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PERFORMANCE INVESTIGATION OF BIPROPELLANT FRACTIONAL-POUND THRUSTERS

by R. James Rollbuhler

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Cleveland, Ohio

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ABSTRACT

The performance capabilities of four fractional-pound-thrust chemical bipropellant rocket thrusters were investigated experimentally. The thrusters were tested with hydrazine-type fuels and nitrogen tetroxide oxidizer in a vacuum of 0.5 to 3.0 torr. Data were obtained to calculate the performance parameters as functions of propellant mixture ratio. Maximum characteristic exhaust velocity ranged from 3400 to 3700 ft/sec (1038 to 1130 m/sec). Peak values of specific impulse varied from 155 to 245 sec. Operational difficulties increased with decreasing propellant injection orifice size (blockage). Hypergolic ignition did not present any problems. Instrumentation limitations prevented the obtaining of pulsing operation data.

PERFORMANCE INVESTIGATION OF BIPROPELLANT FRACTIONAL-POUND THRUSTERS

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SUMMARY

An experimental investigation was conducted with four fractional-pound-thrust engines to evaluate the performance of low-thrust chemical bipropellant rocket thrusters for potential use in space-vehicle reaction-control systems. The engines, or thrusters, were designed to use hydrazine-type fuels and nitrogen tetroxide oxidizer and to be radiation cooled. An instrumented altitude test facility was used to test each of the thrusters over a range of oxidizer-to-fuel ratios and at different total propellant flow rates. The vacuum pressure environment ranged from 0.5 to 3.0 torr.

The maximum specific impulse, about 78 percent of the theoretical specific impulse, was obtained using the thruster designed for the highest thrust operation. The other thrusters, which were designed to operate at lower thrust levels, had maximum specific impulses of 50 to 60 percent of theoretical. For three of the four thrusters the nozzle efficiency was very high while the combustion efficiency was very low.

Because of injector-port blockage, steady-state propellant flow rates and oxidizer to fuel weight flow rate ratios could not be maintained consistently. Vacuum environment, hypergolic ignition was not a problem with these thrusters.

INTRODUCTION

Spacecraft and satellites are becoming larger in size and are requiring greater accuracies in positioning, trajectory control, and attitude maintenance. Secondary propulsion devices to do these tasks on current spacecraft generally feature the positive expulsion of a medium such as nitrogen gas. A cold gas system has the disadvantage that the thrust obtained per unit mass of propellant is very low, less than 100 pounds (444 N) thrust per pound (4.53 kg) of gas expelled each second. Thus, the gas system becomes very bulky and heavy with increasing system reaction control requirements.

Liquid chemical reaction systems (mono- or bipropellant) can provide a higher specific impulse and have a relatively low system volume and mass associated with large total impulse requirements.

Precise control requirements necessitate accurate impulse correction firings - the smaller and shorter in time each impulse bit can be, the more precise can be the control. Slow chemical system rates, however, limit the minimum effective impulse operating time to about 0.01 second (ref. 1). Therefore, the only means of decreasing the impulse per pulse bit is to reduce the reactor thrust. This need for small impulse bit output devices has led to an increased interest in chemical thrusters providing approximately 1 pound (4.4 N), or less, of thrust.

In reference 1, results were presented for bipropellant thrusters operating at 1 or 5 pounds (4.4 or 22.0 N) steady-state thrust. The performance efficiency of the 1-pound thruster was equal to that of the best 5-pound thruster. The question arises as to whether even lower thrust size engines would have the same steady-state performance efficiency as the larger (1- to 5-pound) thrusters. Steady-state efficiency is that obtained after 0.5 second, or more, of continuous operation. Results in reference 1 indicated that the steady-state efficiency is the best obtainable from a given thruster and as the operating test time is decreased, this efficiency drops.

The thrusters used in this investigation were built to NASA specifications, were designed for laboratory-type testing, and were not of a flight-type configuration. The propellants used in the thrusters were a fuel mixture of 50 weight percent hydrazine - 50 weight percent unsymmetrical dimethyl hydrazine (A50) and nitrogen tetroxide (NTO) oxidizer. The results from this test program are presented in terms of specific impulse, characteristic velocity, and thrust performance, each as a function of the propellant oxidizer to fuel flow rate ratio. Also discussed are ignition characteristics, as well as the instrumentation and operating problems encountered. All the testing was done in a vacuum environment (0.5 to 3.0 torr abs).

SYMBOLS

A_t	thruster nozzle throat cross-sectional area, in.^{2}; cm^2
C_f	dimensionless thrust coefficient, F/P_cA_t
C^*	characteristic exhaust velocity, $P_cA_tg/\dot{w}_p$ or $P_cA_t/\dot{m}_p$, ft/sec; m/sec
F	thrust, lb force; N
g	gravitational acceleration constant, 32.2 ft/sec^2; 9.81 m/sec^2
I_{sp}	specific impulse, $F/\dot{w}_p$ or $F/\dot{m}_pg$, sec
L^*	characteristic thruster length, in.; cm

$\dot{m}_p$ propellant mass flow rate, slugs/sec; kg/sec

O/F ratio of oxidizer flow rate to fuel flow rate

P_c thruster combustion-chamber pressure, lb force/in.^{2}; N/cm^2

$\dot{w}_p$ propellant weight flow rate, (slug-ft)/sec^3 or lb/sec; (kg-m)/sec^3

APPARATUS AND PROCEDURE

Test Hardware

Four thruster assemblies, which included the propellant control valves, were tested in this program. Each was designed to produce a particular thrust, as is shown in table I. The most important criterion in their design was the ease of testing, with little regard for size or mass. Therefore, each thruster is more representative of a laboratory test item than a unit that might be used for a space mission. Careful design could eliminate at least half the mass of each thruster without adversely affecting the performance parameters.

