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EXPERIMENTAL INVESTIGATION OF HYPERSONIC RAMJETS

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INTRODUCTION

Hypersonic ramjets are potential power plants for both hypersonic vehicles boosting satellites into orbit, and for long range cruise aircraft. At present several research groups are investigating the problem of designing satisfactory engines, using both theoretical and experimental approaches. One such group is the Hypersonic Propulsion Research Group at McGill University, Montreal. The aim of the present paper is to present the work of the McGill team, much of which has not been published in the open literature. Due to the limited resources of this team, all aspects of the problem have not been followed. However, to make the paper a more complete presentation of the current status of the field, relevant data from other sources have been included. A complete survey of the work of all groups is unfortunately precluded by national and proprietary security.

The McGill team was initiated in the Autumn of 1958, and the first months were spent in a theoretical investigation of hypersonic ramjets. This study showed that conventional ramjets do not give significant performance above Mach 10. However, if the thermodynamic cycle is changed from subsonic to supersonic combustion, then good performance is obtained from Mach 8 to satellite speed.

At that time it was felt that most other groups in the field would adequately investigate ramjets from the then current speed of Mach 4, to the lower hypersonic Mach numbers (i.e., Mach 5 to Mach 10). The McGill group therefore concentrated on the range from Mach 10 to Mach 25, since, unlike other groups, we merely wanted to investigate the problems, and not construct engines.

BACKGROUND THEORETICAL WORK

THERMODYNAMIC CYCLE CALCULATIONS

Figure 1 shows the model of the hypersonic ramjet, which is used for the theoretical analysis. This consists of an air intake diffuser, a supersonic combustion chamber and an exhaust nozzle. The efficiency and thrust produced by the engine has been published by several workers, e.g., Refs. 1, 2, and 3, and the method will not be repeated here. The conclusions are, however, directly relevant to the present discussion, since they determine the test conditions required for experimental investigations. A brief summary of the findings is therefore given as follows:

Figure 2 shows typical values of the overall efficiency and fuel specific impulse, versus flight Mach number. From this graph the Mach number range of interest, and the potentially high performance, are both immediately obvious.

The internal Mach numbers and static temperatures are important to the present study since they determine:

(*a*) the amount of diffusion which the intake must achieve
(*b*) the operating conditions of the combustion chamber, and
(*c*) the entry conditions of the propelling nozzle

These parameters are plotted as a function of flight Mach number on Fig. 3.

The fuel used for these studies is hydrogen, and it will be shown later that this is an almost unequivocal choice.

The first conclusion that we may draw from these figures is that the combustion chamber entry temperature is above the spontaneous ignition limit of hydrogen (i.e., 600°C) at all Mach numbers greater than 8, thus the fuel ignition should present no problem. The stagnation temperatures necessary to achieve this condition are shown in Fig. 4, where it can be seen that temperatures in the range 3000°K to 10,000°K are necessary. The difficulty of achieving these temperatures on a test facility is immediately apparent.

A second conclusion can be drawn from the high diffuser exit Mach number. By avoiding the large losses which occur in the transonic region of a diffuser, very high efficiency intake diffusers should be possible. The effect of intake recovery on the overall efficiency is shown in Fig. 5 for a flight Mach number of 15.

Problems of expanding the combustion products efficiently in the exit nozzle are considerably simplified by the high nozzle-entry Mach number. The dissociation level of the gas entering the nozzle can be controlled by suitable variation of the combustion chamber area. Thus losses due to nonequilibrium gas flow behavior can be minimized. Since the gasses spend a very short period in the engine, those components of the gas which have long relaxation times, such as

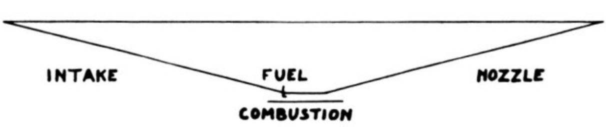

Fig. 1.

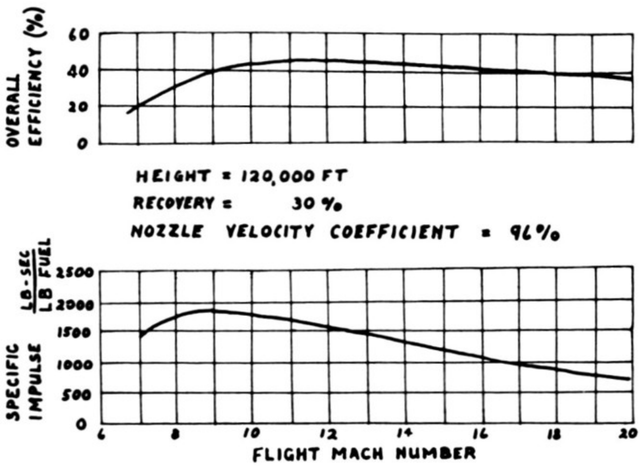

Fig. 2.

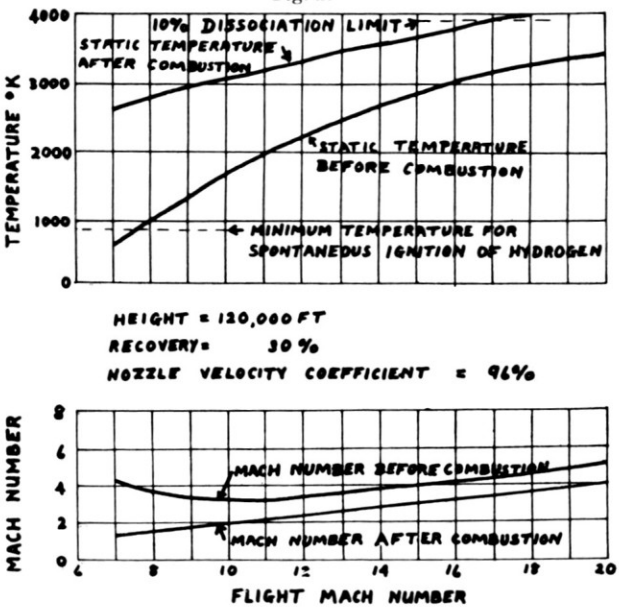

Fig. 3.

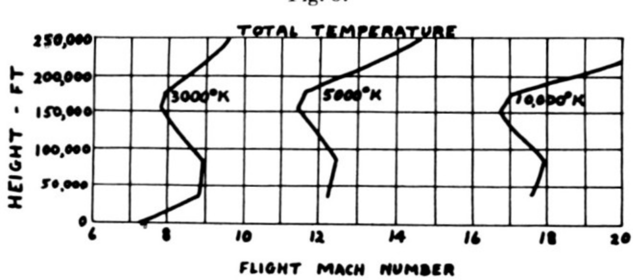

Fig. 4.

nitrogen, do not have time to dissociate. The problem of recombining them in the nozzle may not therefore arise.

The elimination of the throat of the nozzle also eliminates the severe heat transfer encountered when a stream approaches Mach 1, hence the nozzle cooling problem is considerably reduced. While we are considering the cooling, it should be noted that cooling the intake flow improves the thermodynamic efficiency of the cycle, besides being beneficial from the boundary-layer stability aspect.

Two modifications of the basic engine cycle may be mentioned briefly here. The first concerns operating the engine at mixtures richer than stoichiometric. This gives considerable increase in the air specific impulse which is particularly beneficial above Mach 15. This mode of operation corresponds to a "mass-addition" engine. However, the optimum amount of mass addition depends on the trajectory, since compromise between the higher fuel flow and the high acceleration is necessary to minimize the total fuel consumption.

