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ANALYSIS OF A PULSED NORMAL DETONATION WAVE ENGINE CONCEPT

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Abstract

A novel multi-mode propulsion system is proposed for potential application to hypersonic and aerospace planes. A unique feature of this concept is the use of pulsed normal detonation waves propagating upstream through a supersonic flow in the combustion chamber as the primary thrust producing mechanism for Mach numbers in the range of about 3-7. The obvious advantage is elimination of the need to reduce the inlet flow velocity to subsonic speeds prior to entering the detonation chamber. This allows the detonation chamber temperature to be kept below the auto-ignition temperature of the fuel-air mixture. Thus it becomes feasible to consider using air-breathing pulse detonation engine concepts for hypersonic flight. Detailed numerical simulations of detonation wave initiation and propagation are presented. Preliminary performance estimates suggest that thrust and specific impulse are comparable or superior to existing RBCC designs.

Introduction

A novel multi-mode propulsion system is proposed for potential application to hypersonic and aerospace planes. The basic concept is illustrated in Fig. 1, and includes the following four modes of operation:

1. An ejector-augmented pulse detonation rocket (PDR) for take off to moderate supersonic Mach numbers,
2. A pulsed normal detonation wave engine (NDWE) mode for operation at flight Mach numbers from approximately 3 to 7, which corresponds to combustion chamber Mach numbers less than the Chapman-Jouguet Mach number,
3. An oblique detonation wave engine (ODWE) mode of operation for flight Mach numbers that result in detonation chamber Mach numbers greater than the Chapman-Jouguet Mach number, and
4. A pure PDR mode of operation at very high Mach numbers and altitudes.

The proposed multi-mode pulsed detonation propulsion system has several features which distinguish it from existing Pulse Detonation Engine (PDE) concepts. The various detonation-based combustion modes are integrated into a single flow path, which should substantially reduce the propulsion system volume requirements. Furthermore, thrust is generated in critical parts of the trajectory by using upstream traveling normal detonation waves in an internal supersonic flow field. This provides the possibility of extending the operational range of pulsed detonation wave engines to much higher flight speeds than can be achieved with conventional PDE concepts. Finally, transition to a steady oblique detonation wave engine mode of operation should occur naturally when the internal detonation chamber Mach number exceeds the Chapman-Jouguet Mach number for the fuel-air mixture.

The overall concept is discussed in detail by Munipalli, et al.[1] They concluded that thrust and specific impulse were comparable or superior to existing RBCC designs.[2] Munipalli also presented detailed performance estimates for Mode 1 (ejector- augmented PDR).[3] The focus of this paper is the performance analysis of Mode 2 (pulsed normal detonation wave engine).

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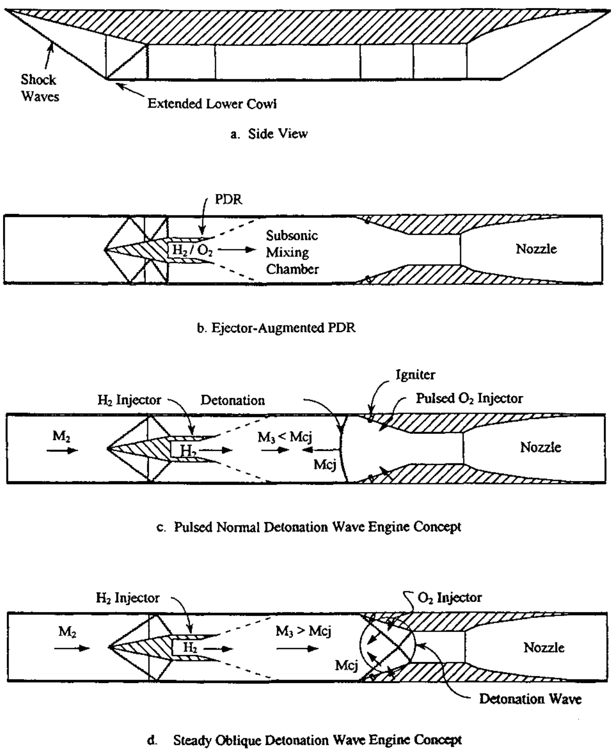

Fig. 1 Proposed multimode propulsion concept

Basic Concept

The pulsed normal detonation wave engine mode (Mode 2) is crucial to the successful operation of the present design concept, and to the knowledge of the authors, has not been dealt with elsewhere in the literature. The unique feature of this concept is the use of pulsed, unsteady normal detonation waves propagating upstream through a supersonic flow in the combustion chamber as the primary thrust producing mechanism. The NDWE is similar in concept to the scramjet, except for the use of pulsed detonation rather than deflagration burning. The obvious advantage is elimination of the need to reduce the inlet flow velocity to subsonic speeds prior to entering the detonation chamber, which allows the detonation chamber temperature to be held to levels below the auto-ignition temperature of the fuel-air mixture. Thus it becomes feasible to consider using air-breathing pulse detonation engine concepts for hypersonic flight.

In the NDWE mode, fuel is injected in a pulsating manner into the supersonic flow field within the detonation chamber. The resulting flow would consist of regions of near-stoichiometric fuel/air mixtures propagating downstream at the detonation chamber Mach number M_3, separated by regions of air. The fuel-air mixture is ignited at a downstream location, producing detonation waves that propagate both upstream and downstream into the fuel-air mixture. Fig. 2 shows an idealized wave diagram of the flow, and CFD simulations of the process for various values of detonation chamber velocity are shown in Fig. 3. These calculations were made with a quasi-one- dimensional CFD code.