Triplet-jet thruster. - This thruster is similar to thruster number 4 of reference 1. The combustor was made of tantalum - 10-percent-tungsten alloy with a proprietary silicide coating diffused into the surface. Shown in figure 1 is the propellant injection pattern - two fuel jets impinging on an axial oxidant jet. The injection-port diameters were 0.009 and 0.012 inch (0.023 and 0.030 cm) for the fuel and oxidizer, respectively. Other details about this thruster are listed in table I.

Vortex thruster. - The vortex thruster design was based on previous work done with swirl-cup type injectors (ref. 2). In this concept the two liquid propellants are injected tangent to the thruster wall such that they both move toward the thruster throat while swirling over each other along the combustor wall (see fig. 2). With such a vortex pattern the combustion chamber length can be very small because the propellant combustion-chamber stay time is a function of the swirl path length rather than of the distance from the injector to the nozzle throat.

The propellant feed tubes between the combustion chamber and the control valve were 0.015 inch (0.038 cm) inside diameter and had a very high length-to-diameter ratio. The same type of torque-operated valve as was tested in the storable-propellant program of reference 1 was used on this thruster. Since the valve was mounted directly above the thruster nozzle, a heat shield was mounted between the two.

The combustor was made of stainless steel, with the inner surface coated with zirconia. The nozzle throat was not contoured but had a sharp-edge entrance. The reason for making such an unusual throat approach was twofold. First, the small nozzle throat (less than 1/8 in. (0.32 cm) in diameter) would have been difficult and time

TABLE I. - THRUSTER SPECIFICATIONS

Thruster	Rated combustion-chamber pressure		Rated thrust		Rated propellant flow rate (a)		Combustor thrust diameter		Nozzle-throat expansion ratio	Thruster characteristic length, L*		Combustion-chamber volume		Combustion-chamber wall material	Nozzle material	Protective inner-surface coating	Number of propellant injection ports	Type of propellant control valving
	psia	N/cm^2 abs	lb	N	lb/sec	$\frac{kg\text{-}m}{sec^3}$	in.	cm		in.	cm	$in.^3$	cm^3					
Triplet jet	50	34.4	1.0	4.44	0.0033	0.0147	0.122	0.645	60	38	96	0.303	4.97	Tantalum-tungsten	Tantalum-10 percent tungsten	Silicide diffusion	2 Fuel 1 Oxidizer	Torque motor
Vortex	50	34.4	.75	3.33	.0025	.0111	.115	.292	40	13	33	.136	2.23	Stainless steel	Tantalum-10 percent tungsten	Zirconia	1 Fuel 1 Oxidizer	Torque motor
Spray jet	50	34.4	.50	2.22	.0017	.0076	.081	.206	74	121	307	.507	8.30	Stainless steel	Tantalum-10 percent tungsten	Aluminide	1 Fuel 1 Oxidizer	Solenoid
Micro-orifice	75	51.7	.25	1.11	.0008	.0036	.061	.155	150	16	41	.049	.803	Titanium-zirconium-molybdenum alloy (TZM)	Titanium-zirconium-molybdenum alloy (TZM)	None	4 Fuel 2 Oxidizer	Solenoid

[a]Based on theoretical specific impulse of 300 sec.

