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Review of Shock-Induced Supersonic Combustion Research and Hypersonic Applications

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This article focuses on research in supersonic combustion and combustion kinetics in high-speed flow between 1959–1968, and the application of the experimental results to hypersonic propulsion. The analysis discusses both advantages and problems for premixing the fuel and employing shock-induced combustion as an ignition method for a scramjet flying at a high Mach number. The experimental tests are discussed, including implications to the chemical kinetics of the high-velocity combustion process. The conditions were confined to relatively low pressure, less than 2 atm (200 kPa). The results were considered to be mainly applicable for high-altitude scramjet flight, at low static pressure, where chemical reaction distances will be long. At these lower pressures, "shock-induced combustion" may be the predominant effect in a scramjet application, and it has some advantages that are discussed. The relation between shock-induced combustion and "detonation" is also discussed. In addition, an attempt is made to resolve the conflicting experimental data published in the 1960s relating to "standing detonation waves" and shock-induced combustion.

I. Introduction

SERIOUS SCRAMJET design attempts and experiments date back to the early 1960s.[1] Waltrip[2] and Billig[3] provide an excellent general background of past and recent scramjet development and discuss many of the problems needing work before this concept can progress to a flying propulsion system. Because of technical difficulties and lack of financial support, the beginnings of the 1960s came to a standstill. With the recent interest in the National Aerospace Plane (NASP), the problems of scramjet propulsion are receiving renewed attention and enthusiasm, of which Refs. 4–6 are typical.

The early thoughts and questions on supersonic combustion (1960s) were based on the work of Avery, Dugger, Nicholls, Gross, Ferri, and others (Refs. 1, 7–10), and are summarized as follows:

1) Because the combustion was presumed to be supersonic, it was assumed to be related to detonation waves. Detonation wave theory and experiments were well known.

2) Because the high pressure and temperature generated by a normal-shock detonation wave can be destructive, the question remained as to whether these effects can be eliminated by controlled fuel injection and mixing.

3) Oblique-shock detonation waves were theoretically possible, but had not been reported in the open literature for continuous flow systems prior to the experimental work of Rubins, Rhodes, and Cunningham (Refs. 14–16).

The classical detonation wave is exemplified by the analysis of Chapman and Jouguet.[11] Static pressure rise behind the detonation wave can be very large, resulting in close coupling of the shock and exothermic reactions. The Zeldovich-von Neumann-Doring (Z-N-D) model offers a graphic explanation.

Consequently, scramjet research by early investigators took the road of injecting fuel in the region where combustion was wanted. This approach has continued for working scramjet engines now under development in the range of Mach 5–10. Here, supersonic flow combustion depends on compressed air temperatures, shocks, hot pilots, or a combination of these to produce ignition. Mixing is obtained by a combination of turbulence, shocks, and diffusion. Interacting aerodynamic and chemical instability create a complex problem in design and performance of a propulsion system, and extensive work has been done to resolve these problems.[3]

However, from a simplified chemical kinetic point of view, we may view the shock wave as a sudden compression that raises the static air temperature to a point higher than the ignition temperature of the combustible material that may be present. When this occurs, chemical activity between air and combustible will begin, according to the chemical kinetics for such reactions. This phenomenon was observed and is the subject of this article. We include a brief description of the research done in supersonic combustion and its potential application to hypersonic propulsion, as discussed in Refs. 13–20.

The work started in 1959 (discussed in Refs. 13–20), used a Mach 3 water-cooled tunnel designed by Gross[12] to operate at temperatures in the range of 3500°R (1944 K). Gross's work with the tunnel was cut short with the closure of his laboratory, and the tunnel was subsequently transferred to another research laboratory.[21]

II. Mach 3 Tunnel Supersonic Combustion Research, 1959–1968

A brief description of the experimental and analytical work performed during 8 yr of research with this tunnel[13–20] follows.

A. Normal Shock-Induced Combustion

The impetus for this initial study was the work done by Nicholls et al.[7] and Gross et al.,[8,9] in which hydrogen fuel was injected in the downstream direction in a heated supersonic

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airstream. On passing through a normal shock, phenomenon were observed that were variously interpreted as supersonic combustion, or a standing detonation wave.

The experimental work by Nicholls et al. employed an underexpanded jet, producing a "shock bottle" and a standing normal shock (Fig. 1). Cold hydrogen fuel was injected and mixed upstream into heated air. The air was heated in a preheated pebble bed, which produced a fine dust and made the combustion zone visible downstream of the normal shock. A well-defined gap between the normal shock and emission zone was always observed, indicating an ignition delay zone.

Attempts to reproduce the experimental phenomenon that Gross reported on,[8,9] using the same tunnel, resulted in similar changes in the observed shock wave shape, as observed by Gross (Figs. 2–4). However, there was a noticed difference. Visible emissions were observed in the experiments that were not observed by Gross[8,9] (Figs. 3 and 4). On investigation, we found that the air supply in the test facility used by Gross provided a clean air supply, with no visible emissions from hydrogen-air combustion, as would be expected. Like the Nicholls et al. experiments, aluminum oxide dust in the new air supply glowed in the heated portions of the flow, providing a clue to the location of the exothermic combustion processes. The glow appeared to originate from an upstream location (Fig. 4). On further investigation it was found that fuel was ignited at the fuel injector, heating the entire airstream. The changes in the shock-wave angles could be explained by calculating Mach number change caused by temperature rise from upstream combustion alone.