The second modification involves adding oxygen at higher Mach numbers. Here it is proposed that the oxygen should be collected from the atmosphere by a liquefaction process, using the excess cooling capacity which the liquid hydrogen fuel possesses at low flight speeds. It is logical to use the intake boundary layer for this purpose, since this has already been cooled, and may be removed anyway to prevent flow separation.

TRAJECTORY CALCULATIONS

A final aspect of the theoretical background which is relevant here, is the flight path or trajectory. References 4 and 5 give some typical results showing 8–12 percent of the takeoff weight can be useful payload in orbit. In general, it is found that the engine is matched to the airframe in that both require operation in the familiar "flight corridor." For the airframe the limits are lift and heating, while for the engine the limits are high pressure and low thrust. Two schools of thought exist at the present time concerning which side of the flight corridor is preferable for a space booster vehicle. One school maintains that the heating rate should be minimized by flying at the upper limit, while the other is in favor of a "quick and hot" flight. The arguments will not be given here, but it is sufficient to conclude that test conditions for experimental investigation of hypersonic ramjets should be limited to the flight corridor.

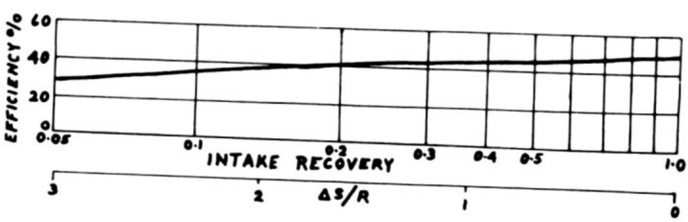

Fig. 5.

GEOMETRY CONSIDERATIONS

To obtain the probable geometry of a ramjet-powered satellite booster, we must first consider the conclusions of the trajectory studies mentioned above. This work showed that the vehicle should have a low value of the ratio of initial weight to the ramjet intake capture area. The lift/drag ratio or aerodynamic efficiency of the vehicle is relatively unimportant, and can be sacrificed in favor of the other design parameters.

Heat-transfer considerations lead to similar requirements, i.e., low weight, low wing loading and low lift/drag ratio operation. In addition, the wings must be blunted and highly swept back in the familiar hypersonic manner. The first conclusion that we draw is that practically all the air entering the projected frontal area must pass through the engine. Thus the whole lower surface of the aircraft is engine.

Provided the upper surface does not present any projected area to the flight direction, its detailed shape will be unimportant, since it operates in a near vacuum.

The low density of liquid hydrogen fuel is perfectly compatible with this geometry, since there is adequate volume available for storage. The high enthalpy per unit mass of hydrogen is also particularly advantageous for an acceleration mission.

The general shape of the lower wing surface thus consists of the engine intake ramp, the combustion chamber, and a two-dimensional exhaust nozzle.

The intake ramp angle can be varied by varying the angle of attack. It is important, however, that the shock wave falls on the lip of the cowl. The shock-wave position is also dependent on the flight Mach number. The cowl must therefore be variable by means of hydraulic jacks or some similar device. The length of the cowl, which contains the fuel-mixing section and combustion chamber, is not known precisely. However, present indications are that 3 ft may be a reasonable estimate for a booster of 15 ft maximum thickness.

The fact that the intake and nozzle are external permits them to radiate heat away, and thus simplifies the cooling problem.

GENERAL EXPERIMENTAL APPROACH

The initial approach to the experimental investigation is to separate the various components, as far as possible, and study them individually. Thus the intake, nozzle, and combustion chamber can be studied independently, using experimental environments corresponding to the range of operating conditions of each one. In many cases, exact simulation of all the variables is not necessary, and this simplifies the test facility requirements.

The main fields of study are:

Materials and Heat Transfer
Intake Research
Fuel Injection

Supersonic Combustion Investigation

Propelling Nozzle Flow Recombination

These various aspects are discussed in detail below; the test facilities, which we constructed for the work, will be briefly described first.

TEST FACILITIES

PEBBLE BED AIR HEATER

The problem of building suitable test facilities for studying hypersonic ramjets stems largely from the high enthalpies encountered. Although it is not always necessary to simulate the temperature, one of the first problems to be investigated was supersonic combustion, and since this depends on spontaneous thermal ignition of the fuel, high temperatures were essential. It was further felt that a running time of many seconds was necessary for combustion research, which precluded the use of millisecond-duration facilities. These considerations led to the construction of a pebble bed air heater in the summer of 1959.

The layout of the pebble bed facility is shown in Fig. 6. The bed itself is a 4-ft-long, 8-in.-diameter cylinder, filled with $\frac{3}{8}$-in. diameter zirconia spheres. These are insulated from the 2-ft-diameter pressure vessel by graded insulating bricks, and supported on a perforated stainless steel disc. The pebbles are heated from above by a propane flame, burning in air preheated to 900°C. The air preheater is a stainless steel heat exchanger, fired with a natural gas burner. The pebbles are heated in this way to a temperature between 2000°C and 2500°C over a period of four hours. When the "front" of hot pebbles reaches the bottom of the bed, the exhaust valve is closed, and the bed evacuated to purge the combustion products. Air at pressures up to 100 psia is then admitted to the bottom of the bed, where it is heated by the pebbles to a temperature of about

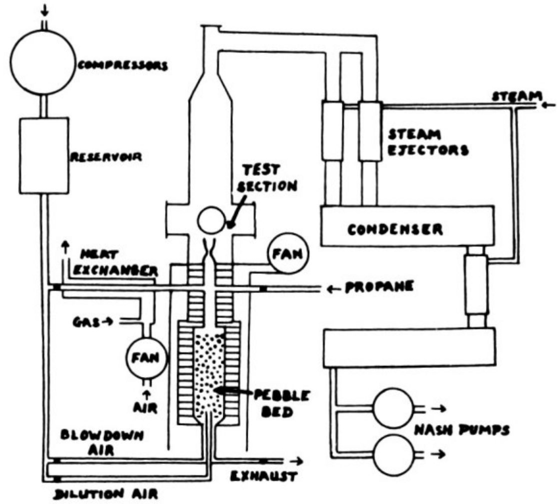

Fig. 6.

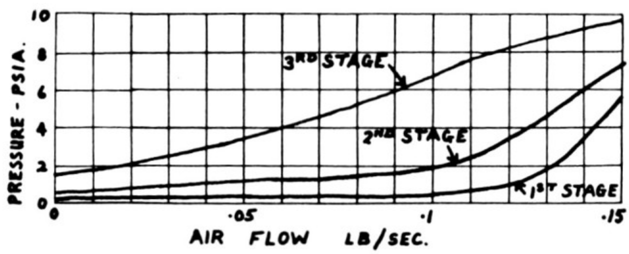

Fig. 7.

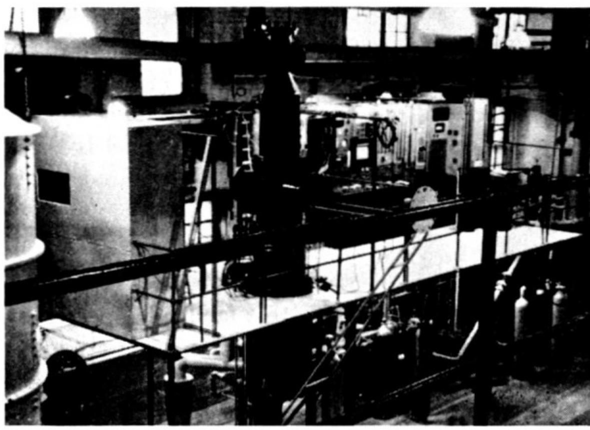

Fig. 7*a*.