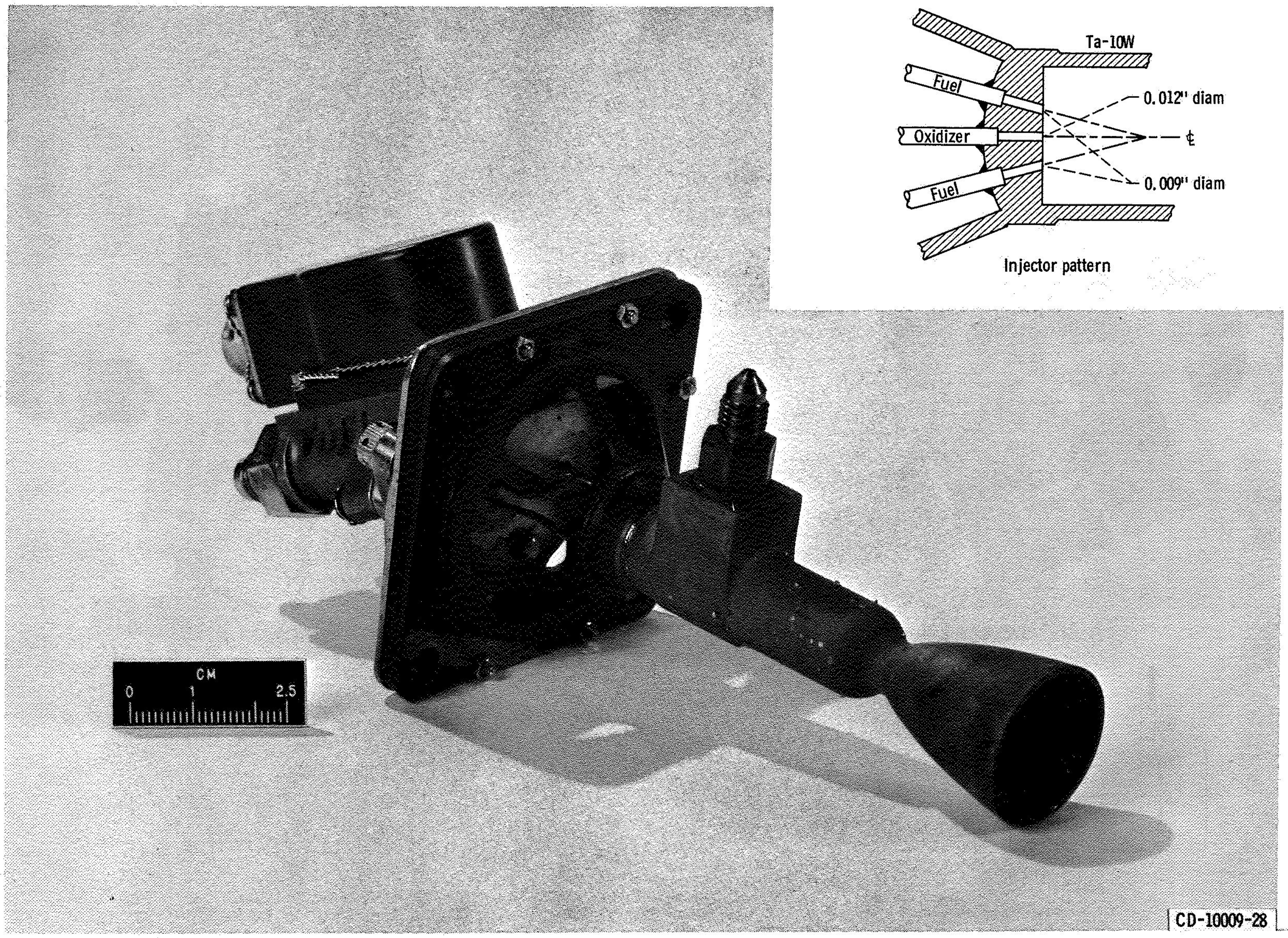

Figure 1. - Triplet-jet thruster.