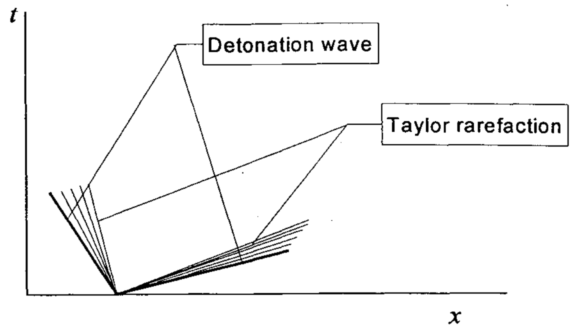

Fig. 2 Idealized wave diagram

For $V_3 = 0$, two opposite-facing detonation waves propagate into the fuel-air mixture, followed by Taylor rarefaction waves. As the detonation chamber velocity increases, the rate of propagation of the upstream facing wave is reduced and wave strength, as indicated by the pressure rise through the wave, increases. In contrast, the downstream facing wave is weakened and convected downstream into the nozzle at an increasing rate. As the chamber velocity approaches the Chapman-Jouguet wave speed, the upstream facing wave becomes a standing wave, and further increases in chamber velocity would cause the upstream facing wave to be convected downstream. At this point, the engine naturally transitions to Mode 3, consisting of a steady oblique detonation wave (ODW) mode of operation (Fig. 1d). Note that for a chamber static pressure of one atm, the static pressure in the region between the two Taylor rarefaction waves is increased to a level of about 5 atm. Furthermore, a significant increase in total enthalpy occurs as a result of the detonation of the fuel-air mixture. Expansion of this high-energy flow through the exit nozzle should provide thrust augmentation over that obtained from a conventional scramjet cycle, which typically operates with combustion chamber pressures on the order of 1 atm or less.

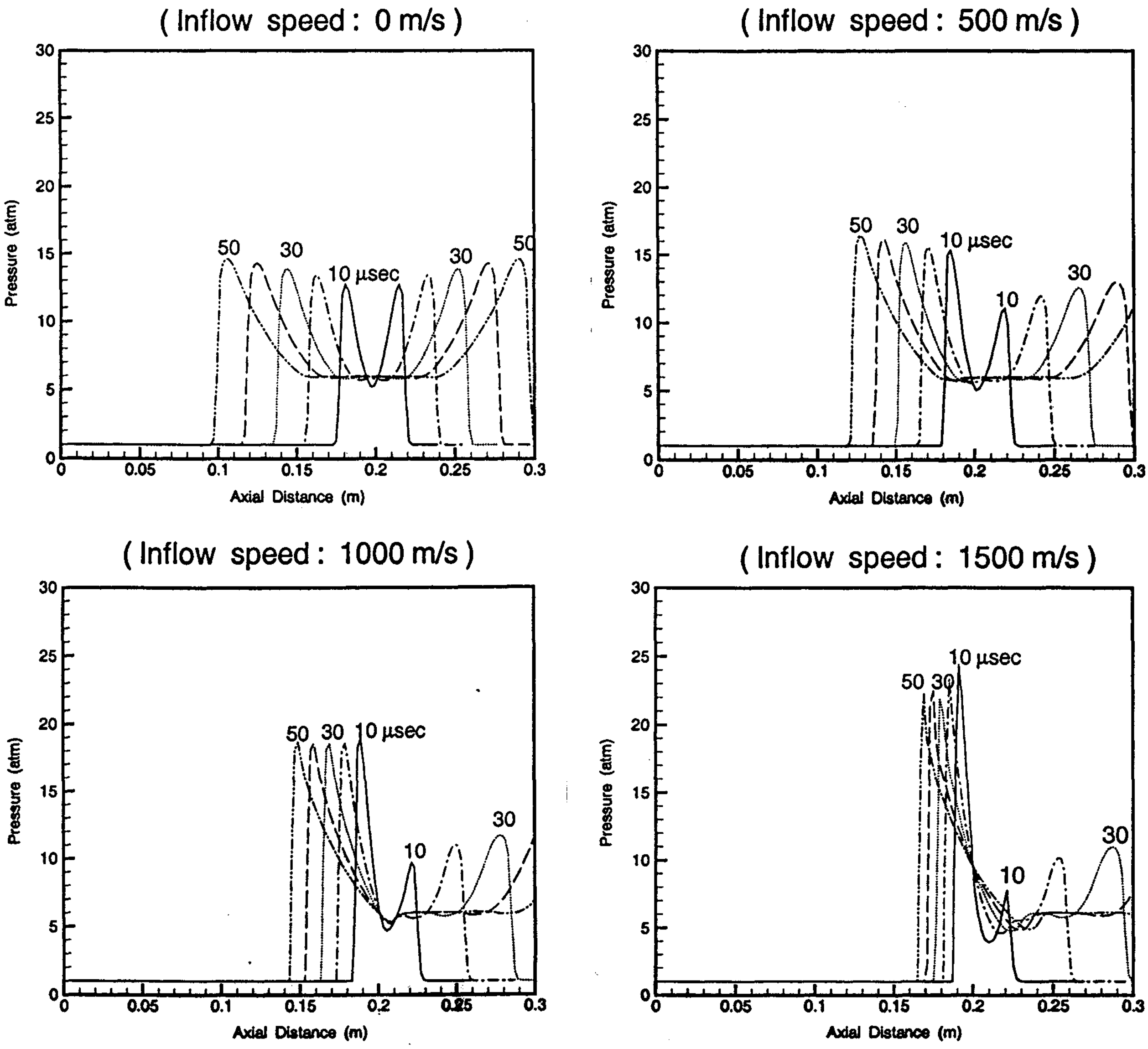

Fig. 3 Effect of detonation chamber velocity on detonation wave propagation

An added benefit the pulsed normal detonation wave engine concept is the self-purging nature of the operation. By proper timing of the fuel injection and ignition pulses, a slug of pure air can be inserted between fuel-air slugs, thus clearing residual combustion products from the detonation chamber before ignition of next fuel-air mixture. Thus, specific impulse penalties associated with separate injection of purge air that are common with conventional PDE concepts are avoided.

Numerical Simulation Model

A two-dimensional time-accurate numerical model to simulate the detonation wave initiation and propagation in a pulse detonation engine was developed by Kim, et al.[4] The time-dependent conservation equations governing inviscid, non-heat-conducting, reacting gas flow in which thermal nonequilibrium is modeled with a two-temperature approximation are written in conservation law form. A discretized set of equations was derived from the governing partial differential equations using the finite-volume method. The advantage of this method is its use of the integral form of the equations, which ensures conservation, and allows the correct treatment of discontinuities.[5] Roe's flux-difference split scheme[6] is combined with the Runge-Kutta integration scheme for an accurate capture of the shock wave both in space and in time.

