A Pulsed Detonation Based Multimode Engine Concept

Ramakanth Munipalli*, Vijaya Shankar †
HyPerComp, Inc., 31255, Cedar Valley Dr., Suite #327, Westlake Village, CA 91361

Donald R. Wilson ‡, Hyungwon Kim¶, Frank K. Lu£
Department of Mechanical and Aerospace Engineering, Box 19018, University of Texas at Arlington, TX 76019

Paul E. Hagseth⊥
Lockheed Martin Tactical Aeronautics Company
Fort Worth, TX

Abstract

A novel multi-mode implementation of pulsed detonation engines is investigated in this paper. The various modes in this proposed concept are (as illustrated in Fig. [1]):

(1) An ejector augmented pulse detonation rocket for take off to moderate supersonic Mach numbers
(2) A pulsed normal detonation wave mode at combustion chamber Mach numbers less than the Chapman-Jouguet Mach number,
(3) An oblique detonation wave mode of operation for Mach numbers in the airbreathing regime that are higher than the Chapman-Jouguet Mach number, and
(4) A pure Pulsed Detonation Rocket (PDR) mode of operation at high altitude.

These various modes utilize a single flow path, in which an array of detonation tubes is placed. From present considerations, 10 tubes will be placed across the width of the engine, which fire sequentially in such a manner as to make the maximum use of the incoming air mass and provide the smoothest possible operation of the device. These tubes could alternately be arranged as load bearing struts that are embedded with rocket chambers. The advantage of such a mode is that a larger exhaust area is available for these tubes when operating in the rocket mode and mixing properties are enhanced in the other modes when these tubes are used for fuel injection alone.

* Member Technical Staff, Member AIAA
† Vice-President, HyPerComp, Inc., Fellow, AIAA
‡ Professor and Chairman, Associate Fellow, AIAA
¶ Postdoctoral Associate
£ Professor, Associate Fellow, AIAA
⊥ Sr. Engineering Specialist

Introduction

This design of a multi-mode pulsed detonation based propulsion system has two principal features which distinguish it from existing designs. Firstly, the design consists of a single propulsive flow path with minimal (or no) moving parts. Secondly, it generates thrust (in critical parts of the trajectory) using upstream traversing or stationary detonation waves in supersonic flow. This involves a minimal loss of total pressure (and kinetic energy), since the supersonic wave is being decelerated at the same time as combustion is occurring in the flow. Resulting thrust and specific impulse values have been found to be comparable or superior to existing RBCC designs. Present research is a collaborative work of HyPerComp, Inc., the University of Texas at Arlington (UTA) and Lockheed Martin Tactical Aircraft Systems (LMTAS) in Fort Worth, TX.

In designing this system, a preliminary SSTO mission has been chosen based on an unclassified study performed by Lockheed Martin. This vehicle is capable of delivering a 40,000 lb. payload to the International Space Station orbit (51.6 degrees, 220 nautical miles). The original version of this vehicle was powered by the NASP propulsion system concept (scaled up to the size required for this 'payload capable' vehicle) and was fueled with liquid hydrogen and oxygen. It had an available fuel fraction of approximately 0.68 and a takeoff gross weight of approximately 900,000 lb. The thrust, drag, fuel flow, and trajectory characteristics have been provided as guidelines for sizing the PDE mixed mode engine. As the latter is a different propulsion system from the original vehicle concept, it may not be possible to exactly match the thrust and fuel flow profiles. An iterative assessment of a final engine system will be performed when confidence is developed in the nature of a multimode engine based upon pulsed detonation. This paper presents a preliminary study based on ideal cycle analysis and some computations of the most significant physical phenomena that render novelty to the proposed concept.

Methods of Analysis

A brief description of analysis and critical results are presented in this section. For detailed discussion of the first two modes please refer [5] and [6]. Modes 3 and 4 are more traditional, and refs. [2] and [7] may be consulted for a contemporary account of their physics.

Mode-1 Ejector Augmented Pulsed Detonation Rocket

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

(i) All effects are one-dimensional. Emanuel [3] presents a quasi-one dimensional analysis of compressible ejector flows and shows that qualitative and important global quantitative data matches the more elaborate Fabri [4] theory without detailed modeling of two

