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A HOT GAS GENERATOR FOR LARGE SCALE SUPERSONIC COMBUSTOR TESTING

by A. J. Metzler and E. A. Lezberg
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TECHNICAL PAPER proposed for presentation at Fourth Propulsion Joint Specialist Conference sponsored by the American Institute of Aeronautics and Astronautics Cleveland, Ohio, June 10-14, 1968

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Abstract

Theoretical calculations indicate a N_2H_4-N_2O_4 rocket operating at an O/F of 3.0 can simulate the temperature and pressure requirements for direct-connect testing of large-scale Scramjet combustors at conditions simulating altitude flight at Mach 8. Combustor inlet gases would contain 21 percent oxygen with nitrogen and water vapor as inert components. Preliminary rocket performance tests indicated C^* efficiency of 96 percent and acceptable temperature and mass flow profiles. Limited data support the feasibility of the gas generator combustor test concept.

Introduction

Ground test facility requirements for airbreathing propulsion testing become increasingly severe as flight Mach number increases into the hypersonic speed range. Pressure, temperature, and mass flow requirements rise rapidly, and, separately or collectively, tax or exceed the supply capabilities of most existing or planned facilities.

Although ceramic storage heaters have been designed to provide high air flow rates at temperatures to 2500° K and pressures exceeding 2000 psia (1.38×10^7 N/m^2) for test periods of 60 to 100 seconds, such units are not yet built. Shock tubes or other pulse systems can duplicate temperatures and pressures for tests at simulated Mach 10 to 12 flight, but these systems are limited to extremely short test time. Arc heaters are capable of continuous testing but are prohibitively expensive for high flow rates. Direct-connect testing of supersonic combustors places less of a burden on air supply requirements than does free-jet testing of an engine since the pressure supply need not reflect the losses associated with the inlet diffusion process. However, stream total temperature must still be duplicated so that the flow capabilities of present facilities for testing supersonic combustors are severely limited and only small-scale hardware may be tested. Such scaled combustor studies do not adequately reflect the injection-mixing-combustion interactions which would exist in full-scale hardware, so that the results of such tests are not directly applicable to the design of combustors for hypersonic vehicles.

This paper is concerned with a new approach to generating hot, high-pressure gas mixtures to simulate air for large-scale supersonic combustor testing. Equilibrium calculations indicate that the exhaust gas of a hydrazine-nitrogen tetroxide rocket operating oxidant-rich can simulate combustor inlet requirements for Scramjet engines up to a Mach number of 8. Although this system involves some compromise of ideal combustor test conditions, it does possess the inherent advantages of economy, simplified facility requirements, and extended test time for large-scale combustors. The feasibility of such a system is supported by the results of limited preliminary rocket tests.

Ground Test Requirements for Flight Simulation

Equilibrium free-stream stagnation conditions necessary for flight duplication are shown in Fig. 1 for flight Mach numbers ranging from 5 to 9 and altitudes ranging from 60,000 to 160,000 feet (18.3 to 48.8 km). Typical cruise and boost trajectories are superimposed on the figure. Thus, free-jet engine testing at Mach 8 for the trajectories indicated requires supply pressures ranging from 1000 to 4000 psia (6.89×10^6 to 2.76×10^7 N/m^2) and temperatures of 2600 to 2800° K.

Large losses in stream total pressure result from the nonisentropic diffusion process of hypersonic inlets. At flight Mach numbers of 6 and 8, total pressure losses of 65 and 80 percent, respectively, would occur in a system having a kinetic energy efficiency of 95 percent and $\gamma = 1.4$. Therefore, the inlet to the combustor of a Scramjet engine would see total pressures of only 20 to 35 percent of the flight stagnation pressure. Facility requirements for test and development of combustors for hypersonic engines are, thus, much less severe than would be required for inlet or complete engine testing. Combustor testing for simulated Mach 8 flight requires supply pressures of only 200 to 800 psia (1.37 to 5.5×10^6 N/m^2) instead of pressures five times greater as required for free-jet tests over the altitude range of the trajectories in Fig. 1.