2000°C, before passing through the test section. The mass flow through the bed is limited to 0.3 lb/sec, as above this flow the pebbles begin to lift.

The manifold surrounding the test section can be evacuated by a pumping system, consisting of two stages of ejectors, followed by a stage of mechanical pumping with Hytor water sealed pumps. This method effectively silences the steam ejectors, without incurring the cost of silencers. The vacuum versus mass flow characteristics of the pumping system is shown in Fig. 7.

The 2000°C hot-air supply is at almost constant temperature for several minutes, and is maintained at a high level for about 20 min.

When higher temperatures are required, an afterburner is added, which introduces hydrogen and oxygen to the hot air. This raises the temperature to about 2500°C before it enters the test section. This technique, of course, adds a few percent of steam into the air.

A general view of the facility is shown in Fig. 7*a*.

SHOCK TUNNELS

In order to test hypersonic ramjet intakes, it is necessary to simulate the flight Mach number, the Reynolds number and the temperature gradients. A shock-tunnel facility can simulate all these variables over a wide range, as shown in Fig. 8, but the running time is only a few milliseconds. This leads to the requirement for special high frequency instrumentation. However, suitable instrumentation techniques have been evolved in other laboratories, and we could benefit from their experience. We decided to build a shock tunnel in the summer of 1961, and followed conventional practice (Fig. 9). The driver is 10-ft long, 4-in. in diameter and rated at 4,000 psi. The interim driven tube is 25-ft long, 1.5 in. in diameter, and rated at 1,500 psi. In operation a mixture of oxygen, hydrogen, and helium is exploded in the driver, and following fracture of the diaphragm, dry air in the driven tube is compressed by the shock wave to the required stagnation conditions. The air is then expanded through a hypersonic nozzle into a dump tank preevacuated to a few microns of mercury. The test section is situated at the exit of the nozzle, and is 3–6 in. in diameter, depending upon the Mach number. This facility is shown in Fig. 9*a*.

For reasons which are illustrated in Fig. 9, very high pressures are often required in the driven tube. A high-pressure shock tunnel is therefore being constructed from tubes rated at 80,000 psi. This can be fired to atmosphere, or to vacuum.

FUEL-MIXING FACILITY

Two dissimilar gases are provided by this facility at pressures up to 100 psia and temperatures up to 100°C. The exhaust is evacuated by the laboratory pumping system described above. With the use of suitable nozzles, we have, in effect, a double supersonic wind tunnel, where the interaction between the two streams can be studied. The general layout is shown in Figs. 10 and 10*a*.

NONEQUILIBRIUM FLOW FACILITY

In order to study the nonequilibrium flow processes occurring in a propulsion nozzle, the recombination reaction of nitrogen dioxide is being used.

The test gas consists of a small proportion of nitrogen dioxide diluted by gases of various molecular weights. This mixture is heated to 100°C, when the N_2O_4 dissociates almost completely to NO_2. As the mixture expands through the nozzle,

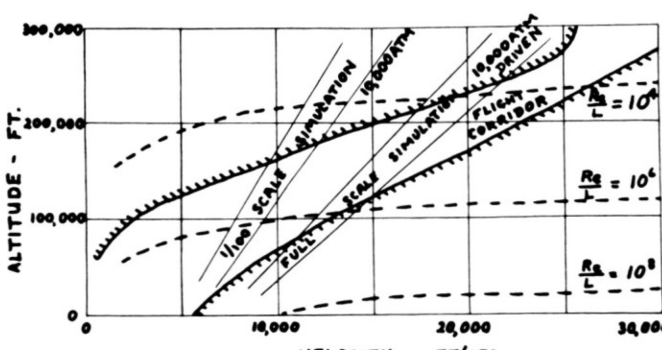

Fig. 8.

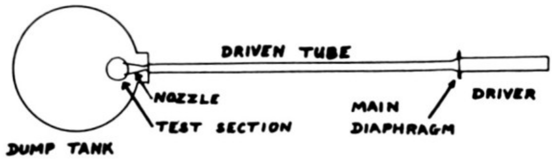

Fig. 9.

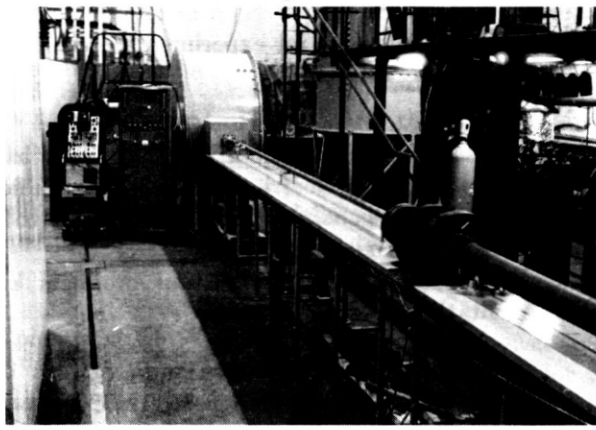

Fig. 9a.

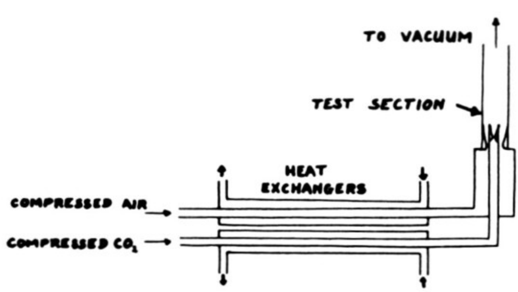

Fig. 10.

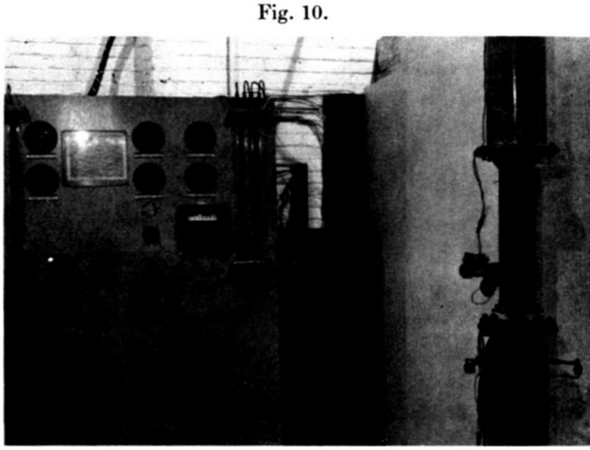

Fig. 10a.

the nitrogen dioxide tends to recombine, at a rate governed by the rate constants of the reaction. This chemical lag gives nonequilibrium flow effects, which can be monitored by analysis of the gas in the nozzle, using optical absorption techniques.

The arrangement of the apparatus is shown in Figs. 11 and 11*a*. Again the exhaust is evacuated by the laboratory pumping system.

The low temperature at which the nitrogen dioxide-tetroxide reaction takes place, makes this reaction convenient for experimental investigation.

OTHER FACILITIES

Several small facilities are also available in the laboratory as follows:

(*a*) High temperature (1800°K) electric furnace of one cubic foot capacity.

(*b*) Radio-frequency plasma generator heating nitrogen to about 4000°K, using 500 watts of power at 27 Mc/s.

(*c*) Trust-measuring facility using combustion products at 2000°K. The thrust instrumentation range is ± 20 lb.

