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EXPERIMENTAL PERFORMANCE OF A WATER-ELECTROLYSIS ROCKET

by R. James Rollbuhler

Lewis Research Center

Cleveland, Ohio

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ABSTRACT

An experimental investigation was conducted to evaluate the performance characteristics of a nominal 1-pound- (4.4-N-) thrust water-electrolysis rocket engine system. The system was tested over a range of propellant flow rates and for various run durations in a vacuum pressure environment. The test results indicated a system specific impulse of 338±17 seconds obtainable at a combustion-chamber pressure of 15 psia (10.3 N/cm^2 abs). Combustion efficiency was greater than 90 percent of theoretical, and combustion instability was not detected in any of the tests.

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SUMMARY

An experimental program was conducted to determine the performance and to investigate the operational problems of a water-electrolysis reaction-control rocket engine system. The system consisted of a water-electrolysis hydrogen-oxygen gas generator, a gas storage tank or tanks, an ignition system, and an altitude thruster with propellant gas control valving. The thruster was rated at 1 pound of thrust (4.4 N) at a combustion-chamber pressure of 15 psia (10.3 N/cm^2 abs) in a vacuum environment. All testing was done with a stoichiometric mixture (oxidizer-fuel ratio, 7.94; 11.19-wt. % fuel) of gaseous hydrogen and oxygen. A fully instrumented altitude environment test facility was used to evaluate the thruster over a range of propellant flow rates and operating test durations.

At 15 psia (10.3 N/cm^2 abs) the specific impulse for the system was 338 seconds, which is approximately 94 percent of theoretical frozen equilibrium performance. The results were accurate within ±5 percent.

Operational difficulties were encountered because of unreliable ignition and occasional flow-line detonations. Sparking ignition became inconsistent as the test environment pressure became less. Flow-line detonations were an occasional problem, occurring when a propellant line valve was actuated or during a test because of possible combustor flame feedback through the injector.

INTRODUCTION

The water-electrolysis rocket is a concept in spacecraft propulsion that combines features of both electrical and chemical propulsion into one system designed for extended operation aboard a space vehicle. The water-electrolysis rocket system generates its chemical propellants, hydrogen and oxygen gases, by electrolysis of water while in flight. Not only can the generated gases be used for propulsion applications, but also the hydrogen can be used in fuel cells and the oxygen used for breathing in a manned spacecraft. For propulsion systems, when the two gases are mixed as they are generated, the mechanical simplicity of a monopropellant rocket system is realized along with the high performance potential of the hydrogen-oxygen propellant combination. The basic components of such a water-electrolysis rocket are illustrated in figure 1. The system is described in greater detail in reference 1.

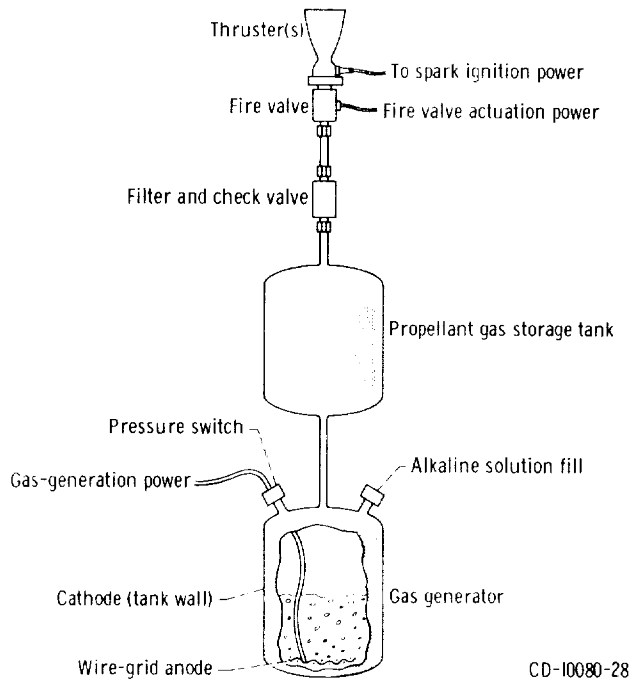

Figure 1. - Water-electrolysis reaction control system (not to scale).

Theoretically, the water-electrolysis rocket offers many advantages over more conventional control systems, some of these being high performance from nontoxic noncorrosive propellants, storage simplicity, state-of-the-art system components, long-life operation potential, and a low input power requirement.

The electrical characteristics of the water-electrolysis system have been investigated in detail in Earth environment conditions (described in ref. 2), but the actual performance capabilities of the system have not been verified in a space environment.

This investigation was initiated to determine the operating performance of such a system in terms of altitude specific impulse, combustion efficiency, ignition behavior, and other operational parameters. All the test data were obtained from a 1-pound- (4.4-N-) thrust engine operating in a vacuum pressure environment.

The results reported herein were calculated from the following experimental data: thrust, combustor-chamber pressure, propellant flow rates, propellant line pressures and temperatures, and thruster wall temperatures. Test durations ranged from 10 milliseconds to 20 seconds.

Numerous modifications were made to the water-electrolysis rocket system as originally procured to cope with ignition and detonation problems.

SYMBOLS

A_e	nozzle exit area, in.^{2}; cm^2
A_t	thruster nozzle throat cross-sectional area, in.^{2}; cm^2
C_f	thrust coefficient, dimensionless
C^*	characteristic exhaust velocity, $P_c A_t / M_t$, ft/sec; m/sec
F	thrust, lb; N
g_c	gravitational constant, 32.2 ft/sec^2; 9.81 m/sec^2
I_{sp}	specific impulse, $F/g_c M_t$, sec
M_t	total propellant mass-flow rate, slugs/sec; kg/sec
O/F	oxidizer-fuel mass-flow ratio
P_a	environmental test pressure, torr
P_c	combustion-chamber pressure, psia, N/cm^2 abs
P_e	thruster nozzle-exit pressure, psia; N/cm^2 abs
ϵ	expansion ratio; ratio of thruster nozzle-exit area to thruster nozzle throat area

Subscripts:

F	frozen equilibrium theoretical conditions
S	shifting equilibrium theoretical conditions

APPARATUS AND PROCEDURE

Test Hardware

The water-electrolysis rocket system studied included a gas generator, a gas accumulator tank, a solenoid gas control valve, a thruster, and an ignition power supply. The components (with the exception of the gas generator) and accessories are shown in figure 2. None of these components were of lightweight construction. They were de-

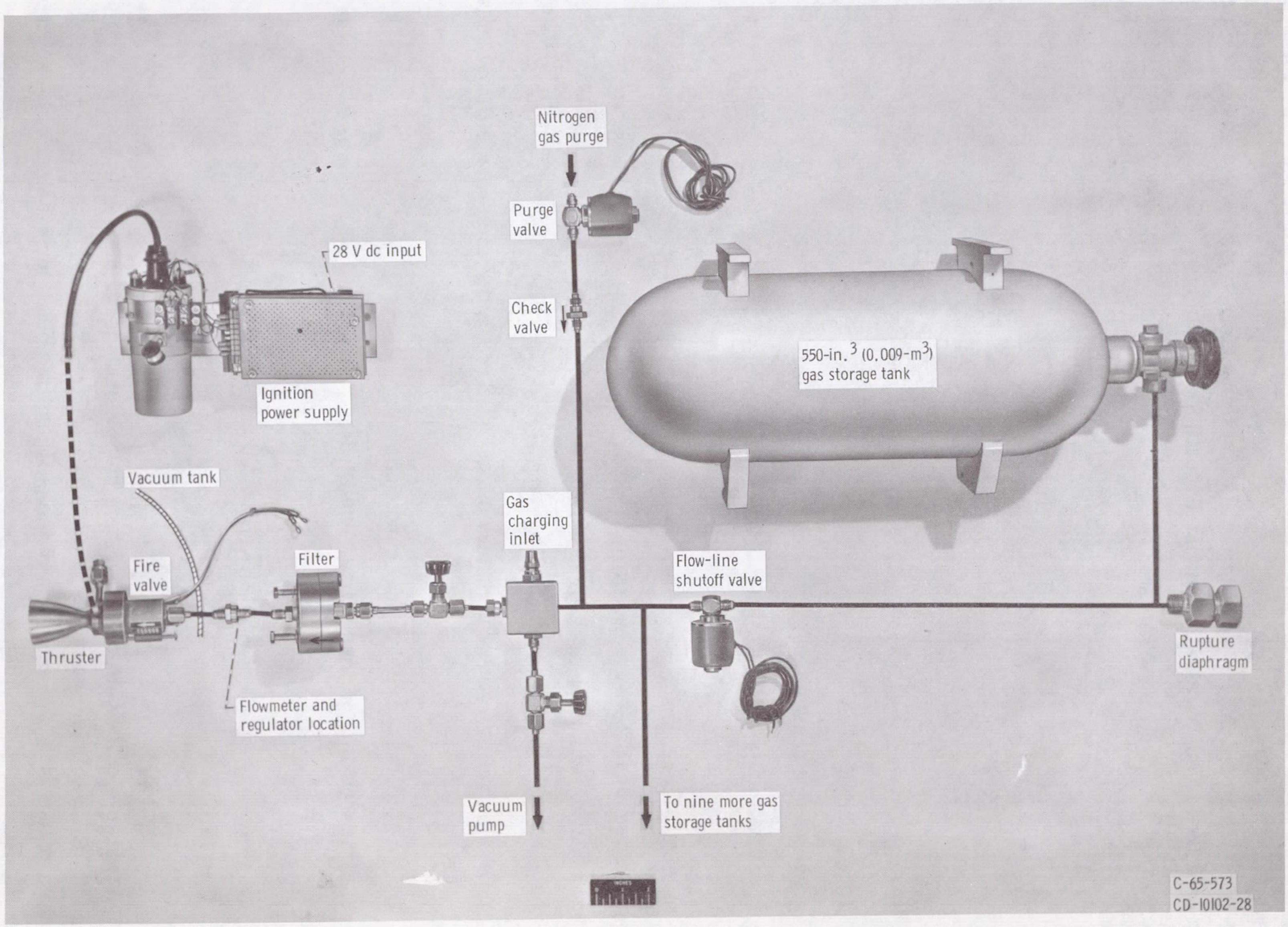

Figure 2. - Water-electrolysis rocket system.