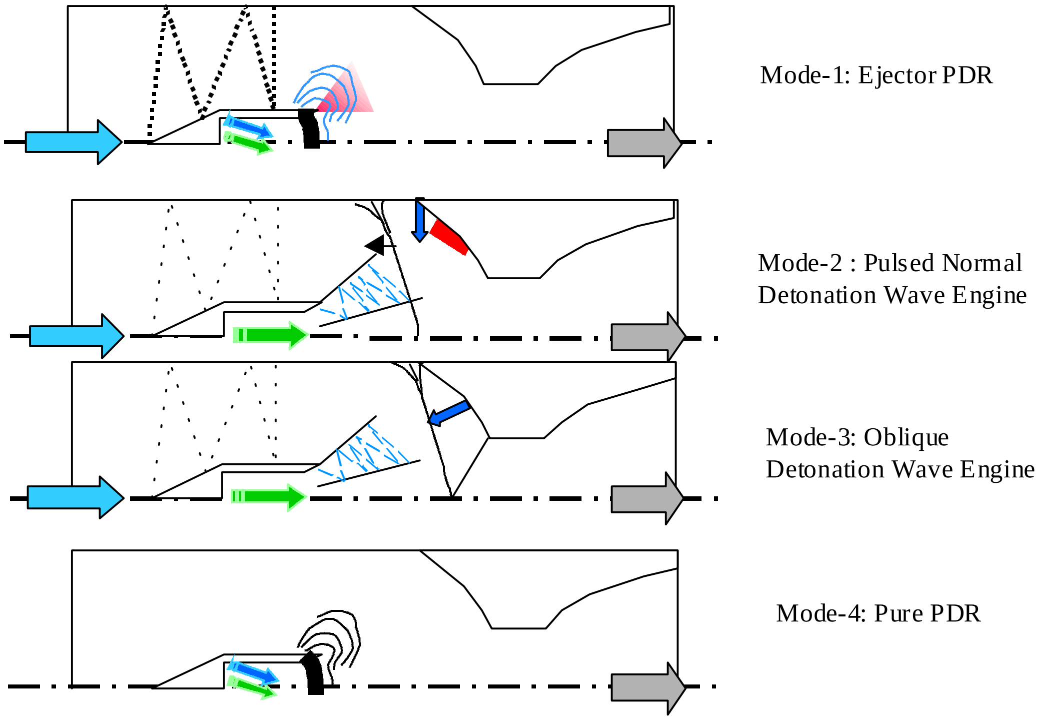


Figure 1: Schematic of Proposed Multimode Engine

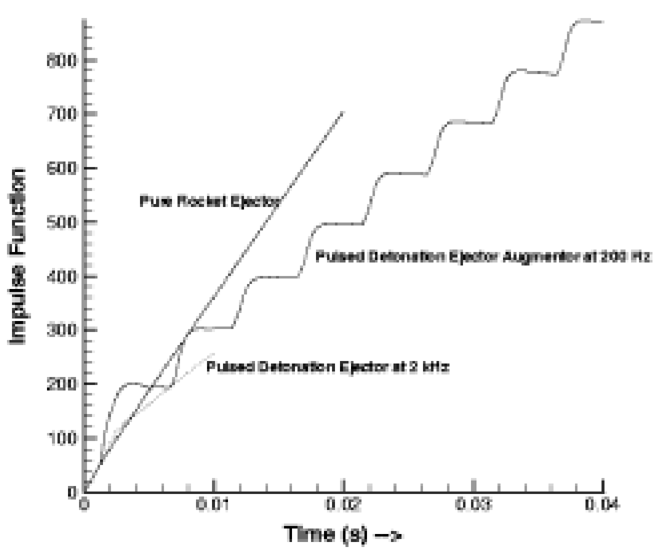


Figure 2: Impulse Variation for Ms = 0.15

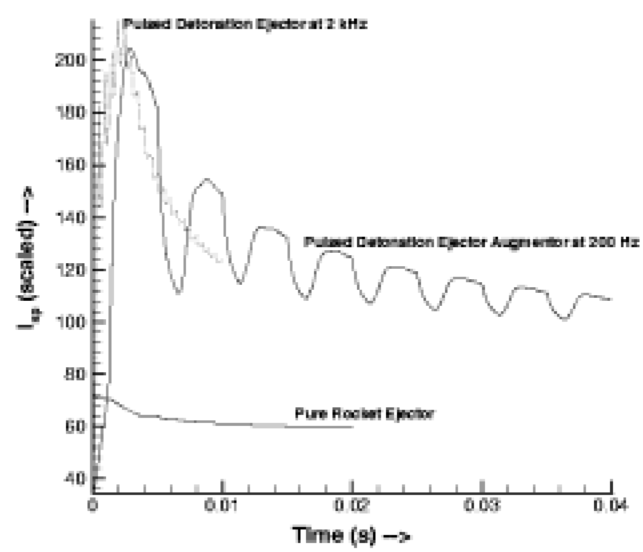


Figure 3: “Isp” comparisons for Ms = 0.15

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(ref. [5]). 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 have been scaled using empirical relations obtained from perfect gas CFD studies. This analysis is based on Ref. [6].

Perfect gas CFD studies have been conducted to obtain preliminary estimates of thrust enhancements due to a pulsed core flow. In this, a PDR exhaust flowfield has been input to the CFD code as an inflow condition and thrust and specific impulse are compared with steady rocket modes. Sample results are shown in Figs. [2] and [3]. Such trends have been correlated to the geometry and trajectory data for the present mission and the resulting specific impulse has been plotted in Fig. [12].

Mode – 2 : Upstream traveling detonation wave engine

This mode represents a crucial part of the present design, and to the knowledge of the authors, has not been dealt with elsewhere in the literature. In this mode, fuel is injected in a pulsating manner in a supersonic combustion chamber flow. The resulting flow can be imagined to comprise of "puffs" of combustible gases propagating at a fixed frequency downstream at supersonic speeds. This mixture is ignited at a downstream location, in a locally Oxygen rich region so as to improve the chances of generating a detonation wave, as has been observed experimentally. The detonation wave propagates into the flow for as long as the fuel concentration is non-zero, and is extinguished and begins to traverse downstream immediately thereafter.

Figure [5] shows the operation of this mode using pressure and hydrogen concentration profiles at various stages of one cycle. Here, a supersonic flow (Mach 3) of air is assumed to have a region of high hydrogen concentration (stoichiometric). Detonation is initiated at the leading edge of this region. The detonation wave progresses upstream, as shown in Figs. [5a]-[5c], until the fuel concentration returns to zero. At this point, the wave recedes, ([5d],[5e]), and is eventually exhausted from the nozzle. The residence time of the detonation wave within the combustion chamber determines the usable frequencies of this mode. These frequencies increase with increase in combustion chamber Mach number.

This mode has a natural upper bound in usable combustion chamber flow velocity, which must always be less than the Chapman-Jouguet Mach number. For higher values of flow velocity, the detonation wave cannot propagate upstream. A steady oblique detonation wave may be setup down stream, which leads into Mode-3. Apart from the Mach number in the combustion chamber, flow temperature is important, in order to prevent ignition prior to proper mixing.

