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ANALYTICAL AND EXPERIMENTAL INVESTIGATIONS
OF THE OBLIQUE DETONATION WAVE ENGINE CONCEPT

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ABSTRACT

Wave Combustors, which include the Oblique Detonation Wave Engine (ODWE), are attractive propulsion concepts for hypersonic flight. These engines utilize oblique shock or detonation waves to rapidly mix, ignite and combust the air-fuel mixture in thin zones in the combustion chamber. Benefits of these combustion systems include shorter and lighter engines which will require less cooling and can provide thrust at higher Mach numbers than conventional scramjets. The Wave Combustor's ability to operate at lower combustor inlet pressures may allow the vehicle to operate at lower dynamic pressures which could lessen the heating loads on the airframe.

The research program at NASA-Ames includes analytical studies of the ODWE combustor using Compuational Fluid Dynamics (CFD) codes which fully couple finite rate chemistry with fluid dynamics. In addition, experimental proof-of-concept studies are being carried out in an arc heated hypersonic wind tunnel. Several fuel injection designs were studied analytically and experimentally. In-stream strut fuel injectors were chosen to provide good mixing with minimal stagnation pressure losses. Measurements of flow field properties behind the oblique wave are compared to analytical predictions.

NOMENCLATURE

C_t	=	Thrust coefficient
I_{sp}	=	Specific impulse
M	=	Mach number
$ODWE$	=	Oblique Detonation Wave Engine
p	=	pressure
q	=	dynamic pressure
R_e	=	Reynolds Number
T	=	Temperature
TAV	=	Trans-atmospheric Vehicle
V	=	velocity
X	=	lateral distance from centerline of strut
Y	=	vertical distance from nozzle floor
Z	=	axial distance fron trailing edge of strut
ϕ	=	equivalence ratio

Subscripts

t	=	total
∞	=	free stream value

INTRODUCTION

The use of detonation waves to initiate and enhance combustion has been proposed since the 1940's[1]. Some analyses have been made using both normal and oblique waves[2,3]. Normal waves are hard to stabilize and they produce higher stagnation

pressure losses than oblique waves. However, it is not clear that stabilized oblique detonation waves have been established in laboratory conditions[4,5]. While free-running normal detonations have been observed to be classsical Chapman-Jouguet waves with sonic gas velocities behind the wave, stationary oblique waves do not necessarily fulfill this condition. We will define an oblique detonation as a wave where the pressure field generated by combustion behind the wave influences the wave itself. This influence is manifested in a rotation of the oblique wave to a more normal orientation. The limiting case is an oblique Chapman-Jouguet detonation where the normal velocity component behind the wave becomes sonic. Further heat addition will cause the wave to detach from the anchoring point and rotate to a more normal orientation.

Several studies have been made recently of the application of oblique detonation waves to hypersonic propulsion[6,7]. These reports have shown that heat addition at an oblique detonation wave can provide combustor performance equal to the conventional scramjet combustor. While the deliberate creation of shock waves may seem to create additional losses, the detonation wave can be considered to be the last wave in a multi-shock diffuser. In addition, there is analytical evidence that heat addition with shocks may not be as inefficient as previously thought [8]. Furthermore, it is highly unlikely that shock waves can be avoided in supersonic combustors with fuel injection and boundary layer regions. Indeed, the design of supersonic combustors should utilize the shock waves to enhance mixing and combustion.

TRANS-ATMOSPHERIC VEHICLE MISSION STUDIES

In order to determine the performance potential of the ODWE, a simulation of a typical single-stage-to-orbit trans-atmospheric vehicle (TAV) mission was completed. Performance and sizing estimations for the TAV were made using a hypersonic vehicle synthesis code for trans-atmospheric designs[9]. Estimates can be made of aerodynamic characteristics, aero-thermal heating, propulsion system performance and structural/subsystem weights. An automated vehicle closure algorithm iterates the trajectory analysis to close the design on both vehicle weight and volume.

To size the vehicles, a mission was selected which carried a payload of 15,000 pounds into a Low Earth Orbit (LEO) of 120 nautical miles altitude. A horizontal takeoff in the easterly direction from Kennedy Space Center was assumed, with an on-station duration of six hours. Two ascent trajectories were studied, with dynamic pressures of 1000 and 2000 pounds per square foot (psf). The flight path was constrained to give 100 pounds per square inch (psi) duct pressure at lower supersonic Mach numbers and a maximum mean surface equilibrium radiation temperature of 2000 F (1367 K) for high Mach numbers. The speed at which the airbreathing engine thrust was augmented by a rocket was optimized to minimize the gross takeoff weight. A descent trajectory was flown near peak L/D to maximize the descent cross-range capability. Fuel reserves of 2% of mission fuel were assumed for the landing maneuver.

General Vehicle Design

The general vehicle configuration, shown in Fig. 1, is a lifting body with aft mounted horizontal and vertical tails. The total propulsion system consists of two airbreathing engines, one for Mach numbers below 6, and a scramjet or an ODWE for the remaining part of the flight. In addition, a rocket engine is used in conjuction with the air-breathing engine for the high altitude, high Mach number portion of the trajectory. Liquid hydrogen is the fuel for all engines.

One-dimensional Engineering Analysis of ODWE

A one-dimensional engineering code was developed to provide the engine data base for an analysis of Trans-atmospheric Vehicles powered by either ODWE or scramjet engines. The engine data was then used in another code for mission analysis studies. The results of these comparisons were presented in Ref.10. The first calculations were made for a scramjet engine. After succesful tests of the modifications, a second version of the code was developed to model the ODWE. Both simulations were one dimensional, and involved many simplifying assumptions which are common in this kind of analysis.

A detailed description of this propulsion model was also presented in Ref.10. For the scramjet case, the inlet operates in a four shock mode which gives good performance over all flight conditions. However, for the ODWE case, the oblique detonation wave acts as a diffuser, so fewer inlet shocks are needed. In this mode, two inlet shocks are sufficient. The viscous and pressure drag forces from nose-to- tail on the underbody or engine side of the vehicle are accounted for in the two engine performance parameters, specific impulse and thrust coefficient. The thrust coefficient is defined as the thrust normalized by the product of dynamic pressure and capture area. Engine specific impulse is obtained by dividing thrust by the fuel weight flow rate. The remaining vehicle drag not accounted for in the thrust coefficient, which includes the top, sides, cowl bottom surface and control surfaces is assigned to the vehicle aerodynamic characteristics. The efficiency of the propulsion system depends on various factors including the flight Mach number, dynamic pressure, forebody shape, fuel temperature and equivalence ratio.

General Engine Performance

The results of the engine performance calculations show that specific impulse and thrust coefficients depend on dynamic pressure, combustion efficiency, fuel temperature and equivalence ratio. Certain trends can be observed. As shown in Fig. 2, it

is evident that higher heat recycling from the engine leads to higher injected fuel temperatures and larger values of specific impulse and thrust coefficient. We assume that the fuel is injected at a constant Mach number of 2.5. As more heat is added to increase the stagnation temperature, significant momentum can be gained from the fuel injection. However, fuel temperature is limited by the amount of heat which can be absorbed from the structure and by the temperature limits of the materials used to store and transport the fuel. In this study, we will assume that 90% of the heat loads have been absorbed by the fuel. The fuel is then heated to a limiting temperature of 1100 K (1520 F), which is representative of the current materials available for fuel storage and transport. If this temperature limit is exceeded, then an amount of fuel in excess of stoichiometric must be used. The resulting equivalence ratio versus Mach number schedule for the scramjet is shown in Fig. 3 for various fuel temperature limits.