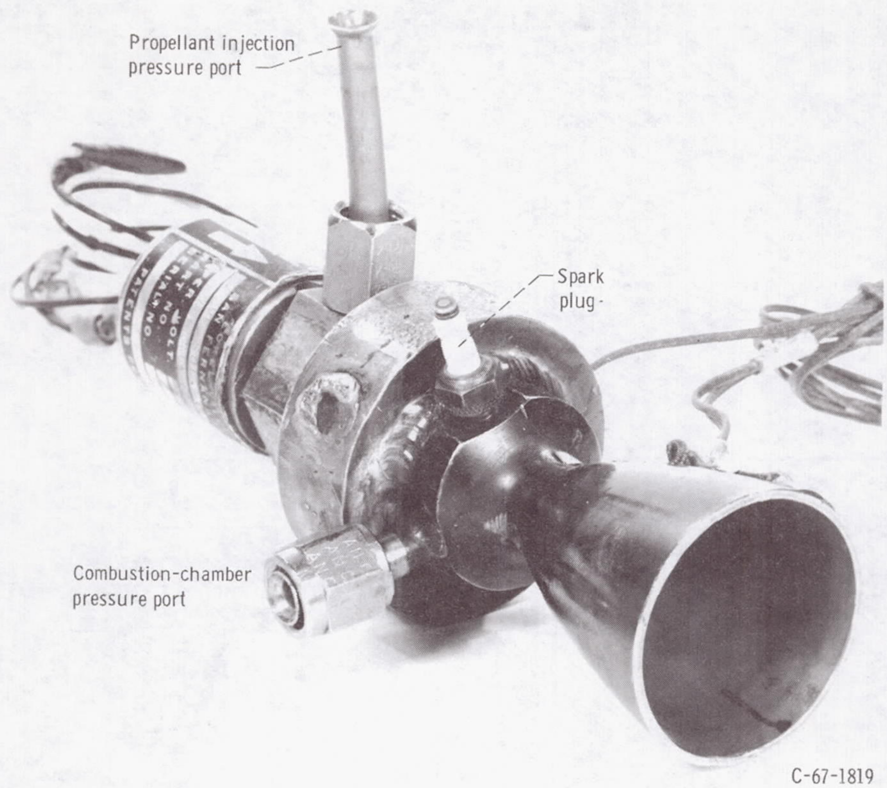

Figure 3. - Water-electrolysis rocket thruster.

signed and built for studying system performance in a laboratory facility, with the potential danger involved in handling a stoichiometric mixture of hydrogen and oxygen gases taken into account.

Thruster assembly. - The thruster and the propellant flow control valve are illustrated in figure 3. The design thrust level of the thruster assembly is 1 pound of thrust (4.4 N) at a combustion-chamber pressure of 15 psia (10.3 N/cm^2 abs) and an equivalence ratio of 1 (stoichiometric mixture ratio). The chamber and nozzle are made of nickel and designed for 20 seconds of full-thrust, continuous operation. The expansion ratio ϵ is 35.4, and the nozzle shape is an 80-percent bell. The nozzle throat is 0.263 inch (0.668 cm) in diameter. The combustion-chamber volume is 0.204 cubic inch (3.34 cm^3).

Thruster propellant injector. - The propellant injector consisted of a sintered disk or disks held in a housing mounted to the combustion chamber. This injector design was intended to provide a uniform propellant flow profile through the sintered disk into the combustion chamber. The thruster manufacturer believed that uniform propellant flow would adequately cool the sintered disk to keep it from burning and that the micropores in the disk would act as a flame arrester to prevent flame flashback from the combustor into the propellant feed system.

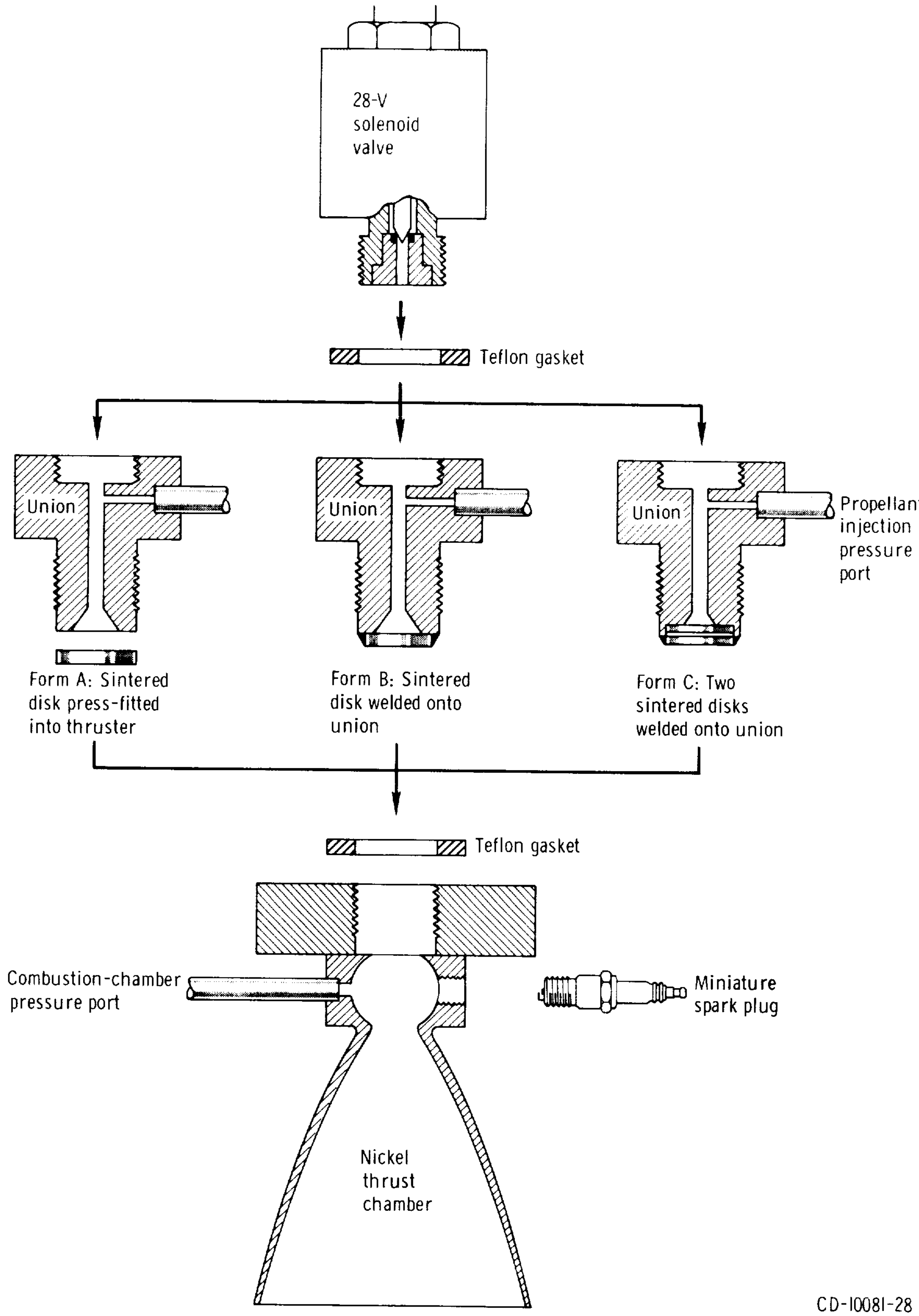

Figure 4. - Hydrogen-oxygen thruster injector modifications (not to scale).

Three injection-disk mounting variations were investigated and are shown in figure 4. As originally fabricated, the disk was press-fitted into a housing or union (form A). After an upstream flow-line detonation, the disk was welded to the injector union to obtain a leak-tight peripheral seal (form B). The final configuration (form C) consisted of two 1/16-inch- (0.16-cm-) thick disks welded in series in the injector union.

During this program, nickel, Inconel, and stainless-steel sintered materials were tried, as well as disk thicknesses varying from 1/16 to 3/16 inch (0.16 to 0.48 cm). Various disk-union sealing materials (Teflon, high-temperature rubber gaskets, and ceramic cements) were unsuccessfully tried before resorting to welding the disk to the union.

Thruster propellant control valve. - The propellant flow control valve was mounted to the thruster directly upstream of the injector. This mounting technique is illustrated in figure 4. The injection disk union, just downstream of the valve, was tapped for measuring the propellant injection pressure and temperature.

Two different valve types were used. The original valve was an in-line fast-response (15 msec) solenoid valve. The replacement was a globe valve, also with fast response, and solenoid-operated. Both valves operated on 24- to 28-volt direct-current power and were of stainless-steel construction with rubber seals and seats. Because the maximum propellant line pressure would be 150 psia (103 N/cm^2 abs), large orifice valves were acceptable. Their use resulted in a low pressure drop for the propellants flowing through them.

In spite of the nearness of the valve to the combustion chamber, heat soakback was not a problem, largely because of the heat-capacity properties of the propellants, the mass of the thruster, and the short duration of each test.