The inherent stiffness in the chemical reaction model is properly taken care of by the point-implicit treatment of source terms, together with the application of a Local Ignition Averaging Model[4] to each mesh where ignition starts. The partition of internal energy is based on the two-temperature model, and the vibrational energy of each species is obtained by subtracting out the fully excited translational and rotational energy from total internal energy. For the temperature range of interest, the rotational mode is assumed to be fully excited and in equilibrium with the translational temperature T, while the electronic excitation and free electron modes can be safely ignored. Thus, the only remaining energy mode that could be in nonequilibrium with translational temperature is the vibrational energy mode. The two-step reaction model proposed by Rogers and Chinitz[7] is used in this code. This model was developed to represent H_2 - air chemical kinetics with as few reaction steps as possible while still giving reasonably accurate global results.

The accuracy of the CFD code was validated by comparing numerical predictions of wave speed and pressure rise with theoretical Chapman-Jouguet (CJ) results calculated from the NASA CEA code.[8] Furthermore, numerical predictions of wave speed and pressure rise for shock-induced detonation were shown to be in excellent agreement with measured values from The University of Texas at Arlington (UTA) detonation-driven shock tube.[9]

Simulation of Detonation Initiation Process

A critical issue for the successful operation of the pulsed normal detonation wave engine is the ability to initiate a detonation wave in a supersonic flow. Kim's detonation initiation model successfully replicated experimental observations of detonation initiation in quiescent fuel-air mixtures.[4] This model was used to investigate detonation initiation for supersonic flows through a constant area detonation chamber, and a typical solution is shown in Fig. 4 for a detonation chamber Mach number of 2.0. The arc is initiated on one side of the chamber, and initially propagates into the chamber as a two-dimensional wave. The wave impacts the opposite side of the chamber within 0.05 msec, and quickly transitions to a near planar wave propagating upstream into the flow. No adverse affects of the supersonic flow on detonation initiation were observed, and the results are very similar to detonation initiation simulations for quiescent mixtures.

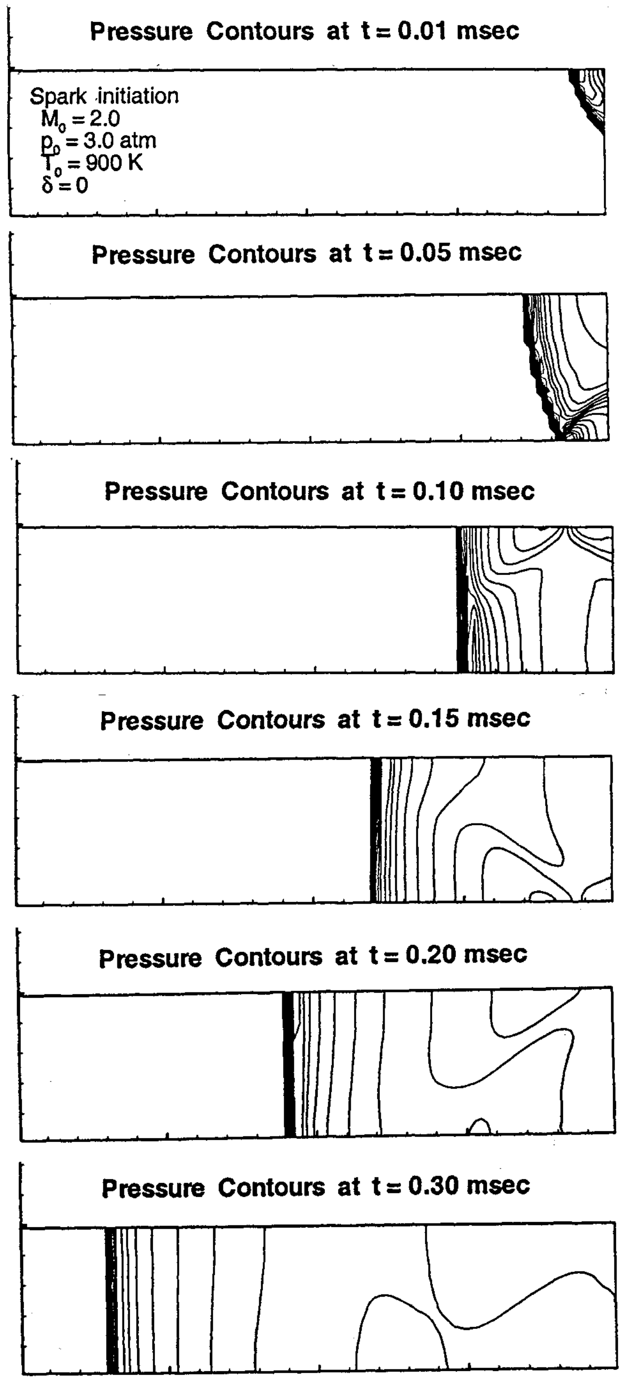

Fig. 4 Upstream propagation of arc-initiated detonation wave in a constant area chamber

For the oblique detonation wave mode of operation (mode 3), a compression ramp is needed to support the oblique detonation wave. Arc-initiated detonation was also investigated for location of the ignition point on the compression ramp. Typical results for arc-induced detonation on a 10-degree ramp for a chamber Mach number of 2.0 are shown in Fig. 5. Again, the initial detonation wave propagates into the flow as a cylindrical blast wave, but quickly transitions to a planar upstream-propagating detonation wave.