The variation of temperature and Mach number in the combustion chamber sized as per the LMTAS data, is

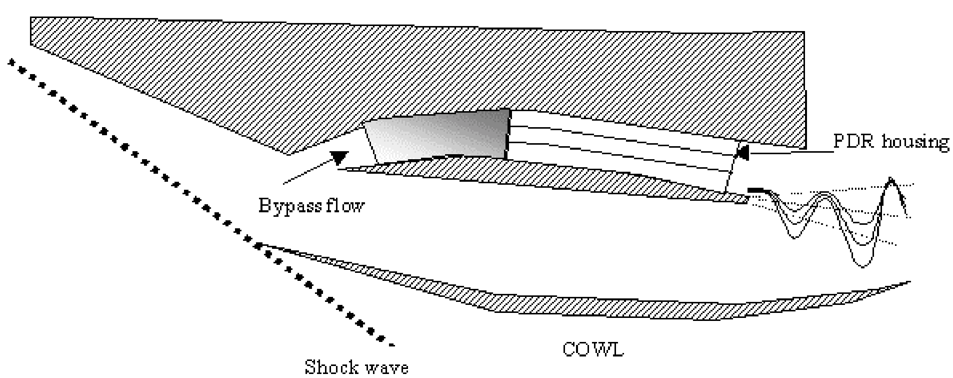


Figure 4: Potential use of a bypass stream to control combustion properties

shown in Figs. [6] and [7]. While the temperature seems to be nominal in the range of interest, an upper Mach number limit of 5.5 – 6 is apparent. This can be mean various things. Firstly, mode-2 may be used in a cruise vehicle at Mach 5 – 6. Second, mode-2 must be operated at a higher altitude such that the combustion chamber Mach number is decreased. Also, the data shown here are for a hydrogen-oxygen mixture. Results would vary significantly for hydrocarbons and liquid fuels. This remains to be studied. Combustion chamber pressures between 0.5 to 1 atm have been deemed as necessary for the successful initiation of detonation waves. A different value, or a multidimensional version of this engine concept could ease the limitation caused by this assumption.

Taking the above considerations to be favorable, the next major problem here seems to be the mixing of fuel and air in a pulsating manner. This is made difficult by the fact that the injection pressure of the fuel is often rather low, and is unable to result in a significant micro-mixing in a supersonic stream. Conventionally, unstable shear layers are seen as a mechanism to achieve the level of micro-mixing required for supersonic combustion. In the present situation, the following design strategy is proposed. In mode-2, the inflow is split into a by-pass stream, by a split intake, or other forms of flow diversion, in such as way as to not lose too much total pressure to internal shock waves. This stream is fed into a continuous stream at a different velocity from that of the primary stream. The size of this by-pass stream is just wide enough to cause a shear layer system which may be made unstable and generate a vorticity field, as shown by the wavy lines in Fig.[4]. Fuel is pulse-injected from the PDR walls, which are blunted for this purpose. The objective is that the pulsed fuel momentum acts as a superposition over an already established flowfield structure. Since the fuel mass fraction is quite small (order of 0.02 – 0.1 typically), the perturbation is likely to result in the expected result that the fuel is trapped and mixed by the vorticity gradients in the combustion chamber. Numerical studies are being performed to test the validity of this concept. At the present juncture, it is assumed to be an acceptable means to generate a pulsed supersonic combustible mixture. In the case where strut based PDRs are used, this mechanism may be achieved via far simpler means.