When the globe-type valve was used, a "tee" line connection was put between the injection union and the valve so a thermocouple could be installed that would touch the upstream side of the sintered injector disk.

Propellant ignition system. - The propellant gas mixture was ignited in the combustion chamber by a miniature spark plug which was recessed in the combustor wall as shown in figure 4. Two different power supply systems were used with the spark plug. Originally the spark power supply was a transistorized network for converting 28-volt direct current into 200-hertz, high-voltage power for the spark plug. Later this transistorized network was replaced by an alternating-current transformer that furnished 60-hertz, 1200-volt power to the plug.

The spark plug cable was electronically shielded, and both the power supply and the spark plug body were electrically grounded.

Test Facility

This investigation was conducted in the facility previously used for low-thrust, Earth-storable, bipropellant thruster testing (refs. 3 and 4). The liquid-bipropellant system was removed and replaced with the monopropellant hydrogen-oxygen gas system. Gas storage tanks were mounted outside the test cell. The gas flow controls and the gas generator were inside the test cell but outside the vacuum environment tank. The rocket motor was mounted on top of the thrust stand and fired horizontally (see fig. 5). The high-voltage ignition power supply was mounted outside the vacuum tank with the high-voltage cable passing into the vacuum tank through a special feedthrough. The vacuum tank seals and fittings were refurbished to attain a vacuum environment around the thrust stand of about 0.5 torr.

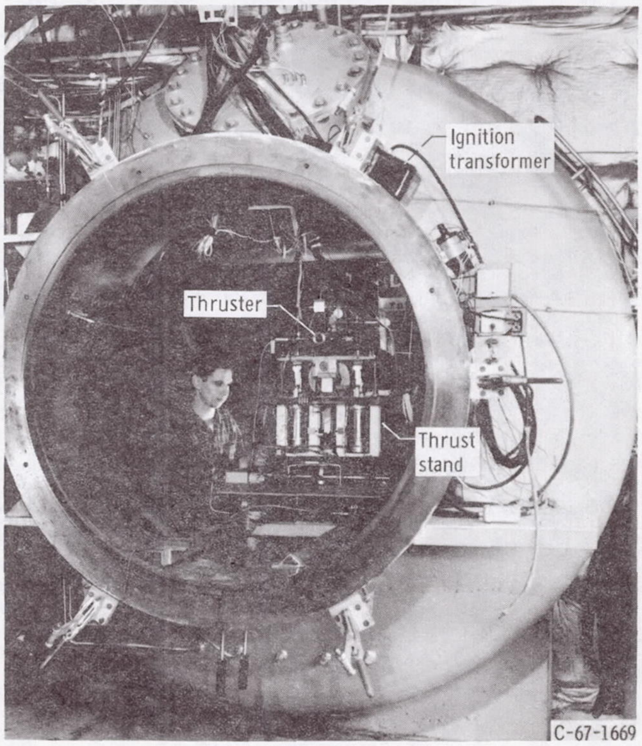

Figure 5. - Vacuum environment run tank.

Propellant gas generator. - The stoichiometric mixture of hydrogen and oxygen gas was generated for use as a propellant by the electrical disassociation of water containing an electrolyte (potassium hydroxide). The reaction at the unit cathode is

$$4H_2O(\text{liquid}) + 4e^- \rightarrow 2H_2(\text{gas}) + 4\ OH^-$$

The reaction at the anode unit is

$$4\,OH^- \rightarrow O_2(gas) + 2H_2O(liquid) + 4e^-$$

These reactions were carried out in an industrial-type oxygen-hydrogen gas generator. The unit was modified to produce gas to a pressure of 150 psia (103 N/cm^2 abs), instead of the 15 psia (10.3 N/cm^2 abs) for which it was originally built. The output gases passed through a moisture trap (with the resulting gas having an effluent dewpoint of 390° R (217 K) or less) before entering the propellant system and gas storage tanks.

Propellant flow system. - The propellant gas mixture of hydrogen and oxygen was stored in 10 tanks mounted outside the test cell. They are shown in figure 6. Each tank

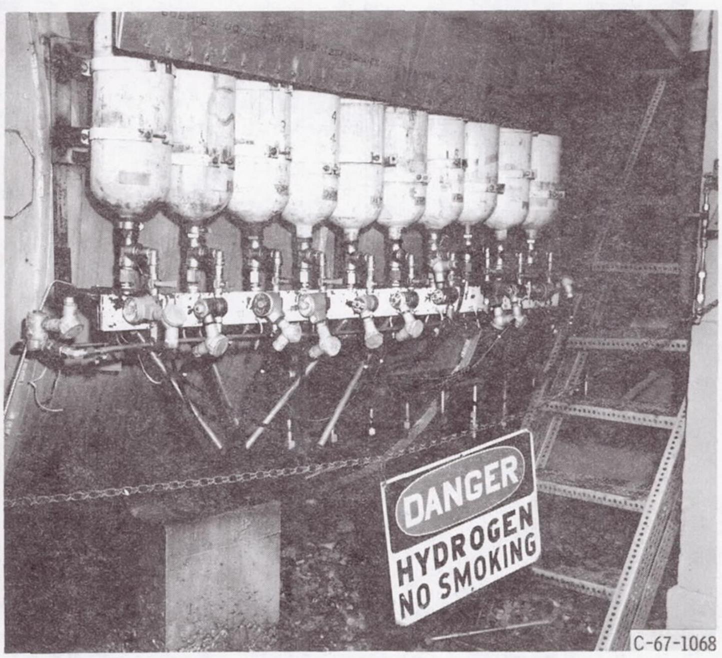

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Figure 6. - Propellant gas storage tanks.

had a volume of 1/3 cubic foot (0.009 m^3), which allowed a blowdown test of about 15 seconds per tank (150 psia (103 N/cm^2 abs) → 75 psia (52 N/cm^2 abs)).

If a hydrogen-oxygen gas mixture is ignited in a closed system, the pressure will increase to 20 to 25 times its preignition pressure; therefore, the storage tanks (and the rest of the flow system) were rated for 6000-psia (4120-N/cm^2 abs) service. Each tank was equipped with a 400-psia (275-N/cm^2 abs) rupture assembly, and hand- and remote-operated shutoff valves. Where possible, connections were welded together.

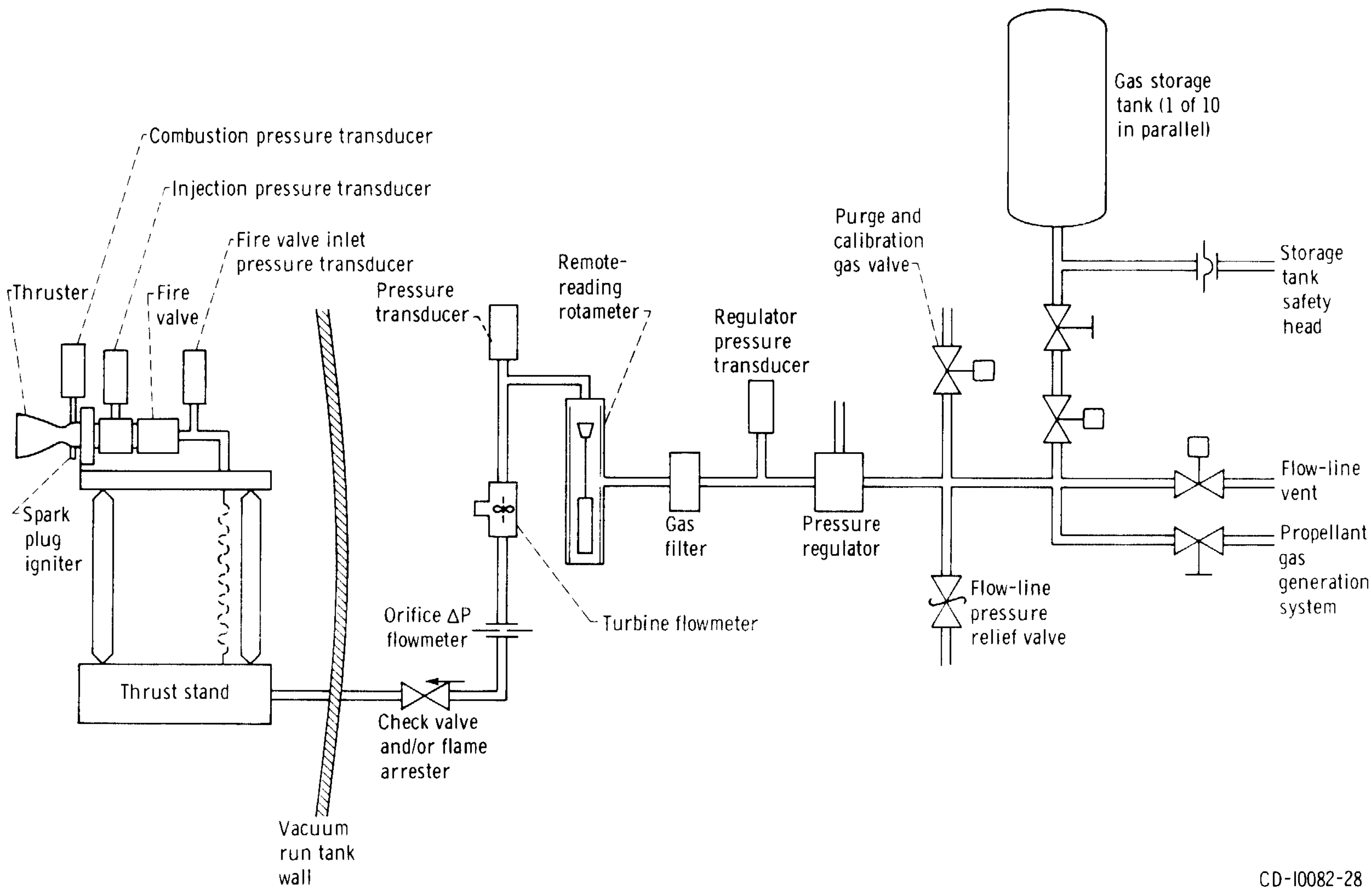


Figure 7. - Schematic of test rig.

