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A Missile Technique for the Study of Detonation Waves [1]

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Problems and effects of stabilizing combustion and detonation against hypersonic flow were investigated by observation of a 20 millimeter spherical missile in a stoichiometric mixture of hydrogen and air at rest. Combustion produced detectable effects on the shape and position of the shock wave at Mach numbers between 4 and 6.5, and above pressures of one-tenth atmosphere. Chemical equilibrium probably was not reached in the time the gas spent near the front of the sphere. One of the factors in the delayed equilibrium was delayed ignition behind the shock wave, which was observed to be between about one and ten microseconds. Ignition delay is explained in terms of chemical kinetic theory and compared with results of experiments in shock tubes. Strong combustion-driven oscillations originated in front of the sphere, with frequencies up to about one-tenth megacycle per second. These were observed when the Mach number was less than 6 at a pressure of one-half atmosphere, and less than 5 at one-quarter atmosphere. A large reduction of the drag coefficient of the missile was noted in one case of intermittent combustion.

1. Introduction

Combustion in hypersonic streams is of considerable theoretical interest, and has a possible application in propulsion of missiles and aircraft. Application would require a knowledge of the problems of stabilizing the combustion against hypersonic flow. Recently some information has become available through the research of Gross [1] [2] on shock-induced detonations stabilized on a wedge. Nicholls et al., [2] studied combustion behind a normal shock wave in a free jet. These experiments were limited in range by problems encountered in generating streams of gas having a realistically high total temperature. In addition, the results may have been influenced by the methods used in mixing the fuel and air.

It appeared that these problems could be avoided by using a hypervelocity missile in a combustible gas at rest. This report presents the initial results obtained from a study of combustion near a missile in a mixture of 30 percent by volume of hydrogen in air. Hydrogen was chosen because a substantial theory exists on its combustion, and because it may be useful in propulsion systems. Wave shape and position were studied by high speed photography to determine the conditions required to induce combustion, and to determine the effect of the missile on the combustion. Combustion is induced by the enthalpy rise of the gas in or behind the shock wave which is dependent only on the speed of the missile, and thus it was possible to create over and under driven detonations for study. Since the missile and wave were viewed through a stationary gas, the observations were not influenced by boundary layers and turbulence as in experiments on detonation in tubes. These advantages permitted observations on the spatial structure of detonation waves, which were interpreted in terms of the mechanism of reaction of hydrogen with oxygen. Although temperature changes would have an important effect, temperature was not changed from ambient in these initial experiments. Pressure was varied from 20 mm Hg to atmospheric to learn of its effects.

2. Experimental Apparatus

The apparatus is illustrated in figure 1 and is composed of: (a) A gun and range which contains the test gas, (b) an electronic time delay ratio generator which triggers a submicrosecond light source (spark) when the missile is expected at the observation station, (c) two electronic microsecond counters, (d) an optical system to make schlieren or shadow pictures of the missile, and (c) auxiliary equipment such as pressure gages, vacuum pumps, etc.

In the experiments to date nylon spheres, 0.785-in. diam (19.94 mm), have been mounted in an electric-primed case containing a fast burning gun powder. This technique of propelling the missile was borrowed from Eckerman [3]. Other techniques may be used, such as the light gas gun [4]. The missile travels through an evacuated range illustrated in the diagram, and enters the test gas after breaking through a thin diaphragm (8).

Signals from the phototubes and ratio delay generator actuate the two time counters, and the measured time intervals are used to compute the velocity of the missile between light beams and in the test gas. The schlieren optical system is illuminated by a spark of about 10^{-7} sec duration, obtained by the discharge of a barium titanate capacitor, and in some early pictures by light from a flash tube of about 0.5×10^{-6} sec duration. These illuminate a piece of ground glass adjacent to four razor blades which outline a rectangular slit $\frac{5}{16}$ by $\frac{1}{8}$ in. This is used as the light source of the schlieren system. A rectangular slit is also used as a schlieren stop to

[1] Sponsored by the Air Force Office of Scientific Research, U.S. Air Force.
[2] Figures in brackets indicate the literature references at the end of this paper.