Since the ODWE combustor is shorter, a stoichiometric mixture can be maintained to a Mach number of 17.5 compared to 14 for the scramjet, for a fuel temperature of 1100 K. While heat recycle increases engine performance for stoichiometric mixtures, the effect of using excess fuel to maintain a specified temperature limit may increase the thrust coefficients but will lower the specific impulses as shown in Fig. 4. It is clear that the cooling requirements seriously affect the performance of the engine at high Mach numbers.

Scramjet Engine Performance

The calculated performance of the scramjet engine is shown in Fig. 4 as a function of Mach number for a dynamic pressure of 2000 psf and an equivalence ratio schedule which maintains the fuel temperature below 1100 K. It can be seen that the specific impulse begins to drop at Mach 14 due to the rise in equivalence ratios necessary to maintain the 1100 K fuel temperature limit.

ODWE Performance

The ODWE performance was also calculated for dynamic pressures of 1000 psf and 2000 psf. In Fig. 4 we compare the performance of both the scramjet and ODWE for the q=2000 psf case. It appears that the ODWE has better performance than the scramjet at high Mach numbers, but has lower specific impulse below Mach 15. The reduced performance at low Mach numbers is due to the steep wave angle of an oblique Chapman-Jouguet (CJ) detonation, and therefore to higher shock losses. The wave angle can be reduced if either the Mach number is increased or the Chapman-Jouguet Mach number is decreased (i.e. the static temperature prior to the detonation wave is increased or ϕ is decreased). Therefore, the ODWE favors operation at high Mach numbers.

The ODWE also takes advantage of a shorter combustor which requires less cooling and less excess fuel at higher Mach numbers than the scramjet. It can be seen in Fig. 4 that the knee in the specific impulse curve, which indicates the start of the excess fueling schedule, begins at a higher Mach number for the ODWE than for the scramjet. Since the problems of mixing and ignition delay impose a long combustor for high Mach numbers, it is clear that increasing the combustor length causes the performance of the scramjet to drop at lower Mach numbers, when fuel must be injected in excess of stoichiometric.

For the ODWE, the benefits of a shorter combustion chamber, which results in a shorter, lighter engine will also be evident in the vehicle size and weight calculations which are discussed later.

Scramjet Vehicle Performance

A scramjet powered vehicle was modeled using the predicted engine performance data for the trajectory of constant dynamic pressure q=2000 psf. Since the scramjet is very inefficient below Mach 6, a hypothetical engine system with an average effective specific impulse of 1000 seconds was used to propel the vehicle from horizontal takeoff to Mach 6. Aerodynamic heating considerations required that the dynamic pressure of the flightpath begins to drop below the specified value of 2000 psf at Mach 17 to about 250 psf at Mach 22. This low dynamic pressure requirement at high Mach numbers necessitates rocket power augmentation which begins at Mach 18. The amount of thrust provided by the rocket is larger than the thrust produced by the scramjet, and the rocket thrust fraction continues to increase until orbital speeds are reached.

The scramjet powered vehicle which flies a 2000 psf trajectory weighs 460,512 pounds and carries a 15,000 pound payload into orbit. The scramjet engine, low speed engine and rocket motors comprise 8.6% of the takeoff weight. For comparative purposes, a vehicle which flies a 1000 psf trajectory was also studied. This TAV is heavier at 623,000 pounds. The main reason for the increased weight is the lower mass capture per unit area of inlet, which requires a larger, heavier engine and associated structure. Also, the lower thrust-to-weight ratio results in a longer flight time to orbit which consumes a greater amount of fuel.

ODWE Vehicle Performance

The hypersonic vehicle using the ODWE has somewhat different weight characteristics. Since the ODWE offers superior performance above Mach 15, the point of rocket turn-on is delayed to Mach 19. The ODWE can operate at higher Mach numbers than the scramjet, and continues to provide a higher fraction of airbreathing thrust to orbital speeds. Therefore, less rocket thrust is

needed and a lower mass fraction of liquid oxygen (LOX) is consumed, 12.5% versus 15.9% for the scramjet. This represents a weight savings of 22,000 pounds compared to the scramjet. In addition, the shorter combustor length provided by the ODWE allows a shorter, lighter engine which saves about 5,000 pounds. The ODWE represents 3.7% of the gross weight, compared to 4.4% for the baseline scramjet engine. While the fuel weight fraction is higher for the ODWE, the actual fuel weight is 14,000 pounds lower. As a result of all these factors, the ODWE configuration weighs 409,500 pounds, some 51,000 pounds less than the scramjet vehicle (for q=2000 psf), and carries the same payload of 15,000 pounds to orbit. Note that the payload weight fraction is increased from 3.3% of the takeoff weight for the scramjet to 3.7% for the ODWE.

Since the scramjet has better performance below Mach 15, and the ODWE above Mach 15, a combination of these two engines may be ideal. This hybrid engine would use a two-shock diffuser for the whole Mach range. At low Mach numbers, the mixing length and ignition requirements are less severe, and a relatively short combustor can be used in a scramjet mode. At higher Mach numbers, the diffusing shocks would move aft into the combustor. The engine would operate in the oblique detonation mode in the aft section of the combustor. Therefore, cooling is required only for a fraction of the combustor, and the drop in performance due to cooling requirements would still occur only at very high Mach numbers. The design of such a hybrid engine would require more sophisticated, 2-dimensional analysis. Work in that direction is progressing.

ANALYTICAL STUDIES OF ODWE

The analysis of the ODWE has been made with levels of sophistication ranging from one dimensional, steady, perfect gas flow[11] to unsteady, 2-dimensional, viscous, shock capturing codes with finite rate chemistry[12,13]. These codes are used to simulate and guide experiments aimed at proving the existance and stability of oblique detonation waves and their use in supersonic combustors.

Proof-of-concept studies of the ODWE are focused on the establishment of stable oblique detonation waves. A NASA-Ames arc heated hypersonic wind tunnel facility has been chosen for the experimental program. This facility can simulate combustor inlet conditions of Mach number and enthalpy. However, it cannot presently reproduce the expected pressures. Therefore it was necessary to determine if the low pressures would prevent the establishment of a detonation wave. This verification was carried out in several ways. The simplest method utilized a 1-dimensional, steady flow, finite rate chemistry program [11] which calculated ignition delays and combustion behavior behind a 30° oblique shock wave. Inputs to this simulation included a hydrogen-air reaction mechanism taken from the literature[12].

The results of these 1-dimensional calculations demonstrated the strong dependence of ignition delay and combustion rate on pressure and temperature. As temperature and pressure are increased, combustion occurs closer to the wave. However, this program did not simulate any coupling between heat release and wave angle so the question remained whether a detonation had been created. There is very little information in the literature on the spacing between the shock wave and combustion zone for a detonation, except that they appear to be almost coincidental. However, some estimates of coupling can be made by generating characteristics in the combustion zone and determining their intersections with the shock. If these characteristics do not intersect the shock within the bounds of the combustion chamber, then there is not enough coupling to be classified as a detonation. Instead, there is shock induced combustion.