A schematic of the propellant flow system is shown in figure 7. Immediately downstream of the storage tanks was a nitrogen gas purge inlet, which was also used to calibrate the flowmeters. The flow line included, at this location, a line vent system and a pressure relief valve (set for 250 psia (172 N/cm^2 abs)). Downstream of the purge and vent systems was a remote-regulated flow-line pressure controller used to control the propellant flow rate to the thruster. After the regulator the propellant gases passed through a gas filter (5-μm rating) and the gas flowmeters.

Just before the propellant gases entered the fire valve at the thruster, they passed through a flame arrester. Various types were tried - check valves, porous ceramic plugs, sintered disks, rolled coils of copper screening, and combinations of these types. None stopped a detonation downstream from going through them into the upstream gas.

Instrumentation

Thrust system. - The thrust stand is pictured in figure 8. It is a compound pendulum type that uses a closed-loop electronic force-generating network in a null-balance system for indicating thrust and total impulse. A detailed description of the system is presented in the appendix of reference 3.

For this program, several modifications were made in the stand which are not mentioned in reference 3. A static deadweight calibration capability was added to check the remote-reading calibration system. A dynamic electromagnetic calibration system was installed to calibrate the impulse-measuring network. Also incorporated into the system was the capability to measure and record impulse and thrust data simultaneously.

Calibration of the thrust system with deadweights indicates the steady-state thrust data error would be ±1.5 percent or less. Dynamic accuracy was very poor. Background noise and electronic interference were a problem in "cluttering" the low-level thrust and impulse signals. To minimize this noise, electronic filtering was used in the thrust system circuits. This reduced the dynamic response readout by a factor of 10.

Static-pressure measurements. - Strain-gage transducers were used to measure pressure in the propellant system. The transducers were calibrated against a master gage in the pressure environment in which they were to be used. The accuracy to which they were calibrated was ±2 percent. The combustion-chamber pressure transducer had low dynamic response characteristics because of the volume in the connecting tube between the transducer and the combustion chamber. This is analyzed in the appendix of reference 3. Other pressure measurements were made at the injector, in the flow line upstream of the control valve, at the flowmeters, and on the inlet and outlet sides of the propellant pressure regulator. These locations are shown in figure 7. Secondary data were the gas generator outlet and the test environment pressures.

Temperature measurements. - The thruster combustor wall temperatures were measured with Chromel-Alumel thermocouples welded to the wall surfaces. The output signal was referenced to a 610° R (338 K) base temperature and calibrated against a standardized millivolt power source. The wall temperature data accuracy is estimated to be $\pm 5^{\circ}$ R (± 2.8 K).

Propellant flow-line temperatures were measured with iron-constantan thermocouples referenced and calibrated in the same manner as the combustor wall thermocouples. The accuracy of the flow-line temperature data is about $\pm 1^{\circ}$. These thermocouples were located in the injector, at the flowmeters, and near the gas storage tanks.

The rear-surface temperature of the injector sintered disk was measured by a thermocouple inserted into the injector propellant inlet (shown in fig. 8).

Flow-rate measurements. - During this program as many as three different types of flowmeters were in series in the propellant flow line. These were an orifice differential pressure type, a turbine type, and a remote-reading rotameter. Prior to a given day's testing, the flowmeters in the flow line were calibrated by flowing known quantities of purge gas through them. Occasionally, a calibration check was made by using a gas-storage-tank temperature-pressure history in conjunction with perfect-gas laws. Cali-

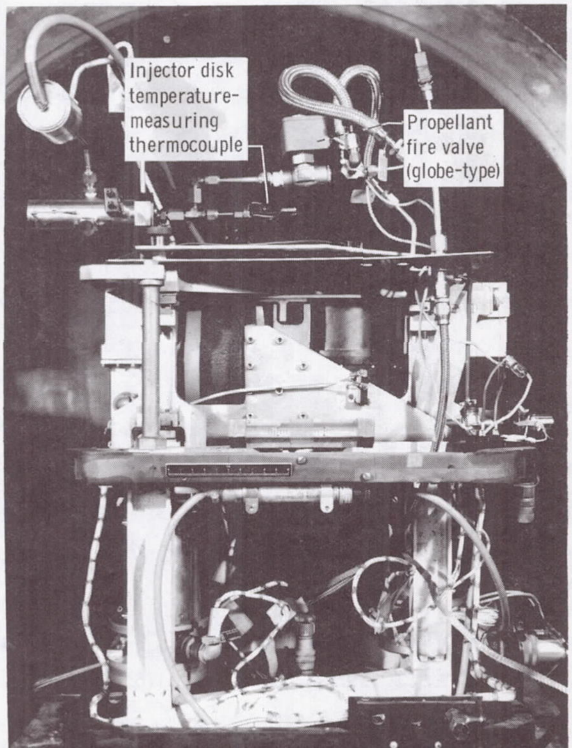


Figure 8. - Millipound thrust stand.

brations indicated the most accurate results could be obtained from the orifice differential pressure data. The chief difficulty was the slow response of the flowmeters; the orifice flowmeter reached stable data within half a second, whereas the other two types took twice that long. Other measuring difficulties were experienced because of the very low flow rates, fluctuations in the propellant line pressure influencing the density of the propellant, and electronic noise interference in the data records. For these reasons the flow-rate data in this report are those from the orifice flowmeter and are accurate to ±5 percent.

Data recording. - All the data signals were recorded on a 24-channel oscillograph recorder, which is described in detail in the instrumentation section of reference 3.

Test Operations

It usually took 1 day to charge the gas storage tanks from zero pressure to 150 psia (103 N/cm^2 abs). At that pressure the gas generator automatically stopped.

Prior to a given series of tests, the thruster environment pressure was reduced to about 0.5 torr. Then the instruments were calibrated and zeroed. For each test a timer automatically controlled the on-and-off sequencing of the instrumentation recorder, the ignition power, and the propellant flow control valve. Test duration, as well as the time between tests, was varied from 10 milliseconds to 20 seconds.

The test duration was one variable, the other being the propellant flow rate. The flow rate was controlled by regulating the propellant flow-line pressure. The propellant mixture ratio was always stoichiometric, that is, with an equivalence ratio of 1.0 ($O/F = 7.94$, or 11.2-wt. % hydrogen).

After each series of tests, or runs, the transducers and flowmeter or flowmeters were recalibrated, and the thruster was examined for erosion, overheating, and/or deposits.

During (or just prior to) several tests, flow-line detonations occurred. Once a flow-line detonation began, generally near the thruster, nothing could be done to prevent the detonation wave from traveling through the flow line to the gas storage tank or tanks. The various flame arresters and pressure-relief valves did not stop the detonation wave; pressure relief in the system was attained only when the rupture disks burst in the safety heads located near the storage tanks. These detonations were loud, but the only system damage was to the burst disks and instrumentation transducers that were over-ranged.

RESULTS AND DISCUSSION

The results of this investigation were obtained from 180 tests. In four of the tests detonations occurred; in 33 of the 180 tests ignition did not occur; and 43 of the tests were of a pulsing nature (pulse durations less than 0.5 second). Therefore, there were 100 out of 180 tests from which steady-state performance data could be obtained. All the tests were made with the same basic thruster but different sintered disk injector designs and different propellant control valves.

Thruster Combustion Performance

The combustion-chamber pressure of this thruster was a direct function of the propellant flow rate, as is shown in figure 9. Two variations of this pressure - flow-rate relation are presented in figure 9, one when ignition occurred and the other when ignition would not occur. Between chamber pressures of approximately 1 and 4 psia (0.7 and 2.8 N/cm^2 abs), the combustion pressure - propellant-flow-rate relation became erratic and below 1 psia (0.7 N/cm^2 abs), it was not possible to obtain combustion.