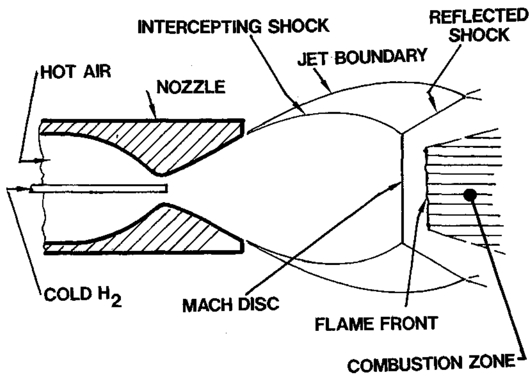

Fig. 1 "Shock bottle" experiment schematic, showing the location of ignition delay gap between the "Mach disc" normal shock and the emission zone for hydrogen-air combustion.[7]

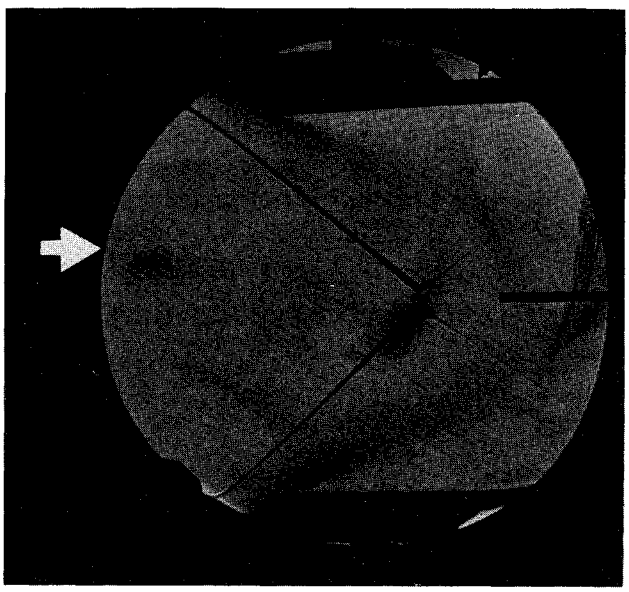

Fig. 2 Reproduction of normal-shock experiment of Gross,[8,9] schlieren photograph of shocks, no fuel injected (per Ref. 21).

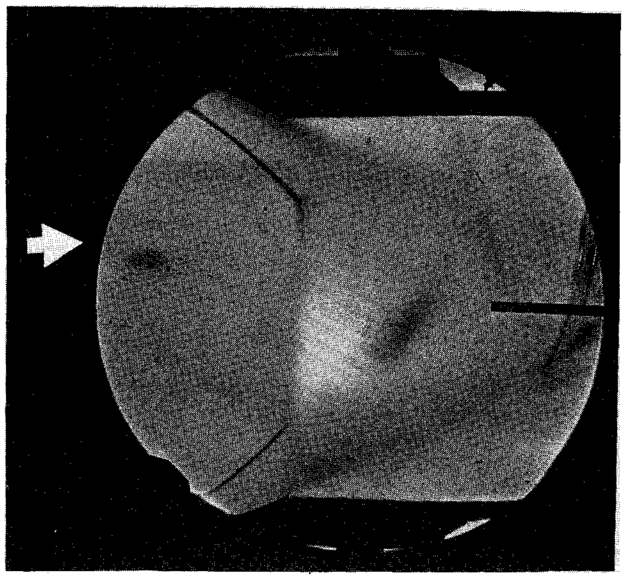

Fig. 3 Reproduction of Gross's experiment,[8,9] schlieren photograph, showing some emission just downstream of the normal shock when hydrogen is injection upstream (per Ref. 21).

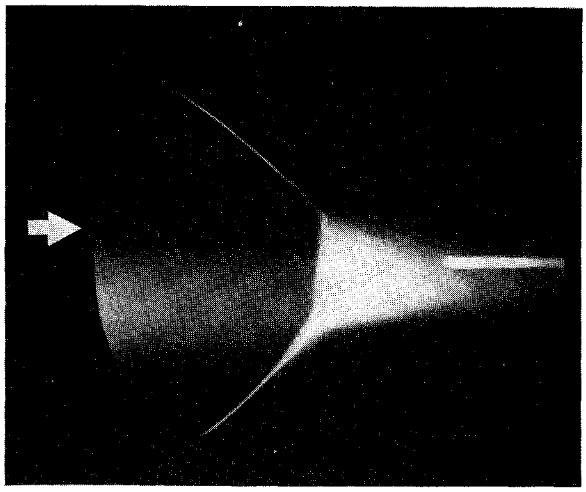

Fig. 4 Reproduction of Gross's experiment,[8,9] emission photograph only, showing emission from both upstream and downstream of the normal shock, indicating that hydrogen is burning at an upstream point.[13,21]

These phenomena were explored further in a series of investigations.[13] Air was preheated to 1700°R (944 K) and further heated to a maximum of 3500°R (1944 K) by burning hydrogen in an upstream preheater. Hydrogen fuel was then injected in the region of approximately Mach 2.0, mixed into the stream by diffusion, accelerated to Mach 3, and then passed through a normal shock formed by the intersection of two oblique shocks (Fig. 5). Static pressure in the combustion region was approximately 0.5 atm (50 kPa). Hydrogen and oxygen concentrations were measured downstream of the shock, using a reaction quenching probe. Both hydrogen and oxygen were depleted as the probe moved away from the shock, indicating chemical reactions in process.[13] This phenomenon could NOT be classified as detonation because the normal shock wave was independently generated by an arrangement of wedges and shock waves, and was not affected by the combustion process. Because it was established that the shock created a sudden static temperature rise in the airstream, and that this static temperature exceeded the ignition temperature of the fuel, it was concluded that the combustion phenomena observed were initiated by shock heating. The term shock-induced combustion was proposed.