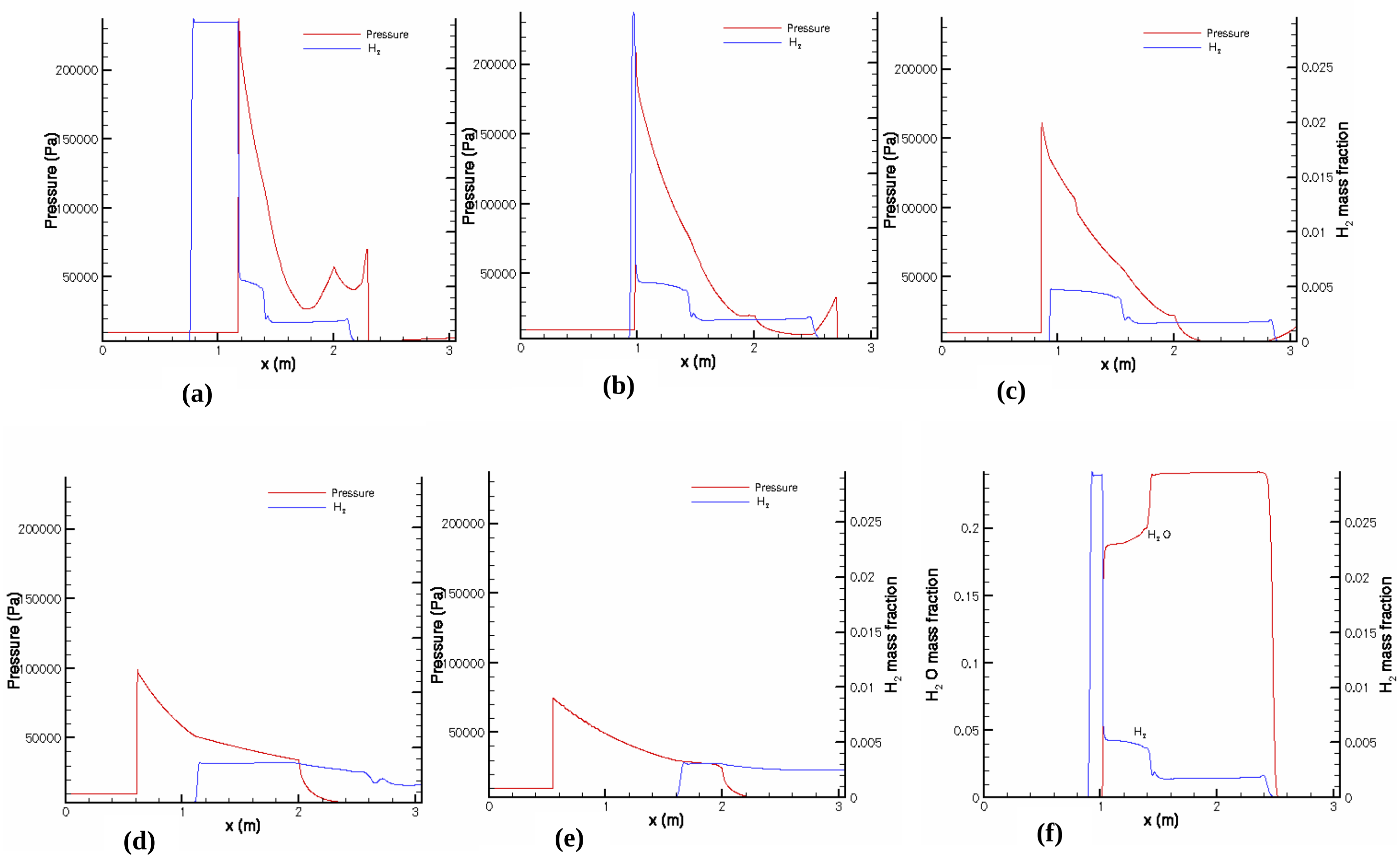


Figure 5: (a) – (e): Sequence of pressure and Hydrogen mass fraction profiles as a normal detonation wave advances into supersonic flow. (f): H2O and H2 mass fractions in the detonation wave