Ideally combustor test facilities should reproduce completely the combustor inlet flight environment since matching temperature, pressure, and gas composition are required to reproduce chemical reaction rates; matching Mach number, Reynolds number, and inlet profiles are important to reproduce viscid-inviscid interaction.

Simulation Attainable by a Rocket Gas Generator

A rocket gas generator supplying a combustor inlet in a direct-connect mode can fulfill many of the requirements for supersonic combustor testing. Although such a system would compromise the matching of inlet gas composition, gases containing 21 mole percent of oxygen at levels of total temperature and pressure suitable for supersonic combustor testing can be supplied within the capability of present rocket technology. Vitiation effects on ignition kinetics may need to be determined in separate clean-air tests, but at static temperatures greater than 1200° K these effects are expected to be minor. At conditions of lower static temperature vitiation effects may be ignored since such conditions would probably require a positive ignition source for combustor operation.

To best simulate the combustor flight environment, two propellant systems containing only nitrogen, hydrogen, and oxygen are considered. These systems are hydrogen-oxygen with nitrogen as diluent, and hydrazine-nitrogen tetroxide with air as diluent. Dilution and O/F can be adjusted to provide inlet gases containing 21 percent oxygen with nitrogen and water vapor as inert gases; gas temperature is variable to simulate a flight Mach number range of 6 to 8. Figure 2 shows the range of theoretical, combustor inlet total temperatures which may be obtained with the hydrogen-oxygen-nitrogen system at a total pressure of 300 psia (2.06×10^6 N/m^2).

Increased nitrogen dilution lowers the temperature and decreases combustor inlet water vapor concentration. The oxygen concentration of the inlet gas stream is maintained constant at 21 mole percent. Inlet temperatures were calculated for the case of gaseous reactants with gaseous diluent and for the case of liquid reactants with gaseous nitrogen diluent. These calculations utilized the thermo-chemical program of Ref. 1 and assumed equilibrium expansion to a pressure of 0.5 atmosphere at the combustor inlet. An all gaseous propellant system using reactants at 298° K will increase the reaction temperature 160° K over that for the cryogenic reactants with gaseous diluent, but the safety and storage considerations would be more complex. Substitution of Argon for some of the nitrogen has been suggested to better match γ and molecular weight of vitiated combustor inlet gases to that of air.

The storable propellant system hydrazine-nitrogen tetroxide is better suited for Mach 8 flight simulation. Theoretical performance calculations assuming 298° K reactants at an O/F of 3.0 and pressure of 300 psia (2.06×10^6 N/m^2) indicate a total temperature of 2680° K and an approximate combustor inlet gas composition of 21 percent oxygen, 40 percent nitrogen, and 39 percent water vapor. The gas composition is relatively insensitive to chamber pressure or exhaust nozzle expansion ratio in the range of interest. For the same gas composition, the stagnation temperature is about 215° K higher than that of the cryogenic hydrogen-oxygen system with gaseous nitrogen dilution. Theoretical exhaust gas compositions and properties are tabulated in table I for the hydrazine-nitrogen tetroxide system at 300 psia (2.06×10^6 N/m^2), O/F = 3.0, and an assumed expansion pressure ratio of 0.0286 for equilibrium, frozen, or kinetic expansion processes. The kinetic calculations assumed equilibrium expansion to the nozzle throat and utilized the calculation method described by Moretti in Ref. 2 from the throat to the exit. The set of reaction equations which was used does not include reactions of the nitrogen oxides so that nitric oxide which is present at the equilibrium starting condition is included with nitrogen and oxygen. The nitric oxide concentration at the combustor entrance would be expected to approach that for a frozen expansion process.

The nozzle exit conditions calculated for the kinetic expansion of table I are compared in table II to the isentropic expansion of ideal air at the same exit pressure and Mach number. The simulated combustor entrance conditions are representative of Mach 8 flight at 115,000 feet (35,000 m) altitude. If a combustor were tested in a rocket gas generator facility at conditions of inlet pressure and Mach number as found in flight, inlet-gas static temperature, uncorrected for heat transfer or combustion inefficiency, would be about 140° K higher than for the corresponding flight condition. To test at conditions matching both flight inlet temperature and pressure would require a higher test combustor inlet Mach number and an excessively high combustor inlet velocity. The lower total temperature of the rocket propellant gases is the result of dissociation together with a higher C_p associated with the water vapor content.