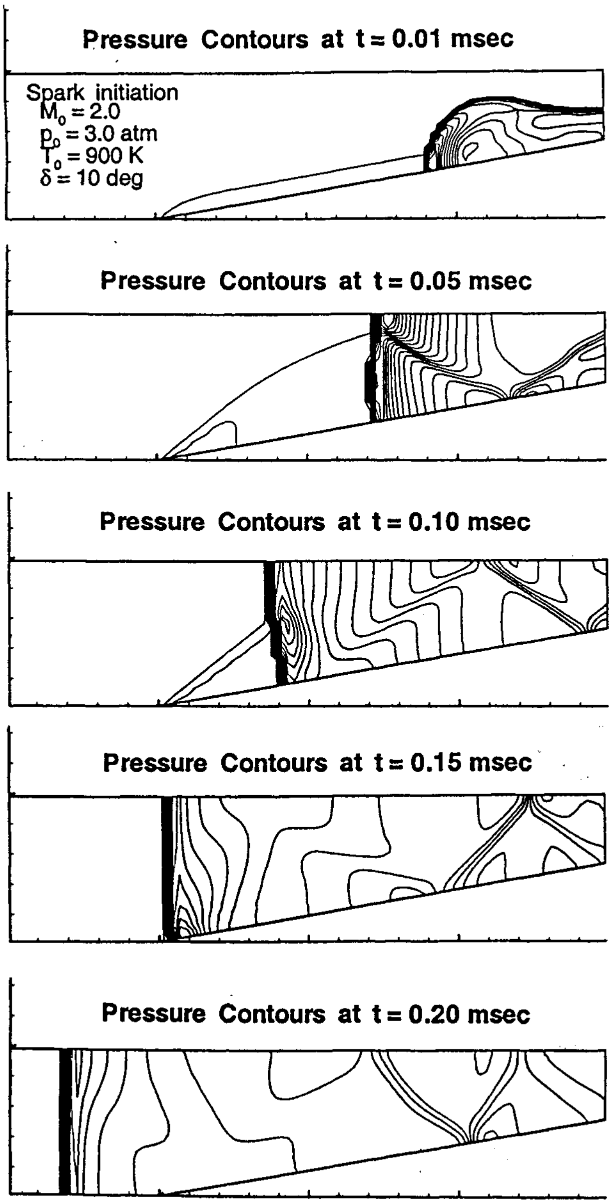

Fig. 5 Upstream propagation of arc-initiated detonation wave on a 10 ^{0} ramp

During the oblique detonation wave (mode 3) investigations,[1] it was observed that at low chamber Mach numbers the oblique detonation wave would detach from the compression ramp and propagate upstream into the flow. Previous experimental investigations showed that shock-induced detonation is an efficient way to initiate strong normal detonation waves.[9] A typical calculation of ramp-induced detonation at a chamber Mach number of 2.0 is illustrated in Fig. 6. Here a Mach 2 inflow impinges on a 10 deg compression ramp. The incoming gas is a stoichiometric hydrogen-air mixture at a detonation chamber temperature of 700 K. As time progresses, an oblique shock wave system is established which triggers the formation of an upstream-propagating oblique detonation wave. The oblique detonation wave quickly transitions to a normal detonation wave that detaches from the ramp and propagates upstream into the chamber. The wave formation is quite similar to that generated by an arc ignition process, and suggests that shock-induced detonation might be used instead of arc-induced detonation. This would eliminate the weight penalty associated with the arc ignition system.

Detonation Wave Propagation

The pulsed normal detonation wave mode is conceived to operate with detonation chamber Mach numbers ranging from about 1.5 up to the Chapman-Jouguet (CJ) Mach number, which is of the order of 4.8 for hydrogen-air mixtures. For Mach numbers greater than the CJ Mach number, the upstream-facing wave will be convected downstream and at that point the operating mode will be switched to mode 3, consisting of a steady attached oblique detonation wave.

One concern with this concept is the possibility of an inlet unstart caused by impingement of the upstream-facing detonation wave on the trailing edge of the shock train in the inlet isolator.[10] CFD simulations of the detonation wave propagation were made for a variety of detonation chamber inflow conditions in order to assess potential adverse operating scenarios and to provide data for subsequent performance estimates. In the present design, fuel is injected into a supersonic flow stream. The subsequent fuel-air mixing process results in a region of varying average equivalence ratio that propagates downstream at approximately the same velocity as the inflow. The actual process was simulated by modeling the flow through the detonation chamber as a mixing