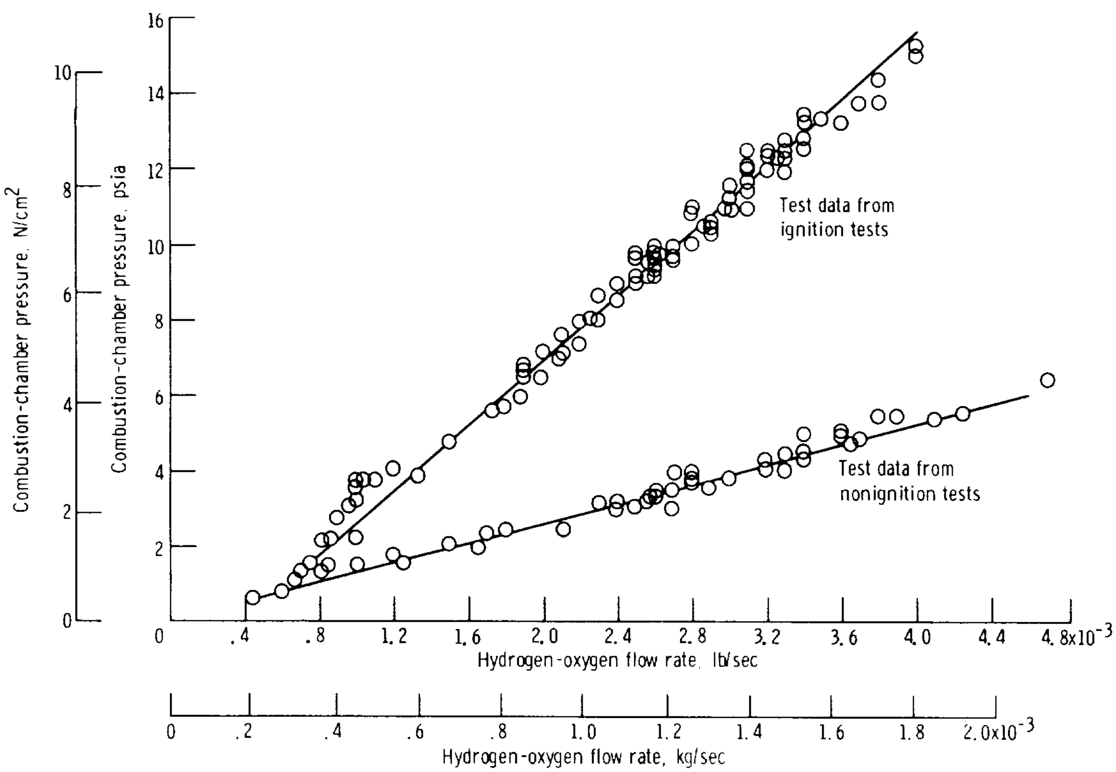


Figure 9. - Hydrogen-oxygen thruster combustion-chamber pressure as function of propellant flow rate.

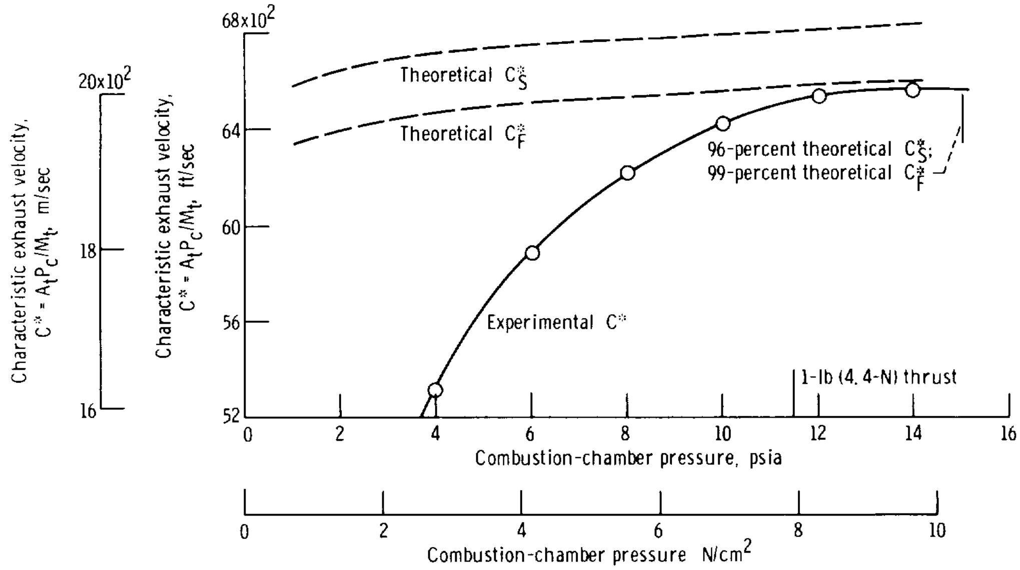


Figure 10. - Hydrogen-oxygen thruster combustion performance. Oxidizer-fuel flow ratio, 7.94.

With the data presented in figure 9, the thruster combustion performance can be calculated in terms of the characteristic velocity C^*. The C^* determined is presented as a function of the combustion pressure in figure 10. Also presented in the same figure are the C_S^* and C_F^* curves. The data, theoretical and experimental, are for stoichiometric hydrogen-oxygen combustion.

The results in figure 10 indicate that at the rated combustion chamber pressure of 15 psia (10.3 N/cm^2 abs) the experimental C^* is about 99 percent of the C_F^* value. As the propellant flow rate, and consequently the P_c, was reduced, the C^* experimental value dropped drastically; at about 1 psia (0.7 N/cm^2 abs), there is no indication of combustion and the efficiency corresponds to that of cold propellant gases expanding as they leave the thruster.

The C^* of this gaseous hydrogen-oxygen thruster is much higher than that obtained from equivalent-thrust-size engines using liquid Earth-storable propellants (refs. 3 and 4). Some of the factors that probably contribute to this difference are that the hydrogen and oxygen are premixed before injection into the combustor, that they are mixed in a stoichiometric ratio, that they are injected as gases, and that the gas mixture enters the combustion chamber as a uniform flow across the injector face rather than as discrete jets.

Although the experimental C^* efficiency is high in figure 10, the C^* value is low compared with what can be obtained at higher equivalence ratios than 1, using the same propellants. The theoretical C^* at an equivalence ratio of 1 is only about 81 percent of that possible at an equivalence ratio of 3.5 to 4.0. This is shown in a plot of theo-

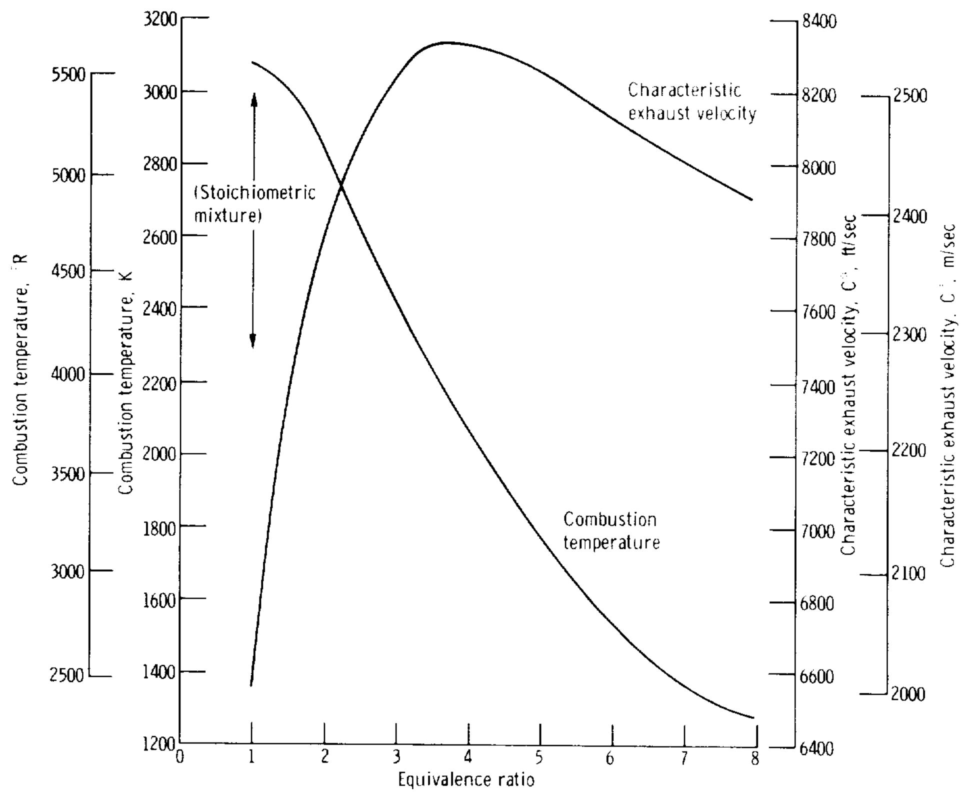


Figure 11. - Theoretical combustion temperature and characteristic exhaust velocity of hydrogen and oxygen gas as function of mixture ratio. Combustion-chamber pressure, 14.7 psia (10.2 N/cm^2); frozen equilibrium conditions.

retical C* as a function of equivalence ratio in figure 11. It can also be noted in the same figure that at equivalence ratios of 3.5 to 4.0, the theoretical combustion temperature is only 61 percent what it is at an equivalence ratio of 1.

The dynamic response characteristics of the system were such that only frequencies of 100 hertz or less could be measured. At no time during the testing was there any indication of combustion instability in this frequency range.

Thruster Nozzle Performance

The thrust data obtained in this investigation are plotted as a function of the combustor pressure at which they were obtained, as shown in figure 12. For the lower thrust values, less than 0.4 pound of thrust (1.7 N), the combustion efficiency is questionable, while at the higher thrust values efficiency was obtained after steady-state operation had been realized during each test. Within the accuracy of the experimental data, the relation of thrust to chamber pressure in figure 12 is a linear function similar to that of the

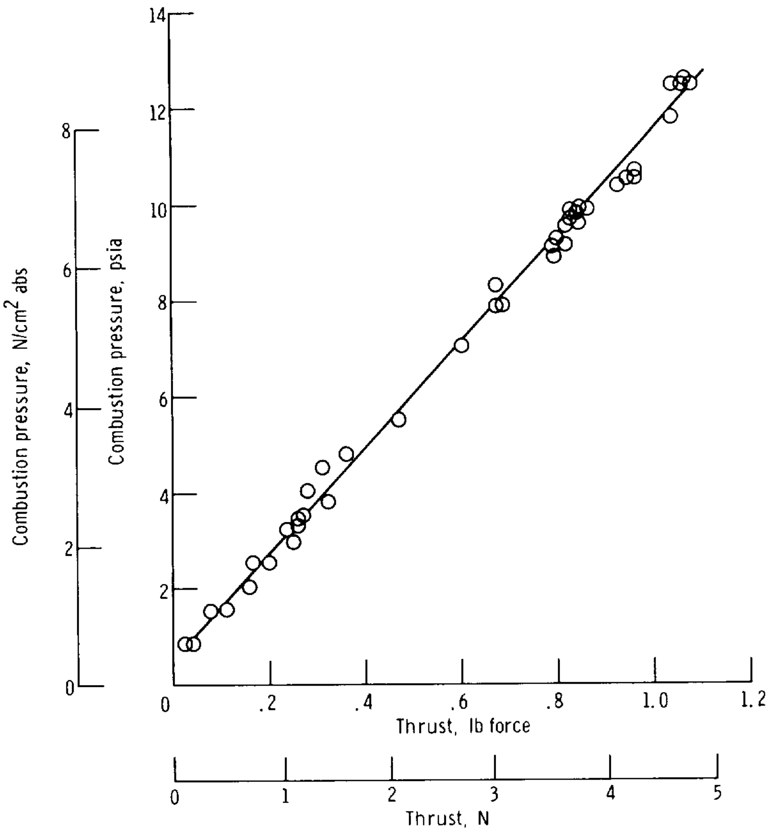


Figure 12. - Hydrogen-oxygen thruster combustion pressure as function of thrust.

thruster results in reference 3.

