Preliminary Design of a Pulsed Detonation Based Combined Cycle Engine

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Abstract

Results of a recent investigation in the design of a single flow path combined cycle engine using periodic detonation waves are presented here. Four modes of operation are used in a sample SSTO trajectory in this preliminary design (sketched in Fig. [1]):

(1) An ejector augmented pulsed 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 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 modes utilize a single flow path, in which an array of detonation tubes is placed. The tubes 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 embedded in load bearing struts. Performance estimates based on the stream thrust approach have been obtained from a time averaged ideal cycle analysis with corrections from CFD results. Suggestions for performance enhancement are outlined.

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) (now Lockheed Martin Aeronautics Company) 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

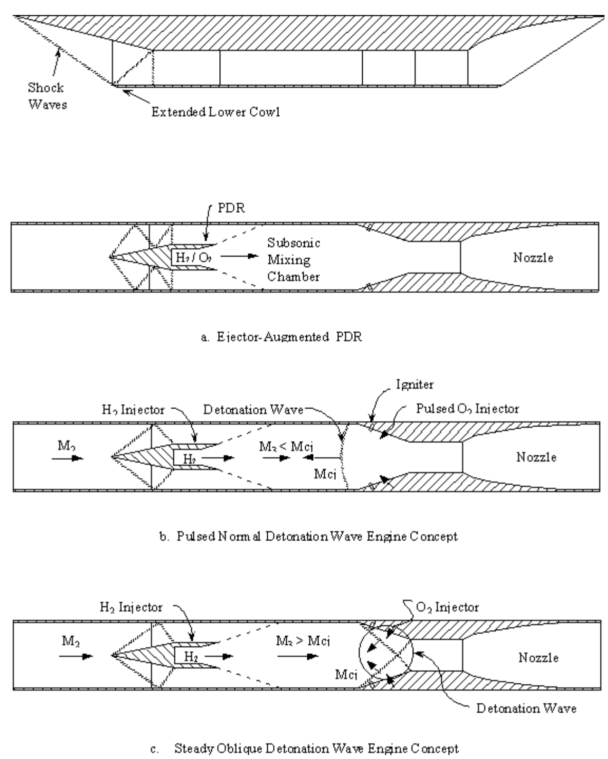

Figure 1: Schematic of proposed multimode engine

* Member Technical Staff
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£ Professor

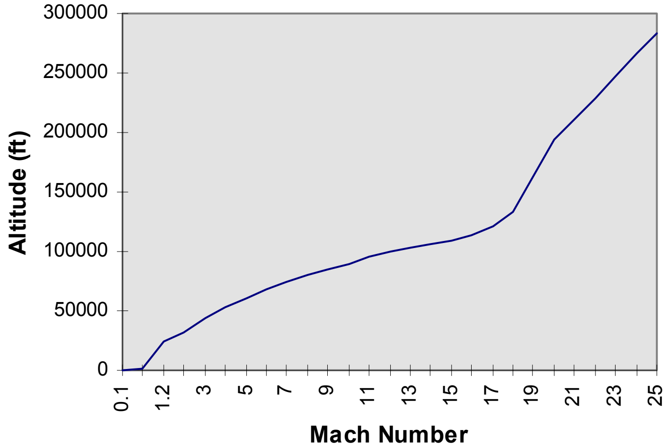

Figure 2: Mach Number Trajectory

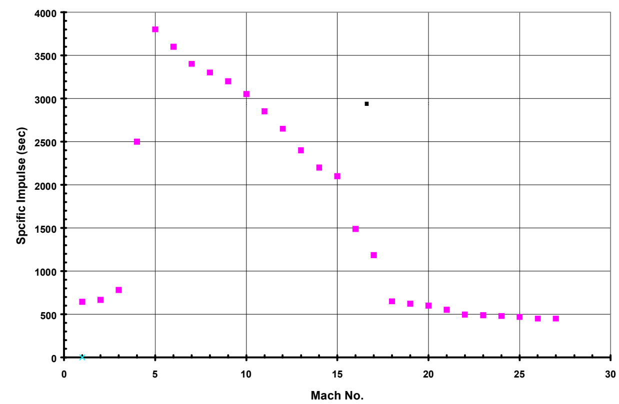

Figure 3: Specific impulse trajectory

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. Mach number and specific impulse for the baseline vehicle configuration is shown in Figs [2],[3].

