experiment, the latter problem was eliminated since the inlet was designed for unity mass flow ratio. Effective internal drag isolation was accomplished with an axisymmetric inlet through the use of metric and nonmetric surfaces. With such a technique, it was found that the associated corrections to the measured drag levels to account for lip bluntness drag and balance cavity internal forces could be made both accurately and consistently. Careful model design can also minimize the magnitude of such corrections. It is to be expected, however, that comparable degrees of internal drag isolation and performance will be more difficult to achieve with three-dimensional inlet designs.

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Shock-Induced Supersonic Combustion in a Constant-Area Duct

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A two-oblique-shock compression was used as a reaction-initiating mechanism for the H_2-air reaction in a constant-area duct while maintaining supersonic flow. Gradual pressure rise from the chemical heat release was observed as the fuel concentration was increased until the duct flow was choked. The aerodynamic, thermodynamic, and chemical kinetic problems are discussed. For the small model used, it was found that the generation of a turbulent boundary layer was necessary to prevent thermally produced pressure rise from causing boundary-layer separation and premature choking. It is concluded that shock-induced combustion 1) is experimentally feasible in ducted flow, 2) can be controlled by varying inlet temperature, pressure, Mach number, or fuel concentration, 3) may be analyzed as a one-dimensional flow for the small models investigated if average stream properties are used, and 4) experimentally produces ignition delay times that agree with existing chemical kinetic flow computations within the accuracy that stream properties can be determined. The possible use of shock-induced combustion for air-breathing hypersonic propulsion is discussed.

Nomenclature

A = area
C_p = specific heat at constant pressure (refers to air when no additional subscripts are used)
D = duct diameter
ER = equivalence ratio
d = ignition delay distance
F = stream force
f = fuel-air ratio
L = duct length
M = Mach number
p = static pressure
p_t = stagnation pressure
p_t' = pitot pressure
Q = heat flux
T = static temperature, °R
T_t = stagnation temperature, °R

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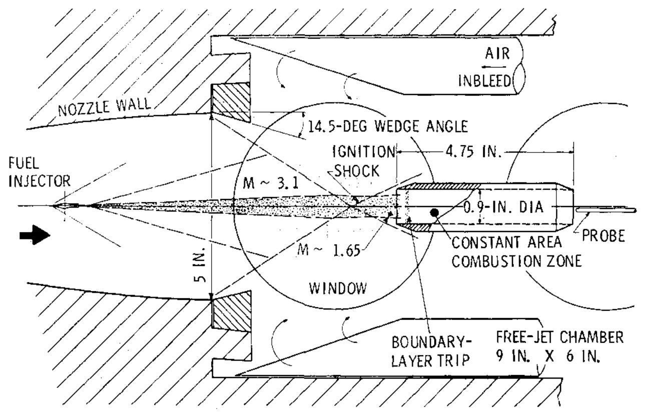


Fig. 1 Schematic of axisymmetric supersonic combustion model installed in tunnel. Tunnel is 6 in. wide. Air inbleed is used to regulate pressure at the model inlet.

V	=	velocity
α	=	wedge angle
γ	=	ratio of specific heats
$\bar{\tau}$	=	average wall shear stress
τ	=	ignition delay time

Subscripts

0	=	condition at time = 0, or reaction starting time
1, 2, etc.	=	condition 1, 2, etc.
A	=	air
a	=	inner stream
b	=	outer stream
f	=	hydrogen fuel
m	=	mixture
w	=	wall

Superscript

*	=	choked flow condition

Introduction

ANALYSIS of hypersonic ramjet cycles shows that supersonic combustion is a requirement.[1-8] Tremendous temperatures and pressures are built up if air is diffused from hypersonic to subsonic velocity, as is done in the more conventional ramjet, and thrust may be impossible because of the high level of dissociation of the gasses. If supersonic combustion is utilized, less aerodynamic diffusion is necessary, pressures can be held to reasonable levels, and dissociation can be reduced.

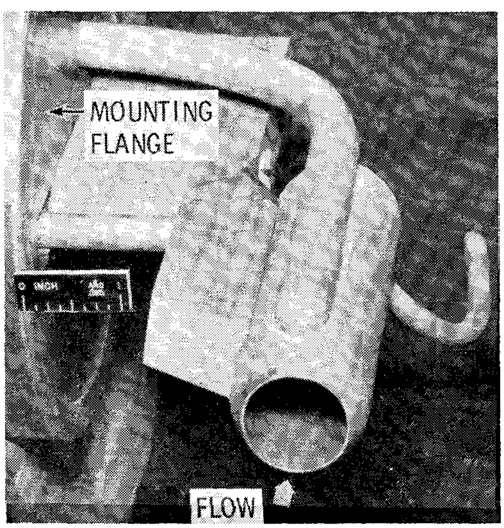


Fig. 2 Constant-area two-dimensional flow supersonic combustion model.