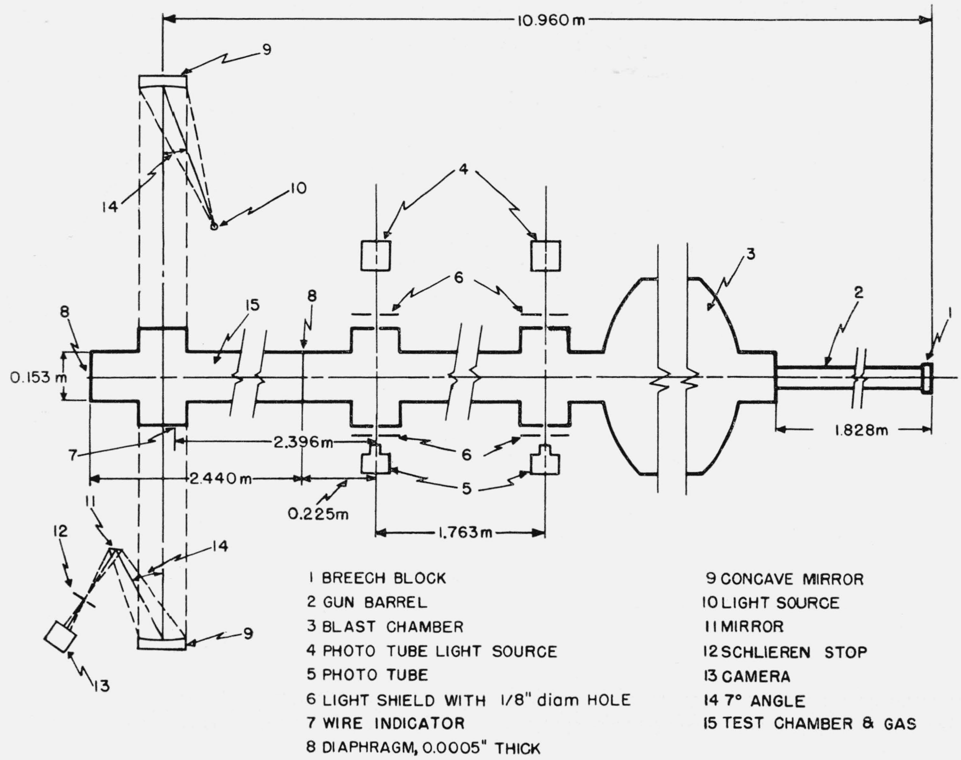

FIGURE 1. *Experimental apparatus.*

reduce effect of light emitted by combustion. Both slits are vertical to measure gradients of index of refraction parallel to the horizontal axis of flight. It was also found necessary to close the shutter of the camera within tenths of a millisecond after taking the schlieren picture to further reduce the effect of emitted light. The apparatus was used to obtain photographs of the missile and the shock and combustion waves that were generated in the combustible gas.

3. Experimental Results

All experiments in this report were made in a mixture of 30 volume percent of hydrogen in air. This mixture was chosen because it may be useful in propulsion systems, and because a considerable theoretical and experimental background exists on its combustion. Preliminary tests were made at pressures, p, from 0.026 to 1 atm absolute and at Mach numbers, M, up to 6.5. These showed that effects of combustion on wave shape and position could be detected at pressures of 0.1 atm and higher, and for a range of M from 4 to 6.5. The upper limit of 6.5 on M was fixed by the gun and missile, while these pressures were selected to partially cover the range that might be found in propulsion systems. Since results between pressures of 0.5 and 1 atm are qualitatively similar, this report describes results at pressures of 0.1, 0.25, and 0.5 atm.

M at the observation point was determined from two speed measurements and the speed of sound, C, where C is 409 m per second in the combustible gas and 347 m per second in air, both at 300 °K. It was estimated from drag coefficient information on spheres, that since the percentage change of velocity of the missile in the test gas was less than 5 percent, the change of deceleration would be less than 10 percent. Therefore, the deceleration was assumed a constant in the derivation of M at the observation point.

3.1. Photographic Results

Experimental results are primarily in the form of pictures obtained by schlieren photography of the region surrounding the missile. These are presented in figures 2 and 3. One shadow photograph is presented in figure 2b. Schlieren pictures of the missile in air, 2a, and in a mixture of nitrogen and hydrogen, 3a, are also presented for comparison with those in the combustible gas. All pictures contain two dark heavy lines, one from a horozintal wire (out of focus) to help delineate the line of flight, and one from a wire perpendicular to the latter to give the position of the missile in the test gas. It is evident that in the range of M between 4 and 6.5, combustion takes place in the shocked gas near the missile, and in many instances has an oscillatory or intermittent property.

At M=4.3 and 0.5 atm, picture 2b shows a discrete region of flame surrounding the sphere. Schlieren