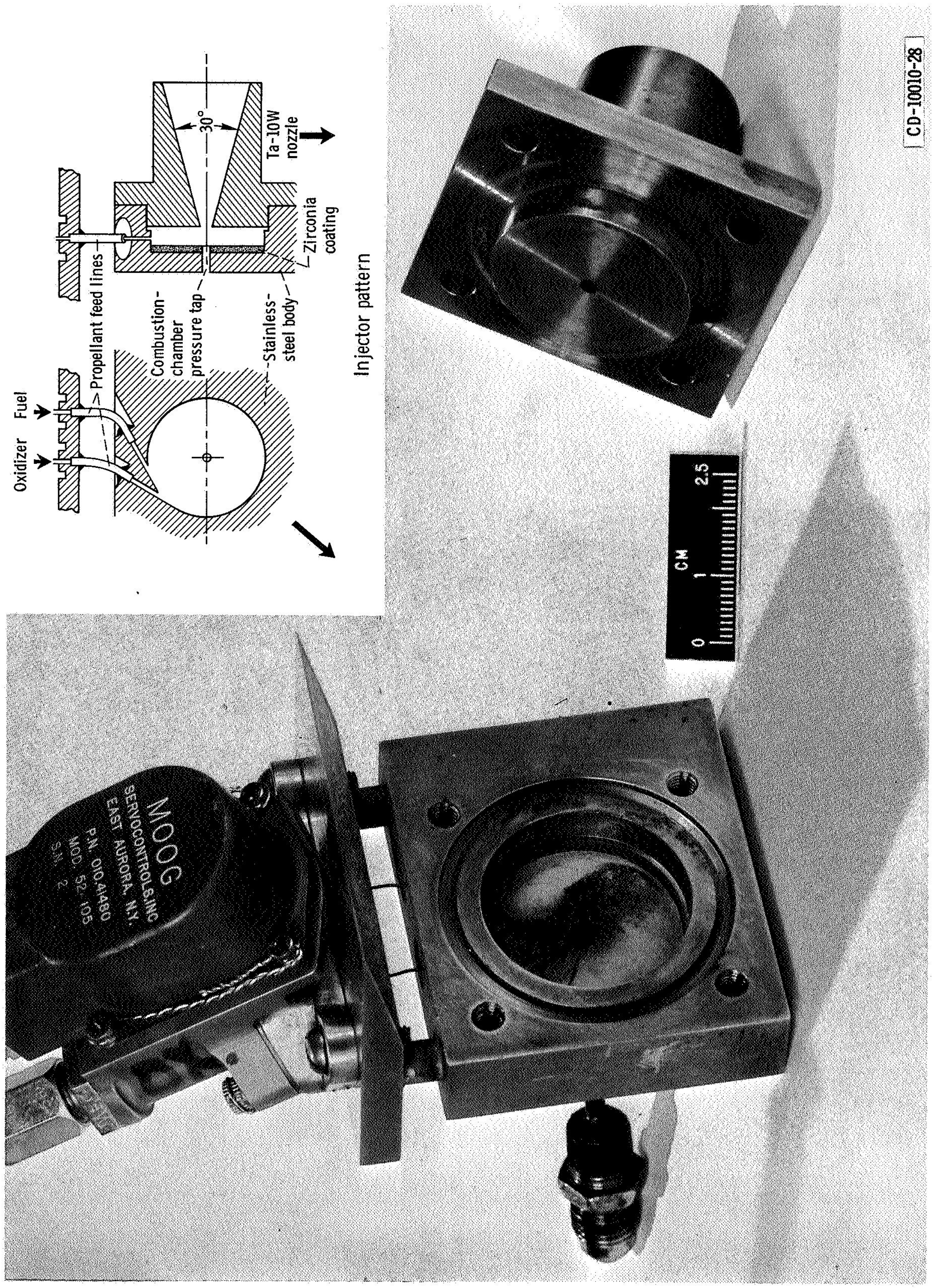

Figure 2. - Vortex thruster.

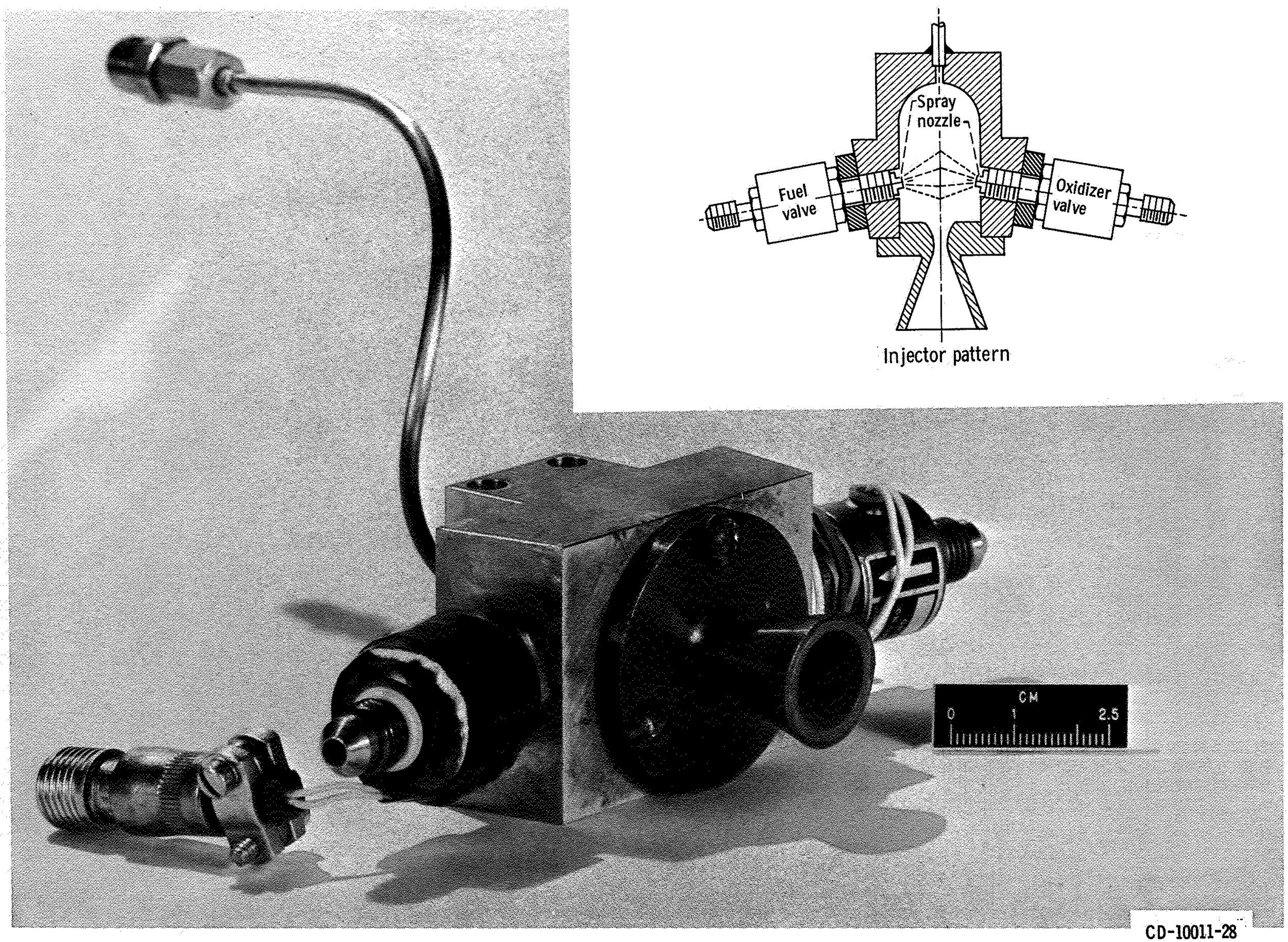

Figure 3. - Spray-jet thruster.