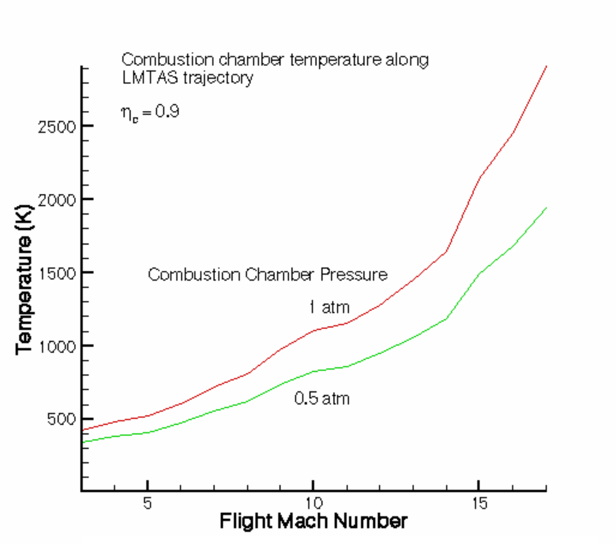


Figure 6: Combustion chamber temperature for the LMTAS trajectory

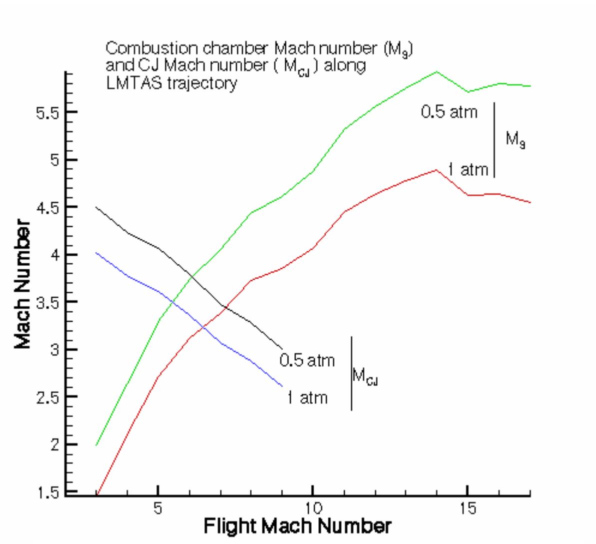


Figure 7: CJ and Combustion chamber Mach numbers along LMTAS trajectory

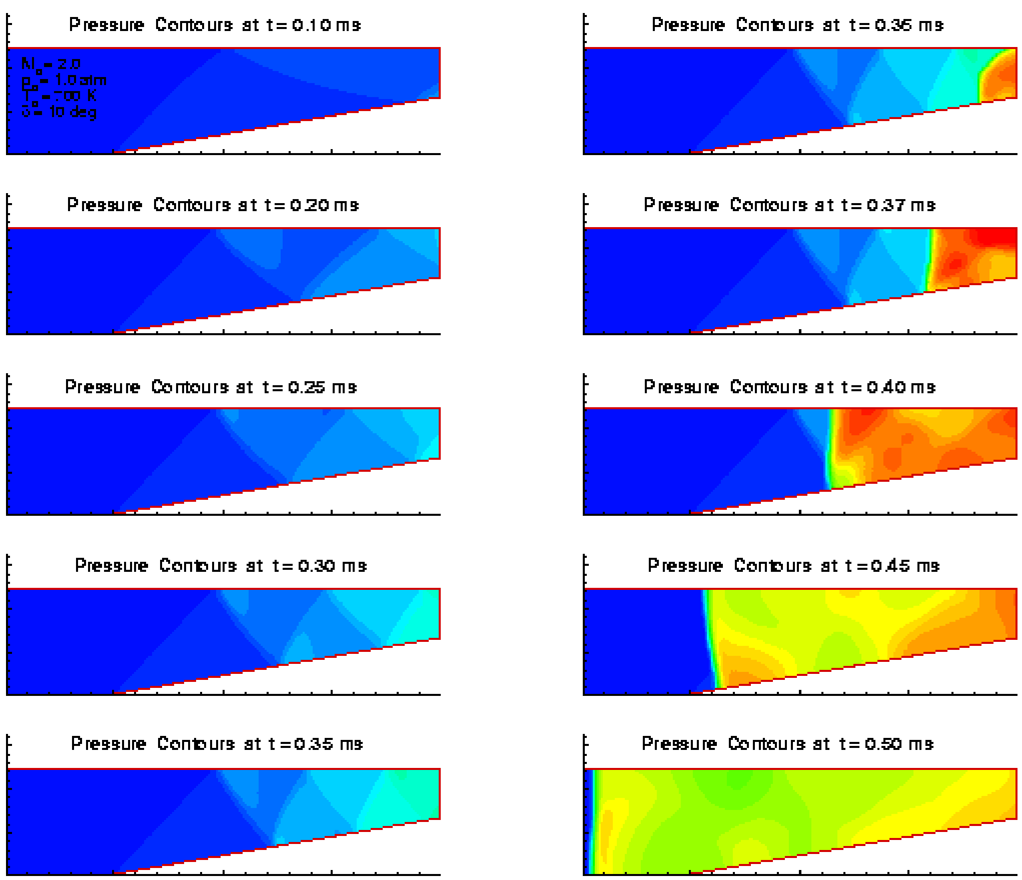


Figure 8: Shock initiation of detonation at low supersonic Mach numbers

Stream thrust analysis has been used here in the following four stages.

1. The stream thrust at the inlet station is computed using LMTAS data.
2. Assuming a pressure close to 1 atm (also 0.5 atm) in the combustion chamber, the static temperature ratio in an inlet with a compression efficiency of 0.9 is obtained. Lower efficiencies result in larger combustion temperatures, which has an effect of decreasing the range of applicability of this mode.
3. Other flow quantities are determined from this, and the effect of an upstream traveling detonation is computed. Typically, this is done by computing cycle averaged values of pressure, temperature and velocity. The NASA-Lewis CEA code is used to compute the temperature of the detonated mixture at the combustion chamber conditions prescribed. Rate constants in exponential curve fits of detonation wave properties are obtained from CFD runs and used to compute cycle averaged pressure, temperature and velocity. These values are assumed to be the post-combustion values (at station 4) in the stream thrust analysis.
4. The exit plane values are computed, assuming expansion to a static pressure equal to the ambient pressure, and an expansion efficiency of 0.9. The equation set presented in Heiser and Pratt [5] is used for such estimates.

Another item of interest with regard to this mode is that unstable shock waves from compression surfaces downstream have a tendency to transition to detonation waves in fuel rich mixtures. While this process is time consuming and can greatly reduce allowable cycle frequencies, it nevertheless represents a potential ignition mechanism for mode-2. A sample oblique shock wave is shown in Fig. [8] to transition to a detonation wave propagating upstream.

Mode –3: Oblique Detonation Wave Engine

Once the chamber velocity reaches the Chapman-Jouguet wave speed, the upstream propagating detonation wave becomes a standing wave. At this point, it should be possible to transition to a steady mode of operation. As the chamber velocity is increased above the CJ velocity, the normal detonation wave will transition to a standing oblique wave that will be stabilized by using a ramp at an appropriate angle to the flow.

Prerequisites are that the fuel and air are mixed to near stoichiometric ratio, and that the condition for the instability of shock waves at the design ramp angle and mach number be satisfied at one of the reflected shocks. This condition is given approximately by (if q is the heat release during the chemical reaction, M_{N1} is the incoming Mach number normal to the shock, γ is the ratio of specific heats of the mixture, C_p is the specific heat at constant pressure, T is the incoming flow static temperature):

$$\frac{q}{C_p T} > \frac{\left(M_{N1}^{\;2} - 1\right)^2}{2(\gamma + 1)M_{N1}^{\;2}}$$

As an alternative to changing the ramp angle, a multiple shock combustion wave is being hypothesized. Preliminary results show that weak oblique shock waves do not easily transition to CJ detonations, being overdriven and becoming successively weaker with each reflection. Mode-3 has been studied extensively in the literature starting from the 1950s. Analytical results describing steady oblique detonation waves are available and agree closely with CFD data. This provides an easier way to analyze this mode. We assume a system in which there is an oblique detonation wave followed by an unreacting shock wave which is canceled on the upper compression surface. This is followed by an isentropic expansion to match the ambient conditions.

Stream thrust analysis has been performed for the oblique detonation wave engine. The trajectory definition provided by LMTAS has been used to determine compression parameters required to attain sufficiently high detonation chamber pressures. The thrust generation mechanism is assumed to consist of an oblique detonation wave followed by an oblique shock wave, which is cancelled without reflection, followed finally by an isentropic expansion process. Closed form relations are used in calculating properties across each of these wave systems. CFD data is used to compute the nondimensional heat release parameter, specific heats and so forth used in the closed form detonation wave relations. The results of this analysis are summarized in the performance Fig. [12-c]. In these calculations, a Mach number range of 7 to 17 was chosen in order to optimally locate the range for Mode-3. Efficiencies of the compression process is fixed at 0.9.