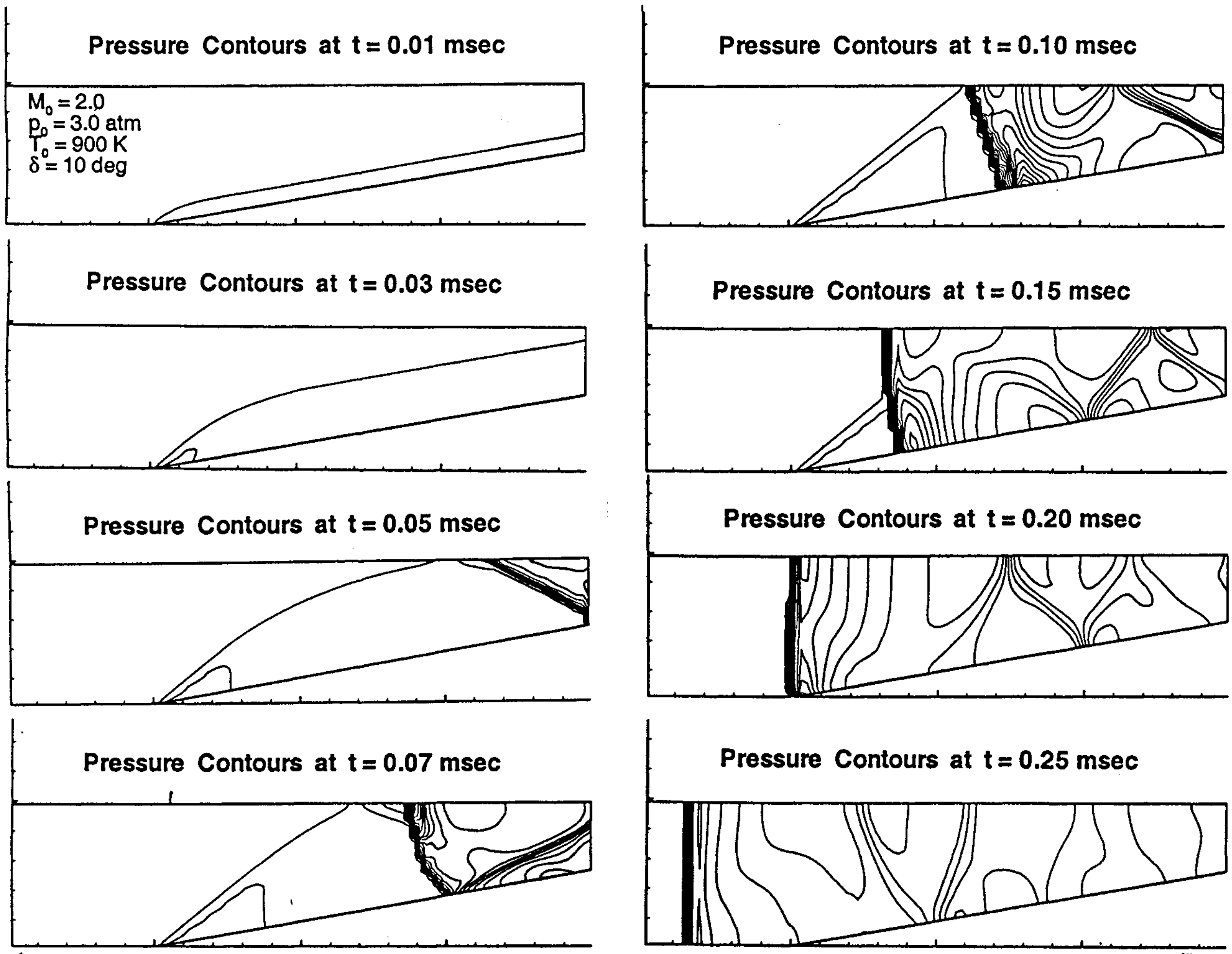

Fig. 6 Upstream propagation of a shock-induced detonation wave

region characterized by an equivalence ratio that varied linearly from 0 to 1, followed by a uniform region with equivalence ratio equal to one. The detonation was initiated at a downstream location, resulting in the generation of both upstream-facing and downstream-facing detonation waves.

Typical results are shown in Figs. 7-8, which illustrates the upstream-facing wave propagation characteristics in the form of pressure vs. distance for different inflow conditions. Corresponding density, velocity and temperature profiles are illustrated in Fig. 9 for the M_3 = 2.0 case. The results show that an upstream propagating detonation wave causes combustion of the fuel-air mixture at the CJ speed. When the entire mixture has been burnt, the leading shock front in the detonation wave degenerates to a shock wave, which rapidly diminishes in strength as a result of the trailing Taylor rarefaction wave system. Ultimately, the upstream propagation of the wave ceases, and at higher detonation chamber Mach numbers the upstream-facing wave reverses direction and is convected downstream with the flow. The time taken for the reversal of this wave seems to depend upon the difference between the fastest local characteristic speed (upstream) and the fastest downstream characteristic speed. The shock wave is accelerated downstream as the flow behind the shock cools and loses enthalpy. Thus, we conclude that for the range of detonation chamber Mach numbers considered, inlet unstart should not be a concern. This conclusion, of course, needs experimental validation.

Performance Assessment

Overall engine performance was estimated by the stream thrust analysis of Heiser and Pratt.[11] In this method, the Stream Thrust function

$$Sa = V\left(1 + \frac{RT}{V^2}\right) \tag{1}$$

is computed at each location in the engine flow path shown in Fig. 10.