For the nominal experimental conditions, the air in the wind tunnel exits the nozzle at Mach 4.6 with a pressure of 0.016 atm and a temperature of 840 K. Combustion behind a 30° oblique wave takes about 0.5 milliseconds corresponding to a distance normal to the shock of approximately 5 centimeters. Raising the pressure by a factor of 5 shortens the distance to about 0.7 centimeters. This coupling should create a detonation. Indeed, more sophisticated analyses employing a 2-dimensional, fully coupled CFD and finite rate chemistry code have shown the existence of a detonation under these higher pressure conditions[13].

A solution would be to raise the pressure or temperature to guarantee a detonation in the experiment. While the temperature can easily be increased, this effect could cause the fuel to ignite prematurely. One method of raising the pressure would be to create a preliminary oblique wave in front of the detonation wave. However, this may not be necessary since the introduction of hydrogen fuel will also create oblique shocks which can have the same effect. These effects will depend on the size, shape and number of injectors and their location in the experimental set-up.

It was possible to approximate the static pressure rise due to fuel injection when some simplifying assumptions were made. For example, if the fuel injection was assumed to occur at constant pressure in an inviscid airstream, then the fluid momentum can be related to the stagnation pressure losses. While stagnation pressure and Mach number are reduced by injection, the static pressure and temperature are increased. These increases will be beneficial to the ignition process behind the oblique wave.

If the injection losses are due only to the streamwise component of momentum, then the stagnation pressure losses can be easily estimated[14]. For the case where M = 4.6, a stoichiometric amount of hydrogen injected at room temperature results in a stagnation pressure loss of 12%. This pressure loss is equivalent to an oblique shock oriented at about 21° to the horizontal which turns the flow at about 9.5°. The downstream Mach number is then reduced to approximately 3.7 and the static pressure

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and temperatures are raised by factors of 2.4 and 1.3 respectively. These higher pressures and temperatures will shorten the ignition distance behind the oblique wave. The pressure field due to combustion should influence the oblique shock wave and create a detonation. In reality, the hydrogen injection will create shock waves which will cause higher stagnation losses than predicted by this analysis along with higher static pressures and temperatures.

While the increased pressures will shorten ignition delays behind the oblique wave, raising the temperatures may create pre-ignition problems prior to the wave. One consideration for injector design and location is premature ignition of the fuel. A study was made of the effects of introducing fuel at various locations inside the wind tunnel nozzle. The results indicated that fuel must be introduced at a location in the nozzle somewhere downstream of the point where the area ratio is 10. However, extensive modifications would be required to inject fuel in the existing nozzle. This result led to the study of strut type injectors which would be located at the exit of the nozzle.

Injection Simulations

In order to verify some of the simplified analyses of fuel injection and combustion behavior, a more sophisticated computer simulation was employed. This code is described in detail elsewhere[13,14]. Many different simulations were performed to validate the fluid dynamic and chemical kinetics portions of this code. Once the code was validated, it was used to guide the experimental program. The first simulation consisted of wall injection through an orifice normal to the air stream. This configuration, which could model injection from a flat plate resulted in an oblique shock ahead of the injected fuel. Unfortunately, the penetration of the fuel jet was poor. A similar result has been observed experimentally, where fuel jet penetrations appeared to peak at a value of about five times the orifice diameter[15].

In an effort to improve the fuel penetration, a projection or finger was added downstream of the fuel orifice. In this case, fuel was forced over the projection further into the air stream. However, a normal shock was also formed upstream of the injector which reduced the flow velocities to subsonic values. Since a detonation can only exist in supersonic flows, this geometry would preclude the establishment of an oblique detonation wave downstream of the injector. A third configuration was examined where the finger was modified to include a ramp on the upstream side. Fuel penetration remained good and the fuel injection shock became oblique. Most of the flow remained supersonic except for a small recirculation zone behind the leward side of the projection. While this configuration appeared to provide improved penetration and supersonic flow downstream of the injection point, this design would have to be installed on a wall where the high temperatures in the boundary layer region could prematurely ignite the fuel. In addition, the boundary layer might decrease the fuel penetration. For these reasons, it was decided to examine strut type fuel injectors located outside the nozzle. Here fuel could be injected by multiple struts into the core flow region where viscous effects are reduced.

In order provide a better model of the detonation process, a 2-dimensional combustion code was also developed. This code uses the same Total Variation Diminishing (TVD) algorithm as the injection model to capture strong shocks without smearing or oscillations. Temperature oscillations could incorrectly predict premature ignition and invalidate the detonation conditions. Finite rate chemistry is incorporated in order to model the heat release of the detonation process. The chemistry is fully coupled to the fluid dynamics so that heat release will couple to the shock front and show the correct rotation of the detonation wave. The fluid dynamics and chemical kinetics parts of this code were verified using many existing data sets and conditions[13].

Simulation of ODWE Experiment

The focus of this work was the simulation of the flow field in the strut region. This was done first with an Euler (inviscid) computation to obtain the position of the reflected shocks. The computations were done for free stream Mach numbers of 4.5 and 5.4. Two values of the vertical separation between the struts were studied (0.67 inches and 0.75 inches). It was apparent from the results that multiple shock interactions occured between the struts, as well as shock impingement on the flat surfaces of the struts. It was clear that in the case of high stagnation enthalpy, extreme care should be taken in avoiding locally high temperatures. In order to model the strut injection and mixing, a series of computations were made with greater refinements, which included blunting the leading edge of the struts and providing a high grid density. The full Navier-Stokes equations were solved for an assumed laminar case. The conditions were $M_\infty = 5.4$, $T_\infty = 42.2K$, $p_\infty = 0.0128$ atm, $Re_\infty \simeq 2 \times 10^5$ per inch. The total length of the strut is approximately 5 inches and transition to turbulence should occur somewhere at the end of the strut. However, because of the leading edge compressive ramp ($7°$) and the porous transpiration plate in the first half of the flat strut section, transition could be expected sooner. There is, however, no definite way to predict the transition with precision and there were no measurements to determine the properties of the boundary layer on the strut. In addition, when fuel injection takes place, the flow obviously becomes turbulent and the algebraic (Baldwin-Lomax) model is then unable to model the correct physics. Ideally a 2-equation model should be used at this point. The development and validation of such a model which uses the turbulent kinetic energy equation is one of the high priority development areas.

An example of the injection patterns for two struts is shown in Fig. 5. This design indicated hot spots on the center strut which caused the fuel to ignite immediately after injection. In fact, it was necessary to inject nitrogen at the tip of the strut to cool the mixture and decrease the oxygen content of the boundary layer[14]. The strut design is discussed in more detail in the next section.

Fig. 5 shows the computed density contours in logarithmic scale for the strut flow field prior to fuel injection. Of special significance is the boundary layer detachment on the top and bottom surfaces, at the start of the trailing ramp section. In addition, weak recompression shocks are seen to originate from the trailing edge itself. This can also be observed in the computed density field, although the pattern is more complex. It appears that weak shocks are thrown off from the pairs of vortices on opposite sides of the strut. The flow between the struts shows a regular diamond pattern from the multiple shock intersections. There is also a recirculation region on the flat plate in front of the first shock impingement. Because of the good resolution of both grid systems and numerical scheme, the pattern of shocks and expansion waves can be observed in detail, especially near the recirculation region.

