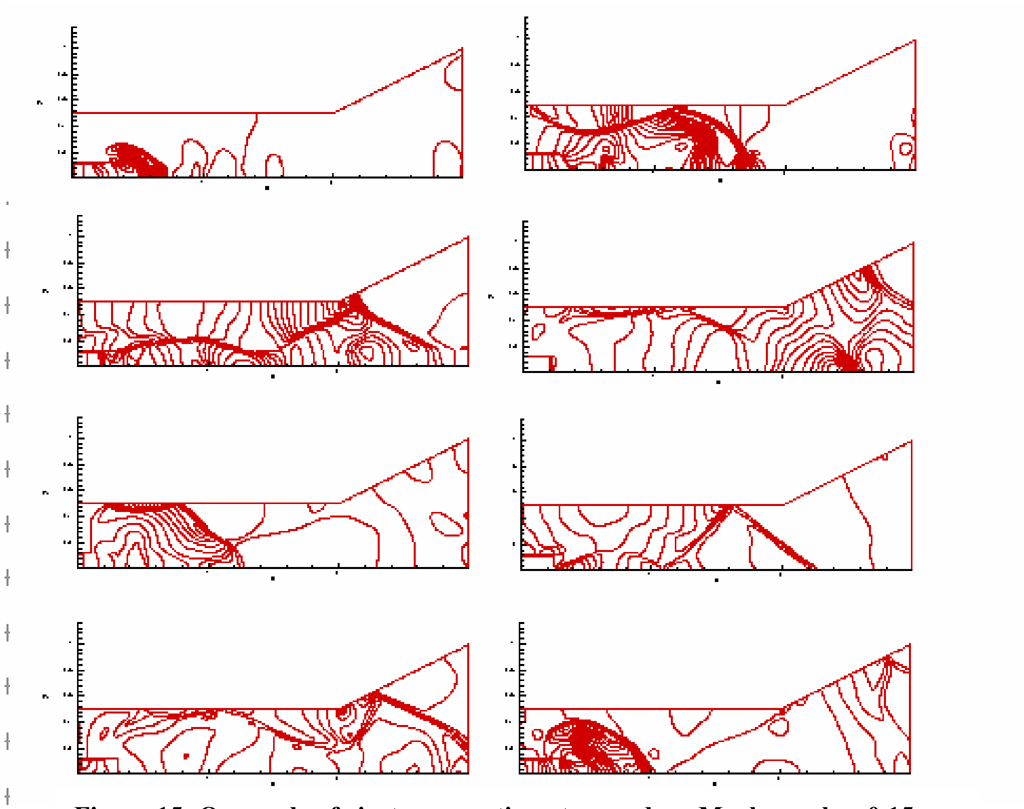

Figure 15: One cycle of ejector operation at secondary Mach number 0.15

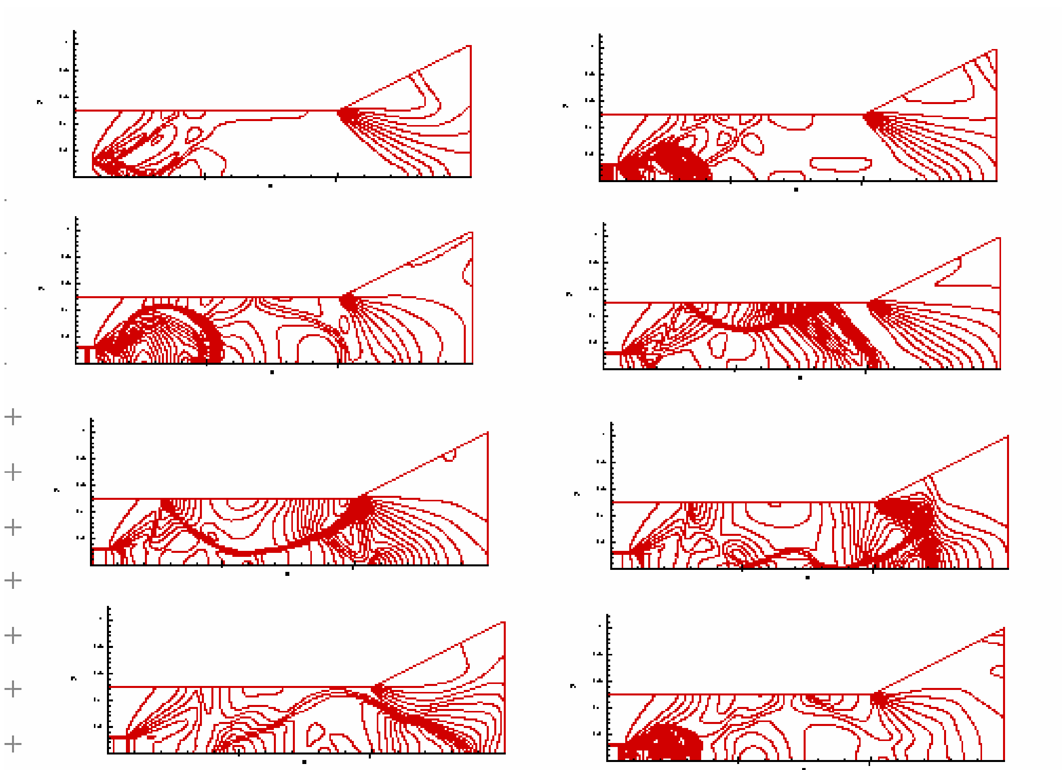

Figure 16: One cycle of ejector operation at secondary Mach number 1.5