Simulation at flight Mach numbers less than 8 may be attained by dilution of the gas generator exhaust with air without changing the oxygen concentration of the exhaust gas. Exhaust gas temperatures and water vapor concentrations are shown in Fig. 3 as functions of air dilution ratio. For simulation of Mach 6 flight, an inlet temperature of 1650° K would be required. This could be attained by dilution to an air-propellant ratio of 1.35 and would result in a water vapor concentration of the inlet gases of 18 percent.

Water cooled gas generator hardware will permit run times limited only by the propellant storage capacity. Chamber pressures to 600 psia (4.1×10^7 N/m^2) and mass flow rates of 200 pps (91 kg/sec), sufficient to supply combustors approaching full scale, are feasible with some extension of present rocket technology. A combustor test facility being developed at the Lewis Research Center will permit testing combustors 18-inch (45-cm) diameter at a total pressure of 300 psia (2.07×10^6 N/m^2) and a flow rate of 100 pps (45.5 kg/sec). A 2 minute run time with a propellant cost of $24 per second is estimated for maximum flow rate tests.

Preliminary Rocket Gas Generator Tests

Large Scale Injector Performance

To be suitable for combustor testing, a rocket gas generator must operate oxidant-rich with a high level of efficiency, since relatively low combustion inefficiencies may result in concentrations of nitrogen oxides in the combustor inlet sufficient to appreciably alter the ignition kinetics of the system.(3) Stable burning is required so that the combustor inlet flow will be free of large amplitude fluctuations. Finally, temperature, composition, and mass flow profiles at the combustor inlet should be flat or predictable. Injectors which will satisfy these specifications are unique in requirement and design range, since injectors designed for storable propulsion systems generally utilize the more stable burning Aerozine 50 (50-50 blend of hydrazine-unsymmetrical dimethylhydrazine) or substituted hydrazine compounds at near stoichiometric O/F. Hydrazine may not be indiscriminately substituted in such injectors without the probability of major stability problems which would be accentuated at the off-design O/F required for gas generator application. Injectors specifically designed to the requirements of gas generator use would therefore appear to be required.

Exploratory gas generator performance tests were conducted in an altitude facility with rocket hardware at a 6000 lb (26,700 N) nominal thrust level. The 10.8-inch (27.4-cm) diameter chambers had an L* = 28 inches, and the nozzle expansion ratio was 1.3. Most of the data were obtained at chamber pressures near 100 psia (6.89×10^5 N/m^2).

The injectors tested were of two basic designs, either fuel-oxidant-fuel triplet or intersecting fan like-on-like doublet patterns, and are shown in Fig. 4. The triplet design had previously demonstrated stable and efficient performance with the Aerozine 50 - nitrogen tetroxide propellant system. Some injectors were designed for an O/F near 2 and were operated off design for these tests, and others were designed for O/F = 3.0. The doublet pattern was designed for an O/F of 3.0.

The results of these tests indicated that with hydrazine at O/F = 3, the triplet pattern injector was unstable. Both low frequency (chugging) and high frequency (screeching) instability were encountered. A single test with a water cooled baffle demonstrated successful damping of the high frequency transverse oscillations. The like-on-like fan pattern injector performed stably at the design chamber pressure of 100 psia (6.89×10^5 N/m^2) but went into screech at 125 psia (8.62×10^5 N/m^2).

The C* efficiency as determined from measurements of flow rate and chamber pressure was 96 percent for stable burning tests at 98 psia (6.76×10^5 N/m^2) at the design O/F of 3. The exhaust gas temperature profile in a plane 1/2 to 3/4-inch (1.3 to 1.9-cm) downstream of the nozzle exit was obtained from a traverse of this position with a cooled gas pyrometer(4) which measured the stream pitot pressure and the temperature of a convectively cooled aspirated gas sample. Gas sample mass flow may be computed from the pitot pressure and sample temperature which were measured immediately upstream of a choked-flow orifice. Exhaust gas total temperature was calculated from the pyrometer indication and gas transport properties computed for an assumed equilibrium free-stream composition. The calculated temperatures are plotted in Fig. 5(a). Since the exhaust gas composition was assumed for the calculation, the temperatures plotted should be considered as relative values only. A rough indication of the accuracy of these temperatures may be obtained by comparison with a single point gas temperature measured with a bare wire tungsten-tungsten rhenium thermocouple. The thermocouple was inserted in the stream for 0.75 second and reached an equilibrium temperature before burnout. The measurement indicated a temperature about 230° K higher than the cooled gas pyrometer, but the measurement may have been influenced by the oxidation of the junction.