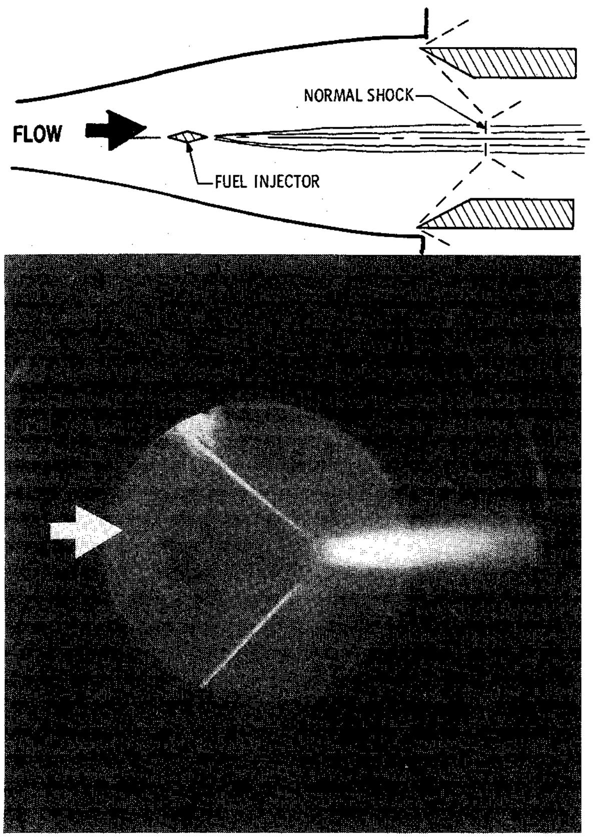

Fig. 5 Schlieren and emission photograph of experiment, in which no upstream combustion was observed, designated as shock-induced combustion. Note normal shock, ignition delay gap, then emissions.[13]

B. Oblique Shock-Induced Combustion

From the results of the first simple normal shock experiments, it was reasoned that a similar experiment could be performed, using an oblique shock to initiate combustion. Because static temperature rise across an oblique shock is less than for a normal shock, a higher total temperature of the airstream is required to produce an ignition static temperature. The pressure and temperature rise can be controlled by controlling shock angle. A simple calculation from available thermodynamic and chemical tables will show that the pressures and temperatures produced by a normal shock at hypersonic Mach numbers are tremendous, and will result in very large structural loads, extreme cooling problems, and a high degree of chemical dissociation that will delay the completion of the exothermic reactions that produce thrust. It is easy to understand why normal-shock combustion was avoided by scramjet proponents. Thus, there exists a real attraction for using oblique shocks for scramjet ignition, if the combustion process can go to completion under controlled conditions.

Rubins and Rhodes[14] injected hydrogen into the supersonic flow upstream of an oblique shock, produced by a 28-deg wedge in the Mach 3 stream. A combined schlieren and emission photograph is shown in Fig. 6. Sampling of the gas downstream of the shock and parallel to the flow demonstrated the gradual disappearance of hydrogen and oxygen, indicating that combustion reactions were occurring (Fig. 7), a result similar to that from the normal-shock experiments. The Mach number was approximately 1.6 in the flow zone parallel to the wedge, and downstream of the shock.

To reach the required inlet air static temperature, the air was partly heated by hydrogen combustion in the upstream plenum. A correction for this vitiated air composition was made for the purpose of correlating experimental data with kinetic calculations.[14]

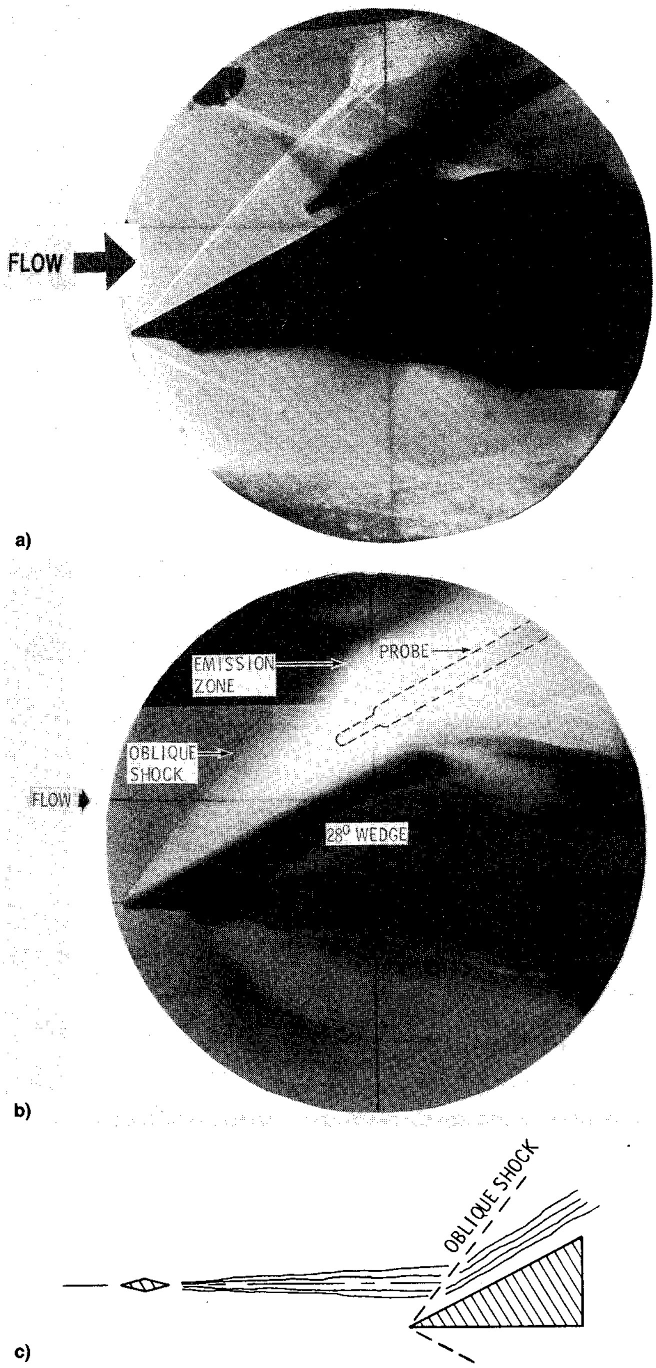

Fig. 6 Shock-induced combustion aft of an oblique shock in a Mach number 3 stream: a) schlieren photograph of the flow without fuel injection; b) combined schlieren and emission photographs when hydrogen fuel was injected; and c) diagram showing a schematic of the location of the fuel injector, wedge, and shock wave.

It was concluded that oblique shock-induced combustion is experimentally feasible, and neither the low molecular weight fuel (hydrogen) nor the reacting mixture made a measurable effect on the appearance of the oblique shock for single-point fuel injection.