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.

Overview of the Design Process

Mission data from LMTAS provided values of thrust, specific impulse and Mach number at various altitudes. Incoming air mass flow rate is deduced from the thrust and specific impulse levels. Assuming a fuel fraction and an expected value of specific impulse, individual engine components may be sized. Actual values of thrust and specific impulse are then iteratively used to refine the design. With an appropriate selection of geometry and staging of the various modes, the thrust generated may be made to match the thrust required.

Closure is obtained when the sizes of the engine components required in all the modes are identical, or stand within the limits of variable geometry design, or thrust augmentation mechanisms such as afterburning. Ideal cycle analysis based upon cycle averaged descriptions of the detonation process has been performed to obtain a baseline value of performance parameters. These values are then refined by CFD simulations wherein corrections are made for the unsteady nature of the primary flow.

Engine Layout

Two primary choices exist for the placement of the pulsed detonation rockets in a combined cycle engine. First, the rockets may be embedded in the engine walls or the cowl, thereby reducing the complexity of their design. In such a design, a provision for mixing fuel and air in supersonic flow is crucial. Fig. [4] shows an option based upon strut based fuel injection. Alternately, the rockets may be placed in the struts themselves, as in the strutjet designs developed by Aerojet (Ref. [9]). Such a design has many advantages at high supersonic Mach numbers, when the fuel injectors located on the side of the struts need to fuel a lesser volume of incoming air, than a comparable design where they are placed along a combustor wall.

Figure [5] shows a proposed layout of a strut-embedded pulsed detonation rocket. The struts serve a dual purpose, by housing the rockets for use in take-off, ejector and trans-atmospheric operation, as well as contain fuel injectors to assist modes 2 and 3 proposed here. A cooling mechanism is required to prevent self-ignition of fuel in hot boundary layers. Strut embedded fuel injectors may inject gas or liquid fuel depending on the flight regime and active cooling requirements in the structure.

Combustor struts leading into a plenum chamber (optionally valved for high altitude operation) where pressure levels are maintained at approximately steady values. A compression ramp at the aft-end of the combustion chamber generates oblique detonation waves in the SSTO application. This design may be used for hypersonic cruise applications, where shock waves generated at this ramp produce unstable