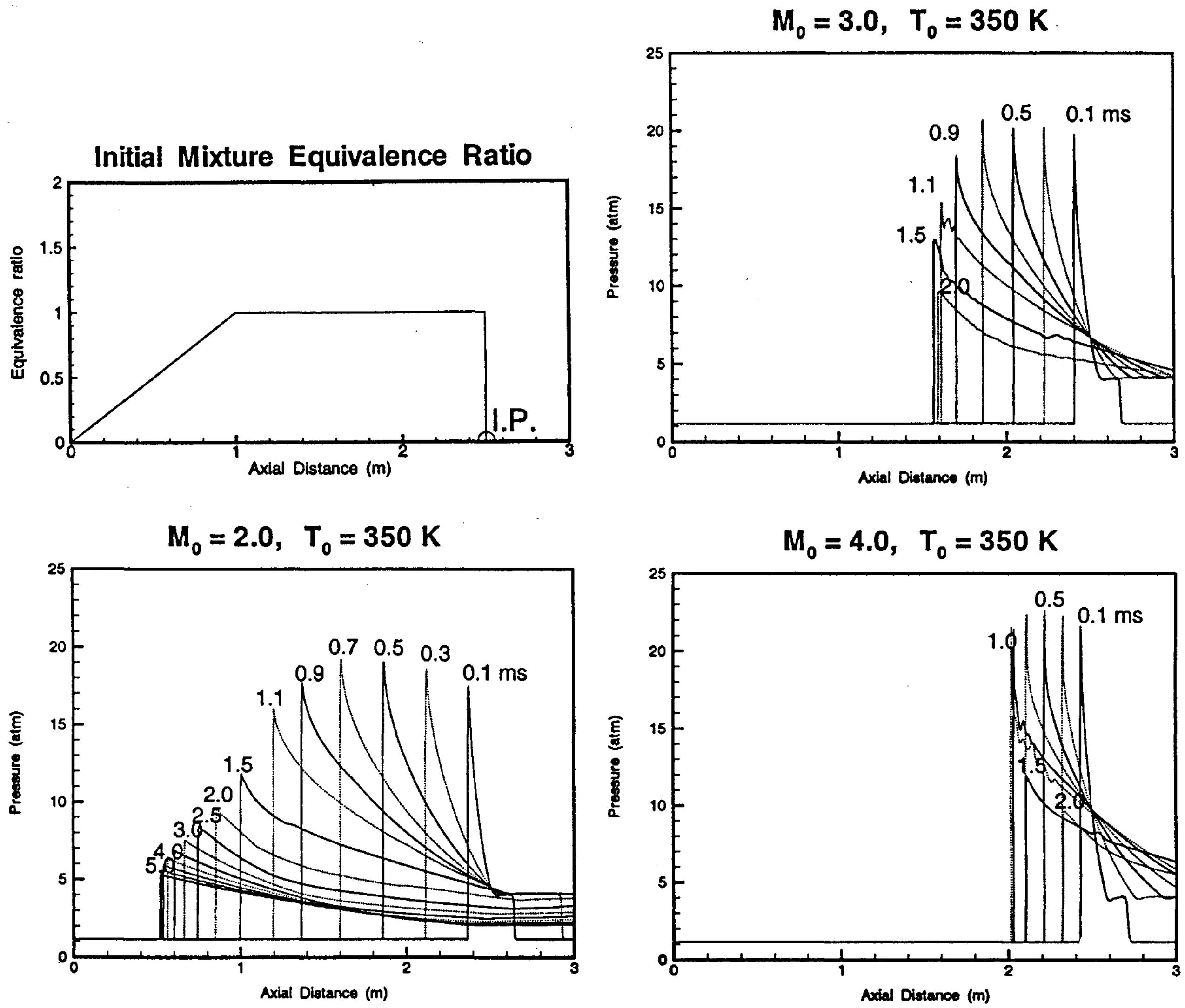

Fig. 7 Detonation wave propagation into a mixing zone for Mode 2, T_3 = 350 ^{0}K

The uninstalled specific thrust produced by the engine is then obtained from:

$$\frac{F}{\dot{m}_0} = (1+f)Sa_{10} - Sa_0 - \frac{R_0 T_0}{V_0}\left(\frac{A_{10}}{A_0} - 1\right) \qquad (2)$$

where f denotes the fuel to air ratio, and A denotes the flow cross sectional area. Specific impulse is obtained from the specific thrust above, using:

$$I_{sp} = \frac{\left(\dfrac{F}{\dot{m}_0}\right)}{g_0 f} \qquad (3)$$

where g_0 is 9.81 m/s^2. The following procedure was used in performing the stream thrust analysis:

1. The stream thrust at the inlet station is computed using trajectory data generated by Lockheed Martin Aeronautics Company – Ft. Worth for a representative SSTO mission.[12]
2. Assuming representative pressures of 0.5 and 1 atm in the detonation chamber, the static temperature ratio in an inlet with a compression efficiency of 0.9 was 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 at station 3 (Fig. 10) are determined, and the effect of an upstream traveling detonation wave is computed. The NASA-Lewis CEA code[8] is used to compute

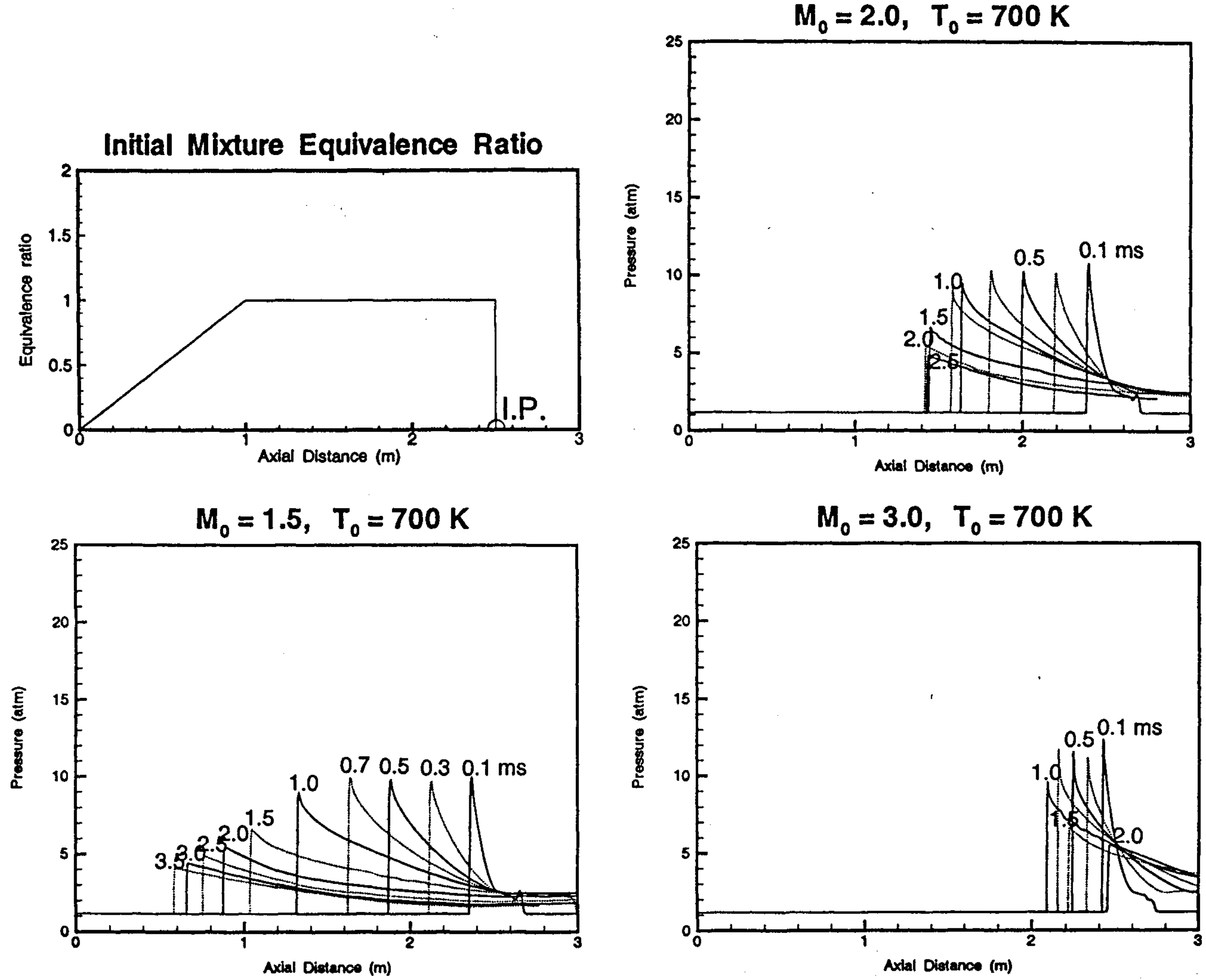

Fig. 8 Detonation wave propagation into a mixing zone for Mode 2, T_3 = 700 ^{0}K

the post detonation conditions for the prescribed detonation chamber inflow conditions, and cycle-averaged values of pressure, temperature and velocity are computed for the remainder of the cycle. 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, using the equation set presented in Heiser and Pratt.[11]

This gives, finally, an estimate for the stream thrust at the exit plane and the exit to inlet area ratios, which are used to compute the specific thrust and specific impulse of this propulsion system across a wide range of Mach numbers. Results are presented in Figs. 11-14.