C. Simulated Scramjet Inlet

The successful demonstration of oblique shock-induced combustion led to the design of small-scale simulations of a scramjet combustion system to demonstrate the feasibility of a hypersonic propulsion application (Refs. 15 and 16, Fig. 8). In each case, the reacting flow is now confined by a constant area duct.

A two-shock inlet was chosen as a simple configuration, and as one with some practical propulsion significance. Two

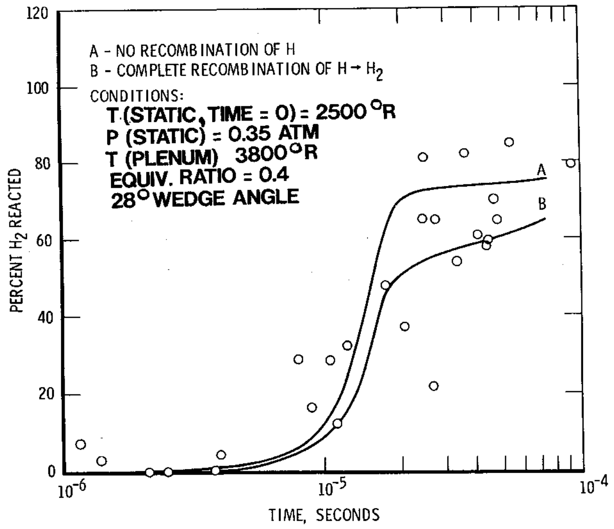

Fig. 7 Comparison of chemical kinetic calculations and experimental data for hydrogen reaction in vitiated air, where an oblique shock was used to initiate combustion.

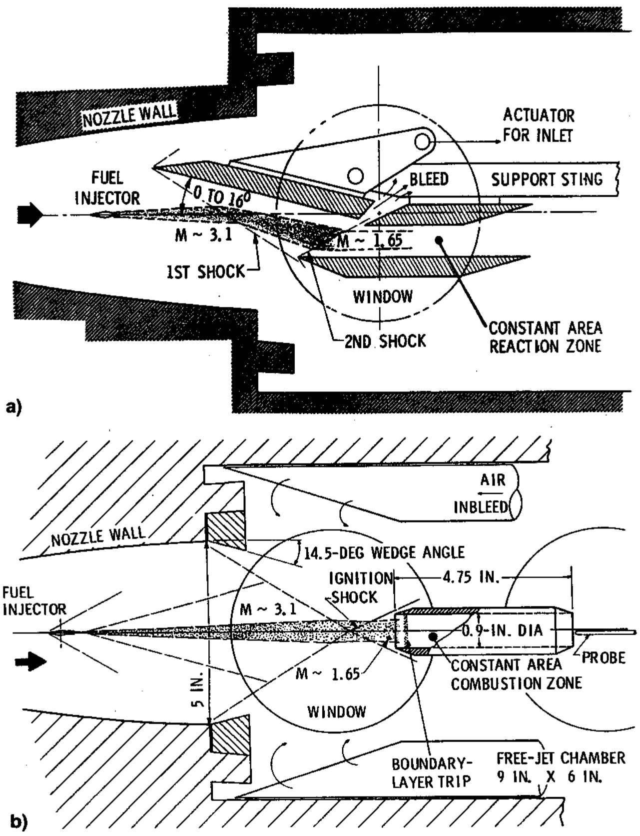

Fig. 8 Schematic of two-shock supersonic combustion models installed in a Mach 3 tunnel: a) two-dimensional model with diffuser inlet ramp and b) two-shock two-dimensional model with cylindrical combustion section.

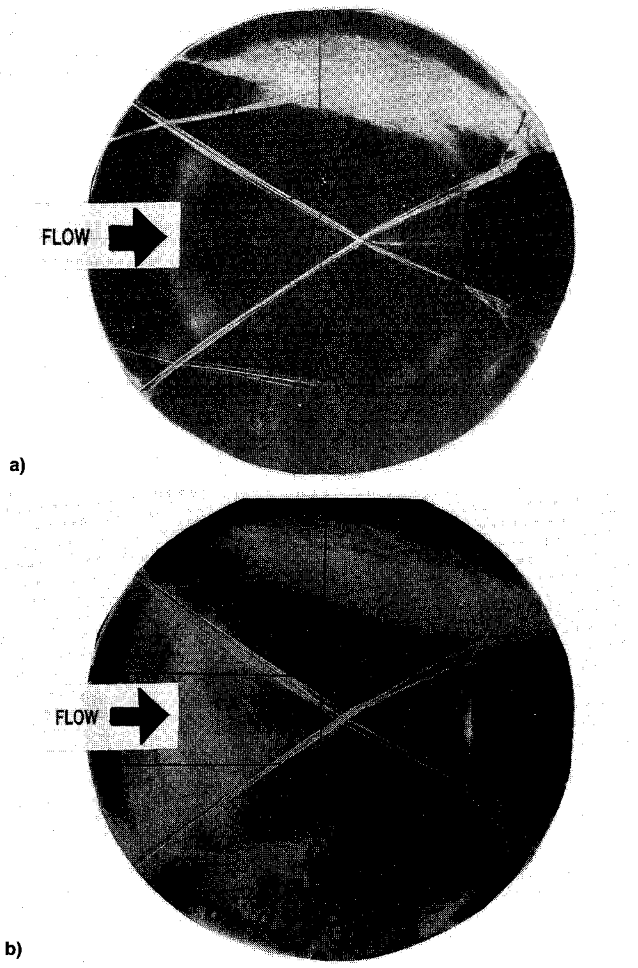

Fig. 9 Schlieren photograph of oblique shocks at the inlet of a cylindrical constant area combustion chamber: a) with low or zero fuel flow and b) with thermal choking and expelled shock.