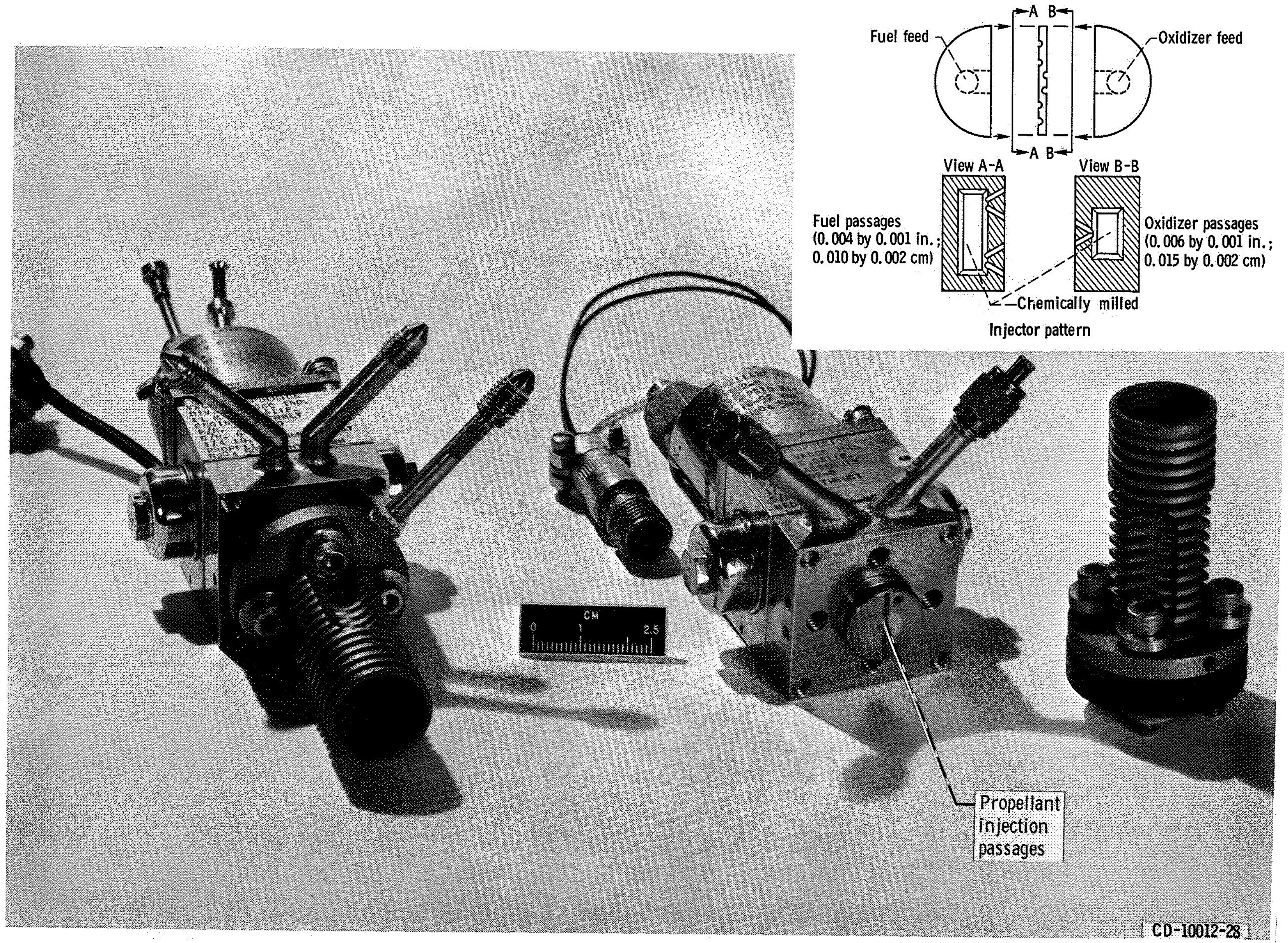

Figure 4. - Micro-orifice thruster.

consuming to machine into a conventional venturi pattern. Second, it was believed that swirling combustion gases in leaving the combustor and entering the nozzle would erode the throat, whereas a sharp-edge entrance would result in a gas-flow vena contracta pattern that would not touch the physical throat.

Spray-jet thruster. - The propellant injection process used with the spray-jet thruster was based on the same design as thruster number 7 of reference 1. The fuel and oxidizer were injected into the combustor through spray nozzles. The spray nozzles were located 180° apart on the combustor wall so that the two sprays would intersect. The spray angle was such that the combined propellant sprays were initially traveling away from, rather than towards, the nozzle throat (see fig. 3). The spray nozzles were commercial types which featured a pintle plugging the flow orifice. Tangential flow passages were machined in the pintle sealing surface so that propellants would eject as distinct spiral jets which broke up into spray droplets upon leaving the nozzle. The nozzles were threaded directly into the solenoid propellant valve outlet. The fuel and oxidizer valves were controlled so that they would open and close together.