Shock-induced combustion occurs where fuel and air mixtures are ignited by a shock that raises the static temperature to the ignition point. It can be used to ignite mixtures over a wide range of static temperature, static pressure, and Mach number by changing the strength of the shock, i.e., by changing the angle of the wedge that generates the shock. This method of ignition could be employed in a hypersonic ramjet by using one of the shocks within the inlet diffuser as an ignition-generating surface. In addition to the use of this technique in a ramjet combustor, it can also be used as a research tool for studies of over-all reaction rates where the shock acts as an ignition plane.

Shock-induced combustion differs from detonations in gases in that the chemical reaction behind the shock does not necessarily affect the shock, but it is similar in that the reaction is initiated by the shock wave, which heats the gas to a reacting temperature.[9-11] The rate of chemical reactions depends on the state of the gas after it has been compressed in the shock and the shape of the confining walls. For very fast reactions where sudden pressures are created or area does not increase rapidly enough, the normal shock or "detonation wave" usually observed in tubes will be generated.

Another method of burning in supersonic flow is termed diffusional burning, in which fuel is injected into the hot air stream within the combustor where it mixes and burns simultaneously. Shock-induced combustion differs from diffusional burning in that fuel is injected upstream of the combustor and mixes with the air in the diffuser, so that a combustible mixture enters a shock wave at the combustor entrance. Although the advantage of either method over the other in a propulsion system has yet to be proved, the shock-induced method appears to be a simpler system to analyze, because fuel injection, mixing, and burning are all separate processes, whereas in diffusional burning they may be simultaneous.

In previous work[9] at the Arnold Center, experiments with single wave, shock-induced reactions were conducted in a constant pressure zone, and the experimental results compared with the results obtained from the chemical kinetic calculations developed by Libby, Pergament, and Bloom.[12] These chemical kinetic calculations utilize a constant pressure, adiabatic flow model in which a homogeneous mixture of hydrogen and air at specified temperature, pressure, and equivalence ratio begins to react at time zero. Good agreement was found between experimental data and the kinetic history imposed on a one-dimensional flow for the ignition delay period as indicated by the hydrogen molecule reaction history. If the temperature rise following the ignition delay as computed from chemical kinetics could also be substantiated by means of experiment, this knowledge could be used in further evaluation of hypersonic ramjet performance.

The purposes of this paper are: 1) to present experimental results where supersonic combustion in a constant area duct is initiated by a shock wave, 2) to evaluate the possibility of analyzing the data with one-dimensional flow equations, 3) to determine how closely current chemical kinetic computations agree with experimental results, and 4) to discuss the experimental and analytical problems encountered.

Equipment and Procedure

The experiments were carried out in a water-cooled, two-dimensional, Mach 3.1 tunnel, which has been fully described in Refs. 9 and 10. The nozzle and test section of the tunnel are shown schematically in Fig. 1. The total airflow was heated to 1000°R with an indirect-fired heat exchanger. Further heating of the flow core up to 3500°R was accomplished by a hydrogen-fueled preheater in the tunnel plenum. Model test fuel was injected downstream from a wedge centered in the two-dimensional flow at the Mach 2.9 section, where static temperatures were low enough to prevent pre-ignition. Two models were used: an axisymmetric combustor, for which the oblique shocks were generated by short wedges in the tunnel (Figs. 1 and 2); and a two-dimensional flow model (Fig. 3). Photographs of the wedge-type fuel injectors used for the two models are shown in Fig. 4. These injectors were cooled mainly by fuel flow, with secondary cooling by conduction and radiation to the cool tunnel walls. The models designed use two oblique shocks rather than one to start ignition, in order to keep the models relatively small and thus avoid blocking the tunnel flow. Model inlet opening size was determined from a knowledge of fuel patterns obtained from previous experiments.