The relative mass flow rate distribution across the stream in the measurement plane is shown in Fig. 5(b) and was deduced from measurements of gas sample flow rate. Gas sample mass flow rate may be expressed as

$$\dot{m} = k \frac{P}{\sqrt{T}}$$

where k is a function of probe inlet and orifice geometry. The capture area ratio of the sample tube was assumed to be constant over the traverse so that the stream mass flow was directly related to $\dot{m}$. Since the exhaust nozzle was short with a divergence angle of 15° and a ratio of throat radius of curvature to diameter equal to 2.0, radial uniformity of mass flow is not to be expected.

Small Scale Injector Performance

A series of small scale injector performance tests were initiated to test proposed injector patterns for stability and C* efficiency. Results of these tests are reported in Ref. 5, so only the major findings will be reported here. The injectors tested had only 3 to 5 thrust elements and were run in heat sink chambers 2 inches (5.1 cm) in diameter over a range of pressures and L*. The basic patterns were fuel-oxidant-fuel and oxidant-fuel-oxidant triplets, like-on-like intersecting sheets, and like-on-like parallel sheets.

Performance and stability levels agreed qualitatively with the results obtained with the large injectors. Triplet patterns were unstable over most of the operational range exhibiting chug at low injector pressure drop and screech at high pressure drop. The occurrence of instability was independent of chamber length. An intersecting sheet pattern was stable over a limited flow range, but a parallel sheet design operated smoothly over the entire range of practical test conditions although chugging was experienced at low injector pressure drops. Injector C* efficiencies were in the range of 90 to 95 percent.

Concluding Remarks

Theoretical calculations indicate a hydrazine-nitrogen tetroxide rocket operating at an O/F of 3.0 can simulate the temperature and pressure requirements for direct-connect testing of large-scale Scramjet combustors at conditions simulating altitude flight at Mach 8. Combustor inlet gases would contain 21 percent oxygen with nitrogen and water vapor as inert components. Limited test data indicated the feasibility of the proposal. Although stability problems were encountered with some injectors, like-on-like injector patterns show promise of minimizing or eliminating high frequency instability. Stable operation with a C* efficiency of 96 percent was obtained with a 10.75-inch diameter injector at a chamber pressure of 100 psia (6.89×10^5 N/m^2). Measured radial exhaust gas temperature and mass flow distributions were adequate to meet combustor testing requirements. It is expected that similar performance at higher pressures can be attained. Gas generator operation at 300 psia (2.07×10^6 N/m^2) and 100 pps (45.4 kg/sec) would be adequate to test near full size combustors at conditions simulating flight from Mach 6 to 8.

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2. Moretti, L., "A New Technique For The Numerical Analysis of Non-Equilibrium Flows," TR-412, DDC No. AD-466922, Mar. 1964, General Applied Science Labs, Inc., Westbury, N.Y.

3. Snyder, A. D., Robertson, J., Zanders, D. L., and Skinner, G. B., "Shock Tube Studies of Fuel-Air Ignition Characteristics," (AFAPL-TR-65-93, DDC No. AD-470239), Aug. 1965, Monsanto Research Corp., Dayton, Ohio.

4. Krause, L. N., Johnson, R. C., and Glawe, G. E., "A Cooled-Gas Pyrometer For Use in High-Temperature Gas Streams," TN 4383, 1958, National Advisory Committee for Aeronautics, Cleveland, Ohio.