The thruster nozzle was made of tantalum - 10-percent-tungsten alloy coated on the inside surface with aluminide. The combustor was made of stainless steel.

Micro-orifice thruster. - The micro-orifice thruster is shown in figure 4. Figure 4 shows one such thruster assembled and another with the combustion chamber and nozzle removed to show the injector face.

The injector was unique in that it consisted of six microsize passages chemically milled in a plate which was perpendicular to the injector face surface. This plate is shown in the sketch in figure 4. The plate was brazed between two semicylindrical pieces to form the injector. The chemically milled propellant-injection passages were arranged so that two outside fuel jets impinged on the thruster wall and each of the other two fuel jets impinged on an oxidizer jet. The result was two inline fuel-oxidizer doublet patterns along with two fuel wall sprays.

Upstream of the propellant-injection passages were 1-micron filters. The propellant-control valves were upstream of the filters. Pressure taps, shown in figure 4, were used for measuring the combustion-chamber pressure and the fuel and oxidizer injection pressure. Other details are given in table I.

Test Facility

The thrusters were tested in the same facility that was used for the investigation of reference 1. The facility consisted of a 1500 cubic foot (42.5 m^3) vacuum pressure tank in which was located the thrust stand. The thrust stand, as well as the rest of the facility, is described and pictured in reference 1

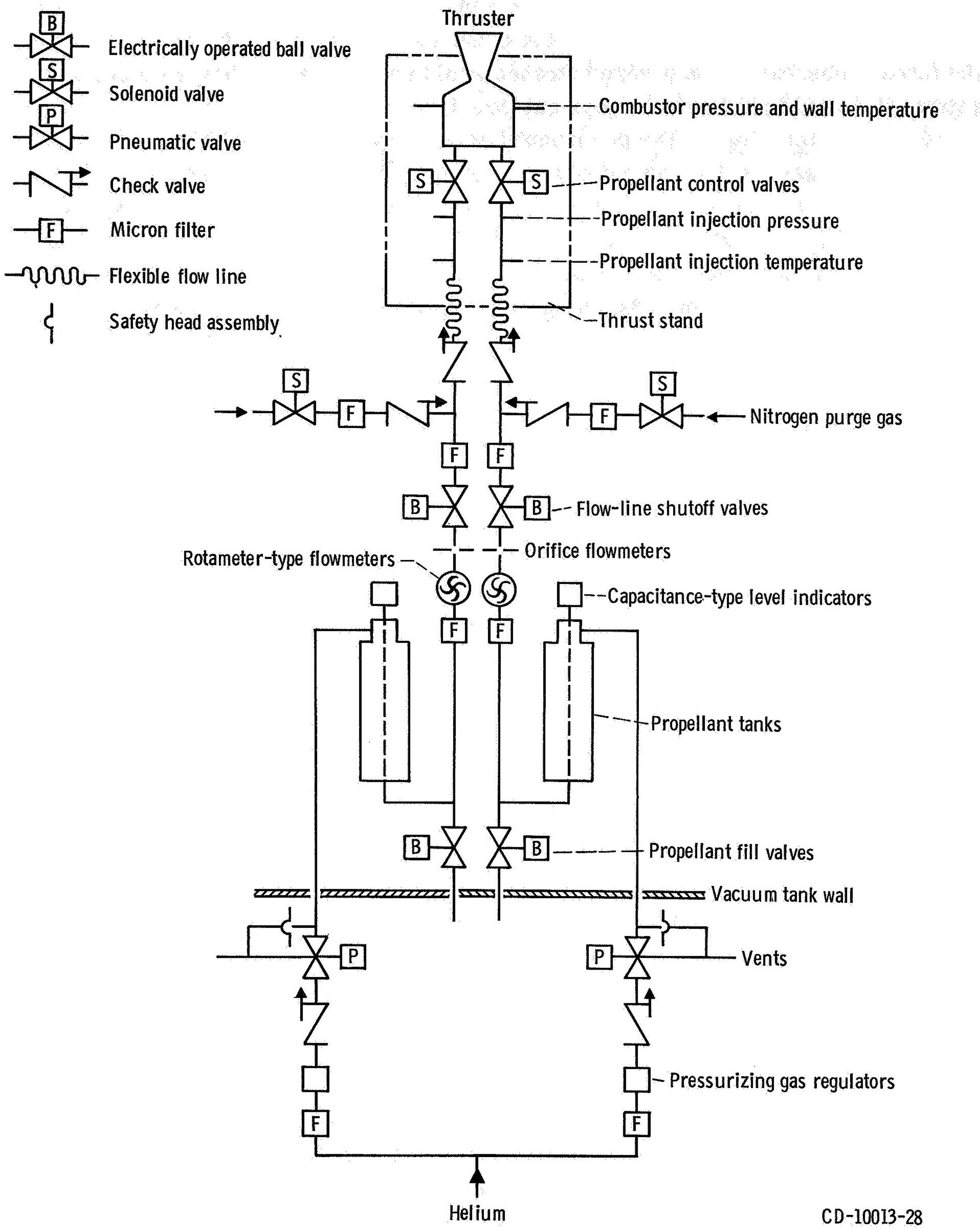

Figure 5. - Flow-line schematic of storable bipropellant test facility.