MATERIALS AND HEAT TRANSFER

The first experimental work carried out on this ramjet research program was a study of materials and techniques suitable for use with high enthalpy air. The following series of Mach 3 nozzles were therefore constructed and tested at

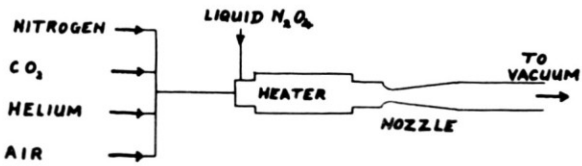

Fig. 11.

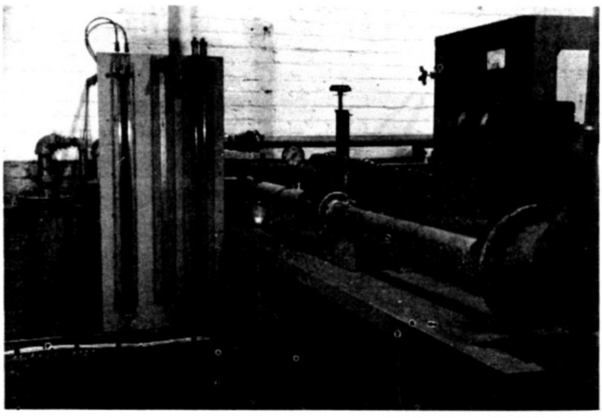

Fig. 11*a*.

standard upstream conditions of 2700°K and 75 psia, using a mixture of air and superheated steam supplied by the pebble bed air heater with the afterburner.

(*a*) ceramic nozzle of zirconia
(*b*) Haveg asbestos phenolic, ablating plastic nozzle
(*c*) alumina flame sprayed onto graphite nozzle
(*d*) uncoated graphite nozzle
(*e*) stainless-steel nozzle
(*f*) water-cooled brass nozzle

The conclusions from these tests indicated the specific suitability of the materials for use on high-temperature laboratory equipment, such as our pebble bed. They can also be interpreted in their application to hypersonic ramjets. These aspects will be considered in the following discussion.

The ceramic nozzle failed almost immediately due to cracks caused by thermal shock. Other ceramics, such as silicon carbide and boron nitride, offer more promise.

The ablating plastic nozzle changed its geometry so considerably compared to the small scale of the apparatus that it is only suitable for larger objects, where dimensional change is insignificant. Thus it may be suitable for certain portions of a satellite-boosting ramjet, where its ability to isolate the structure from the high enthalpy environment may be invaluable.

Thin coatings of alumina flame sprayed onto graphite performed very satisfactorily with little erosion.

The performance of uncoated graphite was very gratifying, as it showed negligible oxidation after several minutes of operation. The application of graphite for both laboratory apparatus and practical ramjet engines is thus very promising, especially as it allows a large proportion of the heat to be radiated away.

Uncooled stainless steel proved to be invaluable for constructing apparatus for use with high enthalpy air (although it would soon melt away if used for a flight engine). This is because the time required to heat up to the limit of structural stength, i.e., 1000°C, can be quite considerable, say one minute for ¼-in. thick material. For the thinner sections, such as must be used for probes, the time in the stream must be limited to one second, however, present-day instrumentation can easily give pressure profiles in this period.

As is well known, water-cooled materials can tolerate extremely high rates of heat transfer, and our experiments verified this fact. With fuels replacing the water, the structure of a hypersonic ramjet can be kept within reasonable limits. Calculations show that fuel flows greater than stoichiometric would be required, if all the heat were to be removed in this way. Some ablation or radiation cooling must therefore be used in addition to fuel cooling. However, determination of the optimum proportion of these various methods depends on the engine geometry, which cannot yet be specified.

A practical difficulty is encountered when liquid hydrogen is used as a coolant. This stems from the tendency for "frost" to form on the cooled surfaces, which would be particularly inconvenient if the technique suggested above of separating the oxygen from the air were attempted. It is perhaps time for the problem to be studied at the fundamental research level in order to find a palliative.

A series of experiments have also been carried out to determine the aerodynamic heating of a swept-back wedge airfoil. This investigation concerned the flat surface, rather than the leading edge, where the effect is known to be considerable. The tests were carried out at Mach 2.6, using an angle of attack of 5° with sweep varying from 0° to 35°. Theoretical and experimental results agreed closely, and showed that the sweep angle had only a negligible effect on aerodynamic heating.

INTAKE RESEARCH

Most groups investigating the hypersonic intake problem started with a geometry which consisted of a single wedge forming a double shock by the reflection of the initial wave from the cowl. This geometry is attractive due to its simplicity; it also gives axial flow after compression, thus minimizing the external drag. The theoretical behavior of this intake indicates that acceptable ramjet performance could be obtained up to about Mach 15, which is sufficient to be of immediate interest. From the University research point of view, practical results are not required immediately, and we could afford to take a more complete approach to the problem. Thus we started with a theoretical study of contoured compression ramps of specified pressure gradient and wall cooling. The theoretical work is now being confirmed experimentally in the shock tunnel facility described above.

The next step will be to study the difficulties known to be inherent in bending the ramp flow back to an axial direction without separating the boundary layer. This is largely a boundary-layer shock interaction problem, since it is the shock from the cowl which initiates the separation. Various techniques to overcome this immediately suggest themselves.

The detailed results of this work will be published in a later paper.

FUEL INJECTION

Perhaps the greatest outstanding problem of hypersonic ramjet research is the fuel injection technique. Before the combustion can be completed, the air and fuel must be mixed together thoroughly, while traveling at velocities generally above 10,000 ft/sec. It follows that the time available for mixing is very short, if the mixing length is to be kept to practical limits.

These phenomena can be investigated by injecting a gas into a wind tunnel and measuring the spread of the gas by sampling techniques, and the total pressure loss by means of probes. This experiment has been performed independently at many laboratories using apparatus similar to the McGill mixing facility described above. Although valuable data, such as the comparison of cross-stream and downstream injectors, can be obtained in this way, the actual mixing distances measured may not be directly applicable to an engine due to interaction with the combustion process. A discussion of some of the practical aspects of fuel injection is therefore included in the combustion section below.

From the theoretical calculations, we know that the height of the combustion chamber is only a few inches for a typical engine, hence the penetration of fuel into the stream does not need to be too great. We can thus inject the fuel from the wall, in a cross-stream direction, in a series of high-speed jets.

An alternative approach is to carry the fuel out into the stream in a thin biconvex fin, and inject upstream or downstream through suitable nozzles.

Further experiments in hot combustion systems are required to resolve which system is best, or indeed if either is adequate.

EXPERIMENTAL INVESTIGATION OF SUPERSONIC COMBUSTION

Many examples of chemical reaction with heat release in supersonic gas streams are already well known, perhaps the most common being recombination of dissociated combustion products in rocket nozzles. The main principle of shock-free supersonic combustion is that the chemical reaction starts further from equilibrium than the rocket nozzle case and will proceed according to the *static* temperature and pressure at any point. In general, mixing of the fuel and air will be a rate-controlling feature of the combustion. Flame holding by means of conventional recirculating baffles is not acceptable, since these could cause shocks, and ignition of the fuel must be achieved spontaneously, either by high static temperature or by the use of highly reactive fuels.

Difficulties are encountered from the following directions:

1. Confining the combustion to the required region.
2. Burning the fuel steadily to prevent shock formation.

Satisfying these requirements is not easy. However, it must be emphasized that there is no apparent theoretical reason why shock-free combustion should not be obtained.