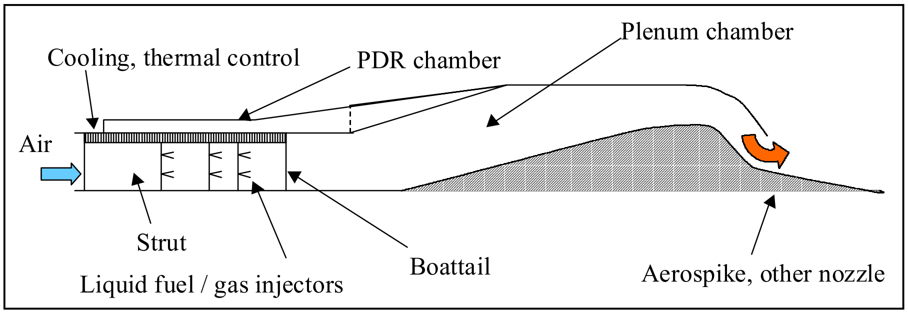

Figure 4: Top-mounted PDR with fuel injection struts

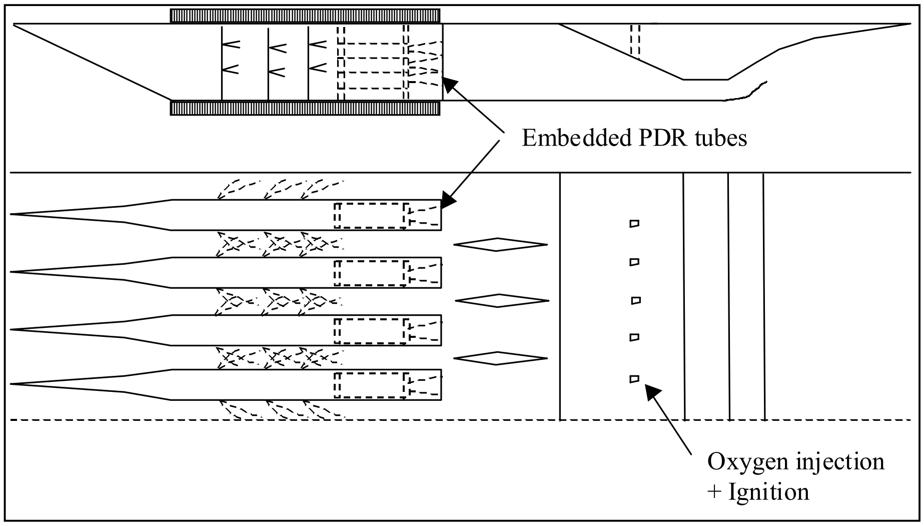

Figure 5: Strut embedded PDR layout

detonation waves which traverse upstream in Mode-2. To assist combustion downstream in supersonic flow, Oxygen ports are located along this ramp, which ease the process of ignition in Mode-2.

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 [8] and [11]. Modes 3 and 4 are more traditional, and refs. [10] and [6] may be consulted for a contemporary account of their physics. Thrust and specific impulse are computed from the stream thrust analysis, an explanation of which has been presented in ref. [5]. A more elaborate presentation of the computational procedure may be found in ref. [7].

<u>Mode-1 Ejector Augmented Pulsed Detonation Rocket</u>

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 ejector rockets have been dealt with in [3],[4]. The analysis of pulsating flows with shock waves is significantly different from that of steady flows. Here, a combined analytical and computational strategy is adopted for this purpose.

Perfect gas CFD studies have been conducted to obtain preliminary estimates of thrust enhancements due to a pulsed core flow. 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. [6] and [7]. 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. [11-a].

An intriguing aspect of pulsed detonation ejector rockets is the potential for thrust enhancement via afterburning. It seems apparent (details shown in ref. [8],) that a series of fuel rich detonating flows emerging into an air stream would ideally suit the conditions for simultaneous mixing and burning in an ejector. Pressure levels immediately aft of the detonation wave are extremely high and cause a rapid transfer of momentum to the secondary flow. While the amplitude of the detonation wave (which transitions to a blast wave) is attenuated rapidly, the temperature changes are much more sluggish. This raises the possibility of detonative afterburning. This will occur for a particular set of conditions when the diffusive mixing of the fuel rich exhaust into the secondary stream occurs more rapidly than the local convective speed of the flow.