A minimum model length in which heat release could be expected to occur was calculated using the chemical kinetic calculations of Ref. 12. Inlet conditions that could be obtained by controlling tunnel total pressure and temperature as well as inlet wedge angles were used for the computer inputs. To reduce the computer time, only a few calculations were made, and the ignition delay lengths of other inlet conditions were compared to these results by the equation

$$\frac{d_1}{d_2} = \frac{M_1}{M_2}\left[\frac{T_{01}}{T_{02}}\right]^{3/2}\left(\frac{p_2}{p_1}\right)\exp\left[\frac{15{,}860}{T_{01}} - \frac{15{,}860}{T_{02}}\right] \tag{1}$$

This equation is a modification of the ignition delay equation

$$\frac{\tau_1}{\tau_2} = \left[\frac{T_{01}}{T_{02}}\right]\left[\frac{p_2}{p_1}\right]\exp\left[\frac{15{,}860}{T_{01}} - \frac{15{,}860}{T_{02}}\right] \tag{2}$$

presented in Ref. 9 and requires the assumption that the specific heat ratio and gas constant are not appreciably different for the two cases considered.

Equation (1) was used to calculate the effect of wedge angle on ignition delay distance with the assumption that stagnation temperature and pressure were held constant at values typical of the tunnel operation (Fig. 5). The ignition delay distance is found to be very sensitive to wedge angle.

Total temperatures of the gas entering the model were calculated from a knowledge of gas composition from samples taken at the model inlet while the hydrogen burning preheater was on, and no test fuel was injected into the model. For this calculation, it was assumed that mass and energy transport were simultaneous and that no heat was lost to the tunnel walls. During fuel injection into the model, a second gas analysis was used to determine the fuel equivalence ratio and the quantity of molecular hydrogen that had reacted.[10] Sampling was done along the model centerline. Pressure measurements were made on a mercury manometer board and recorded photographically.

Discussion and Results

At first it was thought that performing shock-induced combustion experiments in a constant-area duct would involve simple changes in equipment and procedures from previous experimental work in a constant pressure field. A successful constant-area experiment would ideally approach a simple one-dimensional flow with heat release and provide information from which a comparison could be made with computed chemical-thermal history. This anticipated simplicity was not realistic.

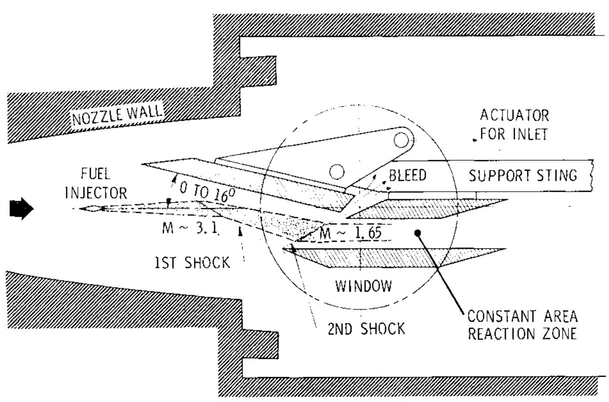


Fig. 3 Axisymmetric constant-area supersonic combustion model.

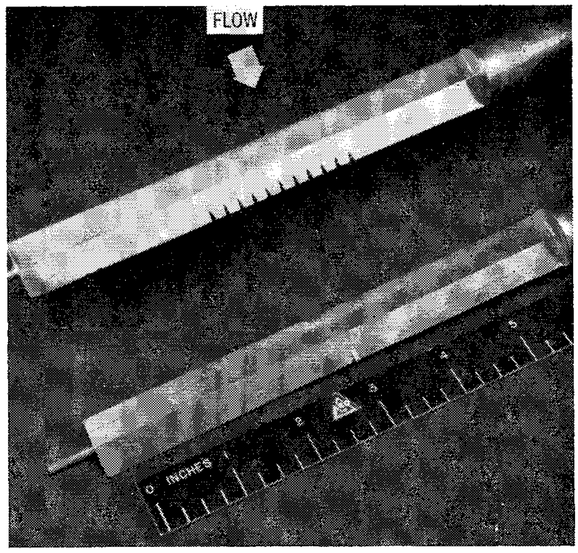


Fig. 4 Double-wedge fuel injectors used in shock-induced combustion experiments: (top) multipoint injector for two-dimensional flow model (bottom) single point injector with nozzle for axisymmetric model.

Experimental Problems

Initial experiments with the two-dimensional model (Fig. 3) gave indications that boundary-layer interactions and possible corner effects prevented achievement of the pressure rise levels that would be expected from one-dimensional theory for heat addition in a constant area duct. A switch to the axisymmetric model yielded similar results. It was then suspected that a laminar boundary layer existed which separated with adverse pressure gradients during combustion and caused the duct to choke when a very small pressure rise was reached. Subsequently, boundary-layer trips were installed near the model inlet. With a turbulent boundary layer in the combustion zone, it was then possible to experimentally produce pressure ratios comparable with those calculated for choked flow from one-dimensional theory.