It is seen that through the initial part of this Mach range, the specific impulse and the specific thrust fall short of the required values. If Mode-3 must be included in the final propulsion system, a redesign of the Mode-3 operation is in order. An alternative to this mode would be an extended use of PDRs in an air augmented mode. Another drawback observed in the present implementation of this mode is that the isentropic expansion required for the largest thrust increment requires large variations in the area ratio of the expansion region. This variation spreads over an

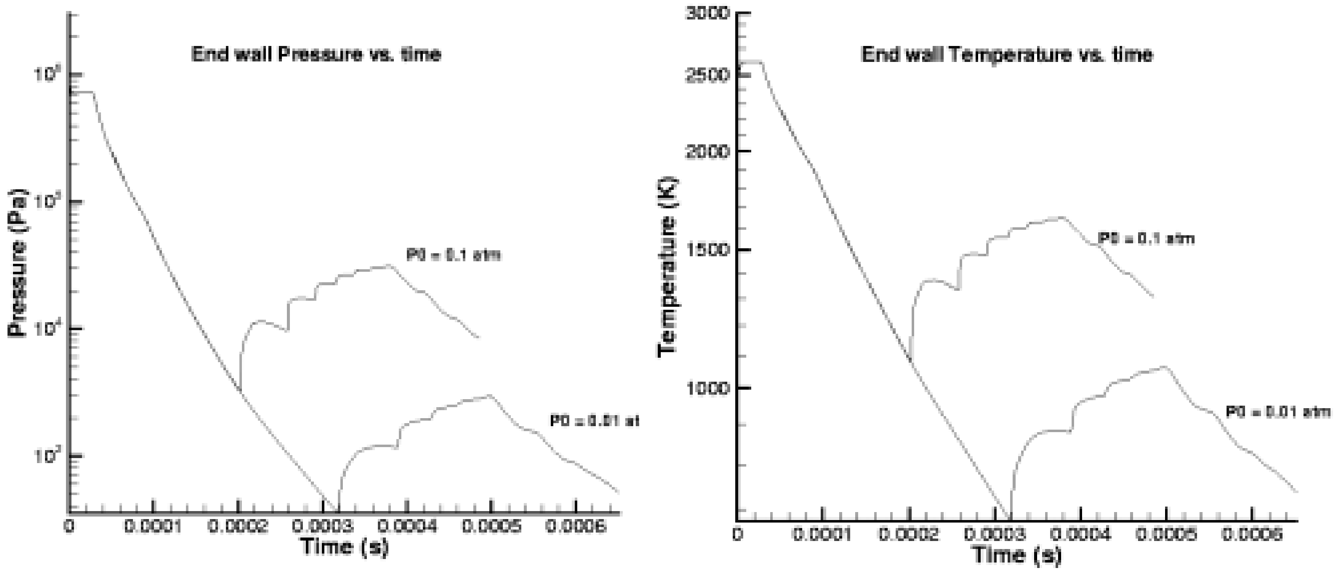


Figure 9: Pressure and Temperature variation in a PDR tube with time

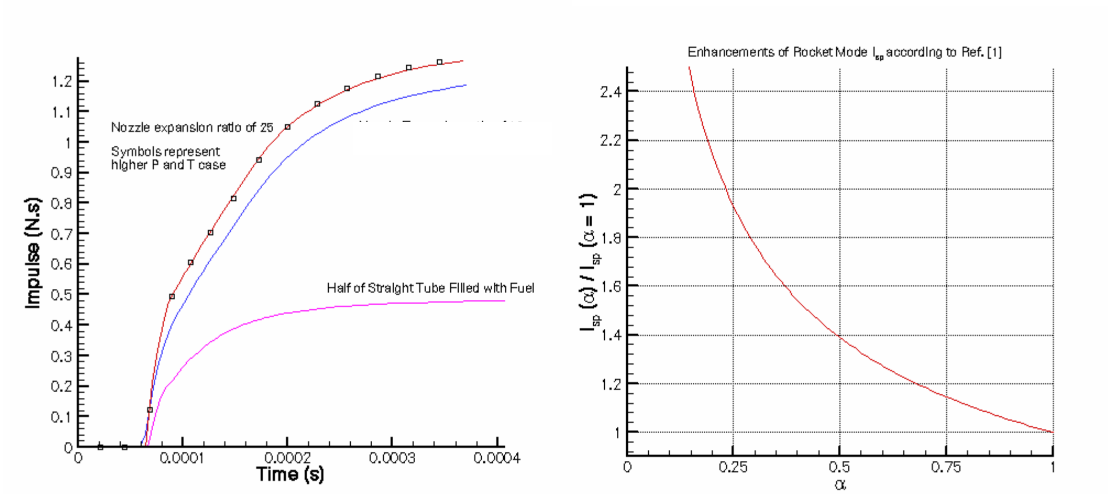


Figure 10: Impulse for various PDR configurations and Isp enhancements from Ref. [1]

order of magnitude, and poses another design challenge. If proper mixing is achieved prior to the location of the detonation wave, a better design alternative might perhaps be to decelerate the flow further, so as to achieve sub-CJ Mach number and use Mode-2. In any case, it is apparent that this mode must be restricted in its range of operation to the very large Mach numbers where the specific impulse seems to be largest. The trajectory redesign must take this in to consideration.

Mode – 4: Pure Pulsed Detonation Rocket

In the upper levels of the atmosphere, a pure PDR will be used. Multiple PDRs will be operated in such a way as to provide a smooth transfer of power to the flight vehicle. Due to the presence of several such engines, this should not be an issue. The challenges in the pure rocket mode center around the demands of high altitude operation of mode 4. High pressure filling of a detonation tube requires the use of end-valves or other mechanisms to prevent losses. Beneficial effects of convergent-divergent nozzles fitted to the end of the detonation tube have been observed computationally. The nozzle reflects a compression wave into the detonation chamber, which in turn assists in retaining the combustible mixture. The purge process is said to be completed when the pressure and temperature levels in the detonation tube fall to a low uniform value at which new combustion gases may be introduced. While tube pressure falls rapidly, temperature in general takes much longer due to the low values of convective speeds at the wall end of the detonation tube. These trends are depicted in figure [9] for a 3 cm long tube as an illustration. Reflected and ingested waves are visible.