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	Expansion process		
	Equilibrium	Frozen	Kinetic
T_s - ^oK	1429	1312	1375
Molecular weight	24.97	24.75	24.91
Specific ht. ratio - γ	1.263	1.275	1.268
Mach number	2.833	2.864	2.870
Mol. fract. O_2	0.2119	0.1938	0.2103
Mol. fract. N_2	0.3979	0.3839	0.3974
Mol. fract. H_2O	0.3895	0.3722	0.3864
Mol. fract. OH	0.0001	0.0221	0.0040
Mol. fract. NO	0.0007	0.0216	------

TABLE I. - THEORETICAL GAS PROPERTIES: HYDRAZINE-NITROGEN TETROXIDE SYSTEM, O/F = 3.0, $T_T = 2680^o$ K, P_T = 300 psia, p_s = 8.58 psia

	Storable rocket gas generator	Isentropic expansion - Ideal air
T_T - ^oK	2680	2790
T_s - ^oK	1375	1237
P_T - psia	300	287
p_s - psia	8.6	8.6
Mach number	2.87	2.87
Velocity - fps	7170	6450

TABLE II. - COMPARISON OF COMBUSTOR INLET CONDITIONS (MACH 8 FLIGHT AT 115,000 FEET ALTITUDE)

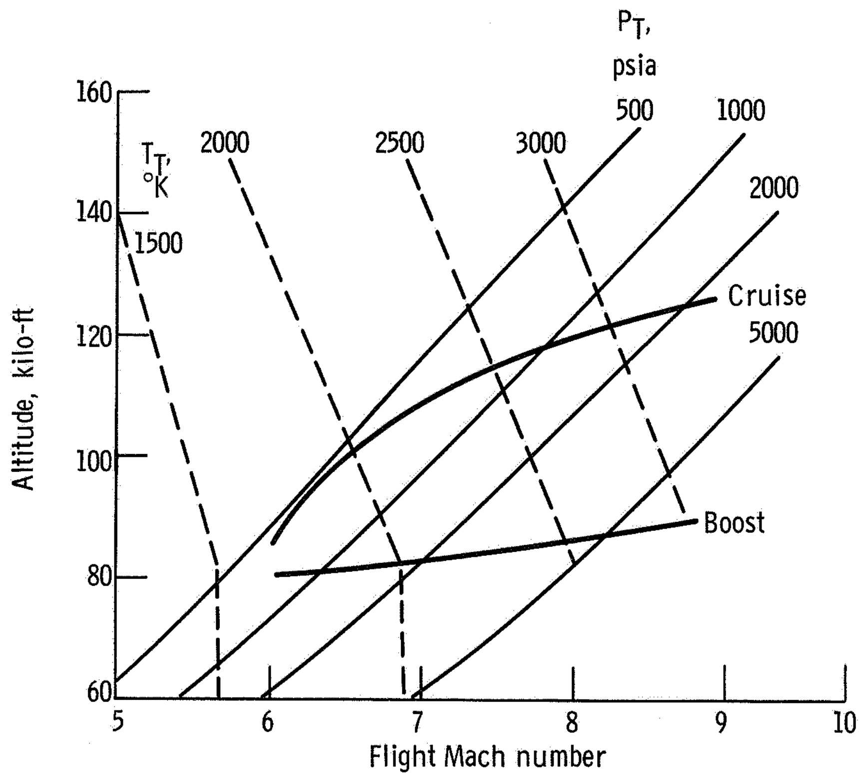

Figure 1. - Equilibrium stagnation flight environment.