Analysis of Pressure Data

In order to analyze the pressure data obtained in the experiments, it was first necessary to consider the nonuniform concentration, temperature, and velocity profiles that existed in the flow entering the model. Temperature and velocity profiles, as well as their fluctuations, may be the result of tunnel

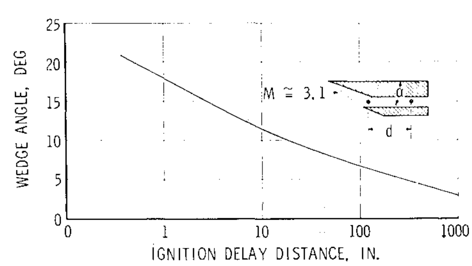


Fig. 5 Effect of increasing wedge angle (α) on ignition delay distance for typical two-shock test conditions, stagnation T = 3000°R, plenum p = 100 psia, test section Mach No. $\cong$ 3.1, d = distance from igniting shock to ignition delay point.

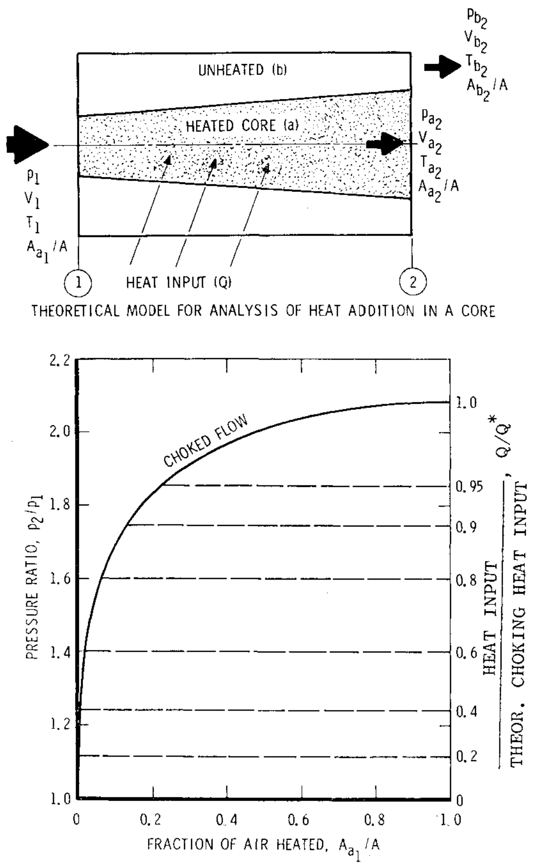


Fig. 6 Effect on pressure rise of nonviscous core heating in a constant-area duct.

aerodynamic and preheater characteristics, as discussed by Morkovin.[13] Possible fluctuations were neglected of necessity since they were so rapid or of such small magnitude that they were not measurable with the instrumentation used. At the high temperatures of the tests, it was experimentally determined that the temperature and velocity profiles at the model inlet when fuel was *not* entering the model were essentially flat. The nonuniform temperature and velocity profiles introduced by model fuel injection were negligible because of the very low fuel concentrations used in the experiments. (In theory, an equivalence ratio of approximately 0.07 could choke the constant-area duct.) It was concluded, then, that only very small errors would be introduced into the analysis if the nonuniform inlet temperature and velocity profiles were neglected.

In contrast to the negligible temperature and velocity profiles, a nonuniform fuel concentration profile could cause

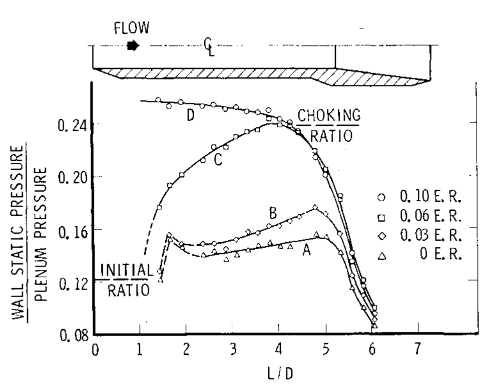


Fig. 7 Wall static pressure profiles with various levels of combustion. Inlet static pressure = 0.81 atm, initial static temperature ~ 1750°R.