As a matter of related interest, the possible values of specific impulse and thrust of a pure PDR were investigated. A PDR with different expansion ratio nozzles was the first subject of investigation. The geometry of the PDR used, comprises of two sections, the first being a constant area (3" dia) tube of length 10 cm, followed by a nozzle of length 10 cm and area ratios of 10 and 25 in two different cases. It was found that by only filling the straight portion of the tube the specific impulse could be increased by about twice. The weight of fuel used in these calculations is given by the volume of the tube multiplied by the density and the acceleration due to gravity. These quantities are:

Volume of the straight portion of tube = 4.5604 e-4 m^2
Density of rocket fuel (stoichiometric $H_2 + O_2$) = 0.49308 kg/m^3
Weight of fuel used per cycle = 2.2206 e-3 N/cycle
Specific Impulse = Impulse / 2.2206 e-3

Using these values, Fig. [10] shows the impulse function for different scenarios. The first, shown as a blue line, is a nozzle of expansion ratio of 12. The peak impulse in this plot is about 1.15N.s., resulting in a specific impulse of 521s. With higher nozzle expansion ratio this is seen to go up to about 1.25 N.s., yielding a specific impulse of about 566 s. These are values based upon peak impulse and will decrease when a multicycle calculations is performed. It will be noted that from the earlier monthly report, the specific impulse value obtained for the case of completely filled tube with nozzle area ratio of 12 was 237 s. This number has been improved by a factor of 2.19. It is interesting to compare these figures with estimates from a recent publication from Aerospatiale (ref. [1]). This publication presents a relation between the specific impulse enhancement due to the partial filling of a detonation tube, wherein:

$$Isp(\alpha) / Isp(\alpha = 1) = \alpha^{(-0.474)}$$ where α is the filling coefficient.

This variation has been plotted in Fig. [10]. For the case above, the filling "coefficient" is approximately the volume filled divided by the total volume of the detonation tube, which is 4.5604e-3/0.00296 = 15.4 %. This corresponds to a value of $\alpha = 0.154$, which should result in a specific impulse enhancement by a factor of about 2.4 compared to the case when the entire tube is filled. This number is quite close to the one computed here (2.19). This leads one to believe that the usable value of the specific impulse for a pure PDR can be rather large for smaller fill ratios, while the thrust levels remain low. This seems to be a multivariate problem, involving detonation tube dimensions, fill fraction, cycle frequency and nozzle efficiency and deserves further investigation in a future study.

Two simple scenarios are considered by way of performance estimates in mode-4. Firstly, a constant specific impulse of 500 s has been assumed for a PDR mode of operation with a mass flow rate of 294.3 kg/s. Since the fuel mass flow rate is held fixed, this yields a constant thrust. When ideal cycle analysis is used, mean quantities for total pressure, total temperature and static temperature in the plenum chamber of a PDR-based engine are computed from cycle averages and an isentropic "mean" flow is assumed. This is certainly an oversimplification of a complex flow process in which supersonic streams are mixing in an unsteady fashion, and provides an upper bound in a sense, of the engine performance. Resulting specific impulse and thrust are plotted in Fig. [12-d] as the blue line. Both of these assumptions yield far less thrust than required by the LMTAS mission, in which high thrust is expended at a low specific impulse at extreme Mach numbers. This again, calls for a redesign of the flight plan and selection of a more evenly distributed thrust profile, as can be provided by the present engine.

Conslusion

A multimode concept involving airbreathing as well as rocket modes is presented here, that uses pulsed detonation waves to obtain mechanical simplicity and enhanced specific impulse. Preliminary performance data is shown in comparison with the LMTAS baseline vehicle in Figs. [11] and [12]. Further innovations in mode-2 shown here could lead to a robust high Mach number propulsion system for hypersonic cruise. An experimental program has been initiated at the University of Texas at Arlington to study pulsed detonation ejectors and their interaction with inlet systems. There is a need to resolve several of the issues raised by the engine concept presented here via computational studies.

Acknolwledgements

This work was supported by an SBIR Phase I contract, F33615-99-C-2923, under the direction of Mr. Glenn Liston of Wright-Patterson Air Force Base. Technical assistance was provided by Mr. Touraj Sahely of HyPerComp in CFD simulations.

References

1. Bouchard, D., Forrat, B., Piton, D., Yvart, P., "AEROSPATIALE and CELERG investigations on Pulsed Detonation propulsion," AIAA paper 99-2658, 35th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Los Angeles, June 1999

2. Cambier, J.-L., Adelman, H.G., Menees, G.P., "Numerical simulations of oblique detonations in supersonic combustion chambers," AIAA Journal of propulsion, Vol. 5, No. 4, pg.482, July 1989

3. Emanuel, G., "Comparison of One-Dimensional Solutions with Fabri Theory for Ejectors," Acta Mechanica, Vol. 44, pp. 187-200 1982

4. Fabri, J., Siestrunck, R., "Supersonic air ejectors," in Advances in applied mechanics, vol. 5, von Mises, R., von Karman, R., eds., pp.1-34, Academic Press, 1958

5. Heiser, W. H., "Thrust Augmentation," Transactions of the ASME – Journal of Engineering for Power, January 1967

6. Heiser W. H., and Pratt, D. T., "Hypersonic Airbreathing Propulsion," AIAA Education Series, 1994

7. Kailasanath, K., "Review of propulsion applications of detonation waves," AIAA Journal, Vol. 38, No. 9, pg. 1698, September 2000

8. Munipalli R., Shankar V., Wilson D. R., Kim H.-Y, Lu F., Liston G., "Performance Assessment of Ejector Augmented PDRs," AIAA paper 2001-0830 to be presented at 39th AIAA Aerospace Sciences Meeting, Reno, NV January 2001

9. Wilson D. R., Lu, F. K., Kim, H.-Y., Munipalli, R., "Analysis of a Pulsed Normal Detonation Wave Concept," to be presented at the 10th AIAA/NAL/NASDA/ISAS International Space Planes and Hypersonic Systems and Technologies Conference, Kyoto, Japan, April 2001