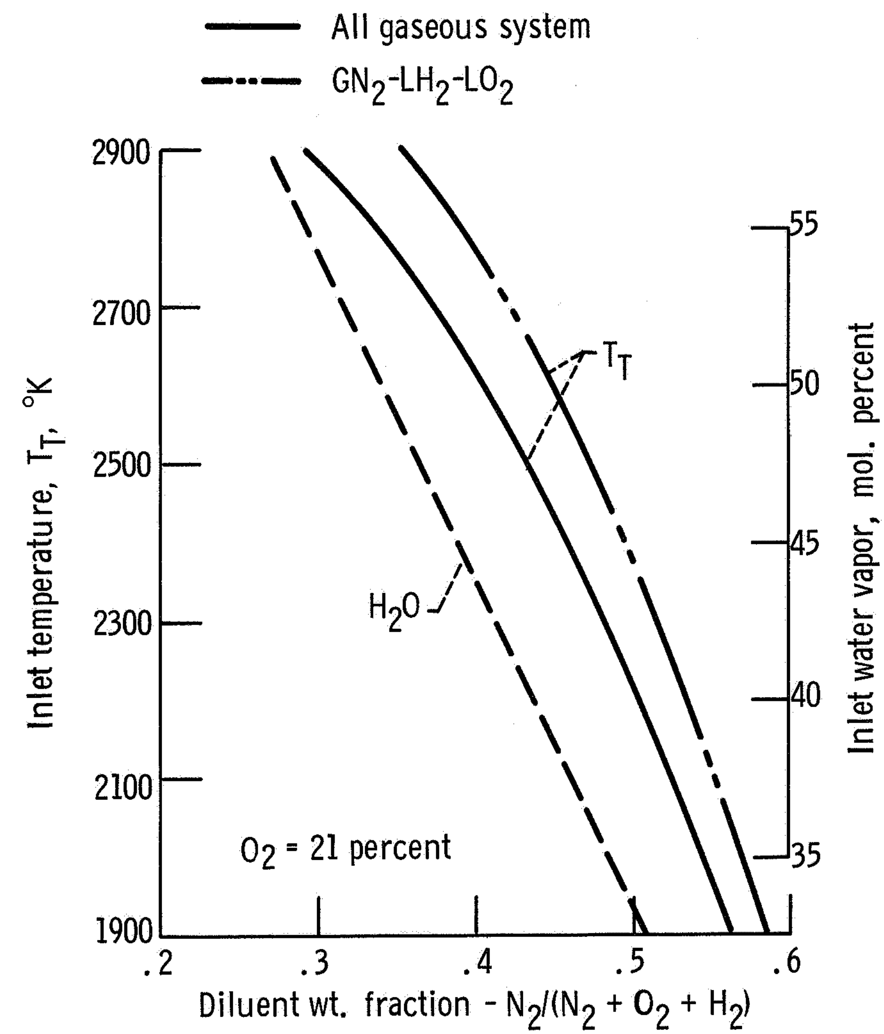

Figure 2. - Theoretical inlet temperature and composition: Nitrogen dilution of the hydrogen-oxygen system.

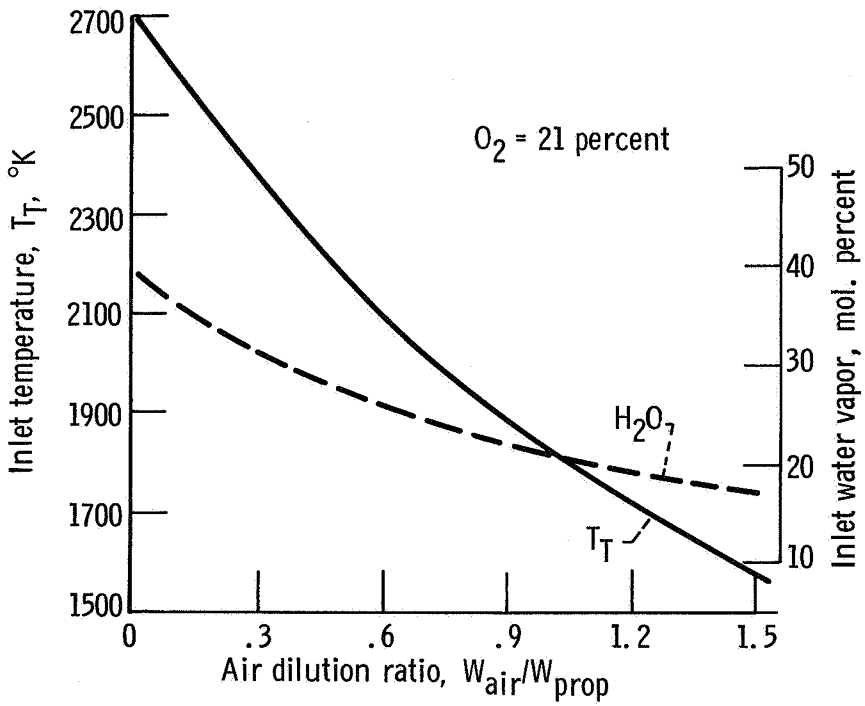

Figure 3. - Theoretical inlet temperature and composition. Air dilution of hydrazine-nitrogen tetroxide system.