drastically nonuniform heat release, or a "core heating" effect. An analysis was performed to determine the magnitude of this effect. A theoretical model was assumed with heat introduced in the core of a constant area duct (Fig. 6). The following assumptions were made: 1) constant static pressure exists across the duct at all times; 2) heat is added uniformly across a core which remains unmixed with the outer stream; 3) the flow is frictionless; 4) constant specific heat ratio and gas composition are maintained in each stream; 5) velocity and temperature profiles were uniform at the inlet; 6) the outer stream is isentropic. Choking conditions were determined from the choking criterion of Pearson et al.,[14] which states that, for two streams of different Mach numbers flowing concurrently in the same duct, choking occurs when

$$A_{a2}\,[(1 - M_{a2}^2)/M_{a2}^2] + A_{b2}\,[(1 - M_{b2}^2)/M_{b2}^2] = 0 \quad (3)$$

The results of this core heating analysis presented in Fig. 6 show that, if the initial core area is greater than 50% of the total inlet area, the choking pressure rise is negligibly different from that for one-dimensional Rayleigh line heat addition. Therefore, it was concluded from this extreme treatment of the problem that possible core heating effects on experimental pressure rise would be negligible. Since it appears from the foregoing that the nonuniformities in the inlet flow have only a small effect on the duct performance, the pressure data were analyzed using one-dimensional equations.

Typical pressure data obtained from the axisymmetric model are shown in Fig. 7. Because of the difficulty of measuring model inlet pressures without the probe disrupting the flow, pitot and wall static pressure measurements for the downstream two-thirds of the model were used to calculate a Mach number profile when fuel was not injected into the model. The pressure and Mach number profiles were then extrapolated forward to the inlet. This procedure produced the inlet static pressure indicated in Fig. 7 and a possible range of initial Mach number between 1.60 and 1.66. Since the experiments were performed at low equivalence ratios, it was assumed that inlet conditions remained constant when fuel was injected into the model.

Curve *A* in Fig. 7 shows the model interior pressure profile without combustion (no fuel injected). It shows some disturbance near the upstream end caused by the boundary-layer trips or the lip shock, followed by a gradual pressure rise caused by wall friction, and the expansion out the conical exit. Curves *B* and *C* show that supersonic inlet conditions exist and also that there is heat addition as reflected by pressure rise until the flow expands out the rear of the model. Curve *C* represents the maximum equivalence ratio for which stable supersonic combustion could be maintained in the duct. At higher fuel flows, a normal shock was expelled from the front of the duct and heat release proceeded in a subsonic stream as represented by curve *D*.

The appearance of the axisymmetric model inlet during low or zero fuel flow is shown in the schlieren photograph (Fig. 8a). When the heat release is large enough to cause severe expelled shock, the inlet appears as shown in Fig. 8b. Some emission from the reaction zone may be seen downstream of the expelled shock. Since H_2-air emission occurs in the ultraviolet wavelengths only, the emission has been attributed to heated dust and foreign particles in the air: spectrographic analysis tends to confirm this.‡

‡ The source of highest intensity emission from an H_2-air reaction for the experiments was the OH ultraviolet bands. A spectrographic analysis detected the OH bands superimposed on a continuum radiation attributed to heated dust particles in the stream. It has been suggested[21] that there are other weak sources of visible radiation that are not detectable in the spectrograph because of its limited sensitivity range, but that may be visible to the eye and camera, such as O_2, H_2, and H.

At the higher fuel concentrations, the flow in the model appears to be choked, with subsequent re-expansion inside the constant-area section. An explanation of this phenomenon is that the boundary-layer growth forms a "throat" at the choke point, followed by an expansion. The expansion is caused by low pressure at the model exit which feeds upstream through the thickened boundary layer. Experimental data obtained by Chapman, Kuehn, and Larson[15] indicate that the transmission of a favorable pressure gradient forward through a boundary layer is not uncommon. A similar experimental effect, flow acceleration at the end of a constant area duct with supersonic flow and chemical heat release, was observed by Valenti.[16] The pressure drop noted in his experiments was attributed to heat transfer. However, heat transfer was not sufficient to explain the sharp pressure drop at the end of the axisymmetric model (curve D).

The choking pressure ratio indicated in Fig. 7 was calculated from a force balance

$$F_2 - F_1 = p_2A_2 (1 + \gamma M_2^2) - p_1A_1 (1 + \gamma M_1^2) \quad (4)$$

Since the area is constant, the total force applied by the duct to the flow is a result of wall shear, so that

$$F_2 - F_1 = \bar{\tau} A_w = A[p_2(1 + \gamma M_2^2) - p_1(1 + \gamma M_1^2)] \quad (5)$$

where A_w is the wetted area of the wall. After substitution and rearranging,

$$p_2 = \left[\frac{1}{\gamma M_2^2 + 1}\right]\left[4\frac{\bar{\tau} L}{D} + p_1 (1 + \gamma M_1^2)\right] \quad (6)$$

The choking pressure was obtained by setting $M_2 = 1$, using initial values of pressure, Mach number, $\bar{\tau}$ determined from fuel-off conditions, and an L/D at which maximum pressure was attained. Good agreement is shown between the computed and experimental values of static pressure at the peak (curve C).