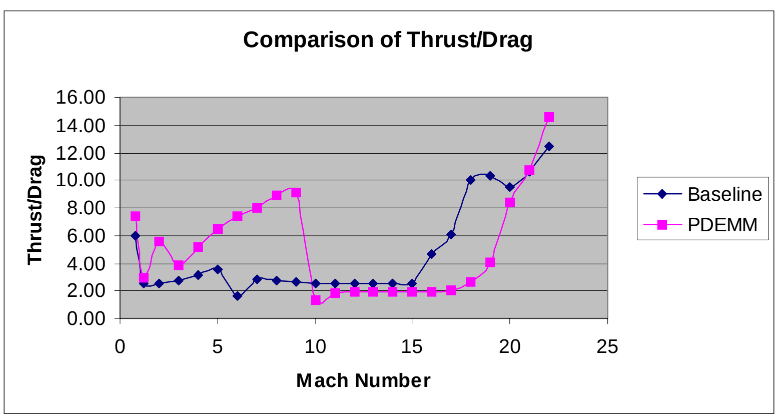
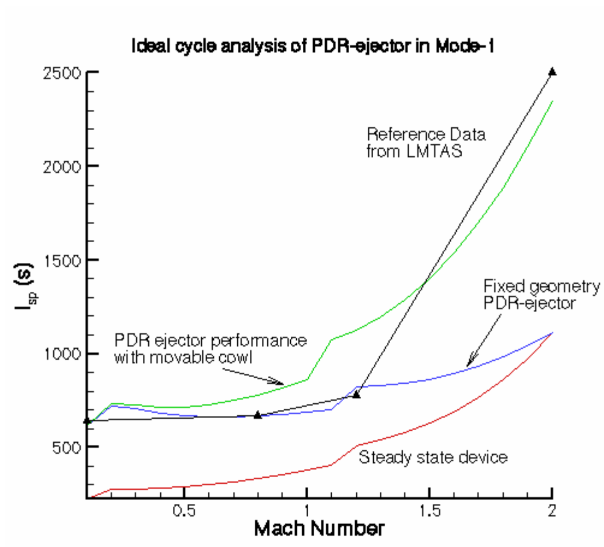
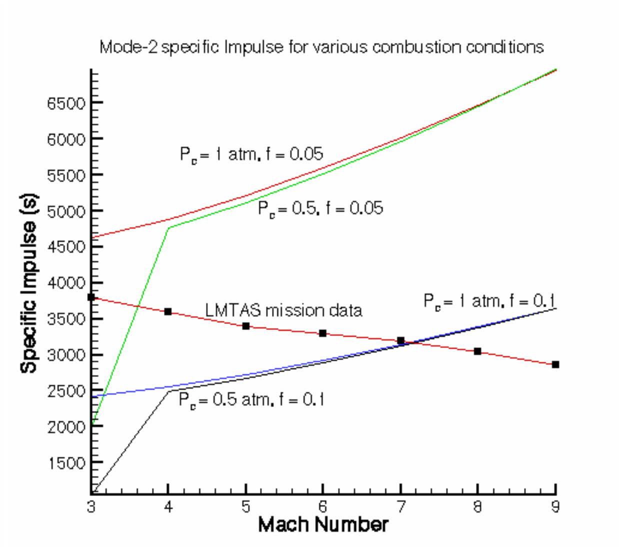


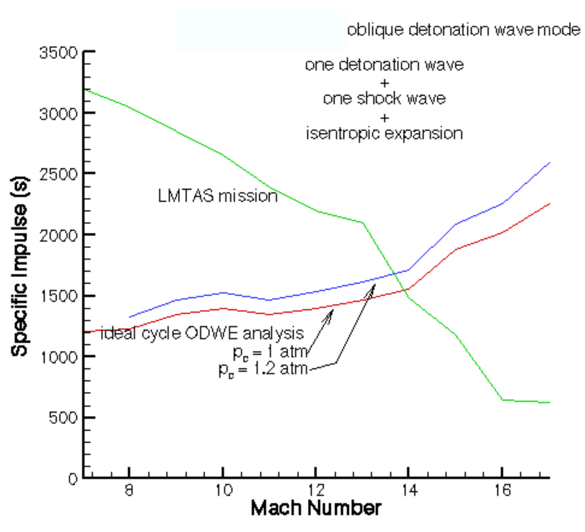
Figure 11 : T/D ratio comparison



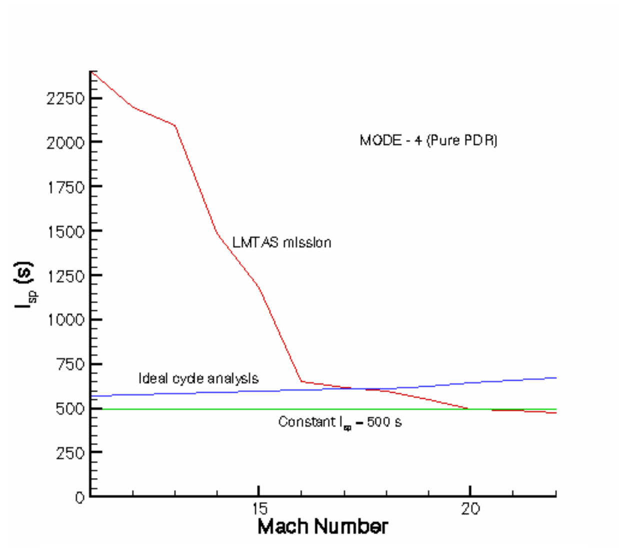
(a) Mode - 1



(b) Mode - 2



(c) Mode - 3



(d) Mode - 4

Figure 12: Ideal Specific Impulse estimates for the present multi-mode engine for a variety of configurations