The corresponding schlieren record is shown in Fig. 6. In this picture, the flow between the two strut surfaces is very complex, and there seems to be larger areas of flow separation and recirculation on the surfaces. It is difficult to obtain clear experimental pictures of the flow. Most of the features of the flow, however, are reproduced by the simulations, especially the strong bow shocks in front of the injectors and the diamond pattern of shock interactions. The mixing predicted by a 2-dimensional simulation is very poor and is below the measured values. This can be explained by the importance of three-dimensional effects with discrete orifice injection, especially longitudinal vortices. In addition, the turbulence levels in the experimental flow are not well known. A more detailed comparison could be obtained only if 3-dimensional computations are performed. These studies are planned for the future.

Once the mixing simulations were completed, efforts were focused on the oblique wave which would be created by the wedge test body. Since the creation of an oblique detonation wave was the goal, the model was extended to include finite rate chemistry and heat release. This code was also validated using existing experimental data. After this code was verified, several detonation cases were simulated. First, an oblique detonation was modeled in premixed hydrogen-air for $M_\infty = 4.2$, $p_\infty = 0.1$ atmospheres and $T_\infty = 700$ K which are close to proposed test conditions. The results presented in Fig. 7 show the oblique shock wave without fuel reaction. When reactions are allowed, detonative combustion results and the wave rotates to a more normal position as shown in Fig. 8.

Since the fuel injection simulations indicated rather poor mixing, a case was studied where a relatively unmixed fuel jet encountered the oblique shock. The results given in Fig. 9 show that the oblique wave, which was straight in the premixed case, has been severely distorted by the fuel jet. The low molecular weight and high speed of sound of the fuel contribute to a lower Mach number flow in the fuel rich areas which results in a more normal wave front.

EXPERIMENTAL STUDIES OF ODWE

Facilities

The arc-jet facility consists of a 20-MW arc heater supplied continuously with high pressure air. The arc chamber can be pressurized to 10 atmospheres. Air leaving the arc heater passes through a semi-elliptical nozzle with an exit area ratio of 36. A schematic of the test configuration is shown in Fig. 10. Note the injectors and test body which will be discussed later. Enthalpies can range from 5 to 35 MJ/kg (2000- 15,000 BTU/lb$_m$) and air flow is variable from 0.05 to 0.68 kg/s (0.1-1.5 lb$_m$/s). Nominal test conditions for the ODWE experiment correspond to maximum pressure and minimum current. Upgrading of the facility from 10 atmospheres stagnation pressure to 40 atmospheres is now in progress. This higher pressure will allow a closer simulation of the conditions expected at the inlet of a supersonic combustor. A five stage steam ejector pump maintains test cell pressures down to 13 mm Hg.

Injector Design

The fuel injection struts are designed to provide good mixing with minimal losses. Analytical evaluations indicated that hot spots and recirculation zones would exist on and around the struts[14]. These regions would be undesirable since the goal is to create a well mixed fuel-air stream which would not ignite before the oblique wave. The simulations indicated that the transpiration of cool(300 K) nitrogen would prevent the premature ignition of the hydrogen fuel. The strut design, shown in Fig.11, has a transpiration area ahead of the hydrogen injection orifices. The porous "felt metal" allows nitrogen to transpire at a rate of 0.015 kg/s per strut, which represents about 5% of the main mass flow rate.

Hydrogen is injected through seventeen 1.52 mm(0.060 in) diameter orifices spaced 1.27 cm(0.5 in) apart. The orifices are drilled at an angle of 30° to the flow direction. Two or three struts will be utilized depending on the results of further mixing and combustion studies. The fuel-air ratio can be varied up to two times the stoichiometric value. The fuel rich condition will ensure that most of the test body will be immersed in a stoichiometric or fuel rich flow. An estimate of the air-fuel mixing was made using a semi-emperical mixing model[16,17]. This model, which is for sonic injection into a two-dimensional duct, accounts for strut separation and orifice size and spacing. Based on this correlation, the mixing efficiency for 30° injection of a stoichiometric mixture is 70% after a distance of 12 inches.

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Test Body

The oblique waves will be created by a water cooled wedge located approximately one foot downstream of the struts in the test section. Optical access is provided by 12 inch windows on either side of the test section and a schlieren system will provide photographic records of the wave angle with and without fuel. Pressure and temperature transducers on the wedge will be used to assess the state of combustion behind the oblique wave.

Mixing Studies

A series of mixing studies were carried out in the hypersonic wind tunnel. The first set of tests were made with two injection struts spaced from 0.5 in to 0.75 inches apart. the extent of fuel mixing was measured by an on-line mass spectrometer. Gas samples were obtained by a probe which was mounted on a traversing table that allowed motion in all three dimensions. Some results of the fuel-air determinations are shown in Fig. 12 for two locations, 0.5 inches and 12 inches behind the strut trailing edge. While mixing is poor at 0.5 inches, it is significantly improved at 12 inches. The further location was representative of the proposed position of the wedge for the detonation tests. Note that the fuel distribution at 0.5 inches resembles the simulated case of Fig. 5 with relatively unmixed jets. The experiment verified the concerns about thermal failure at the areas of shock impingement on the struts. Further mixing tests with multiple struts were carried out only with cold flow to avoid overheating while hot flow tests were run with a single strut.

Oblique Detonation Wave Studies

After the mixing studies were completed, the wedge test body was installed in the wind tunnel. While the original plan was to locate the wedge 12 inches downstream of the struts, this required the fabrication of new doors for the wind tunnel test section to place the windows in the proper location for viewing. Unfortunately, there was insufficient time to fabricate these doors, so the wedge was located in the field of view with the struts. Only 1.0 inches separated the trailing edge of the strut and the front edge of the strut. While this placed the strut in a relatively unmixed region, it was thought that combustion could occur behind the oblique bow shock of the wedge.

Tests were run with both helium and hydrogen injection to determine the effects on the wedge shock. The effects of fuel injection can be seen by comparing Figs. 13 and 14 for the cases of no injection and injection, respectively. It was observed that the injection of either combustible or inert gases caused a similar displacement of the bow shock. This was due to the low molecular weights and high speeds of sounds of hydrogen and helium. The effect is to lower the Mach number of the flow and cause the oblique wave to be more normal. During one test run, an increase in pressure was observed on the wedge with hydrogen injection, indicating combustion. However, in the limited time remaining for the tests, this phenomenon was not repeated.

CONCLUDING REMARKS

An experimental and analytical program has been undertaken to study the characteristics of stable oblique detonation waves in a NASA-Ames arc-jet wind tunnel. The analytical models have been used extensively to aid in the experimental design and to ensure a successful experiment.

The existance of stable oblique detonation waves has been predicted previously for premixed hydrogen-air in supersonic flows. However, complete mixing of the fuel and air streams is not possible within reasonable distances in supersonic combustors. Therefore, it is necessary to introduce the fuel in a manner that provides good mixing in short distances with minimal losses. Several injector designs were examined analytically and a strut type was chosen for its ability to introduce the fuel in the nozzle free jet. The mixing characteristics and the effects of incomplete mixing on the detonation wave are still being studied.