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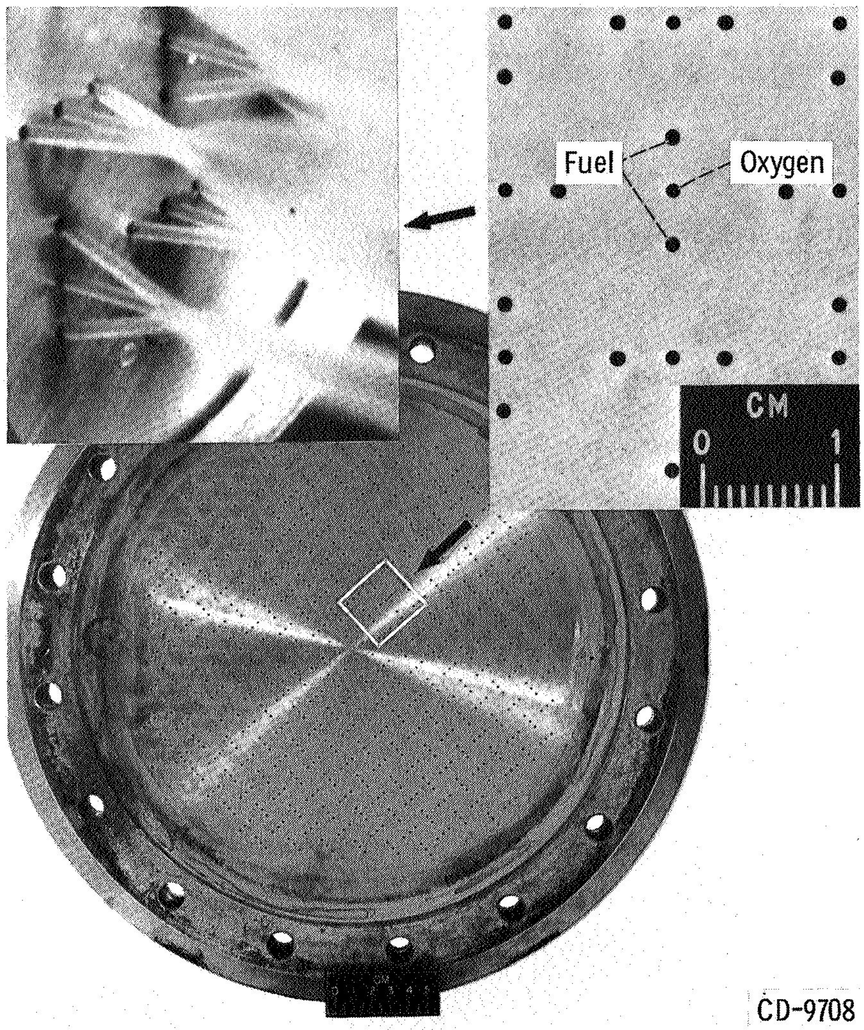

(a) Basic triplet pattern.