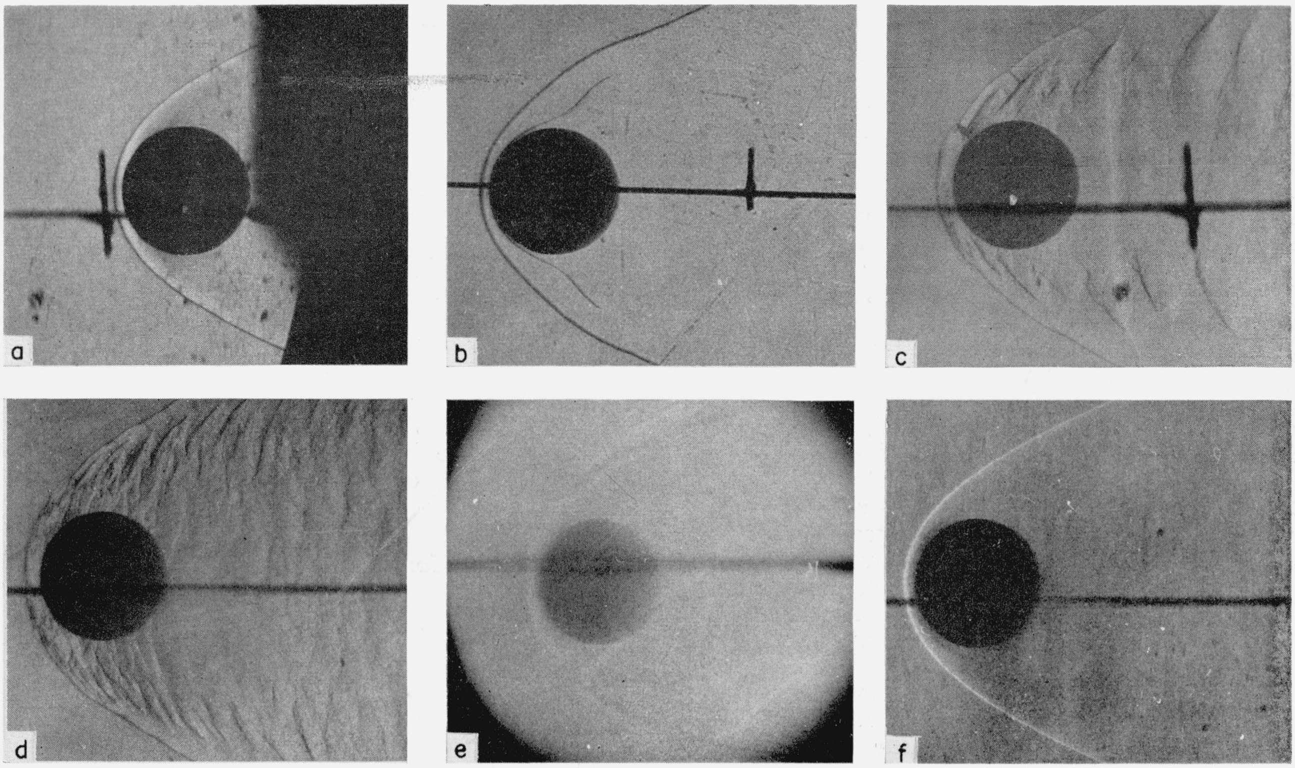

FIGURE 2. *Shadow and schlieren photographs of missile.*

(a) M=5.4; p=0.5 atm; Air
(b) M=4.3; p=0.5 atm
(c) M=4.8; p=0.5 atm
(d) M=5.5; p=0.5 atm
(e) M=6.3; p=0.5 atm
(f) M=5.0; p=0.1 atm

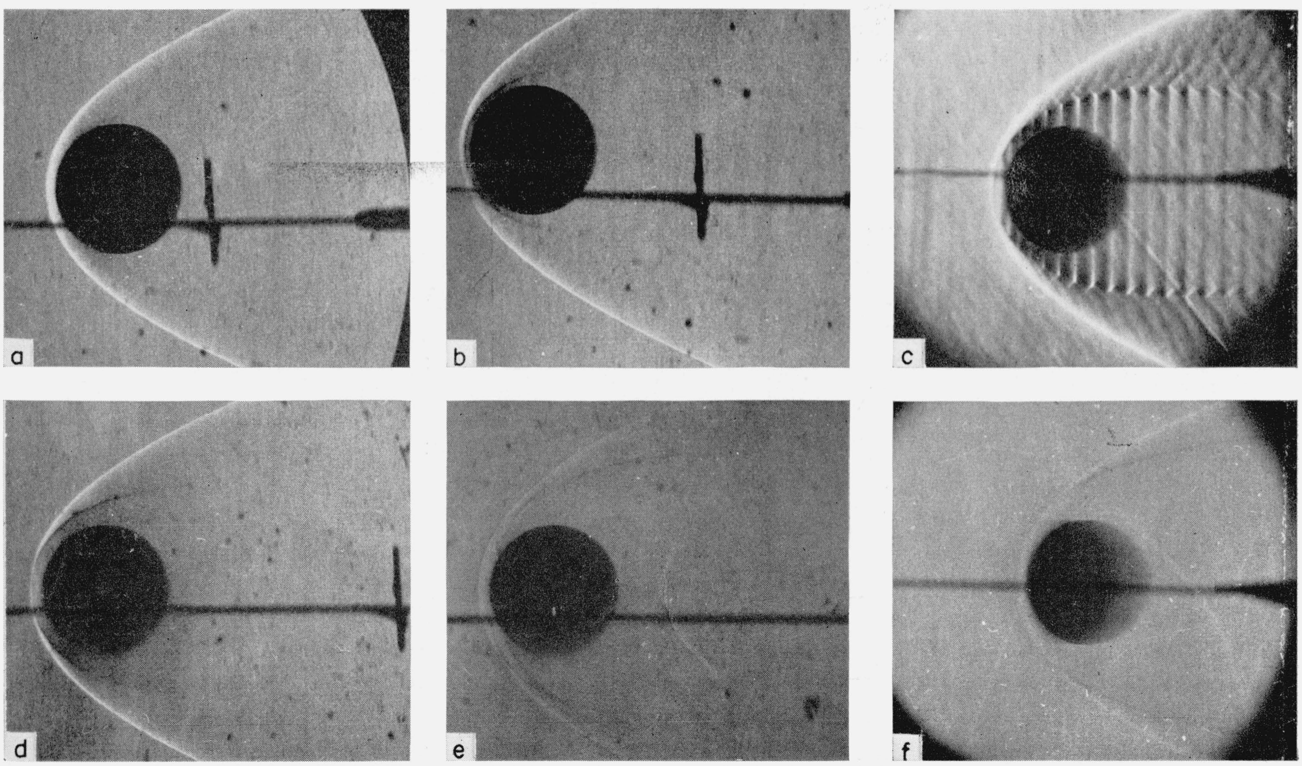

FIGURE 3. *Schlieren photographs of missile, p=0.25 atm.*

(a) M=4.4; $.3H_2+.7N_2$
(b) M=4.5
(c) M=4.9
(d) M=5.1
(e) M=5.9
(f) M=6.5

pictures of the region behind the sphere at this value of M showed discrete regions of burned or burning gas spaced about three sphere radii from center to center and two sphere radii thick along the axis of flight. The regions of combustion overlap at higher M, and finally merge to give a continuous flame front at M of about 6, as in picture 2e. Only one picture at 0.1 atm is presented, 2f, in which a combustion wave is demonstrated between the front of the sphere and the shock wave at $M=5.0$. Other pictures above $M=5$ did not demonstrate combustion waves that could be measured.