Fig. 11 shows the variation of static temperature in the detonation chamber along the Lockheed Martin trajectory for assumed values of the combustion chamber pressure of 0.5 and 1 atm. At higher Mach numbers, the chamber temperatures exceed the auto-ignition temperature of the fuel-air mixture, and operation in the detonation wave mode is unlikely. At this point, operation in a pure PDR mode will be required.

Fig. 12 shows a method to compute the limits of

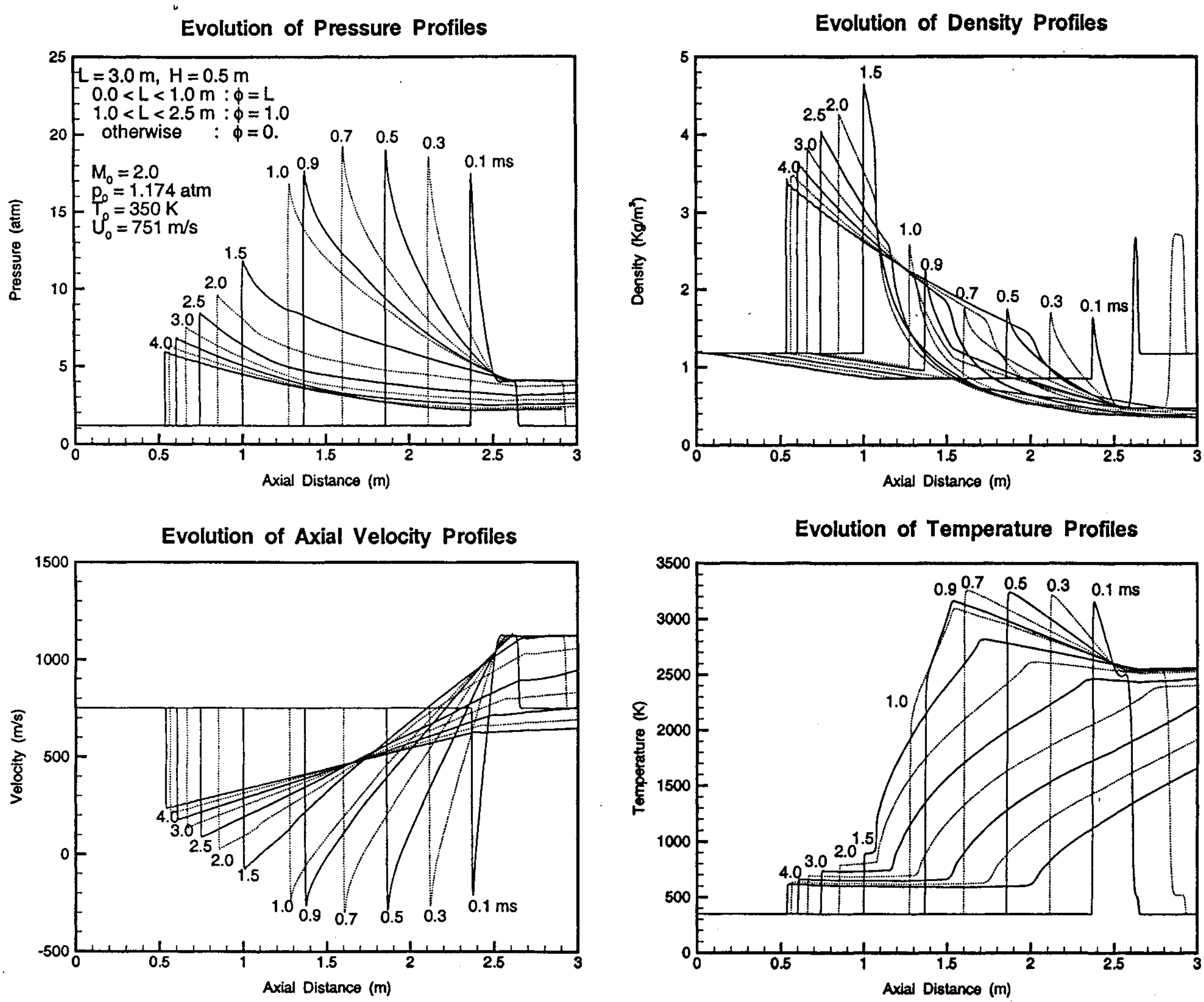

Fig. 9 Pressure, velocity, density and temperature profiles; $M_3 = 2$, $T_3 = 350\ ^0K$

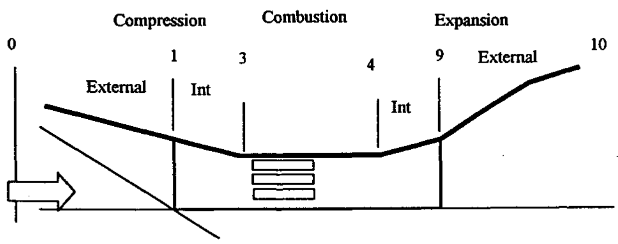

Fig. 10 Engine schematic for stream thrust analysis

operation of the pulsed normal detonation mode. While the lower limit of operation is based upon the extinction distance of the detonation wave, it is possible to tailor the fuel injection process such that upstream traveling detonation waves are extinguished rapidly after they enter the region of zero fuel concentration. The upper limit occurs when the combustion chamber Mach number M_3 matches the Chapman-Jouguet value for the Lockheed Martin trajectory. Fig. 13 shows the location where these Mach numbers become equal (around 6.5 for 0.5 atm, and 7.0 for 1 atm.) Above these flight Mach numbers, the engine should transition to Mode 3 (steady oblique detonation wave engine).