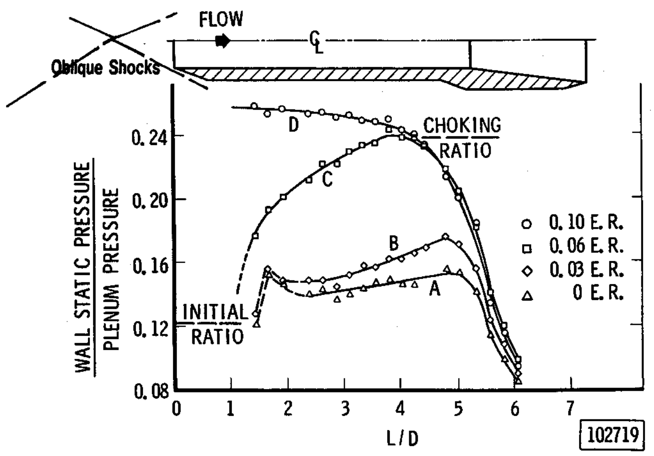

Fig. 10 Wall static pressure profiles in a cylindrical constant area supersonic flow combustion chamber, downstream of a two-shock inlet in a Mach 3 stream. The experimental choking pressure ratio agreed very closely with that calculated.

configurations were used to test the two-shock concept. A rectangular configuration (Fig. 8a) was tested, and reasonable data were obtained.[15] The preheated air passed through the two shocks, which increased static temperature to ignition level as the flow entered the constant area duct. Wall static pressure data and gas sampling demonstrated that combustion had occurred, even though boundary-layer separation caused a few problems in the adverse (increasing) pressure flowfield.

A somewhat different second configuration (Fig. 8b) used an exterior generated two-shock system to simulate a scramjet inlet. A schlieren photograph of the shock pattern at the round duct inlet is shown in Fig. 9. Wall pressure taps were located inside the constant area duct to measure pressure change caused by chemical reaction heat release (Fig. 10). Fuel-air equivalence ratio was varied from 0 to 0.1. Thermal choking was observed at approximately 0.06, which was near that predicted from calculations.[11]

For the case of exothermic reactions in supersonic flow, the pressure rise will produce an adverse pressure gradient, and possible flow separation. Because the effective flow area is reduced, "unstarting" and expelled normal shock at the inlet may result (Fig. 10). This problem was resolved by installing a boundary-layer "trip," near the inlet of the constant area portion. By doing this, fuel flow could then be increased until thermal choking was reached. With additional fuel flow, the duct flow became subsonic and a normal shock was expelled and visible at the inlet (Fig. 9b).

Although the Mach number in the test section downstream of the shocks was only 1.6–1.7, the test demonstrated that the oblique shock can indeed be used to initiate the chemical reactions of supersonic flow entering a confined duct, producing pressure rise and potential thrust in an aircraft. In both of these experiments, the wall pressure data agreed with the predictions of the Z-N-D model.

D. Studies of Global Chemical Kinetics in High-Speed Flow

It became obvious during the course of this research that reaction kinetics were becoming of overriding importance in analyzing and predicting performance and designs of supersonic combustion flows, particularly under conditions of high altitude and high Mach number, where static pressure will be low. Although a simplified reaction kinetics computation of ignition delay and recombination[22] is useful to evaluate some of the problems of supersonic combustion at various flight conditions, more precise chemical kinetic rates would be needed for designing combustion chamber and exit nozzle shape for optimum scramjet performance.[23]

Based on these observations, the concept of using a normal shock standing wave, simulating the classic Chapman-Jouguet wave, was developed, using the Mach 3 tunnel for the study of global kinetics (Fig. 11, Ref. 24). The mass flow of the entering heated vitiated air was controlled with a movable exit plug. Variable quantities of fuel could be introduced at the upstream injector. The exit position of the movable plug was then adjusted, while maintaining a normal shock on the lip of the tube entrance. Wall static pressure profiles could then be used to study "global" combustion kinetics. The main portions of the combustion could be broken down into 1) ignition delay equations, and 2) partial equilibrium equations.[18] Global reaction kinetics for hydrogen-air were studied. Preliminary experiments with methane-air indicated that this technique could also be applied to combustion of other gaseous fuels.[18]

The flow-restricting, movable exit cone plug was designed to locate the normal shock EXACTLY on the lip of tube (Fig. 11). As fuel flow was increased and combustion heat was produced inside the tube, the cone was retracted, while holding the shock on the edge of the lip. At the theoretical Chapman-Jouguet condition, exothermic heat would be expected to drive the normal shock, the exit flow would be expected to be choked, and the cone completely retracted. However, this condition could not be reached because the flow became unstable and developed strong oscillations at some fraction of the choked flow heat release. Extensive oscillation phenomena in shock-tube experiments have been observed by several investigators (i.e., Strehlow et al.[25]), and may be related to these standing wave observations.

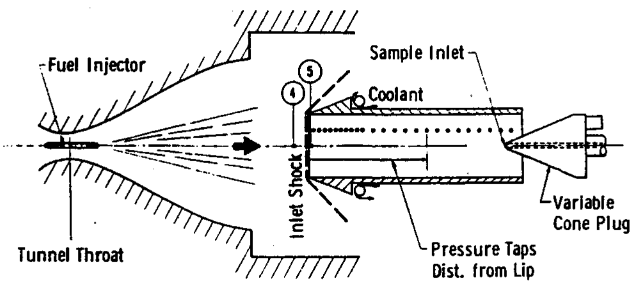

Fig. 11 Schematic of combustion tube used for normal-shock, standing-wave global chemical kinetic studies. The tube is 0.9 in. diam and 7 in. long. The adjustable exit cone plug was used to adjust the normal shock precisely on the lip of the tube.

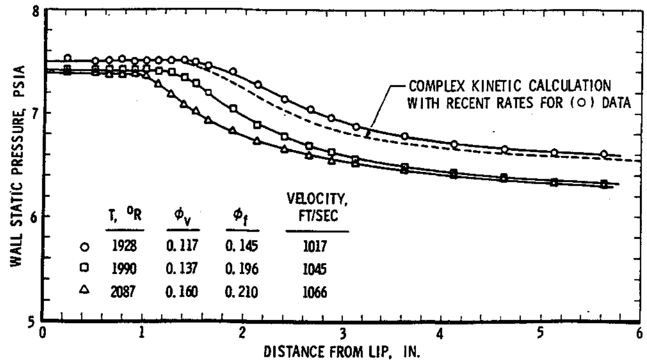

Fig. 12 Combustion-generated pressure profiles measured downstream of the normal shock inside the tube, for hydrogen-air reactions at approximately 0.5 atm. Ignition delay is indicated by the distance of the bend in the curve from the lip of the tube. The effect of increased inlet air temperature is shown.