The test engine used in this program had an expansion ratio ϵ of 35.4 and a combustion pressure P_c of 15 psia (10.3 N/cm^2 abs). It is designed to have the greatest nozzle efficiency at an environmental pressure of 1.45 torr for $(C_f)_F$ conditions or 2.74 torr for $(C_f)_S$ conditions. At other environmental pressures the thruster is experiencing nonideal gas flow through the nozzle; that is, the nozzle is flowing either under- or over-expanded.

In this investigation it was impossible to maintain a constant environmental pressure during a given test. By the time steady-state operating conditions had been realized during a test, the environmental test pressure had increased from 0.5 torr to as much as 4.0 torr by the end of a long test. Also the combustion-chamber pressure was varied intentionally between 1 and 15 psia (0.7 to 10.3 N/cm^2 abs) from test to test, thereby creating new theoretical environmental pressure requirements. All of these variations add to the difficulties in analyzing just how efficiently the thruster nozzle is performing.

The nozzle performance efficiency is measured in terms of a thrust coefficient C_f as

$$C_f = \frac{F}{P_c} A_t = I_{sp} \frac{g_c}{C^*} + \frac{A_e}{A_t} \left(\frac{P_e}{P_c} - \frac{P_a}{P_c} \right)$$

Ideally, the nozzle exit pressure P_e should equal the environmental pressure P_a, and the last term of the thrust coefficient equation will drop out. But for the operating conditions of this investigation the effect of this term influences the nozzle performance. The theoretical C_f for this thruster is presented in figure 13 as a function of both the combustion pressure P_c and the environmental pressure P_a for both $(C_f)_F$ and $(C_f)_S$ theoretical performance.

The experimental C_f is calculated from the data presented in figure 12. The C_f is equal to the ratio of thrust to chamber pressure divided by the nozzle throat area. The nozzle throat area was a constant for all the tests, so the ratio of thrust to chamber pressure alone determined the experimental C_f. This C_f is presented as a function of P_c in figure 13. Most of the thrust data were obtained after the thruster had operated

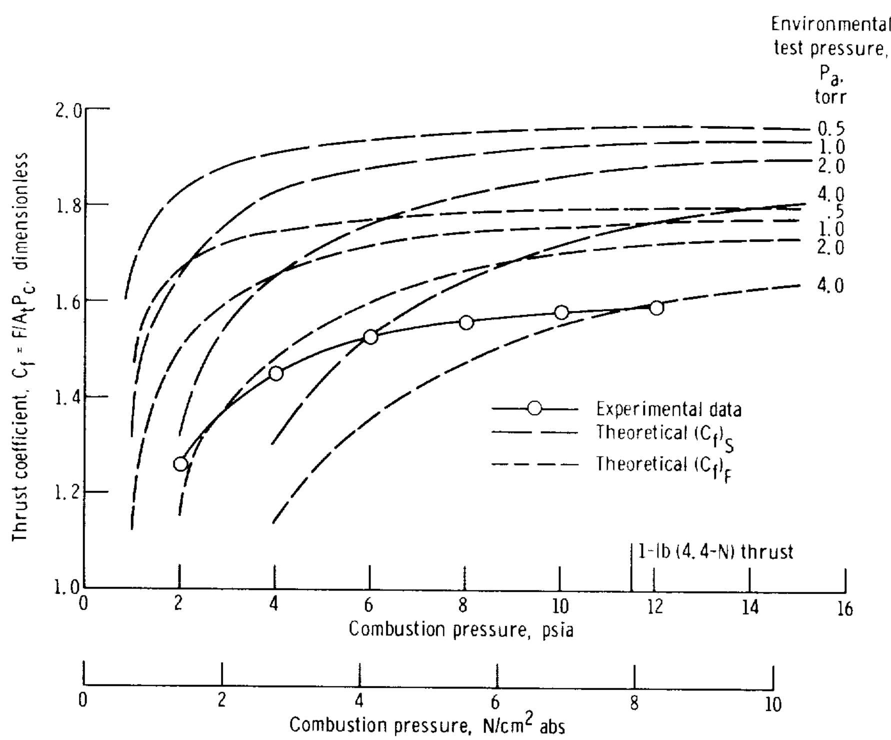


Figure 13. - Hydrogen-oxygen nozzle performance. Expansion ratio, 35.4; oxidizer-fuel mass-flow ratio, 7.94; nozzle throat cross-sectional area, 0.0543 square inch (0.35 cm^2).

long enough in each test to raise the environmental pressure to about 2 torr. At that environmental pressure and at 1-pound- (4.4-N-) thrust conditions, the experimental C_f is about 93 percent of the theoretical frozen C_f. At lower chamber pressures, the C_f, both experimental and theoretical, drops off. Also for lower-chamber-pressure tests, the environmental pressure remained at lower values longer than it did at 1-pound- (4.4-N-) thrust run conditions.

Thruster Specific Impulse Data

The thruster specific impulse I_{sp} was calculated by using the average experimental propellant flow rates and experimental thrust data. The results are plotted as a function of the combustion chamber pressure in figure 14. At a combustion pressure of 11.5 psia (7.9 N/cm^2 abs), thrust equaled 1 pound (4.4 N) and I_{sp} was 328 seconds. As the propellant flow rate W_t was reduced to obtain lower thrust and combustion pressure values,

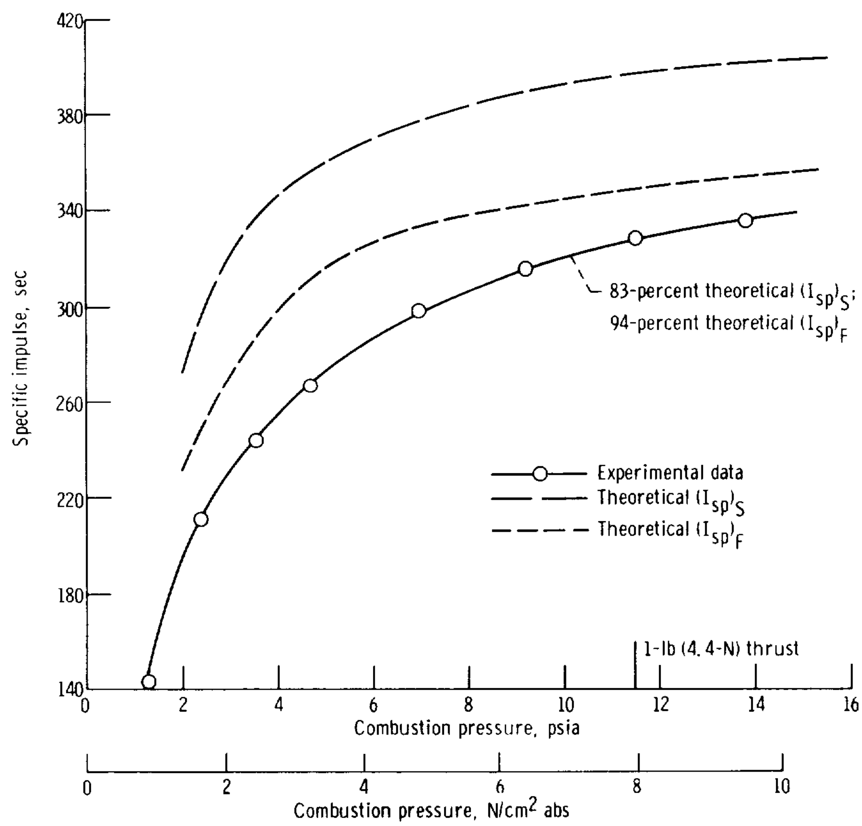


Figure 14. - Hydrogen-oxygen thruster specific impulse as function of combustion pressure. Ambient pressure, 2 torr at data reading; expansion ratio, 35.4; oxidizer-fuel flow ratio, 7.94.

the I_{sp} decreased rapidly. At 0.1-pound- (0.4-N-) thrust, the I_{sp} was about 143 seconds.

Also shown in figure 14 are the theoretical $(I_{sp})_F$ and $(I_{sp})_S$ values. The theoretical data were calculated from the theoretical C^* and C_f data (figs. 10 and 13) by using the relation that $I_{sp} = C_f C^*/g_c$. Theoretical C_f was chosen at an environmental pressure of 2 torr, as this is the pressure at which most of the steady-state thrust data were obtained. At 1-pound- (4.4-N-) thrust the experimental I_{sp} was 94 percent of the theoretical frozen equilibrium I_{sp}.