The simulation of the strut flow field in the ODWE experiment provided great detail on the shock-shock interactions and shock-boundary layer interactions. Notably, the flow structure near the injector is particularly detailed (shock, Mach disk). The results agree reasonably well with the experimental schlieren records.

A mission analysis study compared the performance of vehicles powered by a scramjet or an ODWE. The results showed that the ODWE had better overall performance than the scramjet. The increased performance allowed the ODWE powered vehicle to weigh less than the scramjet powered vehicle for the same payload weight.

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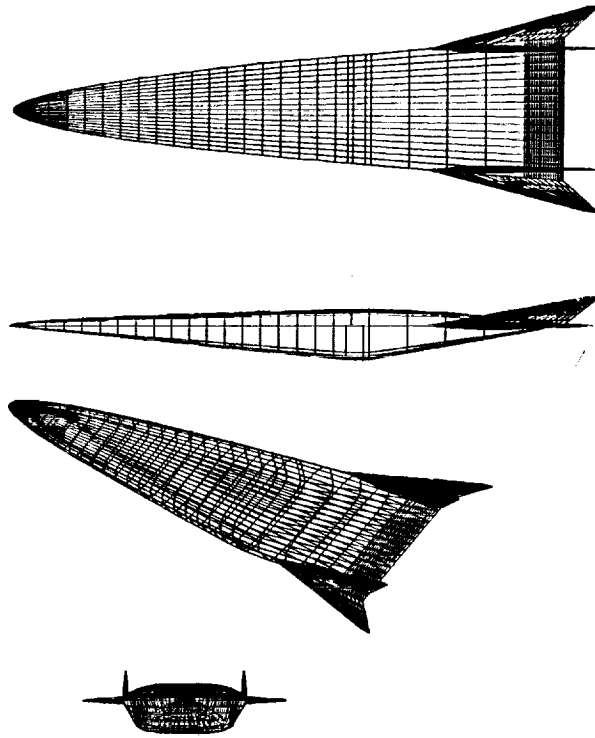
Fig. 1. Schematic of generic hypersonic
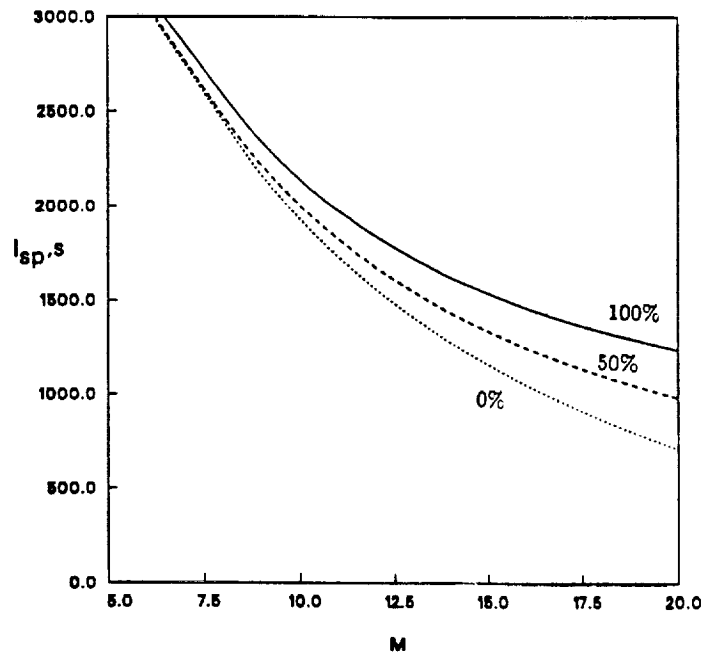trans-atmospheric vehicle used in mission
analysis study.

Fig. 2. Specific impulse versus Mach number for
scramjet engine (q=2000 psf, ϕ=1). Cases
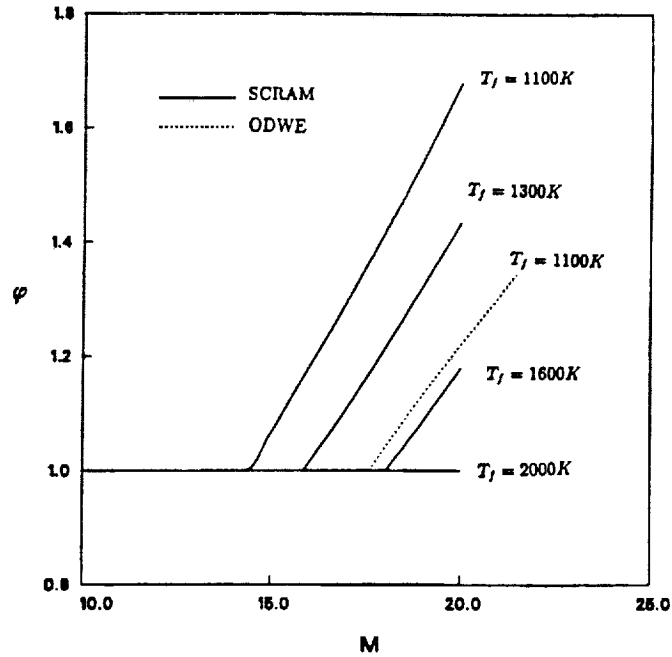shown are for 0%, 50% and 100% of the heat
load absorbed into the fuel.

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Fig. 3. Equivalence ratio versus Mach number for scramjet and ODWE engines at q=2000 psf. ODWE results are shown for a fuel temperature limit of 1100 K while scramjet results are shown for a range from 1100 to 2000 K (1520 to 3140 F).
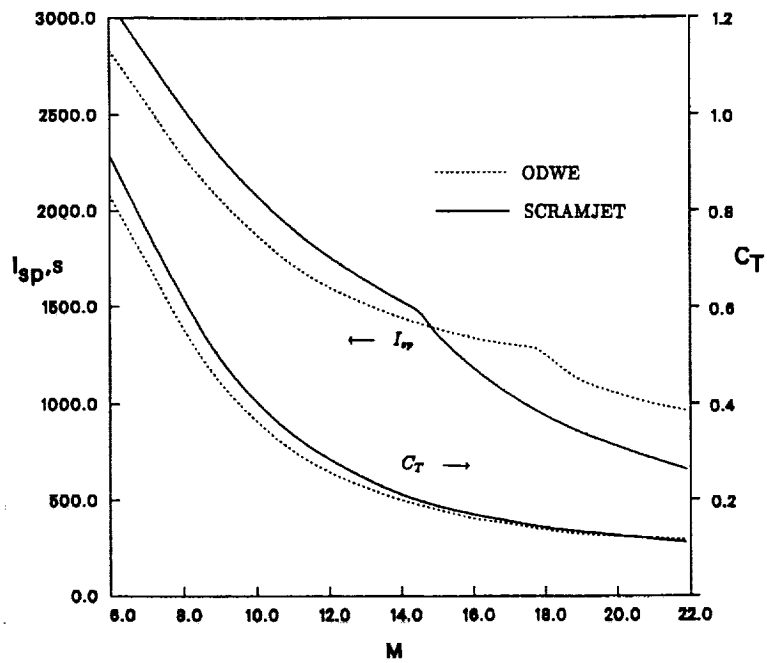