The following series of experiments were carried out using the pebble bed air heater facility to investigate the behavior of supersonic flames.[6]

SPONTANEOUS COMBUSTION OF HYDROGEN UP TO MACH 1

A relatively simple experiment was carried out to obtain some preliminary "feel" for this problem, and also to confirm that the accepted spontaneous ignition data applies to flow systems. The apparatus used is shown in Fig. 12,

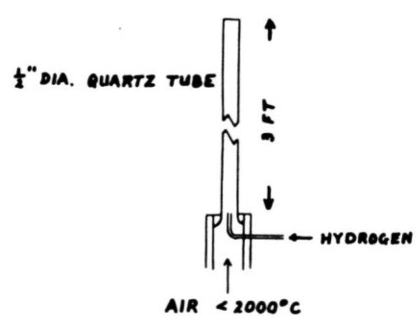

Fig. 12.

where it can be seen that hydrogen is injected downstream at the throat of a crude nozzle. Without the quartz tube, the Mach number could be increased to sonic at the injector, by applying a suitable pressure ratio to the nozzle. The results of these tests showed that the hydrogen would ignite spontaneously at all stream static temperatures above 600°C, and this figure has been verified on many subsequent experiments.

With the quartz tube in position, thermal choking occurred at some fuel flows and pressure ratios. The purpose of using the tube was to visually locate the flame, so that the conventional method of measuring the ignition delay could be employed. The usual difficulty of this method was observed: i.e. visual location of the "flame front" was not sufficiently accurate to give quantitative results over a wide range of pressure and temperature.

At temperatures close to the ignition limit, an interesting effect was observed. This consisted of an intermittent "cracking" type of combustion. The inference from this phenomenon is that the flame is completely self sustaining, and not significantly affected by heat or radical diffusion from the flame already present. Thus there is no evidence of hysteresis in the spontaneous ignition of hydrogen.

IGNITION DELAY OF HYDROGEN-AIR MIXTURES AT HIGH TEMPERATURES

The gas velocity in a supersonic combustion "chamber" is so high that the gas would pass through a 10-ft-long chamber in one millisec. Thus fuel mixing, ignition delay, and complete combustion must be carried out in less than a millisecond, if the length of the combustion chamber is to be reasonable.

Ignition delay studies of hydrogen/air mixtures have been carried out in many laboratories, e.g., Ref. 7. Some results from Ref. 7 are included in Fig. 13, and it can be seen that delay is very sensitive to temperature, although the maximum temperature used did not give delays short enough for our purposes. It was therefore decided to extend the ignition delay data for hydrogen to the higher temperatures available from the McGill test facility. As a result of the difficulty of accurately locating the flamefront, a new technique was developed. The method and results are briefly outlined here, and details are available from Ref. 8.

The technique depends on the fact that heat addition to a subsonic gas stream in a constant-area duct results in local pressure gradients dependent on the rate of heat release at each point. A 2-in.-diameter combustion tube was, therefore, instrumented with 50 static pressure taps, and from the pressure profile, the region in which heat was being released could be determined precisely. The apparatus was bolted directly onto the high temperature air facility.

Hydrogen fuel was injected radially from six orifices located on the wall at the entrance to the combustion tube.

The results are plotted in Fig. 13 for pressures down to 2 psia and temperatures up to 1400°K. This data verifies that combustion and mixing can be completed in much less than one millisecond at the high static temperatures encountered at hypersonic flight speeds.

The tendency for the delay to become almost constant at the higher temperature is interpreted as being partly due to "mixing" delay, which could not be resolved by the apparatus. Other methods have shown that the ignition delay continues to decrease rapidly with increasing temperature.[9] The results also showed that ignition delay is almost independent of equivalence ratio.

Recently a complete theoretical solution[10] to the ignition delay problem of hydrogen/air mixtures has been carried out, using chemical kinetic rate constants. The results agree with experimental data given here, and confirm that combustion can be completed in microseconds at the higher temperatures.

Since the combustion period was less than a millisecond, the possibility of carrying out supersonic combustion experiments in a shock tunnel (with a few milliseconds running time), was immediately apparent.

BOUNDARY-LAYER INJECTION OF HYDROGEN AT MACH 2.1 AND 2.6

Hydrogen was chosen as the fuel for these supersonic combustion experiments due to its probable practical utilization, its high thermal stability and its suitability as a coolant for hypersonic aircraft. It was decided that boundary-

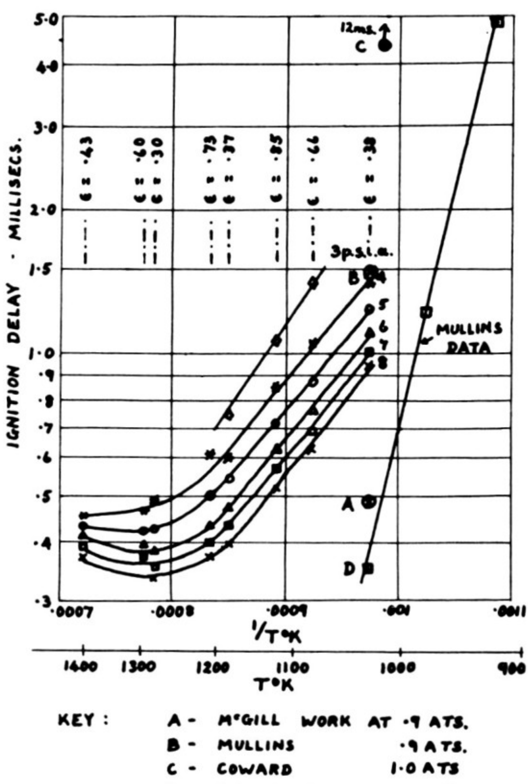

Fig. 13.

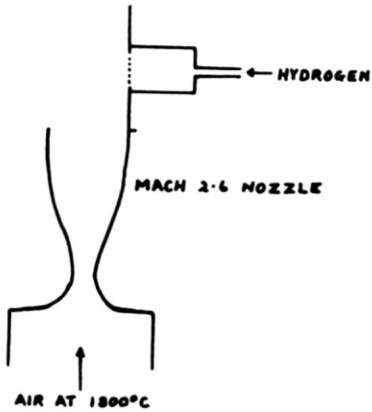

Fig. 14.

layer injection was a possible technique, as this would fulfill the cooling requirement, and did not require the presence of a physical or gaseous fuel jet in the stream. Such an obstruction would necessarily produce unwanted shocks.

The apparatus shown in Fig. 14 was therefore built of stainless steel with an interchangeable boundary-layer injector mounted on a flat plate.

It was found that the uncooled stainless steel nozzle could be used at air temperatures of 1800°C for a 45-sec test period. The air supply would then be shut down, just before the apparatus melted.

Several tests were required to find a satisfactory porous injector, since they tended to burn out. The final arrangement had five holes of 0.010-in. diameter drilled in a 1/2-in. diameter stainless steel disc, and this proved to be satisfactory, as it would operate over a wide fuel flow range, without overheating.

Figure 14*a* shows this injector operating at Mach 2.6 in an initially laminar boundary layer. Lower Mach numbers were obtained by tilting the plate to introduce an oblique shock upstream of the injector.

At low fuel flows, visual inspection of the flame zone showed that combustion started *upstream* of the fuel injector, except when the temperatures were marginal for igniting the fuel. At higher injection velocities the flame was in the turbulent mixing zone well above the nozzle.

The interpretation, which we gave to the upstream burning, assumed that the boundary layer had separated, and that the flow was recirculating in the separated region. At these Mach numbers the pressure gradient through a normal shock intersecting the boundary layer is much more than sufficient to cause such separation. It was assumed that the shock was maintained by heat addition from the fuel. It follows that this mode of fuel injection does not give shock-free supersonic combustion. (A similar phenomenon has also been observed with water injection.)