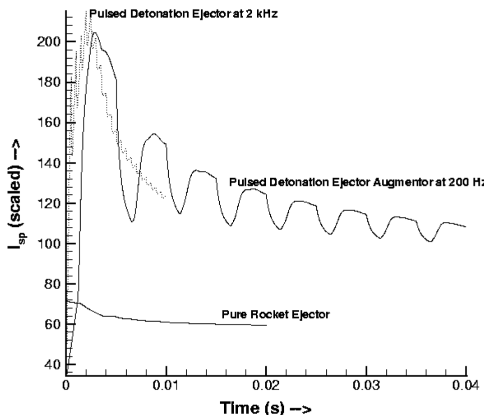

Figure 6: "Isp" comparisons for Ms = 0.15

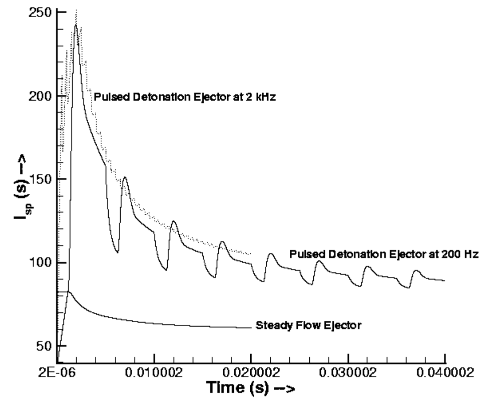

Figure 7: "Isp" comparison for Ms = 1.5

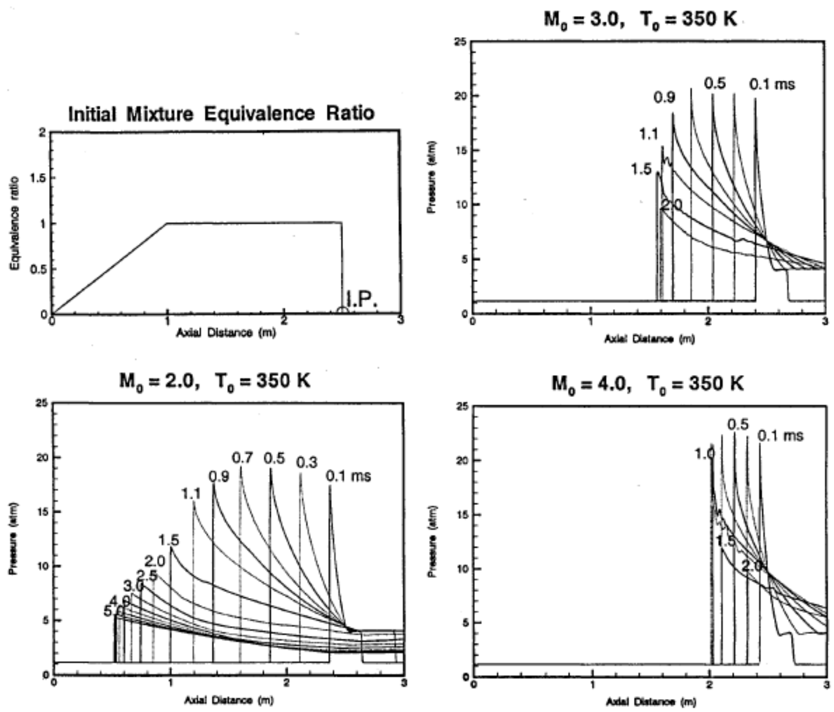

Figure 8: Detonation wave propagation into a mixing zone for Mode 2, T3 = 350 K

<u>Mode – 2 : Upstream traveling detonation wave engine</u>

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 [8] shows the operation of this mode using pressure 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. [8], until the fuel concentration returns to zero. At this point, the wave recedes, and is eventually exhausted from the nozzle. The residence time of the detonation wave within the

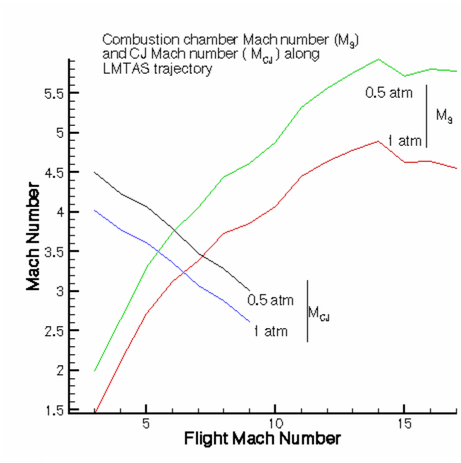

Figure 9: CJ and Combustion chamber Mach numbers along LMTAS trajectory

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 Mach number in the combustion chamber sized as per the LMTAS data, is shown in Fig. [9]. 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. A strut based fuel injector system described earlier is presently thought to be sufficient for this purpose.

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. [10] to transition to a detonation wave propagating upstream.