When the pressure was 0.25 atm, as in figure 3, a dark combustion wave is plainly visible in picture 3b at $M=4.5$, and there is evidence of a weak oscillation in the combustion zone. Amplitude of the oscillation became large enough to be plainly demonstrated by the photography when M was 4.9, picture 3c. Above $M=5$ the combustion zone is enclosed in a smooth flame front as illustrated in pictures 3d to 3f. Picture 3e has additional waves that apparently emanate from the front surface of the missile. Breakage of the missile was troublesome during these experiments, and it is possible that a small chip caused these waves.

There are other differences between the results that should be explained. First, pictures 2e, 3c, and 3f were taken using light from the flash tube with relatively poor picture quality. Second, the shock waves are bright in figure 3 (except 3e) compared to the dark waves of figures 2a to 2d. Large positive density gradients in the shock at 0.5 atm move the image of the light source, to the left as illustrated in part (a) of figure 4, to produce a dark wave. A smaller density gradient at 0.25 atm gives less movement, as shown in part (b), and results in a bright wave. A negative density gradient in a combustion wave would have the effect, as shown in part (c), to produce a dark wave. In air, and in some instances with the combustible gas a black and white shock wave is produced, as illustrated in figures 3e and 4. When the slit was replaced by the ordinary knife stop all shock waves were bright, as in figure 2e. However, the bright wave of figure 2f is a result of the low pressure which caused the effect illustrated in part (b) of figure 4. In all cases it is evident that a substantial part of the total compression in the shock wave takes place before combustion (or detonation) moderates the density rise of the shock wave. Picture 3c appears to have three lines radiating from the sphere; these lines are the results of imperfections present in one of the windows used in early experiments.

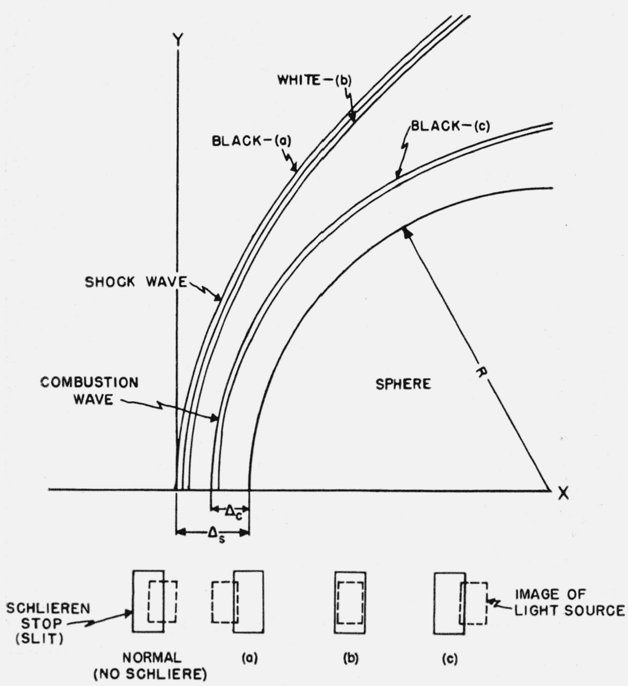

FIGURE 4. *Image position of light source for various regions of the schlieren picture and wave coordinate system.*

3.2. Effect of Combustion on Wave Shape and Detachment

Since combustion can be expected to have an appreciable effect on the density of the gas between the wave and sphere, it would also be expected that combustion would affect the shape and detachment of the wave from the sphere. Thus a measure of he extent of combustion in the vicinity of the sphere may be derived from these quantities. A projector was used to enlarge the image on the negatives from one-half to 7.64 times actual size, and the image of the waves and missile were traced on coordinate paper containing ten lines to the inch. Coordinates of the leading edge of the waves were read from the drawings and plotted to determine their shape.

It was found that in the immediate vicinity of the sphere the shock waves could be represented by parabolas, and the combustion waves (at 0.25 atm) by ellipses. No attempt was made to measure the shape of the combustion waves at 0.5 and 0.1 atm. Equations of the waves, in the coordinate system of figure 4, are given below:

$$\text{shock wave:}\left(\frac{y}{R}\right)^2=m\,\frac{x}{R},\quad \frac{x}{R}<0.5$$

$$\text{combustion wave:}\left(\frac{y}{R}\right)^2=a\left(\frac{x}{R}-\frac{\Delta_s-\Delta_c}{R}\right)-b\left(\frac{x}{R}-\frac{\Delta_s-\Delta_c}{R}\right)^2,\quad \frac{x}{R}<0.9.$$

The constants m and Δ_s were determined and used to specify the shape and detachment of the shock wave, while the constants a, b and Δ_c were determined and used to specify the shape and detachment of the combustion wave. These quantities are plotted in figure 5 for the shock waves at 0.5 and 0.1 atm and in figure 6 for both waves at 0.25 atm, all against M. Both figures contain results of these measurements in air and Eckerman's [3] measurements in oxygen which he obtained with an 8mm sphere. His values of Δ_s are less than the present results in air, possibly because he used a rotating mirror and a fast spark to provide a stopped image of the sphere and wave.