Future combustion experiments, therefore, used fuel injection techniques which avoided this problem either by downstream injection, or by injection penetrating the boundary layer.

DOWNSTREAM INJECTION OF HYDROGEN IN A SLOTTED NOZZLE AT TRANSONIC MACH NUMBERS

The Mach number at the exit of a slotted nozzle can be varied continuously from subsonic to approximately Mach 2, by varying the pressure ratio across the nozzle. Although this technique is usually used for testing aircraft models in the transonic regime, it provided a convenient method of investigating combustion phenomenon in the same regime.

A stainless steel slotted nozzle was constructed with a hydrogen injector along the axis as shown in Fig. 15. Static pressure taps were provided to indicate

Fig. 14a.

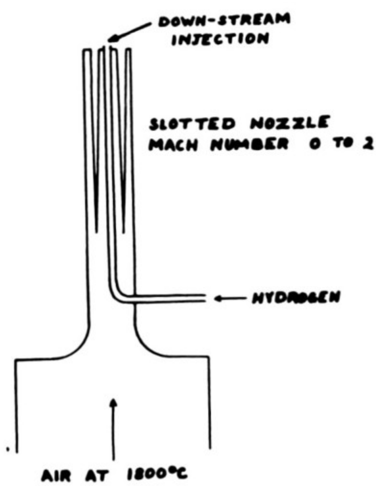

Fig. 15.

correct operation of the nozzle, and as usual, visual observation and direct color photography were used. (It should be mentioned here that the Schlieren equipment gave disappointing results with this and the previous test, probably because of the strong convection currents circulating in the hot test section.)

The test technique consisted of setting the nozzle upstream pressure at the required level (e.g., 20 psia), then varying the downstream pressure to obtain the required range of Mach number. There was no noticeable change in the flame behavior as the Mach number varied through sonic, although small luminous lines, which were attributed to weak shock waves, could be seen in the flame at higher Mach numbers (Fig. 15*a*).

The ratio of the length of the flame to the diameter of the jet was about 12. This ratio could be compared with the ratio predicted by the theory of supersonic combustion controlled by a turbulent mixing process. The theory of this important type of supersonic combustion is presented by Dr. Libby in Ref. 11.

In this particular experiment, a boundary layer had already been established on the injector tube, and this, together with the low hydrogen velocity and high hydrogen injection temperature, contributed to the short flame which was achieved. It will also be noted that the combustion was at approximately constant pressure rather than constant area.

For stoichiometric hydrogen fuel at moderate temperatures, the heat release is sufficient to choke the flow in a constant area tube up to about Mach 6. Thus, provided the air were sufficiently hot to ignite the fuel spontaneously, confined shock-free combustion could be obtained more easily at higher Mach numbers. To obtain Mach 6 air at about 600°C static temperature poses a severe problem and alternative approaches were proposed for subsequent experiments.

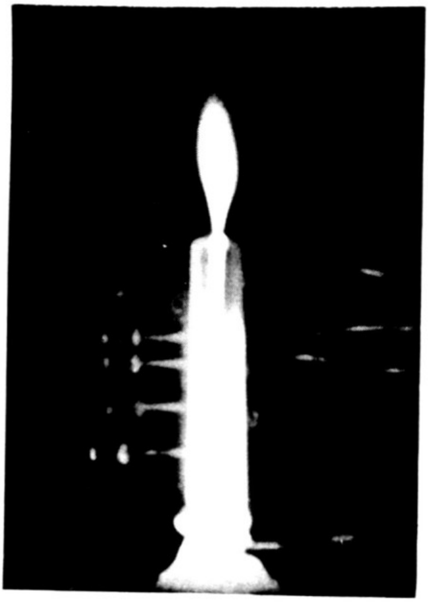

Fig. 15*a*.

The first technique is to increase the Mach number as high as possible, and compensate for the lower static temperature by the use of a more reactive fuel. The second method is to use a small heat release in the combustion zone. The heat release can be controlled both by changing the fuel, and by preburning some of the oxygen in the air before it enters the test section.

DOWNSTREAM AND CROSS-STREAM INJECTION OF TRIETHYL ALUMINUM (T.E.A.) AT $M = 3.8$

Following the above discussion, the Mach number of the tests was increased to 3.8 and T.E.A. was obtained from the Ethyl Corporation. T.E.A. is a highly reactive fuel, which will ignite easily with air at room temperature. A series of experiments was carried out using different injectors, in order to evolve a satisfactory fuel injection technique.

The customary precautions had to be taken with this fuel, such as flushing the lines with nitrogen before and after each run. It was found that injector holes less than .040 in. in diameter tended to plug, in spite of extreme care.

The first injector consisted of a simple tube, injecting fuel cross-stream at the center of the $M = 3.8$ nozzle with 2000°C total temperature. The fuel burned behind the injector, showing that the turbulent wake in this region was playing a significant part in the behavior. The experiment showed that there was no difficulty in burning this fuel at this Mach number, and it gave us practical experience in handling techniques.

A more sophisticated injector was then made with a thin (⅜-in. chord) biconvex profile. This was drilled to give downstream injection from the trailing edge. Although satisfactory combustion could be obtained on occasion, plugging of the holes frequently occurred, with thermal decomposition of the fuel to aluminum. A simple cross-stream injector with a biconvex airfoil section proved to be the most practical arrangement.

The partial success of these preliminary experiments with T.E.A. encouraged us to build an apparatus to determine the detailed flow field in the combustion region.

CONSTANT AREA COMBUSTION OF TRIMETHYL ALUMINUM (T.M.A.) AT $M = 3.8$

T.M.A. is rather more reactive than T.E.A. and was therefore selected for this experiment. The apparatus is shown in Fig. 16. This consists of an $M = 3.8$ axisymmetric stainless steel nozzle followed by a stainless steel fuel injection section and combustion tube. The fuel was injected cross-stream through three biconvex stub jets, each about ¼ in. long, thus penetrating the boundary layer. The fuel injector was water cooled to prevent thermal decomposition of the fuel and hence plugging of the jets. The constant area combustion tube was connected directly to the injector and was fitted with static pressure taps every half inch.

Several tests were carried out with this apparatus at a variety of pressures and temperatures. Provided the air stream was hot enough (above 1400°C) no difficulty was encountered in igniting the fuel. The heat release zone in the combustion tube was to be identified by the change in the wall static pressure. The results of the test were disappointing as the static pressure holes choked almost immediately with aluminum oxide powder, and unequivocal conclusions could not be reached. There were, however, indications that a strong shock formed at the fuel injection station, and this has been attributed to the use of a liquid fuel.

The experimental difficulties plus the small practical significance of T.M.A. and T.E.A. led to renewed efforts to use hydrogen in these experiments.

CONSTANT-AREA COMBUSTION OF HYDROGEN AT MACH 3.8 USING CROSS-STREAM INJECTION

The apparatus described in the previous experiment was also used for this test. The afterburner was added to the pebble bed to increase the temperatures as far as possible, and up to half the oxygen in the air was removed by the afterburner hydrogen. A further addition to the apparatus consisted of three uncooled pitot probes which could be inserted rapidly into the combustion tube, and thus indicate the pitot pressures.

The results of these tests indicated that supersonic combustion of hydrogen was obtained in the wake of the fuel jets.

Fig. 16.