Thruster Pulsing Performance

The thruster steady-state performance tests were generally between 3 and 5 seconds in duration with off-time durations of several minutes for reducing the environmental test pressure to 0.5 torr. However, 43 tests were made in which the thruster was pulsed; that is, the individual pulse on-time was 10 to 500 milliseconds and the off-time was about 100 to 5000 milliseconds (10-percent duty cycle). Each test consisted of 10 to 100 pulses, depending on their duration.

During such pulsing tests the ignition spark was operating during both the on and off portions of the pulsing mode. Pulsing could be continued at the same propellant flow rates until the flow-line pressure regulator upstream pressure became equal to that set into the regulator.

For the longer duration on-time pulses (more than 100 milliseconds), the data signals stabilized enough that engine performance could be calculated. Within the error of the calculations the pulsing performance was equal to that presented in figures 9 and 12 (pp. 14 and 17).

For pulses of less than 100 milliseconds on-time, the data signals did not reach an equilibrium value, and the resulting data are considered to be transient. In reference 3 it was possible to integrate the short-pulse thrust and flow-rate signals to get average-pulse results, but in this investigation it was impossible because of background electronic "noise" that saturated the data signals. Filtering the signals removed not only the noise but also the signals for short-duration pulses. The pulses of a given unfiltered test sequence were consistently repeatable. The thrust, combustion pressure, and propellant flow-rate data signals were of the same amplitude from one pulse to the next.

In these pulsing tests the thruster wall temperatures would increase in step fashion from one pulse to the next during the testing sequence. As the pulse on-time became shorter, the step increase in temperature became less. For pulses of less than 50 milliseconds on-time, there was no increasing temperature trend, and, for these very short pulses, it was not possible to determine if ignition was occurring (the exhaust flame was almost invisible for all the tests) or if the thruster was only experiencing cold-gas thrusting.

Propellant Flow-Line Detonations

During this program, four flow-line detonations occurred during the testing, and two others occurred during pretest procedures. Detonations were not totally unexpected with such a propellant combination, and damage was minor to the system.

The exact cause of the detonations is unknown. The pretest detonations might have been initiated by energy resulting from actuating components in the flow line. At the instant of explosion, internal metallic and/or nonmetallic parts in the flow-line valves (which had an explosion-proof rating) were moving across each other. It is possible conditions were just right for generating a static charge.

The timing of the flow-line detonations during a given test was unpredictable; it varied from 1 to 5 seconds after a test started. The only common feature apparent in the predetonation data of these tests is an unusual propellant injector pressure behavior. This is illustrated in figure 15 for each of the tests in which detonation occurred. In all the testing the propellant flow rate would be increased by a predictable increase in the injector pressure; however, at or just prior to detonation this relation did not hold. The propellant line pressure would change while the propellant flow rate and combustion pressure were remaining fairly constant. For example, in test 714 the line pressure was much higher for a given flow rate than it had normally been. This pattern might be an indication of face-burning of the sintered injector disks or of sintered disk pore blockage. Examples of this are shown in figures 16(a) and (b).

An inline thermocouple was used to measure the temperature on the upstream side of the injector sintered disks. Between tests this temperature would increase to 710° R (394 K) or more, but when propellant began flowing through the disks, the temperature rapidly decreased to that of the propellants (ambient). If and when detonation occurred, the disk temperature was the same as that of the propellant. Only after a detonation did this temperature increase extremely rapidly.

Other than the noise, the first indication of a flow-line detonation occurring was a sudden increase in the flow-line pressure and temperature in, and directly upstream of, the injector assembly. The combustion pressure, the flowmeter pressures and temperatures, and the gas storage tank pressures and temperatures were driven, in progressive order, off scale on the high-speed oscillograph record. The detonation velocity was calculated to be approximately 8000 feet per second (2440 m/sec) based on a knowledge of the distance between the various transducer locations.

The instrumentation capabilities and test-time limitations of this program did not

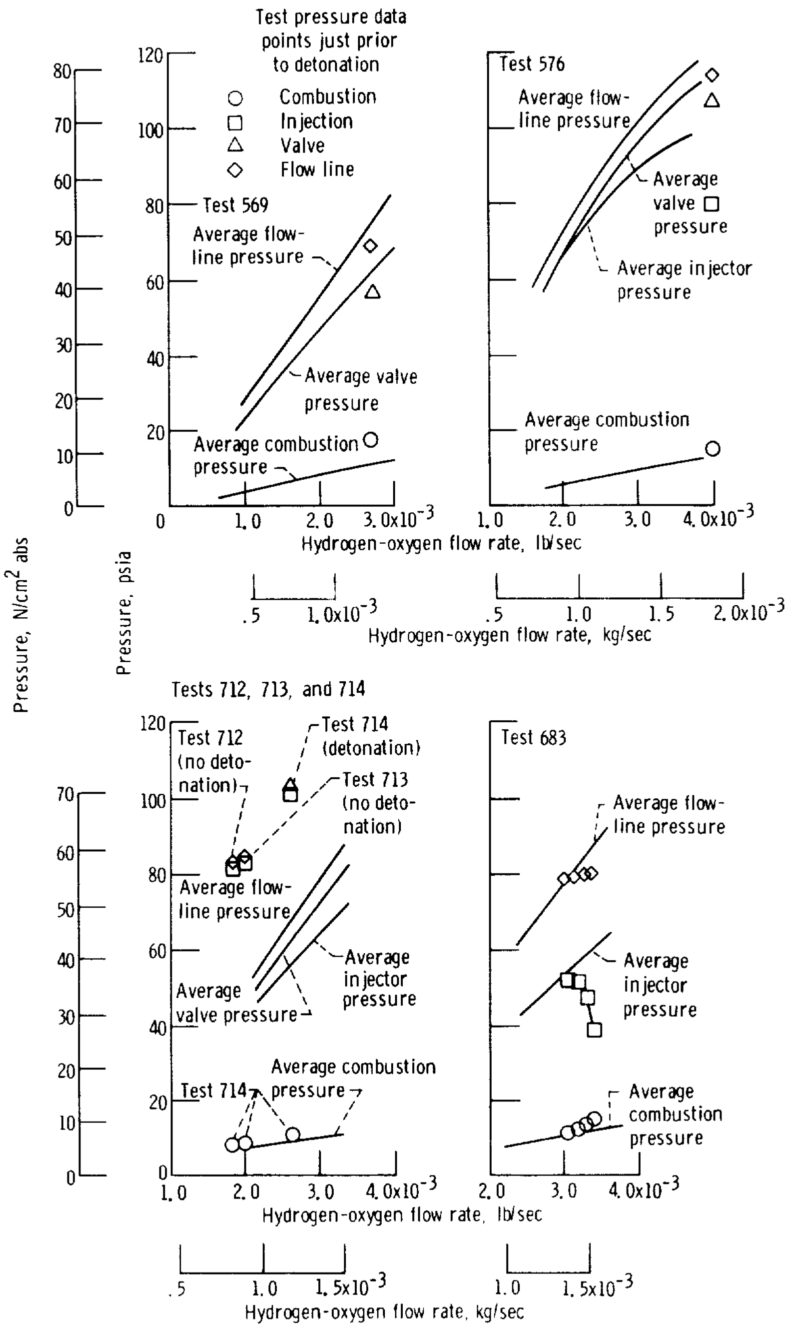
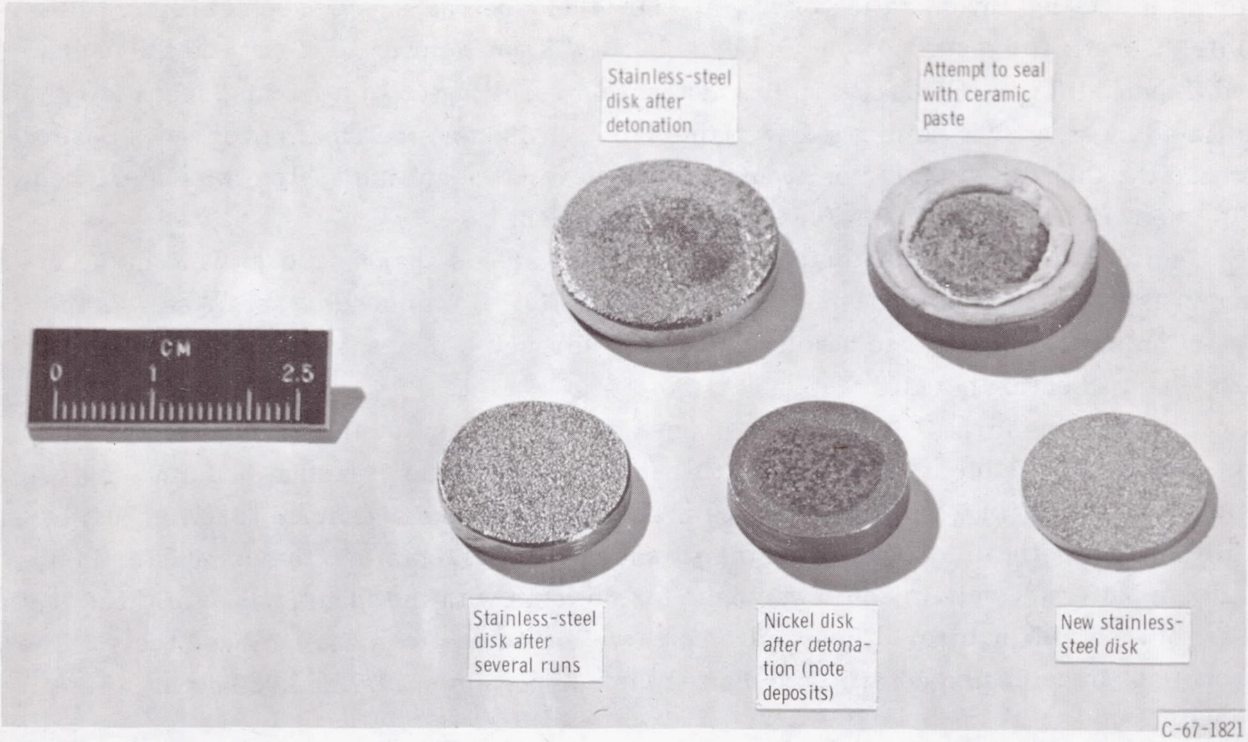
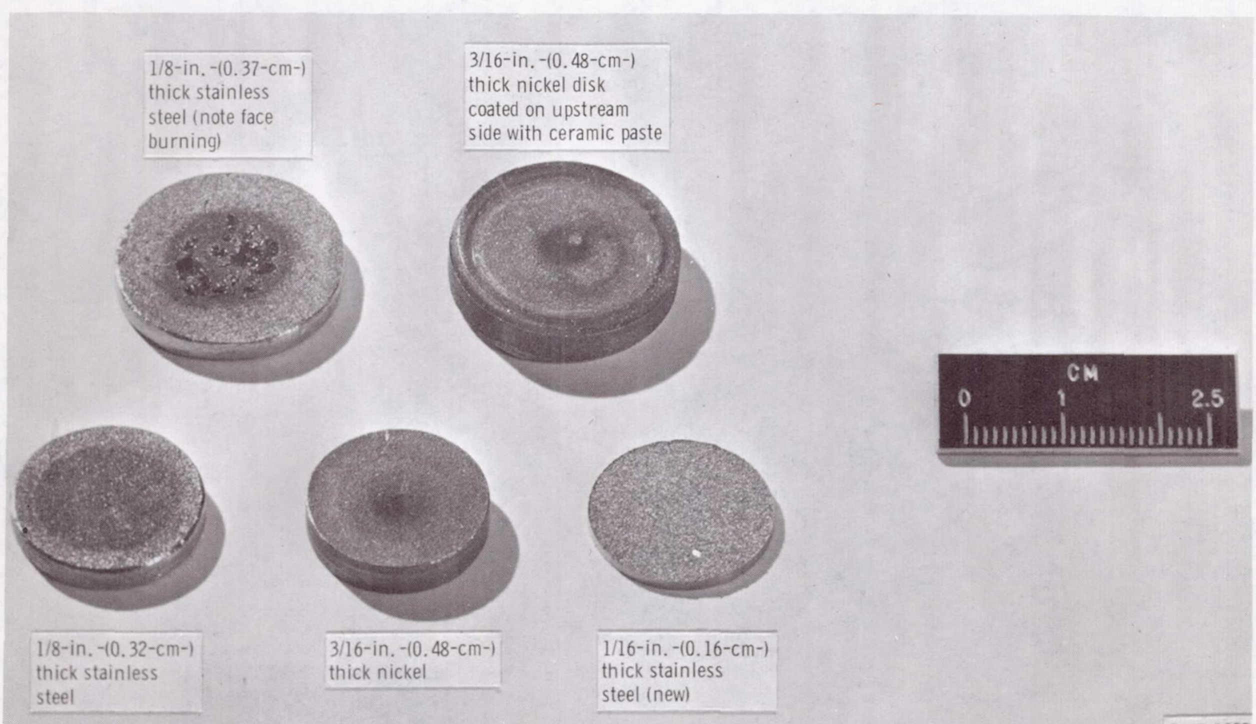