Figs. 13-14 show computed values of specific thrust and specific impulse as a function of flight Mach number for several values of detonation chamber pressure and fuel/air ratio. The curve marked LMTAS shows the required values for the Lockheed Martin SSTO mission trajectory.[12]

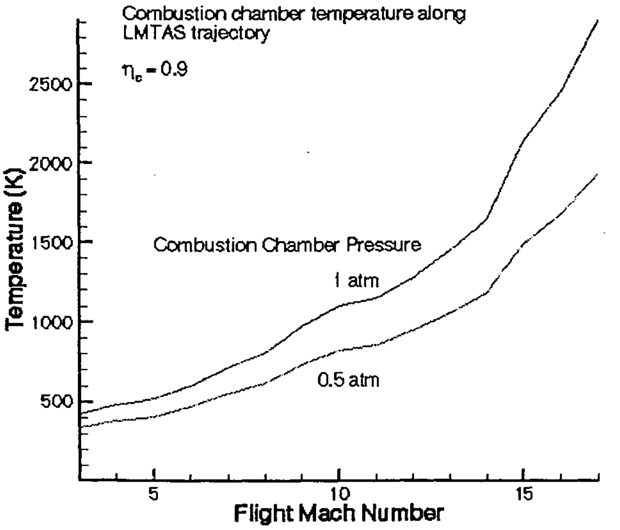

Fig. 11 Detonation chamber temperature variation

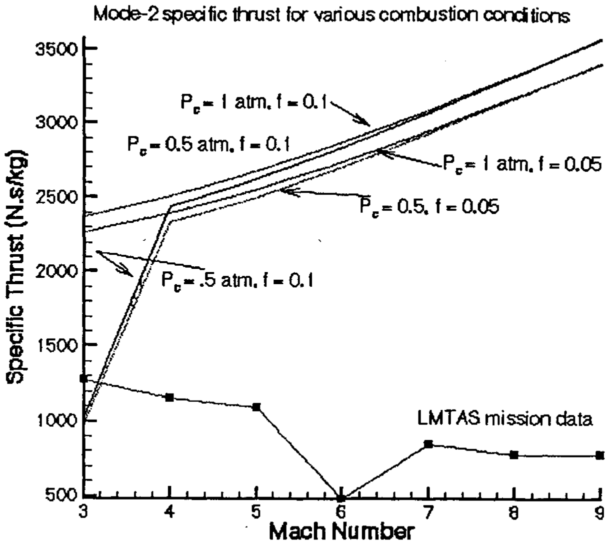

Fig. 13 Mode 2 Specific thrust

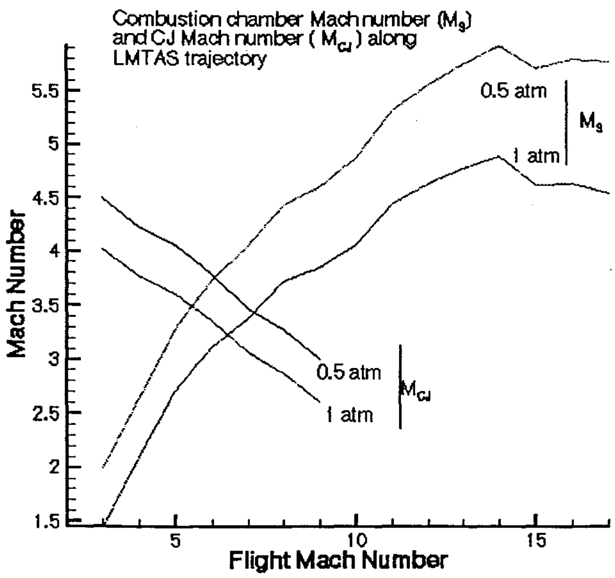

Fig. 12 M_3 and M_{CJ} variation

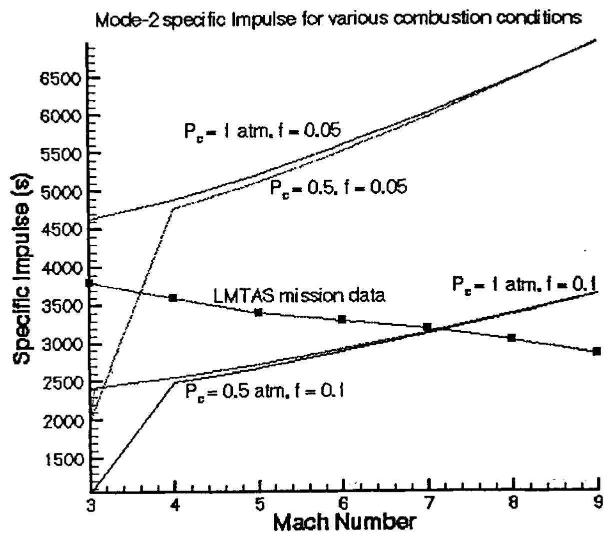

Fig. 14 Mode 2 Specific impulse

Specific thrust values are clearly in excess of those required to perform the mission, and specific impulse values also exceed the required levels for the lower values of fuel/air ratio.

Concluding Remarks

In conclusion, the proposed Mode 2 appears to be a high performance alternative to conventional scramjets or rocket-based combined cycle propulsion systems. The integration of this mode into the engine hinges on the ability to achieve pulsed micro mixing of fuel and air at supersonic speeds. Preliminary calculations indicate that the residence time within the engine is sufficient for mixing caused by vortical mechanisms. It is suggested that a non-reacting shear layer be set up prior to the injection of the fuel from PDR edges. This would entrain the fuel flow and cause mixing in the same time scales as an equivalent supersonic scramjet flow. However, the specific impulse will be much larger due to the detonative nature of the combustion. The operation of Mode 2 has been observed to be superior at higher altitudes. This also suggests the possibility of using Mode 2 for a hypersonic cruise vehicle operating at about Mach 7 at an altitude of over 85,000 ft. A test program is currently in progress at UTA to validate the performance of the Mode 1 engine concept, and funding is being sought to start a second experimental program to validate the Mode 2 performance estimates.

Acknowledgements

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