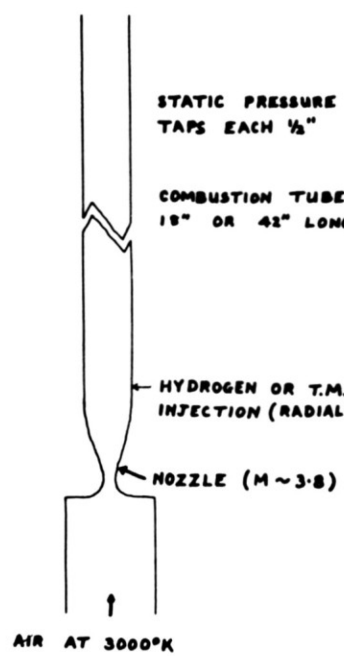

Fig. 16a.

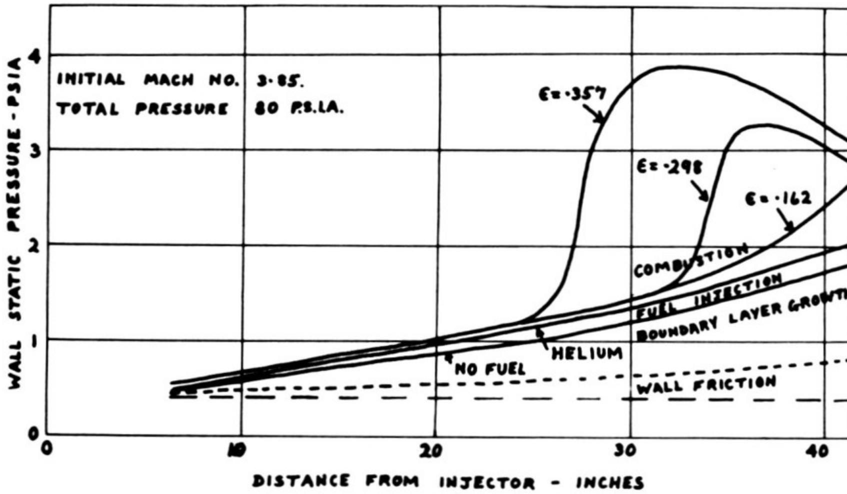

Fig. 17.

A further interesting and significant effect was observed, when the temperature of the air was just below the ignition limit of the hydrogen. The introduction of one of the pitot probes would cause ignition and subsequent combustion, in the wake of the jet, of the hydrogen.

This suggests that small cooled tubes could be used as wake generating flame stabilizers in a supersonic combustion ramjet. It is possible that they could be helpful if hydrogen fuel were to be used below a flight Mach number of 10.

A modified fuel injector was used for a second series of tests with this apparatus. This consisted of four choked jets injecting radially. Fig. 16*a* shows this chamber operating at a stagnation temperature of 3000°K.

Typical results are plotted in Fig. 17, where it can be seen that some supersonic heat release was obtained at the lower fuel flows, and thermal choking of the duct occurred at higher flows. Following thermal choking, the flow reaccelerated to sonic at the exit of the tube. The position of the thermal choking was determined by a combination of mixing, ignition delay and boundary layer effects. It is likely that ignition delay was not a major factor, however, since the location of the thermal choking did not vary significantly with a 200°C variation in air temperature.

The causes of pressure rise in the tube are analyzed on Fig. 17 where it can be seen that diffusion of the air due to a thickening boundary layer was an important factor, with smaller contributions from friction and mixing. Details of this experiment are available in Ref. 12.

A further point of interest was observed on one modification of the experiment in which a Vycor glass tube was used in place of the stainless steel tube. The main flame was seen at the point of maximum heat release shown on Fig. 17, but a thin flame estimated at ¼ in. in diameter could be seen in the tube extending down to the fuel injector. This was attributed to a supersonic turbulent mixing flame of the type proposed in Ref. 11.

DISCUSSION OF SUPERSONIC COMBUSTION EXPERIMENTS

From the foregoing it can be seen that several types of supersonic combustion can be identified:

1. Boundary-layer separation stabilized combustion, obtained with the effusion injector system.
2. Shock-stabilized combustion or "detonation wave" as observed by Gross[13] and Nichols.[14]
3. Turbulent mixing or "wake" stabilized combustion.
4. "Thermal choking" system, probably stabilized by boundary-layer interaction.

The detonation-wave engine has been treated elsewhere and will not be discussed here, although the comments concerning mixing will also apply.

For a practical engine combustion system the mixing process will generally be critical and this suggests many small jets should be used. Alternatively some means such as a wake should be provided to increase the local mixing and temperature.

Since the performance of an engine with a "thermal choking" combustion would probably be satisfactory from Mach 7 to Mach 10, this mode of operation could be used instead of a wake producer until the stream temperatures were high enough to give satisfactory turbulent mixing operation. It is possible that no change of geometry would be necessary during the transition.

PROPELLING NOZZLE FLOW RECOMBINATION

As discussed above, at the high combustion temperature associated with the hypersonic ramjet, the working fluid is partially dissociated, and the specific impulse is greatly affected by the degree of recombination achieved in the propelling nozzle.

At the time of our decision to investigate this problem, very little was known of the phenomenon either from an experimental or theoretical point of view.

The theoretical aspects of flow recombination in nozzles have now been published by several researchers, e.g., Ref. 15, and the solution is relatively straightforward with the aid of a digital computer. Essentially this involves the simultaneous solution of the gasdynamic equations of the continuity of mass, momentum, and energy, together with the rate reactions for the several reactions possible in the system.

These solutions are rigorous, but their application to real flow systems depends on the accuracy of the rate constants used in the chemical reactions. In many cases the constants are known to be inaccurate, although there are some indications that they may be satisfactory for the special case of a hydrogen/air system.[10] Nevertheless it is clear that experimental work is still required before we can predict the performance of the propelling nozzle at any given conditions.

Our approach to the experimental work was based on an investigation by Dr. Wegner,[16] in which the recombination of nitrogen dioxide in the presence of a single diluent, nitrogen, was studied.

We have extended the investigation to the case of diluents of different molecular weights, and to mixtures of diluents. Some results of this work are plotted in Fig. 18 where it can be seen that the flow departed from equilibrium part way down the nozzle, when nitrogen was used as a diluent. With helium diluent the results indicate that the flow was frozen, while with carbon dioxide the flow remained in equilibrium. Details of the experiment are available in Ref. 17.

In order to investigate the flow of actual combustion products through a nozzle, they must be heated to hypersonic enthalpies. This is achieved most easily by shock-heating the products in a shock tube and allowing them to expand through a nozzle. Optical methods may then be used to determine the concentration of atoms and radicals in the nozzle, thus indicating deviations from equilibrium flow.

FUTURE RESEARCH

From the foregoing it will be seen that the current status of hypersonic ramjets is such that we should be able to look forward to the construction of a development engine within the next year. The performance and uses of the engine

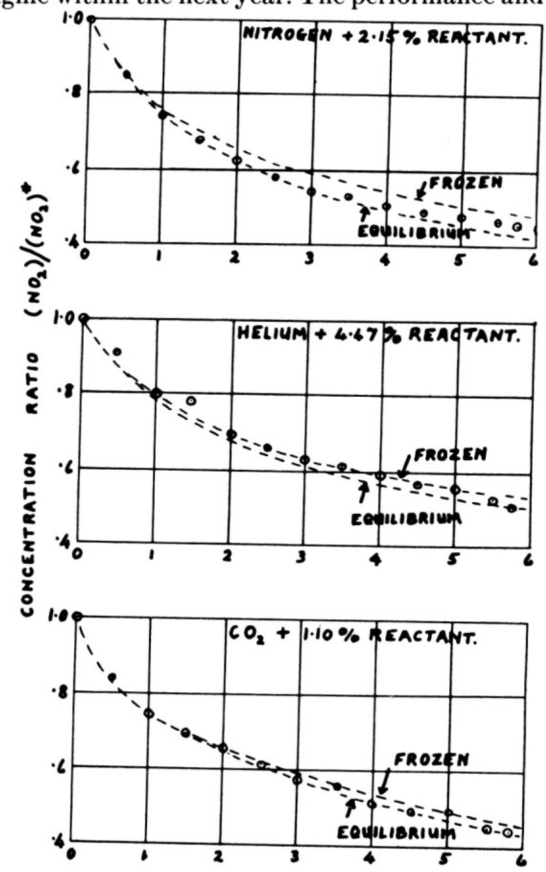

Fig. 18.

appear sufficiently promising to justify the work involved in making it a practical unit, although the work will probably take several years.