A calculation of Mach number at the end of the constant-area section of the model was made for curve C. It was assumed that the flow was choked at the peak pressure point, that the reaction was quenched when the expansion started, and that the gas expanded isentropically to the wall static pressure at the exit of the constant-area section. The calculated exit Mach number was 1.20 as compared to a value of 1.24 found from the experimental data. This agreement supports the contention that the flow chokes within the constant-area section.

Correlation of Theory and Experiment for Ignition Delay

By using the onset of experimental pressure rise as a measure of ignition delay, a comparison can be made with the ignition delay predicted from one-dimensional chemical kinetic computations where the same inlet conditions are used. However, there may be effects that prevent this comparison from being a valid one.

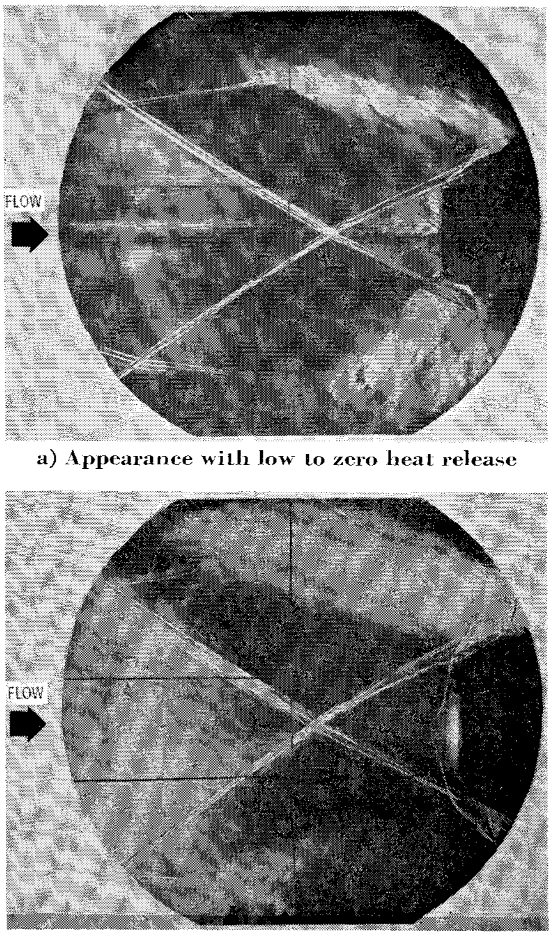
Data were used from the axisymmetric model. For supersonic flow in an axisymmetric duct, the lip shocks inside the duct tend to coalesce into a small normal shock or Mach disk in the center of the duct.§ Shock waves are reflected from this Mach disk and impinge on the interior walls, as indicated by the data in Fig. 7, which show a rapid compression followed by an expansion near the inlet (curve A). Because of the short time duration of the compression and expansion (a chemical speed-up followed immediately by a slow-down), the total effect of this shock on the chemistry was assumed to be small.

Boundary-layer ignition was regarded as a possibility in the model and could seriously affect any attempt to measure ignition delay. However, a calculation of temperature profiles using the method of Chapman and Rubesin[17] indicated a boundary-layer temperature slightly higher than freestream static only at a point very close to the leading edge. Since no fuel was present in the boundary layer near the leading edge, the possibility of boundary-layer ignition was discarded. However, this mode of ignition is a distinct possibility at higher Mach numbers where temperatures within the boundary layer can easily exceed the stream static temperature.

Since vibrational energy requires a longer time to reach equilbrium after passing through a shock wave than translational and rotational energy, the translational and rotational modes were considered to contain an energy increase for the duration of the vibrational relaxation. The additional rotational and translational energy may tend to increase the speed of the reaction, but the effect is relatively small and within the experimental accuracy. However, vibrational energy relaxation time may be of importance at higher Mach numbers or when greater shock strengths exist.

As was pointed out earlier, the fuel mixture enters the model as a core, and initial heat release occurs only in the central portion of the flow. However, the relatively low supersonic duct Mach numbers will not support an appreciable lateral pressure gradient, so that with heat generated in the core, little change in the location of the point at which wall static pressure begins to rise would be expected.