The fuel flow system and the oxidizer flow system were adjacent to the thrust stand. Each system consisted of a supply tank connected to the thruster control valve by tubing which contained the propellant flowmeters. The relation between the various flowline components is shown in figure 5. The flow rate of the propellant was controlled by varying the supply-tank pressure.

Prior to this investigation, the test facility described in reference 1 was changed by the installation of a better vacuum environment pumping system, installation of a water scrubber for the venting combustion gases, and use of a monitoring television test cell camera and a control room receiver.

The vacuum pumping improvements featured a cryogenic cold trap between the vacuum tank and the pump. By using this trap and improved tank-port sealing, it was possible to reduce the environment pressure around the thruster to less than 0.5 torr.

The television monitoring system proved to be the best method of knowing when combustor ignition occurred, and it also showed malfunctions and propellant leaks around the thruster.

Instrumentation

The test parameter data were obtained using the compound-pendulum type thrust stand, pressure differential orifice propellant flowmeters, strain-gage pressure transducers, and thermocouples. The same type of instrumentation, with minor variations, was used for the investigations of reference 1.

The pressure transducers were calibrated against precision, direct-reading instruments such that the transducer accuracy was ±1 percent. The frequency response of the transducers was only 150 to 175 hertz because of connecting-tube restrictions, as described in reference 1. The temperature data are accurate to $\pm 1^{\circ}$ for the propellants and to $\pm 5^{\circ}$ for the thruster wall temperatures.

The flow data are primarily those obtained by measuring the millipound differential pressure across the propellant orifice meters. As backup propellant flow rate indicators, remote-reading rotameters were used in each propellant line. The orifice flowmeter was a faster-responding instrument than the rotameter; and once steady-state flow was obtained, accuracy was estimated to be ±5 percent. It was possible to measure the total flow per test by comparing the propellant supply tank level (as measured by capacitance-type level indicators) before and after testing.

The flow rates were difficult to measure because they were so small - 3×10^{-7} slug per second (4.5×10^{-6} kg/sec), or less - and the flowmeters which were used had a very low frequency response. As a suggestion to others planning to do low-thrust testing, consideration should be given to using a constant-temperature anemometer flowmeter or a microflow turbine flowmeter. Anemometer flowmeters with frequency responses of 1000 hertz or better are made for use with these propellants. Unfortunately, calibration difficulties delayed the use of such instrumentation in this test program until after the testing portion of the program was completed.

Another difficulty was obtaining accurate thrust data. It was much more of a problem in this investigation than in the larger-thrust program of reference 1. The mass of the thruster and associated hardware of this program was the same as in reference 1, but the thrust was only 1/10 that of reference 1. Because of this changed thrust-to-mass ratio, the thrust-measuring-system spring constant decreased by at least a factor of 10. The very small deflection to thrust signal had to be amplified to get resolution and at the same time electronic interference and background vibration noise were also amplified. Zero thrust shifts due to thermal expansion, system friction changes, flow-line drag, and electrical line drag became apparent during tests. Minor problems in the program of reference 1 became major ones in this program.

Test Operations

Prior to testing any of the thrusters with A50 and NTO propellants, each thruster flow system was extensively calibrated with water over a range of injection pressure drops. From these calibrations, and after making density difference corrections, the propellant-system pressure was determined for obtaining the desired propellant flow rate.

Testing of a given thruster, using the actual propellants, was done in the sealed vacuum pressure chamber. Test control and recording of data were carried out from a control room remote from the vacuum chamber. For each test the sequence of events was automatically controlled once the test programmer was started. The thruster firing time was 5 to 25 seconds for each test. During the test, the vacuum chamber pressure would increase slowly from about 0.5 to 3.0 torr.

To determine the maximum thrust for a given total propellant flow rate, the fuel and oxidizer system pressures were varied to change the O/F. However, an attempt was made to keep their combined weight flow rate $\dot{w}_p$ near a constant value for each test with each thruster.

To minimize thrust-system background vibrations, the vacuum-chamber pump was stopped during each test; then, between tests pumping was resumed to reduce the pressure back to about 0.5 torr.

Because low-thrust engines are generally intended for reaction-control systems, they usually must operate in not only a steady-state but also a pulse-mode manner. It was intended in this program to operate the test thrusters in a variety of pulse-mode cycles; but the limited pulse testing undertaken resulted in unacceptable data. The reasons were fluctuations in the low thrust signal, slow responding propellant flowmeters, lack of dynamic instrument response and accuracy, and "noise" (electronic and background) in the data records. Therefore, only the steady-state test data are reported herein.

RESULTS AND DISCUSSION

The data of this investigation were obtained from 98 test firings of the triplet-jet thruster, 22 firings of the vortex thruster, 40 firings of the spray-jet thruster, and 6 firings of the micro-orifice thruster. These tests were all made in a vacuum environment. In addition, a great number of water flow calibration tests were made with these thrusters in a 1-atmosphere environment.