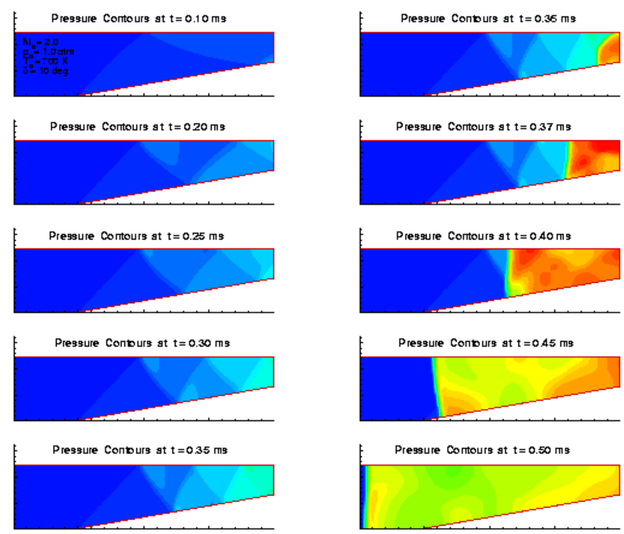

Figure 10: Shock initiation of detonation at low supersonic Mach numbers

<u>Mode –3: Oblique Detonation Wave Engine</u>

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. [11-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 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.

The attainable values of thrust and specific impulse of a pure PDR were investigated. A PDR with different expansion ratio nozzles was the first subject of investigation. While nozzle expansion ratio seems fairly influential on the performance of a PDR, the exact shape of the nozzle contour does not have much impact. Laying aside thermodynamic considerations on the limitations of chemical rockets, specific impulse per se, may be improved by partial filling of detonation tubes, as discussed in ref. [1]. A relation between the specific impulse enhancement due to the partial filling of a detonation tube, is given by (α is the filling coefficient):

$$\text{Isp}(\alpha) / \text{Isp}(\alpha = 1) = \alpha\text{\^{}}(-0.474)$$

The filling "coefficient," taken as the volume filled divided by the total volume of the detonation tube. Numerical experiments have been performed with results that are rather close the predicted values of specific impulse enhancement (see ref. [7] for details). 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. [11-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.

Conclusion

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 Fig. [11]. 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, funded by the state of Texas, using the wind tunnel shown in Fig. [12a] for ejector testing and the arc heater facility depicted in Fig. [12b] for high Mach number (Mode 2, etc.) testing. Of particular interest is a study of pulsed detonation ejectors and their interaction with inlet systems. There is a strong need to resolve several of the issues raised by the engine concept presented here via computational studies. It is likely that a two mode engine (modes 1 and 2) would perform ideally in a hypersonic cruise vehicle.

Acknowledgements

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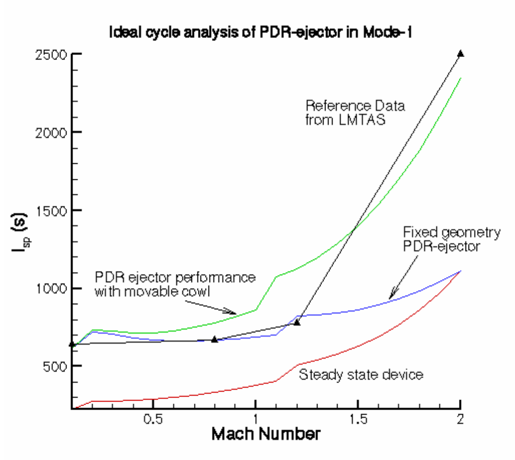
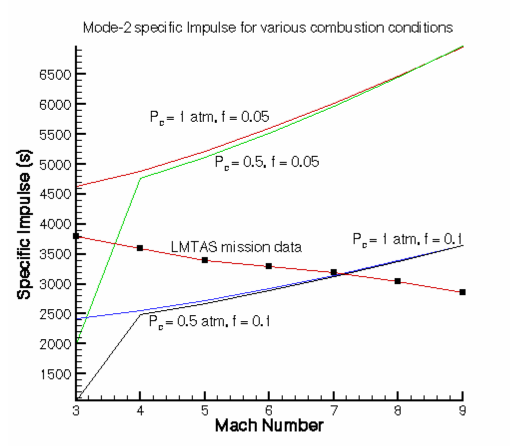
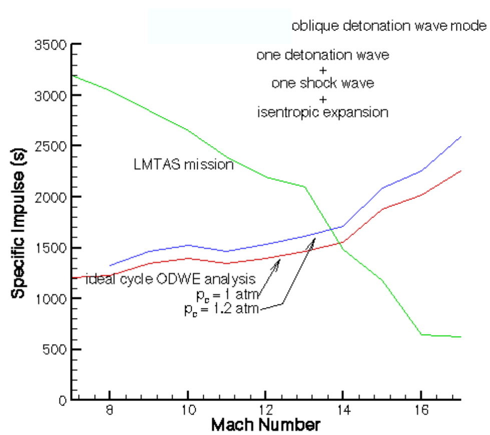