These effects (lip shock, boundary-layer heating, vibrational energy relaxation, and core heating) could cause error in the experimental determination of ignition delay. However, the analyses indicated that these effects were insignificant. In contrast, possible large errors could be introduced in determining *input* conditions for the chemical kinetic calculations corresponding to the experiment. It was therefore



a) Appearance with low to zero heat release

b) With expelled shock as a result of excessive heat release

Fig. 8 Schlieren photo of constant-area combustion model inlet.

§ See footnote on page 204.

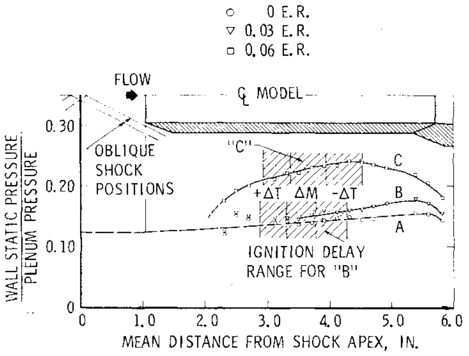


Fig. 9 Pressure rise in constant-area axisymmetric model compared with ignition delay distance predicted by chemical kinetic calculations.

necessary to include the effects of model test fuel and preheater generated free radicals when determining these input conditions.

Free radicals entering the shock-induced reaction have a pronounced effect on ignition delay.[9] Rhodes[18] has further shown from an analysis of computed reaction histories that ignition delay is a strong function of the initial free radical concentration for a wide range of temperature and pressure. In the experiments, free radicals were produced from combustion in the preheater. Their concentrations were calculated for each preheater equivalence ratio by assuming frozen flow from the nozzle throat to the combustion-inducing shock. These concentrations were then used as the inlet gas composition for the chemical kinetic calculations that were compared to the experimental results.

Since the interior of the model could not be probed during combustion, the model inlet conditions were determined for the fuel-off case only. The effects of fuel addition on model inlet conditions are small at the low equivalence ratios used and could be neglected for pressure measurements. However, it may be seen from Eq. (2) that ignition delay is very sensitive to inlet initial temperature. Therefore, when inputs for the chemical kinetic program were selected, the model inlet conditions for the fuel-off case were not applicable. Instead, the model fuel concentration was experimentally determined for each test point and was used to correct the inlet temperature. This corrected temperature was then used as the initial temperature fed into the chemical kinetic computer program. Earlier experiments with the two-dimensional model had shown that inlet velocity and static pressure were essentially unchanged at equivalence ratios less than 0.1; hence, it was necessary to calculate only the effect of fuel concentration on static temperature.

The known quantities in the problem were: 1) stream properties without fuel, 2) fuel/air ratio in the center of the stream at the model inlet, and 3) fuel stagnation temperature and pressure. It was first necessary to solve for the specific heat and stagnation temperature of the mixture:

$$C_{pm} = [1/(1+f)](C_p + fC_{pf}) \tag{7}$$

$$T_{tm} = \frac{T_t + fT_{tf}\,[C_{pf}/C_p]}{1 + f[C_{pf}/C_p]} \tag{8}$$

The assumption that mass and energy are transported by the same mechanism is implicit in the equation for total temperature. Since it was assumed that $V_m = V_a$,

$$C_{pm}(T_{tm} - T_m) = C_p(T_t - T) \tag{9}$$

Equation (9) may be rearranged to solve for mixture static temperature:

$$T_m = T_{tm} - (C_p/C_{pm})(T_t - T) \tag{10}$$

Mixture temperature was determined for each test condition and was used as the initial temperature for the corresponding chemical kinetic computation.

The comparison between experimentally determined ignition delay and that predicted by the chemical kinetic calculation of Ref. 12 is presented in Fig. 9. (Curves A, B, and C correspond to the same curves in Fig. 7.) The ignition delay distances for curves B and C predicted by the chemical kinetic calculations are shown as shaded ranges that include Mach number and static temperature uncertainties. It is evident that curve B begins to diverge from curve A within the calculated range. This would seem to give experimental verification of the ignition delay predicted by the chemical kinetic program in Ref. 12.

The lack of agreement between experiment and theory shown by curve C is inconclusive. At the higher equivalence ratio of curve C there are several possible effects that could cause the point of initial pressure rise to occur further forward in the duct. They are: 1) a thicker boundary layer, caused by the greater adverse pressure gradient, which conducts downstream pressure further forward along the duct walls, 2) decreased inlet Mach number caused by increased fuel concentrations which increases the size and effect of the shock disk formed by the intersection of the lip shocks, and 3) interaction between the thickened boundary layer and the shocks reflected from the shock disk, which causes boundary-layer separation.¶ Since the change in inlet Mach number was very small between curves B and C, effect 2 is probably not important. There is no estimate as to the magnitude of the effects of 1 and 3, but it is possible that they could significantly affect the experiments.