Results from tests with both light sources are included in the figures. Spark data on Δ_s are represented by closed circles, and flash tube data by closed triangles. Smaller values of Δ_s result from the spark data, perhaps for the same reason that Eckerman's results are smaller than ours in air. Values of wave shape m are also differentiated the same way, but values of Δ_c and constant a are not.

Constant b (not plotted) varied linearly between a value of 0.7 at $M=4.5$ and a value of 0.58 at $M=6.1$ when the spark pictures were evaluated, and was about 0.15 smaller at corresponding M when the flash tube pictures were evaluated. All data at 0.1 atm were obtained with the faster spark.

Inspection of figures 5 and 6 shows that the effect of combustion is to move the shock wave away from the sphere, and that the effect increases as pressure increases. The large effect of pressure, and the maximum in the curve of wave detachment, suggest that chemical reaction is not complete near the wave front. Thus the situation probably can be described as a shock wave followed by chemical reaction with relaxation time comparable to or larger than the time the gas spends near the front of the missile. Duff [5] has calculated the progress of the chemical reaction behind shock waves, and Bird [6] has outlined a graphical-numerical technique of handling the equations of motion of the gas between the wave and sphere. It is theoretically possible to combine these calculations to derive the shape and detachment of the waves. Alternatively, the measured shape and detachment may be used to check the validity of the chemical rate processes. Any computation of this kind would be long and involved. For the time being an estimate of the situation is all that will be given.

Eckerman [3] observed that the shock-detachment distance in air varied approximately inversely with the density ratio ρ_2/ρ_1 across the shock wave, where subscript 1 represents the condition before and 2 the condition after the wave. This relation was used to estimate the wave detachment to be expected in these experiments. Therefore, at a given M, wave detachment with combustion would be approximated by

$$\Delta_s=\Delta_{s,\,\mathrm{air}}\frac{\rho_2}{\rho_3},$$

where subscript 3 represents the condition behind the detonation wave. Density ratios ρ_1/ρ_2 of adiabatic shock waves in the mixture of air and hydrogen were calculated using real gas properties, and calculated values of ρ_1/ρ_3 across detonations at chemical equilibrium were given by Eisen and coworkers [7]. They showed that an initial pressure change from one to 10^{-3} atm changed the density ratio across Chapman-Jouguet detonations by about only 6 percent. Thus it seemed that their calculated

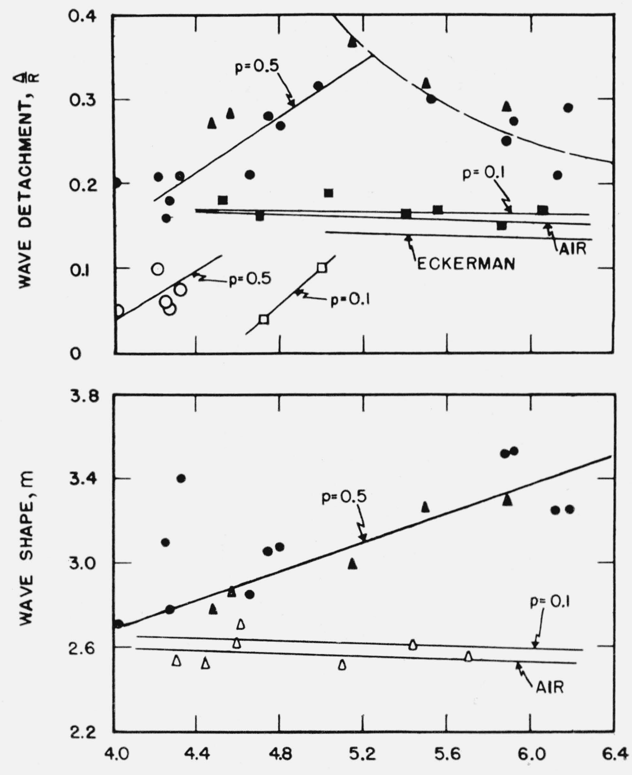

FIGURE 5. *Effect of Mach number on wave detachment and shape for a mixture of 30 volume percent* H_2 *in air at pressures of 0.1 and 0.5 atm.*

Shock wave ●▲■ for H_2 and air; △ for air.
Combustion wave ○□.