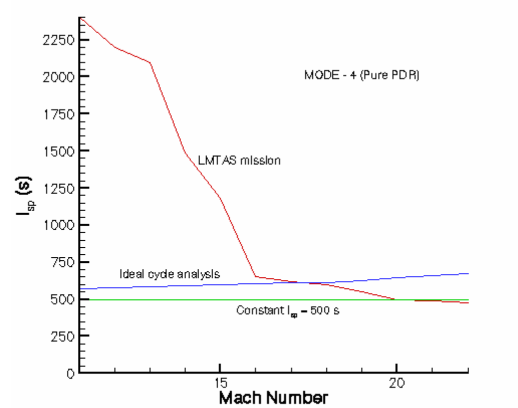
(a) Mode - 1



(b) Mode - 2

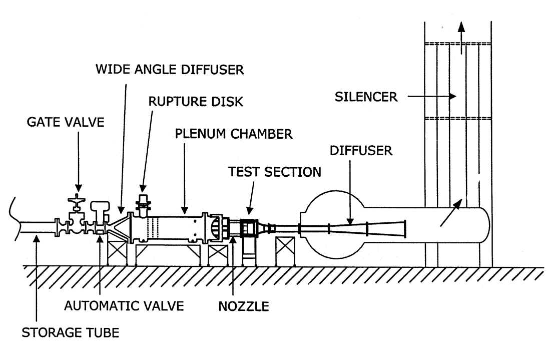


(c) Mode - 3



(d) Mode - 4

Figure 11: Ideal Specific Impulse estimates for the present multi-mode engine



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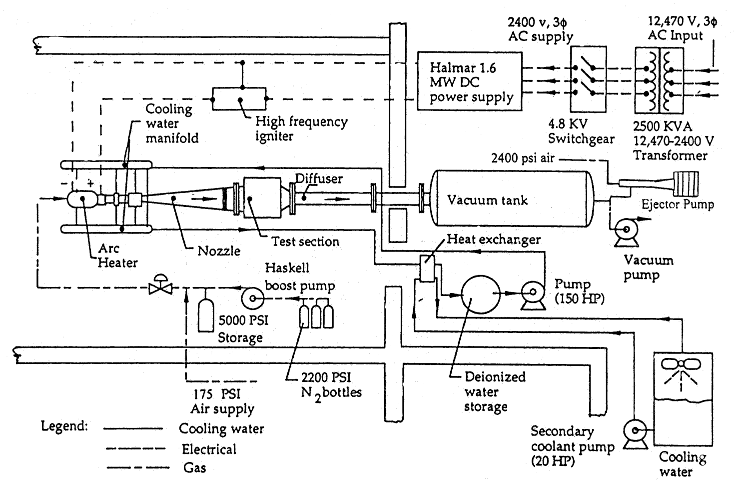

(b)

Figure 12 : **(a)** UTA Aero-Propulsion Tunnel, **(b)** Arc heater test facility