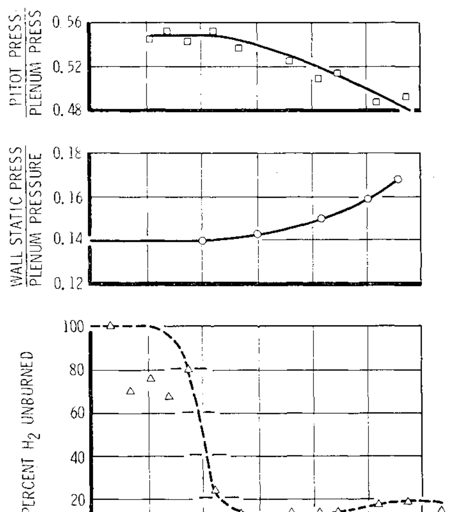


Fig. 10 Measured parameters in supersonic heat addition (H_2-air combustion) in a constant-area duct at p_0 = 0.7 atm, *ER* ~ 0.08.

Correlation of Sampling and Pressure Data

Figure 10 shows pitot and static pressure data obtained concurrently with gas sampling for molecular hydrogen concentration.[9, 10] Ignition delay has been defined in various ways[19] but, in general, refers to that portion of the chemical reaction marked by rapid decrease in molecular hydrogen concentration, rapid increase in free radicals, and the begin-

¶ The suggestions that the lip shocks could form a normal shock disk and that the shocks reflected from the disk could cause boundary-layer separation were made by R. J. Lane of Bristol Siddeley Engines Ltd.

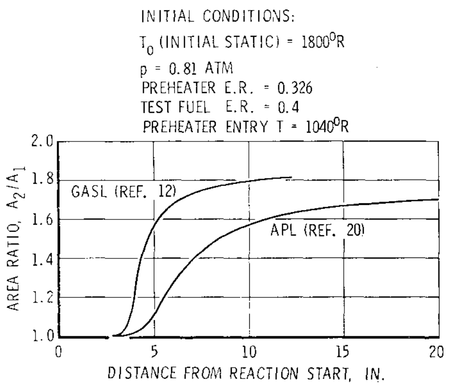


Fig. 11 Comparison of theoretical area change for H_2-air reaction at constant pressure; velocity = 3385 fps.

ning of temperature and pressure rise. The data show that the molecular hydrogen "fast" reaction period and the start of pressure changes coincide, as would be expected from a chemical kinetic history. These data were obtained by probing the two-dimensional model containing a slightly divergent combustion chamber.

Comparison of H_2-Air Reaction Kinetics Computer Programs

Two current H_2-air reaction computer programs[12, 20] use somewhat different reaction rates for the sixteen forward and reverse intermediate H_2-air reactions.[9] A comparison was made between the constant pressure calculations of both programs with identical input temperature, pressure, velocity, and composition for each (Fig. 11). Agreement was found for ignition delay distance. However, the recombination reactions exhibit a definite tendency to diverge. This is shown in Fig. 11 by the plot of area change for the constant pressure reaction. The significance of the comparison is that it shows the need for more accurate chemical kinetic rates for the recombination processes. Any attempt to design a supersonic combustion chamber or nozzle must depend on accurate rate values, or realistic experimental evaluation of the over-all reaction, which can be used in some type of rate equation. Future experiments of the type reported herein may prove useful in providing better information on chemical reactions in high-speed flow.

Concluding Remarks

The experiments have demonstrated that heated and premixed hydrogen and air can be ignited by an oblique shock and that the reaction can be contained in a constant-area duct. The rate and quantity of heat released can be regulated by governing the inlet temperatures, pressure, fuel concentration, and Mach number. There appear to be logical explanations for the observed phenomena (based on the pressure data) in terms of one-dimensional flow analysis.

Ignition delay as determined by the onset of pressure rise has shown reasonable agreement of the limited data recorded with those predicted[12, 20] by chemical kinetic calculations. However, further work is necessary to experimentally verify over-all reaction rates for the chemical reactions. If this can be done, chemical kinetic calculations can be used to design the contours of hypersonic ramjet combustors and nozzles. Alternatively, it might be possible to use models of the types discussed here to study reaction kinetics. Although the experiments were confined to reactions in a Mach 1.6 to 1.7 stream and a particular small model configuration, some of the problems considered, such as boundary-layer separation and boundary-layer ignition, portend similar problems at the higher Mach numbers and temperatures which are characteristic of hypersonic flight.

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