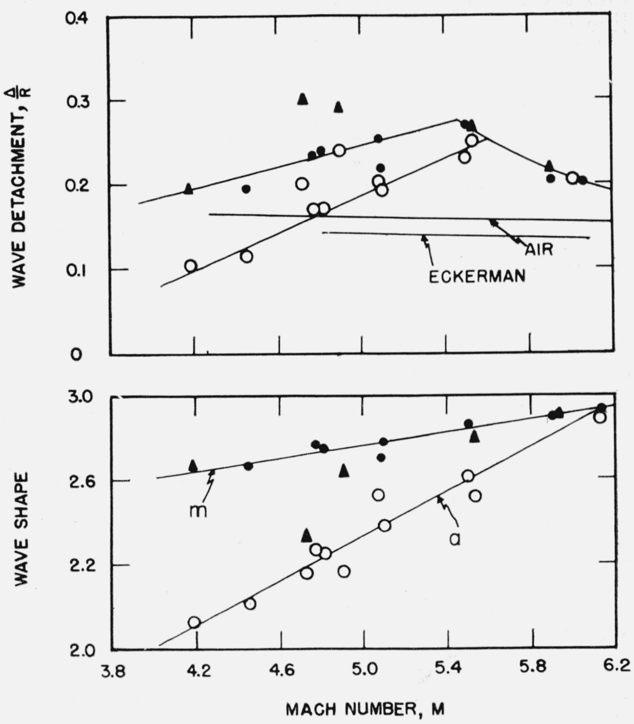

FIGURE 6. *Effect of Mach number on wave detachment and shape for a mixture of 30 volume percent* H_2 *in air at a pressure 0.25 atm.*

Shock wave ●▲.
Combustion wave ○.

values of density ratio across strong detonations at 1 atm could be used for comparison with results of these experiments. These calculated density ratios, and observed values of $\Delta_{s,\mathrm{air}}$, gave estimates of Δ_s for detonations which are plotted as a dashed line in figure 5. Agreement between experiment and estimate above $M=5.5$ indicates the possibility that chemical equilibrium was approached near the front of the sphere. However, two experiments at $M=5.3$ and 6.0 at a pressure of 1 atm showed an increase of about 20 percent in Δ_s over the values at 0.5 atm. This indicates that the estimated curve would probably be corrected to larger values of Δ_s by an exact fluid-mechanical calculation. Thus, chemical equilibrium is probably not reached under any condition in these experiments near the front of the missile, especially below M about 5.5.

3.3. Ignition Delay and Chemical Kinetics

One factor involved in delayed equilibrium is the observed delay in ignition behind the shock wave. A picture such as figure 3d could be used to determine the state of the gas and the time spent between shock and combustion waves along any selected streamline through fluid-mechanical calculations as outlined in reference [6]. Comparison of the time intervals and the state of the gas for different streamlines might furnish information on the factors that control the onset of combustion. In this report we use the observed separation between shock and combustion waves to estimate the state of the gas and the ignition delay time, τ, on the central streamline (axis of flight).

The time spent by the gas between the shock wave and x is

$$t(x)=\int_0^x \frac{dx}{V}$$

where V is velocity on the central streamline, and x is defined as in figure 4. Bird [6] stated that the velocity decay to zero at the sphere is approximately linear. If it is also assumed that the flow between waves is the same as that behind a shock wave of the same Mach number and shape without subsequent combustion, velocity V is given by the equation

$$V=V_2\left(1-\frac{x}{\Delta_{s,\mathrm{air}}}\,\frac{m_{\mathrm{air}}}{m}\right).$$

V_2 is the velocity behind the shock in the real combustible mixture, and m is the shape of the shock wave and is proportional to the radius of curvature at the origin. Insertion in the integral and integration from zero to $x=\Delta_s-\Delta_c$ gives

$$\tau=-\frac{\Delta_{s,\mathrm{air}}}{V_2}\,\frac{m}{m_{\mathrm{air}}}\ln\left(1-\frac{\Delta_s-\Delta_c}{\Delta_{s,\mathrm{air}}}\,\frac{m_{\mathrm{air}}}{m}\right).$$

Values of Δ_s, Δ_c, $\Delta_{s,\mathrm{air}}$ and m were taken from figures 5 and 6, and calculated values of τ, which is considered an ignition delay, are plotted in figure 7. Spark data only are used here since it is considered more accurate than the "slower" flash tube data. Ignition delays cover a range of zero to 10 μsec. Curves in the figure are derived from a correlation to be described. Portions of the curves are dotted to indicate extrapolations.

An attempt was made to explain these ignition delays through considerations of chemical kinetics, and to compare the results with those of other research workers. Lewis and von Elbe [8], Schott and Kinsey [9], and Duff [5] have discussed the kinetics of the hydrogen-oxygen reaction. Their explanation of the factors that are involved in the initial stage of the reaction is used. Five important reactions, and their contribution to the rate of formation of the hydrogen atom, are listed below:

(0) $H_2+I=2H+I$	$[\dot{H}]_0=2k_0[H_2][I]$
(1) $H+O_2=OH+O$	
(2) $O+H_2=OH+H$	$[\dot{H}]_{1-3}=2k_1[H][O_2]$
(3) $OH+H_2=H_2O+H$	
(4) $H+O_2+I=HO_2+I$	$[\dot{H}]_4=-k_4[H][O_2][I]$

Inert species are indicated by I, concentrations are indicated by brackets, and k's are rates of reaction. Reactions (2) and (3) are considered very fast compared to the others, and their effect is to form three hydrogen atoms for every one consumed by (1). Total rate of formation of [H] is the sum of the three rate equations, if it is assumed that there is no reverse reaction of any significance during the induction period. The temperature behind the shock wave, T_2, and therefore k, as well as the concentrations of H_2, I, O_2, are probably unchanged during the induction period. Therefore the rate equation can be integrated to give

$$\ln\frac{[\dot{H}]_t}{[\dot{H}]_{t=0}}=2k_1[O_2]\left(1-\frac{k_4[I]}{2k_1}\right)t.$$

Although the particular chain initiation reaction chosen is open to question, other plausible reactions essentially would modify only the denominator of the left side of the above equation.