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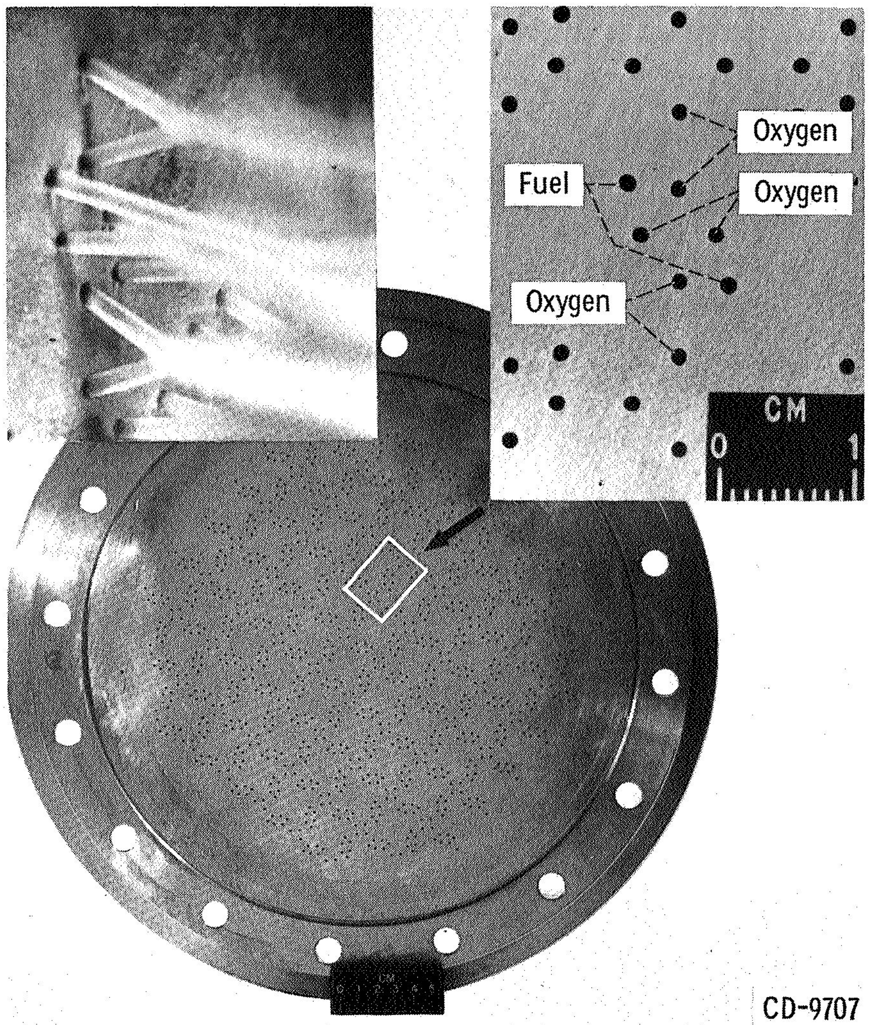
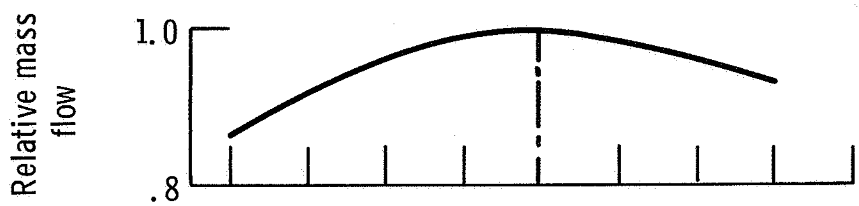
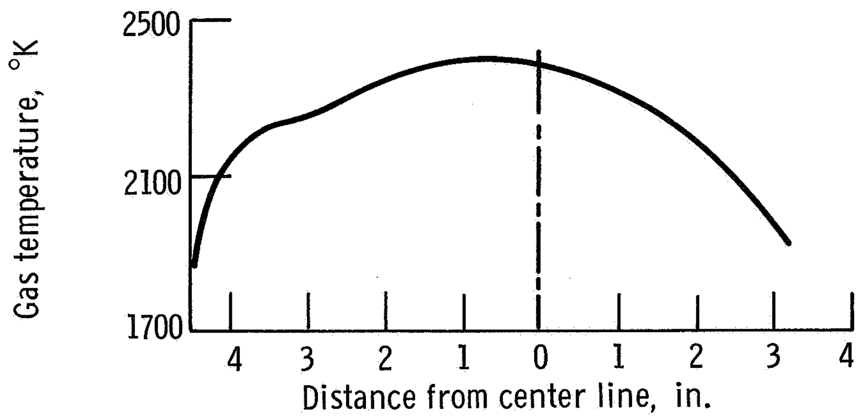


(b) Like-on-like intersecting fan.

Figure 4. - Hydrazine-nitrogen tetroxide injector.



(b) Mass flow distribution.



(a) Temperature distribution.

Figure 5. - Measured nozzle exit flow characteristics: hydrazine-nitrogen tetroxide propellants, O/F = 3.