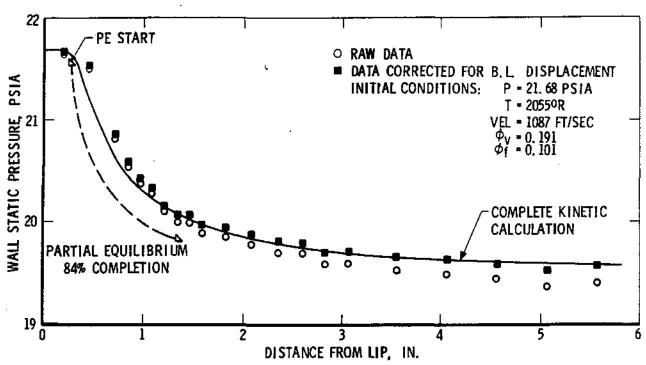

Fig. 13 Combustion-generated pressure profiles measured inside the tube, for hydrogen-air reactions at approximately 1.5 atm. Note reduced ignition delay distance at the higher pressure.

Experimental data from the tests are shown in Fig. 12, showing pressure drop as the gas reactions become exothermic. Ignition delay was defined by the point at which the pressure drop begins. These data were compared to existing hydrogen-air kinetic rate computer programs, and showed good correlation with the selected kinetic mechanisms. From these data, we conclude that ignition delay is reduced as inlet air temperature increases, as expected from theory.

Additional data recorded with this device, but at 1.5 atm (150 kPa), produced shorter ignition delay, as would be expected (Fig. 13, Ref. 18). A comparison of Figs. 12 and 13 will illustrate the shortening of the distance from shock to exothermic reactions as pressure is increased. One may envision shorter and shorter ignition and reaction distances as pressure increases, until the phenomenon becomes "detonation."

These experimental data, both fluid dynamic and chemical, could be analyzed and correlated using the one-dimensional compressible flow equations with heat addition.[11] Nothing in the experimental observations or calculations indicated any difference in the mathematical treatment of shock-induced combustion and detonation phenomena for a one-dimensional flow, constant area duct.

III. Application of this Combustion Research to a Propulsion System

A. Introduction and Literature Review

Scramjet propulsion applications can be divided into two parts: 1) real systems that have been under development for many years, mainly for application to missiles, generally below Mach 10 velocities; and 2) futuristic calculations and research aimed for the Mach 10–25 range.

Waltrup[2] and Billig[3] describe the excellent work performed for many years to develop usable products for defense purposes in the lower Mach number range, 5–10.

We have looked at shock-induced combustion (described by some investigators as detonation wave combustion) from the standpoint of application to higher Mach number flight. The literature is repleat with papers, mainly computer calculations, which predict the type of phenomenon that may be expected at high Mach numbers. They appear to be divided into two categories: 1) those that utilize or assume very fast chemistry, and 2) those that consider that chemical rates may require relatively long distances, particularly at high altitude and low pressure.

A few of the papers reviewed for application to the higher Mach number range are discussed in the following paragraphs:

1) Anderson[26] cites the need for combustion data for high-speed flows, and the need for adequate chemical kinetic and flow computations.

2) A review of Refs. 27–32 will lead to the conclusions that a) high-pressure and low-altitude flight will produce "fast" chemistry, and close coupling of the shock and chemical reaction, and b) low pressure will produce slower chemical reactions and decoupling of the shock and chemical reaction exothermic reactions. This is in agreement with our experimental results.[13–20]

Aerodynamic tests of inlet diffusers in the range of Mach 16–26 are reported in the work of Nagamatsu et al.[33,34] These data point out that viscous effects produce serious pressure losses above a Mach number 15, for a relatively simple two-shock inlet. Although total enthalpy for atmospheric flight and wall cooling could not be simulated, the data indicate the extent of some of the real problems for hypersonic diffusers.

The end conclusions are as follows:

1) The data from Refs. 13–18 agree quite well with the chemical kinetic computations, which predict a well-defined ignition delay and recombination time for the low pressure and stream static conditions under study.

2) An extrapolation of the chemical kinetic computations to higher pressures will produce fast reactions and close coupling of the shock and exothermic reactions, similar to those observed in shock tubes.

3) Fluid mechanical computations that use equilibrium chemistry are valid in the high-pressure range only, assuming fast reactions.

B. Application of Experimental Research to Scramjet Propulsion

Based on experimental results in supersonic combustion research and combustion kinetics,[13–18] Rubins and Bauer[19] proposed an application to a high-altitude scramjet, assuming premixing the fuel and using inlet diffuser oblique shock waves for ignition.

This discussion is mainly concerned with the feasibility of generating stable supersonic combustion in the low-pressure, high-altitude, high Mach number region, 10–25. Willbanks[35] examined shock-induced combustion from a flow sensitivity point of view. He concluded that exothermic reactions without strong initiating shocks are feasible, with the proper aerodynamic design. Pratt et al.[36] have analyzed oblique detonation waves (ODW) and the conditions under which they may occur, and have indicated that shock-induced combustion may result when ignition delay creates a separation of shock and exothermic reaction zone.

As pointed out by Pratt et al., closed-coupled and near-closed-coupled exothermic reactions will affect the shock shape and strength. For a close-coupled oblique shock, the term ODW applies when the downstream pressure increases from the exothermic reactions, and causes the wave to become steeper.

The critical conditions that determine close coupling are those that control chemical kinetic rates in a prescribed flowfield. These include pressure, temperature, area change, boundary layer, etc.