Fig. 4. Comparison of scramjet and ODWE performance characteristics. Shown are I_{sp} and C_T profiles for q=2000 psf, 90% of heat loads carried by fuel and 1100 K fuel temperature limit.
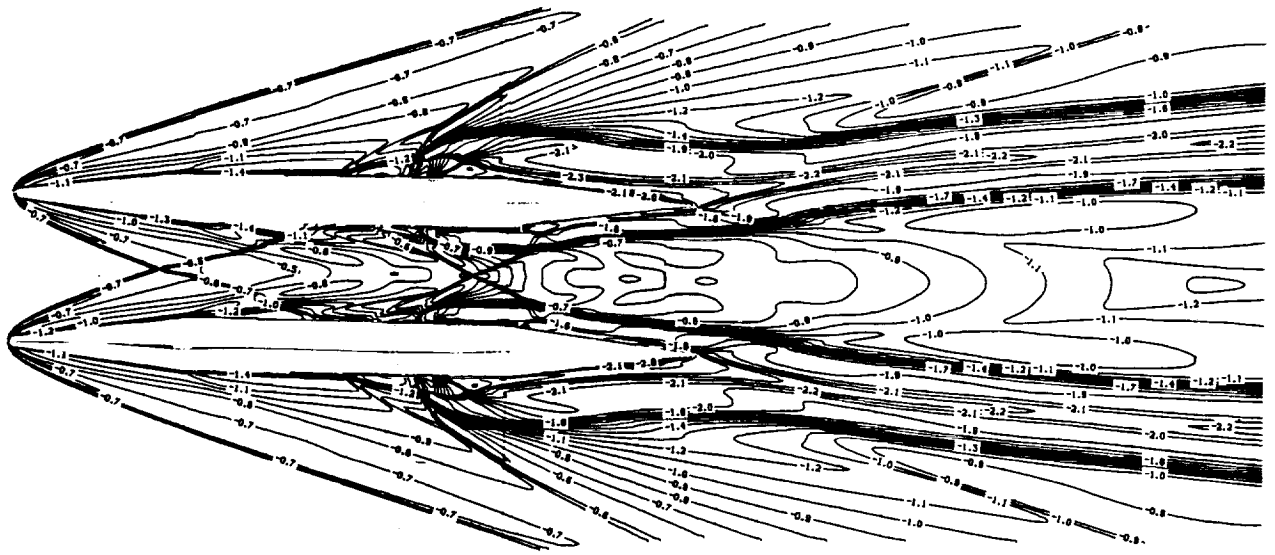
Fig. 5 Predicted logarithmic density contours for fuel injection from two struts in Mach 4.5 flow.
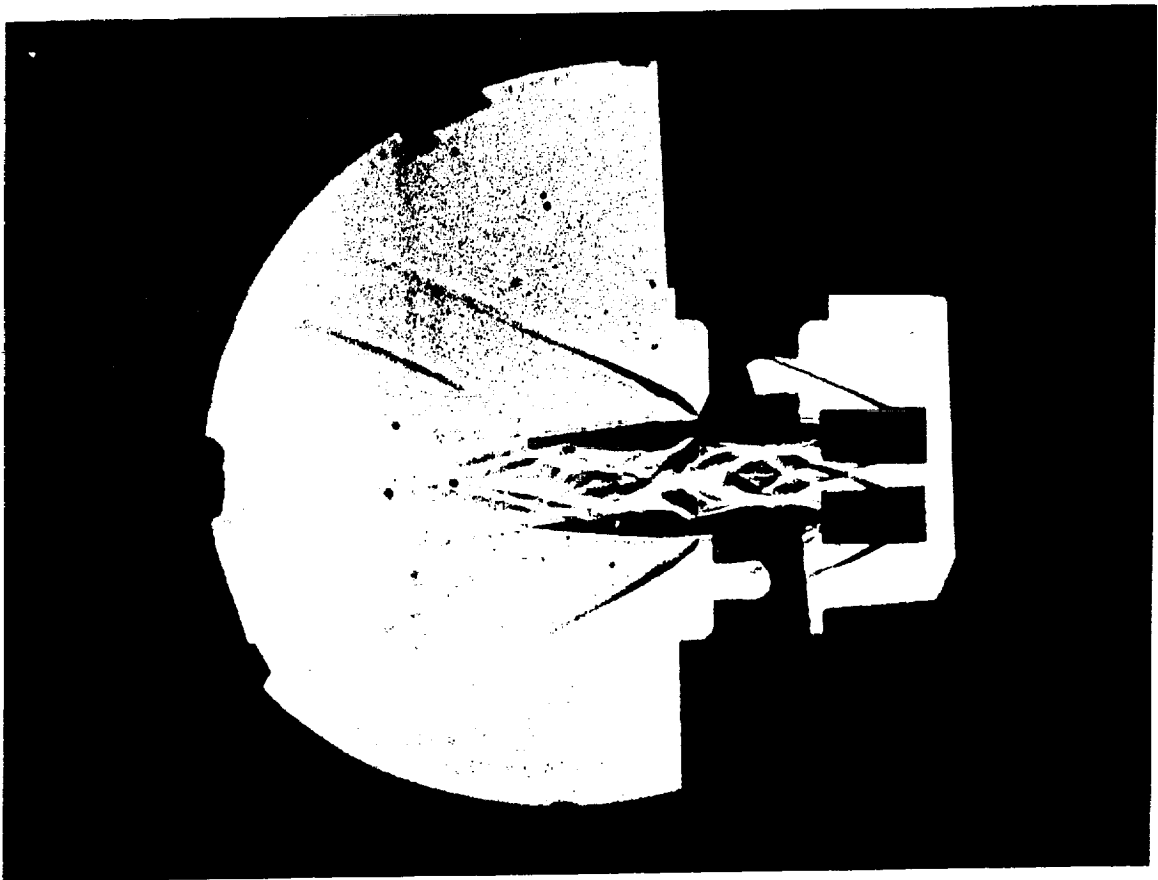
Fig. 6. Shadowgraph of two fuel injection struts in Mach 4.5 flow.