Schott and Kinsey [9] tested a part of the above equation over a range of time intervals from 5 to 500 μsec for combustion behind incident and reflected shock waves in tubes. A straight line was obtained when $\ln[O_2]\,\tau$ was plotted against T_2^{-1}, where T_2^{-1} is proportional to $\ln k_1$. This implies that the ratio of $[\dot{H}]$ is always the same at ignition, when $t=\tau$, or that changes of $\ln\ln([\dot{H}]/[\dot{H}]_{t=0})$ were not significant over the range of the experiments. Their intervals were measured to the appearance of the OH radical. Strehlow [10] measured ignition delays behind reflected shocks in tubes over a range of 12 to 880 μsec. His time intervals were from 20 to 100 percent larger than those observed by [9], possibly for the reason that ignition is that much later than the appearance of OH. Both omitted the effect of reaction (4) in their correlation, possibly due to the low pressures of their experiments.

Ignition delays presented in figure 7 are plotted in figure 8 using the correlation given by the above equation. $[O_2]$ was calculated from p_2 and T_2 immediately behind the shock front in the real combustible gas. The temperature rises by about two percent and pressure by about 15 percent between the shock and combustible waves. Therefore the error is less than 15 percent when shock pressure and temperature are used. Information is presented by [8] on the rates of reactions (1) and (4) which can be used to derive the effect of reaction (4). For these experiments it is given by

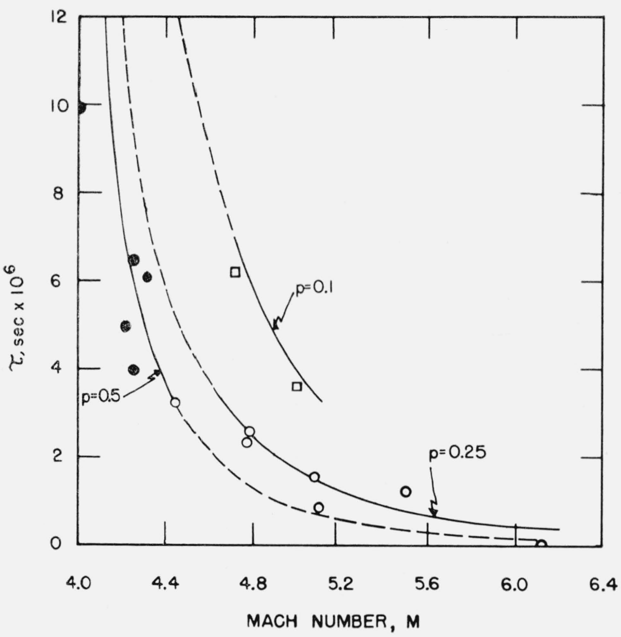

FIGURE 7. *Effect of Mach number on ignition delay behind the shock wave.*

●p=0.50 ○p=0.25 □p=0.10

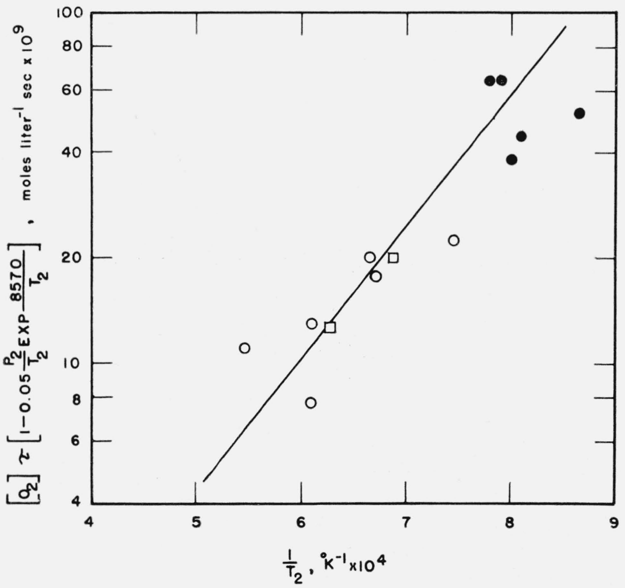

FIGURE 8. *Correlation of ignition delay.*

●p=0.50 ○p=0.25 □p=0.10

$$\frac{k_4[I]}{2k_1}=0.05\frac{p_2}{T_2}\exp\frac{8570}{T_2}$$

where 8,570 °K is the activation temperature of reaction (1), and the constant 0.05 °K atm^{-1} is one-quarter of that derived from the information of [8] for this combustible mixture. Since our data are not comprehensive enough to fix the activation temperature, the constant 0.05 was used in order that the data would give an activation temperature of 8,570 °K. This lies between that of 8,300 °K given by [10] and 8,800 °K given by [9]. Inclusion of the effect of reaction (4) in this manner modified only the time results of experiments at 0.5 atm, in which the pressure was high enough and temperature low enough for reaction (4) to be significant.