It is thus time to look at the outstanding problems, and the facilities which will be necessary to complete the program.

The biggest unsolved problem is the performance of the combustion chamber with its fuel-injection system. If, as has been indicated, a shock tunnel can be used to give realistic enthalpies for testing this system, little difficulty should be encountered in achieving an optimum design for any particular application.

The next problem is controlling the separation of the boundary layer in the inlet. This again can be tested with a shock-tunnel facility, alternatively gun tunnels, high-enthalpy tunnels or helium tunnels can be used.

For hydrogen/air mixtures the recombination problem in the propulsion nozzle is nearing solution; however, for certain applications hydrocarbon or other fuels may be used. The rate constants for the various reactions encountered with hydrocarbon fuels are not known with sufficient precision at present. These data can probably be obtained by chemical kineticists when required although the experiments should be planned now so that no delay will occur at a later date.

Looking next at the engine development stage, the main requirement is for air at the correct pressure and enthalpy. The shock tunnel is again the most promising facility, supported by high enthalpy techniques such as were used in the development of hypersonic reentry nose cones.

An interesting characteristic of a two-dimensional supersonic combustion ramjet is that if we take Mach lines from the rear edge running forward, then within the area between them, the flow can be unaffected by the fact that the engine has a finite width. These Mach lines are generally separated by about 20°, so that in order to test, for example, a 10-ft-long engine we would only need a 3½-ft-wide test section. Shock tunnels with 20-ft-diameter test sections are already in existence, so we see that the testing requirements for the engine can be quite modest.

Two alternatives are available for the next stage of development. These are either a continuous-arc heated or magnetohydrodynamic-accelerated wind tunnel giving correct simulation of temperature and pressure, or a free-flight technique. The MHD-heated facility would be extremely expensive—say £10 million—but would have many other uses, such as airframe testing. The free-flight technique is much less expensive in capital outlay, however, the cost is more difficult to estimate as the number of tests depends on the difficulties encountered. It is expected that a free-flight test vehicle would be rocket boosted to the required speed.

The final stage of development of ramjet powered satellite boosters would benefit greatly from the experience gained on rocket programs which have already been carried out. The winged orbital vehicle would thus follow a similar development program until it became a standard vehicle for ferrying goods into orbit.

CONCLUSIONS

1. Theoretical studies of hypersonic ramjets have shown that they are potentially very economical power plants for satellite boosters and long-range cruise aircraft.
2. Experimental techniques for investigating the performance of the various component parts have been evolved by the Hypersonic Propulsion Research Group in the Mechanical Engineering Department of McGill University.
3. A "zirconia" pebble bed has been used for studying materials and heat transfer. This work showed the value of graphite, liquid-cooling, and ablation techniques for flight use. It was also found that uncooled stainless steel could be used for sufficiently long periods for laboratory use.
4. The pebble bed has also been used for supersonic combustion experiments with hydrogen and pyrophoric fuels. The feasibility of supersonic combustion has been demonstrated, and four types of supersonic combustion identified. Suitable fuel injection techniques are suggested.
5. A shock tunnel has been used for studying the performance of the initial ramp of a hypersonic ramjet intake. Accurate simulation of the enthalpy is not required for intake testing provided the Mach number, Reynolds number and temperature gradients are reproduced.
6. The problem of nonequilibrium flow in propulsion nozzles has been studied by means of the nitrogen dioxide-tetroxide reaction. This permits the fundamental chemical/aerodynamic behavior to be investigated at relative low temperatures.
7. A double wind tunnel has been built to investigate the mixing of two supersonic streams.
8. The shock tunnel provides a convenient method of obtaining high enthalpy air for intake, combustion and nozzle research, and is therefore likely to become a basic facility for hypersonic ramjet development. Arc heated and magnetohydrodynamic accelerated air tunnels and free-flight testing techniques provide the necessary facilities for completing the development of a hypersonic air-breathing satellite booster.
9. The present status of hypersonic ramjets is very encouraging. In the few years of research which have been completed, many doubts have been answered in the affirmative, and there is every reason to believe that development of a practical engine could commence almost immediately.

ACKNOWLEDGMENT

The results reported here include the work of many research students whose contribution to the program is sincerely appreciated.

The support of fellow staff members is also acknowledged, in particular, Dean D. L. Mordell and Mr. S. Molder have made considerable contributions.

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Discussor: Prof. H. Wittenberg, Technological University of Delft

(*a*) In your paper you mentioned four types of supersonic combustion. I should like to know what types of combustion will be most suitable for the hypersonic ramjet. Is that possible to indicate in a general way at the moment?

(*b*) My second question is about the speed sensitivity of the hypersonic ramjet. Will it be possible to design a ramjet of the type you described, for a large Mach number range as necessary for the launching of satellites without considerable change of geometry of the complete engine?

Author's reply to discussion:

Of the four types of supersonic combustion described, it would appear that those systems which introduce adverse pressure gradients in the combustion region tend to cause shocks. This leads me to suggest that a constant-pressure combustion chamber is most likely to be successful, although it is not the most efficient from the thermodynamic viewpoint.

With reference to the speed sensitivity of the engine, the most sensitive component here is the intake. From Mach 12 to above Mach 25 the geometry change necessary is very small, while it is not excessive at the lower speeds. These considerations are discussed in Intakes for Hypersonic Ramjets by S. Molder, Mechanical Engineering Research Laboratories, Report Number 62–6, McGill University.

Discussor: R. Legendre, ONERA

Distingue les statoréacteurs hypersoniques a combustions subsonique de ceux ou la combustion est supersonique. Les premiers sont apparemment hors de l'objet de la conférence, mais, d'une part, les problemès qu'ils posent intéressent les seconds et, d'autre part ils sont seuls été éprouvés en vol en France. La combustion supersonique n'est étudiée qu'en laboratoire.

Le nombre de Mach ne caractérise pas un statoréacteur. Si l'accélération doit être raisonnable il faut que l'altitude soit limitée supérieurement et l'échauffement est sévère à nombre de Mach élevé, même si la combustion est supersonique. Le materiau protectoire ne doit pas seulement être refractaire—il doit être isolant en raison de la durée de vol notable. Un nombre de Mach de 15 n'est pas prochainement accessible.

La pénétration du combustible, sans chocs prohibitif, sa diffusion, sa combustion, sa mise en équilibre pendant la détante exigent une tres grande longueur de foyer et de fuyère.

L'ONERA ne vise pendant les prochaines années que l'execution d'expérience en vol a un nombre de Mach s'élevant de 5 jusqu'à 6 ou 7. L'étude de la combustion supersonique sera poursuivie en laboratoire à la température d'arrèt la plus grande possible.

(*Author did not reply.*)