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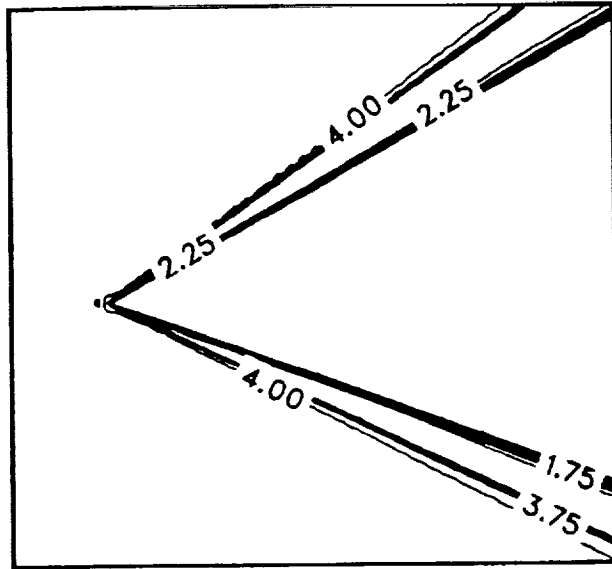
Fig. 7. Mach number contours for non-reacting
stoichiometric air-fuel mixture flowing over
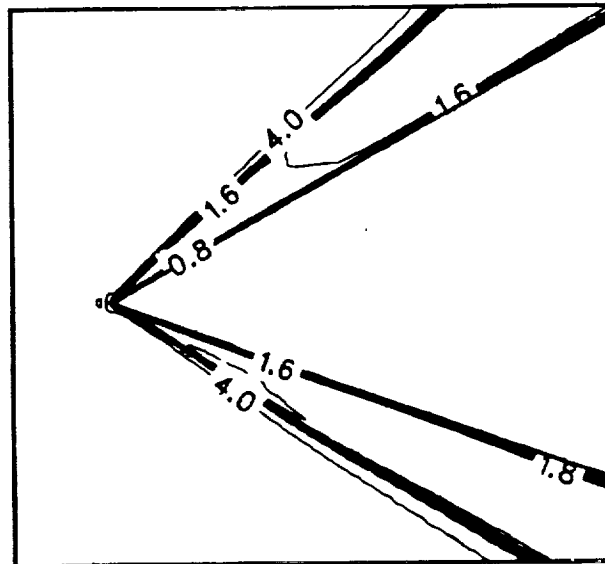a wedge at Mach 4.2.

Fig. 8. Mach number contours for reacting
stoichiometric air-fuel mixture flowing over
wedge at Mach 4.2. The rotation of the wave
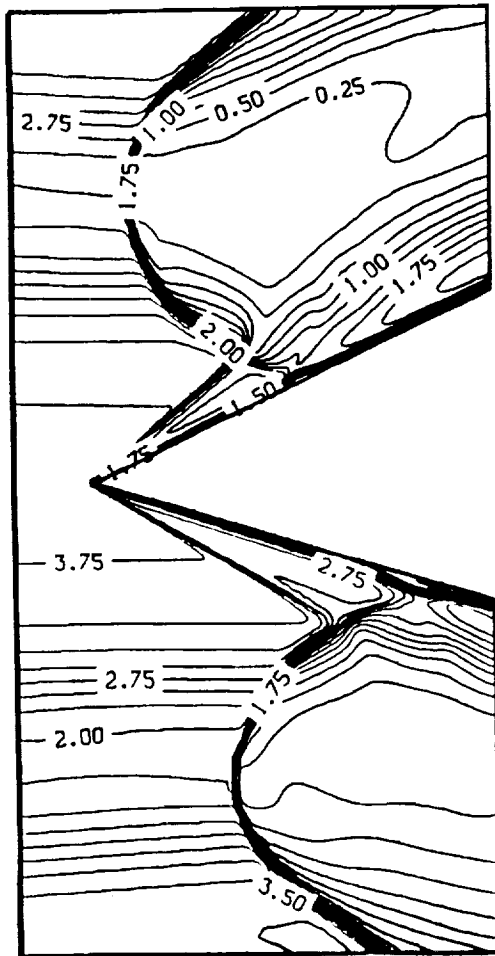with combustion indicates a detonation.

Fig. 9. Mach number contours for relatively
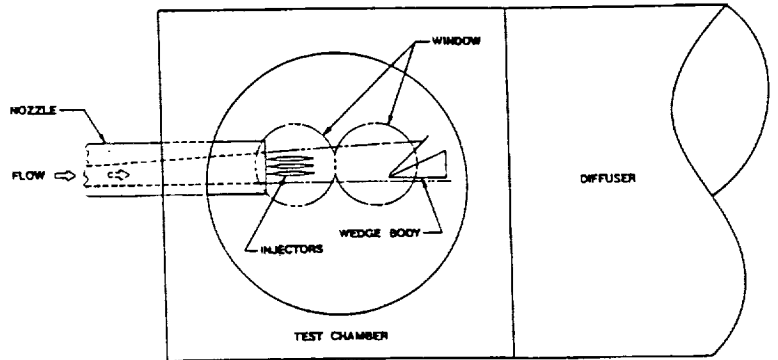unmixed fuel jet flowing over wedge.

Fig. 10. Schematic of test set-up in 20 MW arc heated
wind tunnel. The strut injectors are shown at the exit
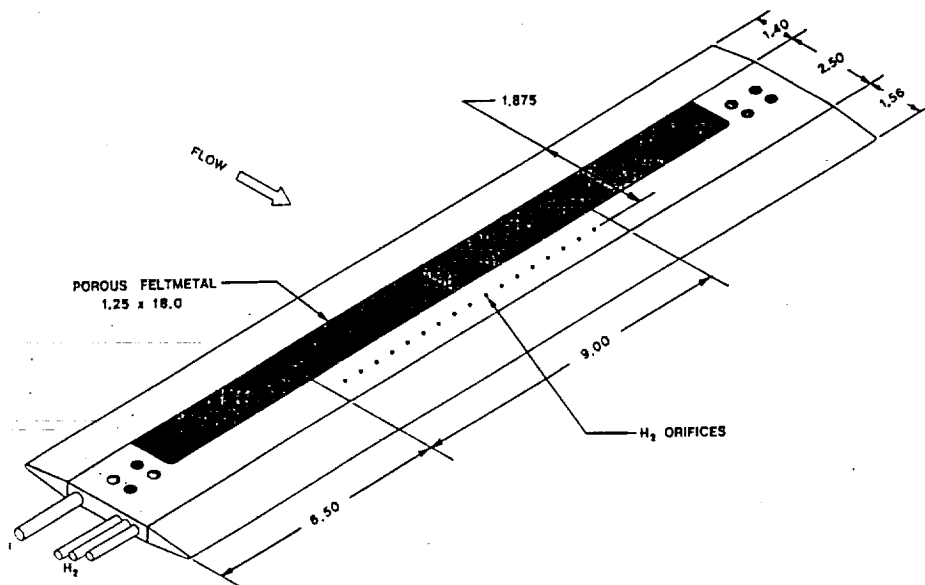of the nozzle and the wedge test body is downstream.

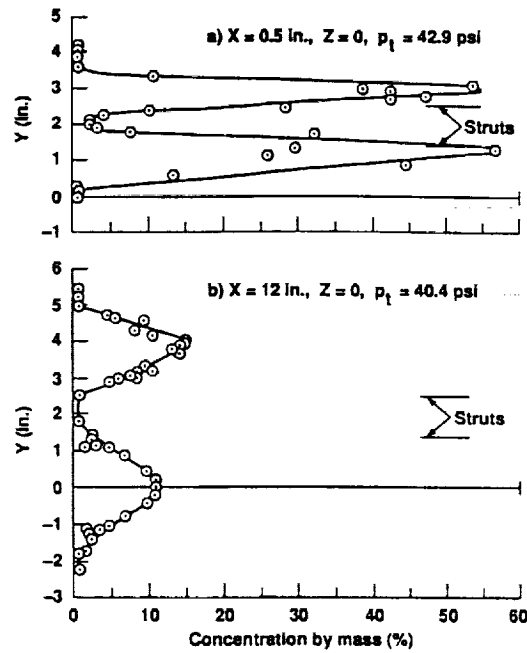Fig. 11. Schematic of fuel injector strut for
ODWE tests.