C. Flight Envelope

A typical scramjet flight envelope is shown in Fig. 14 (Refs. 10 and 19). The envelope indicates severe limitations for scramjet propulsion, based on aerodynamic flight limitations of combustion chamber temperature and pressure, as they affect the combustion process:

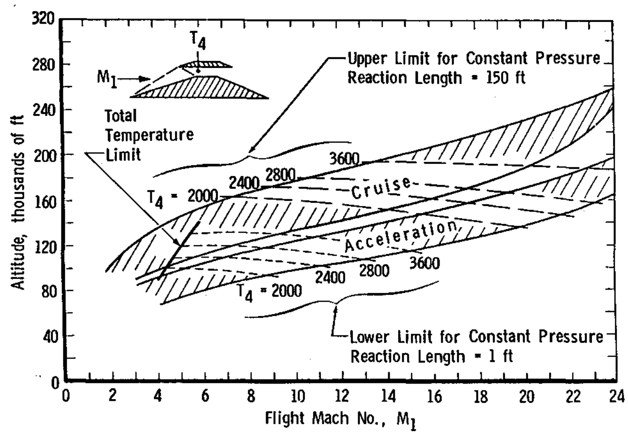

Fig. 14 Hypersonic ramjet flight envelopes, shown with the limitations of hydrogen-air combustion kinetics, for a two-shock inlet diffuser.

1) We have arbitrarily defined a lower altitude Mach number limit when the fuel exothermic heat release can be accomplished in a combustion length of 1 ft (0.3 m) or less, i.e., fast reactions.

2) The upper flight limit was chosen at 150-ft (45.7-m) maximum combustor-nozzle length, to complete the recombination reactions for hydrogen-air combustion.

3) A minimum diffuser exit static temperature of 2000°R (1111 K) was selected as a limit for hydrogen-air ignition. A temperature lower than 2000°R (1111 K) could be used if a "hot pilot" is available.

4) The combustor static inlet temperature of 3600°R (2000 K) was chosen as an upper limit because dissociation losses increase quite rapidly as temperature increases beyond this point.

The calculations of ignition delay and recombination heat release time were made, using the approximate equations of Pergament,[22] modified to more nearly correspond to the results from our hydrogen-air data.[13–18] More recent chemical kinetic rates may be available, but it is expected that the trends will be similar. By this, we mean that there is an extreme sensitivity of nozzle thrust to reaction kinetic rates and the nozzle contour.

We conclude that flight envelopes indicate finite and severe limitations to the altitude and Mach number of scramjet propulsion aircraft; and different approaches to the combustor design will be required, depending on whether the Mach number and altitude result in fast or slow chemistry in the combustion zone.

D. Inlet Diffuser

To determine feasibility of a premixed fuel scramjet, a simple two-shock inlet configuration was analyzed (Fig. 15). Combustor inlet static temperatures (downstream of the second oblique shock) were selected at 1800, 2000, 2400, 2800, and 3600°R (1000, 1111, 1333, 1555, and 2000 K). Combustor inlet temperature was assumed to be controllable by a combination of flight Mach number and inlet diffuser shock strength. In order to evaluate the effect of friction on flight performance, inlet diffuser pressure losses were calculated for zero skin friction, laminar and turbulent airflow, and laminar and turbulent flow with hydrogen in the boundary layer, where hydrogen was injected from upstream fuel premixing ports.

Above Mach 12, total pressure loss for either turbulent or laminar boundary layer increases very rapidly, dropping from 20 to approximately 4% recovery at Mach 22. However, with a hydrogen boundary layer, pressure recovery actually improves about 4% if the flow is laminar, and shows a small improvement for turbulent boundary layer.

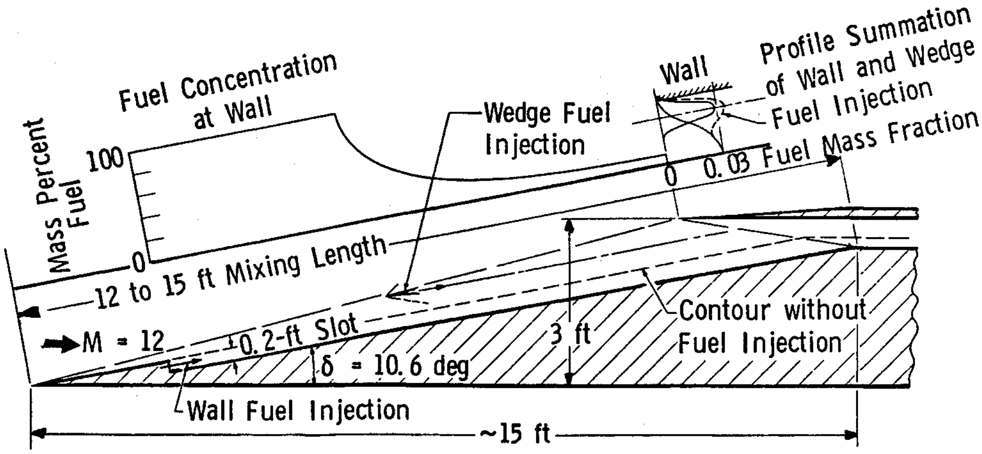

Fig. 15 Schematic of Mach 12, 2-shock inlet, with premixed fuel injection, showing calculated hydrogen fuel diffusion-mixed profiles.

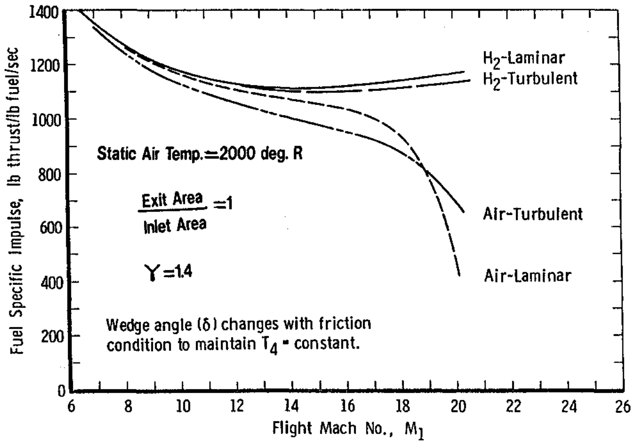

Fig. 16 Calculated scramjet specific impulse for two-shock hypersonic inlet showing the effect of hydrogen in the diffuser boundary layer.