The equation of the line in figure 8 is

$$2.3\log_{10}\left([O_2]\,\tau\left[1-0.05\frac{p_2}{T_2}\exp\frac{8570}{T_2}\right]\right)=\frac{8570}{T_2}-23.49.$$

Since the values of $[O_2]\tau$ are only 10 to 20 percent larger than those given by [10], it appears that this method of correlating ignition delay is valid over a range of time from about 10^3 to about 1 μsec. Curves in figure 7 were derived from the above equation, and it appears that these experiments have permitted observation of ignition delay behind shock waves with a time resolution of about a microsecond.

3.4. Combustion Oscillations and Drag Coefficient

Many of the pictures of figures 2 and 3 exhibit oscillations that are driven by combustion, and in one case (picture 2b) an intermittent combustion as demonstrated. At this Mach number of 4.3 and pressure of 0.5 atm, the data on Δ_s and m show large variations, which suggests that the oscillations originate at the front of the sphere. These oscillations may contribute to the scatter of the data on ignition delay described in the previous section, since some of the data were taken from pictures showing oscillations. It is likely that oscillations will appear in applications of hypersonic combustion as it does in most other forms of combustion.

In pictures 3b and 3c, the oscillation appears as a regular wave motion, probably originating in the region ahead of the sphere. Longitudinal oscillations in rocket motors also exhibit this characteristic. They can be described approximately as an acoustic oscillation in a tube with both ends closed, since expansion waves cannot be reflected upstream against the sonic flow in the throat of the nozzle [11]. If longitudinal oscillations between the sphere and wave front contribute to the observed oscillation, it would be expected that they would be comparable to those in a tube with one end open, since in this case expansion waves may be reflected from the shock front. The period of acoustic oscillation, τ', would be given by

$$\tau'=\frac{4\Delta_s}{nC_3}$$

where C_3 is the velocity of sound between the wave and sphere. In picture 3c the velocity of sound would be between 890 and 1,130 m per second. These values are those behind an adiabatic shock wave at $M=4.9$, and behind a detonation at $M=4.8$ at chemical equilibrium, respectively. The latter value was computed by [7]. Using these values and a value of $0.24R$ for Δ_s gives a period between 8 and 11×10^{-6} sec for the fundamental oscillation ($n=1$). This period is, and probably should be, larger than the ignition delay of 2 μsec at this Mach number of 4.9. An accurate analysis of the motion probably would need to take account of a three-dimensional propagation of waves of finite amplitude with a periodicity in the rate of heat release.

Since deceleration of the missile was computed to determine M at the observation point, the drag coefficient, C_D, was also evaluated in each test. Over a period of many tests it became apparent that the time measurements were not reproducible enough to give consistent results. C_D is derived from a small difference between two large velocities, and an error of several microseconds would have a large effect on the difference, but not on the absolute velocity. Since C_D was unusually small in the first test at $M=4.3$, pressure 0.5 atm, a series of eight tests was made to attempt a tentative evaluation of the effect of combustion on the drag coefficient. It was hoped that the system would be reproducible in this short interval of eight tests. At these conditions picture 2b demonstrates intermittent combustion. A rapid lateral expansion of the burning gas at the rear half of the sphere is also evident. This may have delayed the separation of the flow from the rear surface, thereby causing the unusually small value of the drag coefficient.

Four tests at a Mach number of 4.6 showed that the drag coefficient of the missile in air was 0.8 ± 0.1, which compares well with results of about 0.9 given by measurements in wind tunnels. Uncertainty of 0.1 in the drag coefficient corresponds to an uncertainty of about 3 μsec in the time measurements. Drag coefficient changes very slowly with M in this range, and this value of C_D can be compared with results in the combustible gas at $M=4.3$. One test before, and three immediately after the tests in air, all at $M=4.3$, gave 0.4 ± 0.3 in the combustible gas. Since there was no known change in the characteristics of the measurement system, the larger uncertainty in the combustible gas may have been caused by variations in the characteristics of the combustion process. Detachment of the shock wave from the sphere was only slightly larger than that in air, and it seems unlikely that reduced pressure on the front of the sphere caused the reduction of C_D. Hence the tentative conclusion may be drawn that the pressure on the rear of the sphere was increased. It was estimated that a time average value of $6p$ over the rear of the sphere would be needed to reduce the drag coefficient by the observed one-half. Pressure on the rear of the sphere in air was estimated to be $0.3p$. A missile with a turning angle less than that of the rear of the sphere would probably respond even more favorably to this effect and perhaps a net thrust would be realized. Although this effect on the drag coefficient appeared when combustion was intermittent, it is not necessarily dependent on the intermittent character of the combustion.

4. Conclusion

It is concluded that observation of a hypervelocity missile is a technique which may be used to study some of the problems of stabilizing hypersonic combustion. Combustion waves are observed in some instances to follow the shock wave generated by the missile. The spatial separation was converted to ignition delay times, which were between about 1 and 10 μsec. These were about one order of magnitude smaller than previously observed by experiments in shock tubes. Satisfactory agreement between these and observations in shock tubes was obtained through chemical-kinetic theory. Thus it is concluded that the method may be used to determine some of the properties of detonation waves. As in most all applications of combustion, oscillations driven by combustion are demonstrated under some conditions. A possible large reduction of the drag coefficient of the missile was noted in one case of intermittent combustion in which the combustion may have delayed separation of the flow from the rear surface of the missile. Fluid-mechanical calculations will be used to extend the information derived from this technique.

5. References

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