Fig. 12. Measured fuel concentrations at 0.5 inches and 12 inches behind fuel injection struts.
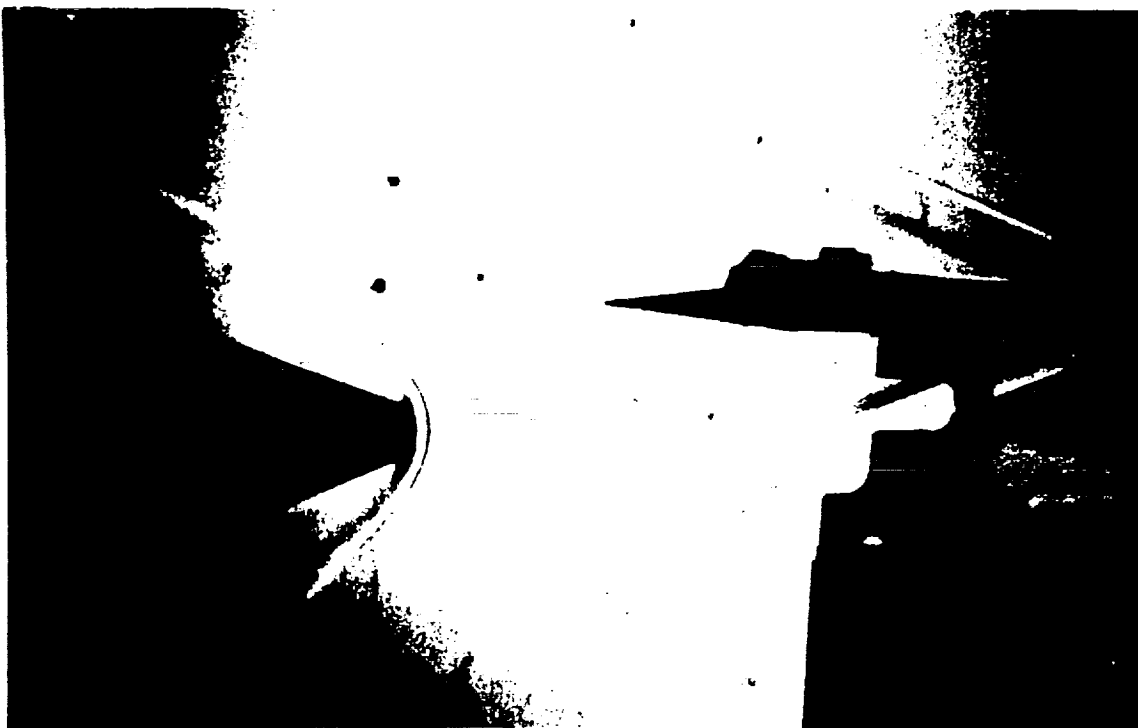

Fig. 13. Schlieren photograph of a shock wave created by a wedge in Mach 4.5 flow. A single strut fuel injector is positioned slightly below the wedge centerline. No fuel is injected in this case.
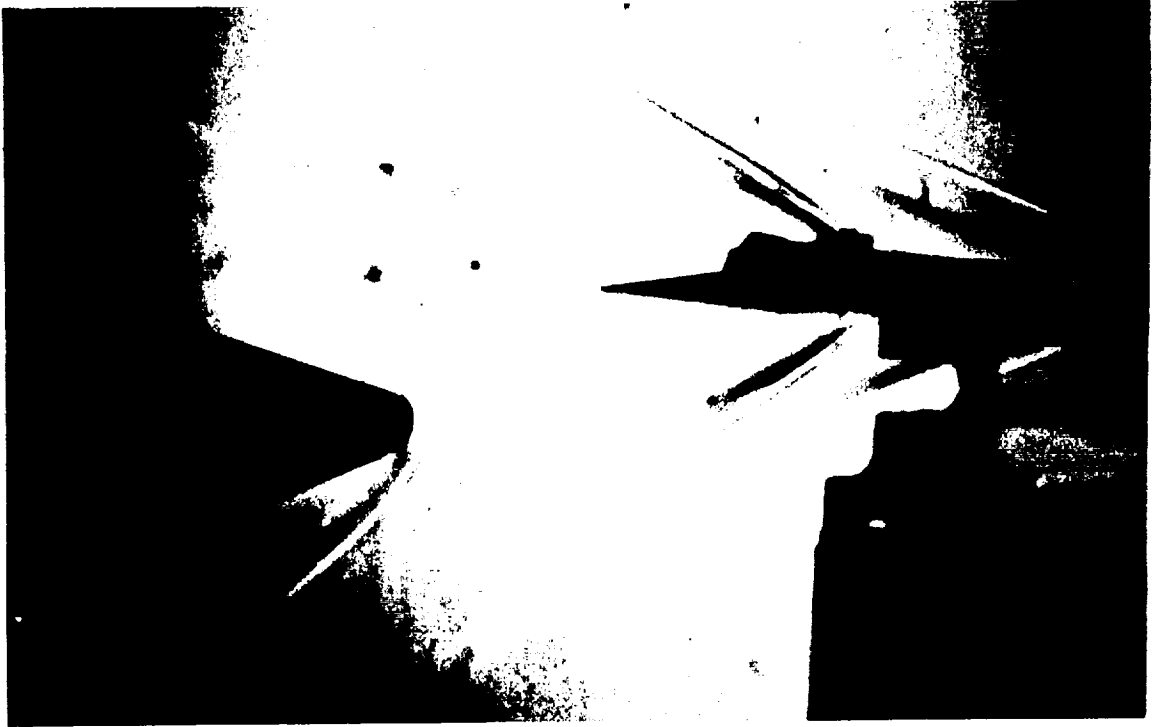
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Fig.14. Schlieren photograph of an oblique wave created
by a wedge in Mach 4.5 flow. Fuel is injected from a
single strut. Note the displacement of the lower portion
of the wave compared to the previous figure.

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Report Documentation Page

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16. Abstract

Wave Combustors, which include the Oblique Detonation Wave Engine (ODWE), are attractive propulsion concepts for hypersonic flight. These engines utilize oblique shock or detonation waves to rapidly mix, ignite and combust the air-fuel mixture in thin zones in the combustion chamber. Benefits of these combustion systems include shorter and lighter engines which will require less cooling and can provide thrust at higher Mach numbers than conventional scramjets. The Wave Combustor's ability to operate at lower combustor inlet pressures may allow the vehicle to operate at lower dynamic pressures which could lessen the heating loads on the airframe.

The research program at NASA-Ames includes analytical studies of the ODWE combustor using Computational Fluid Dynamics (CFD) codes which fully couple finite rate chemistry with fluid dynamics. In addition, experimental proof-of-concept studies are being carried out in an arc heated hypersonic wind tunnel. Several fuel injection designs were studied analytically and experimentally. In-stream strut fuel injectors were chosen to provide good mixing with minimal stagnation pressure losses. Measurements of flow field properties behind the oblique wave are compared to analytical predictions.

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