Figure 15. - Hydrogen-oxygen thruster detonation test data.



(a) Upstream side.



(b) Downstream side.

Figure 16. - Sintered disk injectors.

permit a detailed investigation of why and how these detonations were occurring. Possible theoretical explanations are offered by Lewis and vonElbe (ref. 5) and Laderman and Oppenheim (ref. 6) in regard to a detonation resulting from a moving shock front initiated by an uneven-laminar-flow flame front. Also an electrical corona was observed around the thruster (in spite of grounding) whenever the ignition system was energized with the thruster in a vacuum (0.5-torr) environment.

In order to alleviate the detonation problem, various changes were made in the injector assembly. Modification form B (shown in fig. 4) was unsuccessful, as was the use of an injector which had a solid metal face through which were drilled sonic-orifice propellant injection holes.

Modification form C (fig. 14) was based on the assumption that previous injector sintered disk pore blockage was resulting in a propellant gas flow that had a nonuniform laminar profile. In form C the upstream sintered disk was presumed to act as a filter, while the downstream disk injected the filtered propellant gas into the combustor in a uniform pattern. This injector was used for 35 tests with the thruster; 22 of these tests were of a pulsing nature. There were no flow-line detonations during these tests. The propellant injector and control valve after the completion of testing are shown in figure 17.

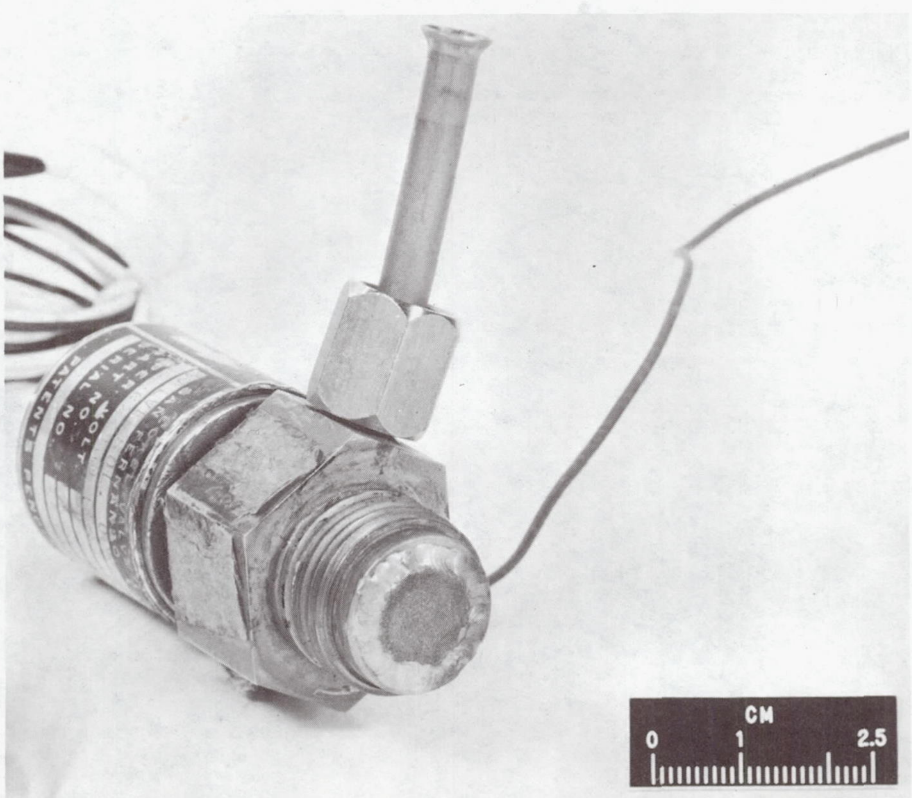


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Figure 17. - Water-electrolysis rocket injector and valve after completion of test program.

Propellant Ignition

The electrical system used to initiate combustion had a strong influence on the ignition characteristics; it also had a disturbing influence on the electrical data signals and could possibly be the cause of flow-line detonations.

The transistorized power supply source furnished with the thruster system resulted in reliable ignition. It delivered a 0.04-joule spark approximately every 0.005 second. The average ignition delay - the time from when the propellant control valve was energized until the combustion pressure started to suddenly increase - was 10 to 15 milliseconds. However, this solid-state device had a limited life which did not last the duration of the program.

Two replacement 10 000-volt alternating-current ignition transformer systems were erratic. Neither a 0.07- nor a 0.16-joule spark (at 60 hertz) would reliably ignite the propellant gas when the environmental pressure was 0.5 torr, although the minimum theoretical energy required for the ignition of a stoichiometric mixture of H_2 and O_2 is less than 0.001 joule (ref. 5). While testing the rocket system in the vacuum environment, a corona was noticeably visible on the combustor spark plug and thruster surfaces, indicating considerable power leakage. In addition, with these transformer ignition systems the electronic data signal noise became very severe and almost completely masked the data signals.

Another method of obtaining ignition is the use of a catalytic ignition scheme. Such a scheme for igniting hydrogen-oxygen is suggested in reference 7. A thruster designed for catalytic ignition was built for this program, but insufficient time prevented the testing of it.

SUMMARY OF RESULTS

In this experimental investigation to evaluate the performance characteristics of a nominal 1-pound- (4.4-N-) thrust water-electrolysis rocket engine system, the following results were obtained:

1. Overall steady-state specific impulse was 83 percent of theoretical shifting equilibrium performance and 94 percent of theoretical frozen equilibrium performance.

2. The nominal 1-pound- (4.4-N-) thrust system was operated at thrust levels as low as 0.1 pound (0.4 N), at which point the thruster performance had decreased to that of a cold-gas hydrogen-oxygen system.

3. The radiation-cooled nickel thruster operated without erosion or valve operation problems in spite of having an oxidizer-fuel flow ratio of 7.94, which resulted in a 5500° R (3050 K) combustion temperature.

4. A sintered porous injector was developed to provide adequate filtering and uniform propellant gas injection profiles. No flow-line detonations were experienced during its use.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, Oct. 7, 1968,
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