In addition, specific impulse above $M = 16$ is considerably higher when hydrogen is injected into the boundary layer (Fig. 16). Specific impulse is shown plotted vs Mach number for combustor inlet static temperature of 2000°R (1111 K). The large viscous losses reported by Nagamatsu[33] could theoretically be reduced by injecting hydrogen along the surface of the inlet diffuser at an upstream location.

We conclude that a two-shock diffuser produces some advantage where fuel is premixed in the diffuser; low molecular weight fuel (i.e., hydrogen) stored on board as liquid will have a very low total temperature, and, if injected as gas in the boundary layer near the diffuser inlet, can be used as a film coolant; and low molecular weight of this gas will generate lower inlet skin friction. This concept has not been experimentally tested, that we are aware of.

E. Fuel Injection

For minimum disturbance to the flow, and lowest possible losses, fuel should be injected in a downstream direction, parallel to the flow (Fig. 15). Obviously, parallel fuel flow injection does not provide the fastest mixing. Mixing rates in this case are the result of boundary-layer turbulence and molecular diffusion. There are three obvious locations for fuel injection: 1) along the inlet diffuser surface, 2) through a thin strut located in the inlet diffuser, or 3) by parallel wall injection in the combustion region. These three are suggested in order to keep pressure losses from fuel injection and mixing as low as possible, especially at the higher Mach numbers where pressure loss may be critical. Fuel cooling properties are also important in locating the injection point and configuration.

F. Ignition

Ignition of the fuel is at hand when the ignition static temperature is reached. However, at the lower limit of ignition temperature, both ignition delay and recombination time may be longer than desirable. In this situation, ignition help may be needed, in the form of a movable wedge, which generates an additional shock wave, or (retractable?) bumps, which produce a strong shock near the wall. One of these devices may be needed in order for the engine to maintain combustion over a wide range of altitudes and flight Mach number. Ignition devices have received considerable attention in combustion research and published literature.[1-3,10]

G. Recombination and Exothermic Reactions

As discussed, constant area heat addition in supersonic flow produces an adverse pressure gradient and a tendency for flow separation. By including a designed area contour, based on fluid mechanics and chemical kinetics, this condition may be averted. However, this area change must be closely controlled for a wide range of altitudes, Mach number, and fuel-air ratio in order to reach maximum propulsion efficiency and optimum combustion conditions. Detailed configuration studies for a specific flight mission are required in order to arrive at the best configurations.

The end result of this study demonstrates that for the lower hypersonic Mach numbers, <10, dependable combustion may require strong shocks, a detonation-type combustion, or a mixture of fuel injection and ignition methods; and for the higher Mach numbers, >10, the reduced pressure losses of premixed fuel combustion, with shock ignition and a controlled area expansion, may be necessary in order to reach a reasonable propulsion efficiency level. The Mach 10 division is not a precise one, but is used to indicate that at some point ignition is the primary problem. At the higher Mach number, pressure and aerodynamic control are important for control of chemical kinetics.

H. Exit Nozzle

At the lower Mach numbers, where reactions kinetics are fast, the exit nozzle behaves much like a rocket nozzle, converting thermal energy to kinetic energy and thrust. However, at the high-altitude, high Mach number regime, the exit nozzle may become a very sensitive pressure-temperature controller that regulates the rate at which the recombination reactions go toward completion. The exit nozzle then requires the ability of the engine to adjust to the chemical recombination rates of the combustion process, a very difficult requirement.

IV. Summary and Conclusions

A number of experimental phenomena related to supersonic combustion were investigated and demonstrated in Refs.

13–18 during the period of 1959–1968. They demonstrated that continuous supersonic combustion can be generated by shock heating an air-fuel mixture to the ignition static temperature of the fuel, or higher, in a supersonic flow. These experiments included the following demonstrations and analysis.

1) A demonstration was done showing that hydrogen or methane fuel can be injected into a supersonic stream, at high total temperature, and at a static temperature less than ignition temperature, without measurable combustion of the fuel.

2) It was demonstrated that the mixed fuel-air mixture can be ignited by passing it through a shock wave, where the static temperature is suddenly increased to higher than ignition temperature. This was termed shock-induced combustion where distance between the shock and exothermic reactions are relatively large, and detonation at higher pressures where the shock and exothermic reactions are coupled.

3) Shock-induced combustion in the downstream region of both a normal shock and an oblique shock was demonstrated.

4) A demonstration of oblique shocks to produce supersonic combustion entering a constant area duct, simulating a scramjet combustor was performed. Heat release up to thermal choking could be produced without generating a high-pressure loss oblique detonation wave.

5) An analysis showed that one-dimensional flow theory with heat addition is applicable to shock-induced combustion,[11] essentially the Z-N-D model.

6) A demonstration was done of a standing-wave normal shock in a tube to study global chemical reaction kinetics.

These experimental results were used to investigate an application to a hypersonic scramjet. Our calculations also indicate that injecting fuel in the diffuser inlet can be used to mix fuel and air, to cool the inlet wall surface, and to reduce diffuser surface friction if low molecular weight fuel (hydrogen) is used.

The most difficult foreseen problem with premixed fuel is the strong sensitivity of the exhaust contour, or nozzle, and expansion ratio to the rate of exothermic heat production, i.e., chemical recombination rate. A controllable "rubber" exhaust nozzle may be required.

Some of the expected characteristics of premixed fuel are 1) better control of fuel distribution by selecting the location of fuel injection at an upstream point in the inlet diffuser; 2) the fuel can be used for cooling the diffuser inlet surface; 3) premixing the fuel can reduce the combustor length; and 4) for a low molecular weight fuel (i.e., hydrogen), the diffuser inlet losses caused by viscous effects can be significantly reduced at the higher Mach numbers, resulting in higher specific impulse.

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