Thruster Operational Characteristics

A major problem in determining the performance characteristics of these thrusters was propellant injection-passage blockage. The severity of this problem varied from thruster to thruster.

The small design thrust of these thrusters requires very low propellant flow rates. This requirement in turn necessitates the use of injection ports of very small cross-sectional area in order to meter the flows and attain high injection velocities. The size of the passages is governed by the thruster manufacturer's ability to form small injector orifices and by the maximum allowable propellant injector pressure drop. For the thrusters tested, the smallest injector orifice, or jet, was 4×10^{-6} square inch (2.58×10^{-5} cm^2) in cross-sectional area, while the largest was 177×10^{-6} square inch (114×10^{-5} cm^2).

The pressure drop across each fluid injection orifice for each thruster was determined as a function of simulated propellant flow rate. The calibrations were made in a 1-atmosphere pressure environment; water was used to simulate the oxidizer and fuel propellants. The pressure drop as a function of the propellant flow rate (density-corrected water flow rate) for each of the thrusters is presented in figure 6. The pressure drop needed to obtain the desired flow rates for some of the thrusters was high (up to 80 psi, 55 N/cm^2), but not so high as to be unobtainable.

In hopes of avoiding the physical plugging of the injection orifices, filter units of various porosities (down to 1 micron) were installed in the propellant feed lines. During water calibration, blockage did not occur and the resulting flow and pressure data were repeatable enough to obtain the curves in figure 6.

When the same thrusters were tested in a vacuum environment, with NTO and A50

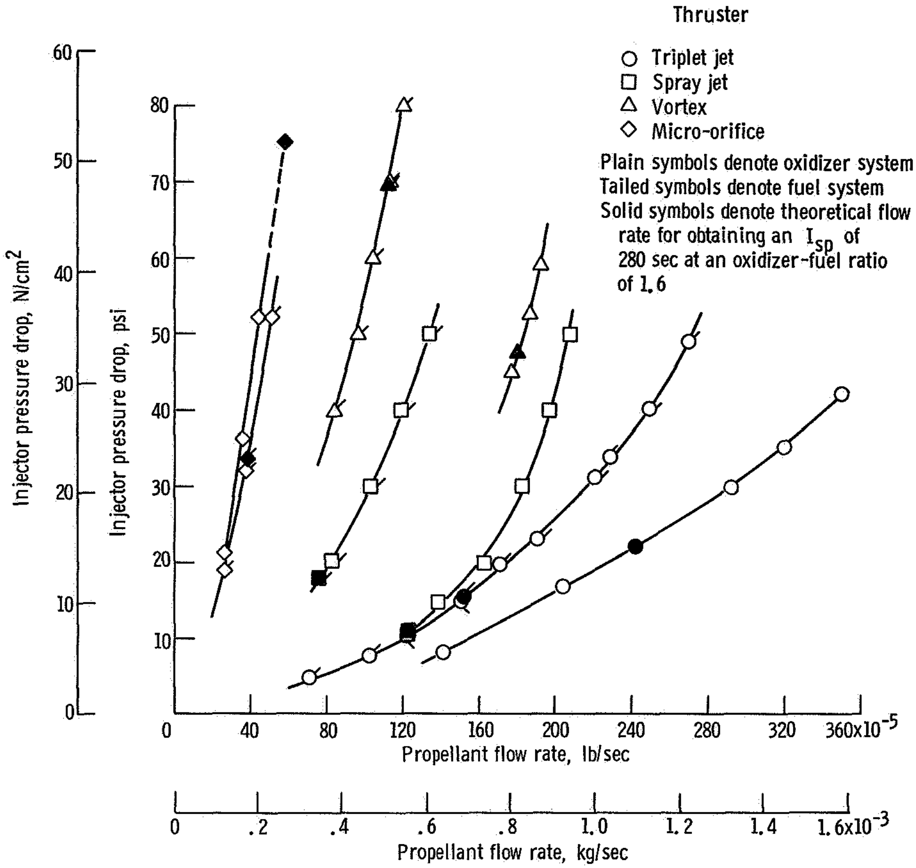

Figure 6. - Thruster propellant calibration with simulated fuel and oxidizer flows.

propellants, propellant injector-passage blockage became a serious problem. This is illustrated for the triplet-jet thruster in figure 7. In this figure the coordinates are the same as in figure 6, and the calibration curve for the triplet-jet thruster is repeated. The fuel flow rate at the beginning of the test started at a pressure drop close to what the calibration had indicated, but as the run proceeded the pressure drop became greater for maintaining a given flow rate. Generally, the pressure drop came close to the calibration curve for the start of each run; then, as the run progressed, the pressure drop began to increase again. This procedure would continue for subsequent tests. The oxidizer flow was not as close to the calibration curve as was the fuel flow. As each test proceeded, the pressure drop for oxidizer flow became larger, and for subsequent tests the pressure drop did not return to the initial run value. Each of the thruster injectors plugged similar to the injector of the triplet-jet thruster. Fortunately, the upstream propellant pressure could be increased enough to obtain satisfactory flow rates for all but the micro-orifice thruster. The micro-orifice thruster propellant passages plugged so fast that meaningful performance data could not be obtained.

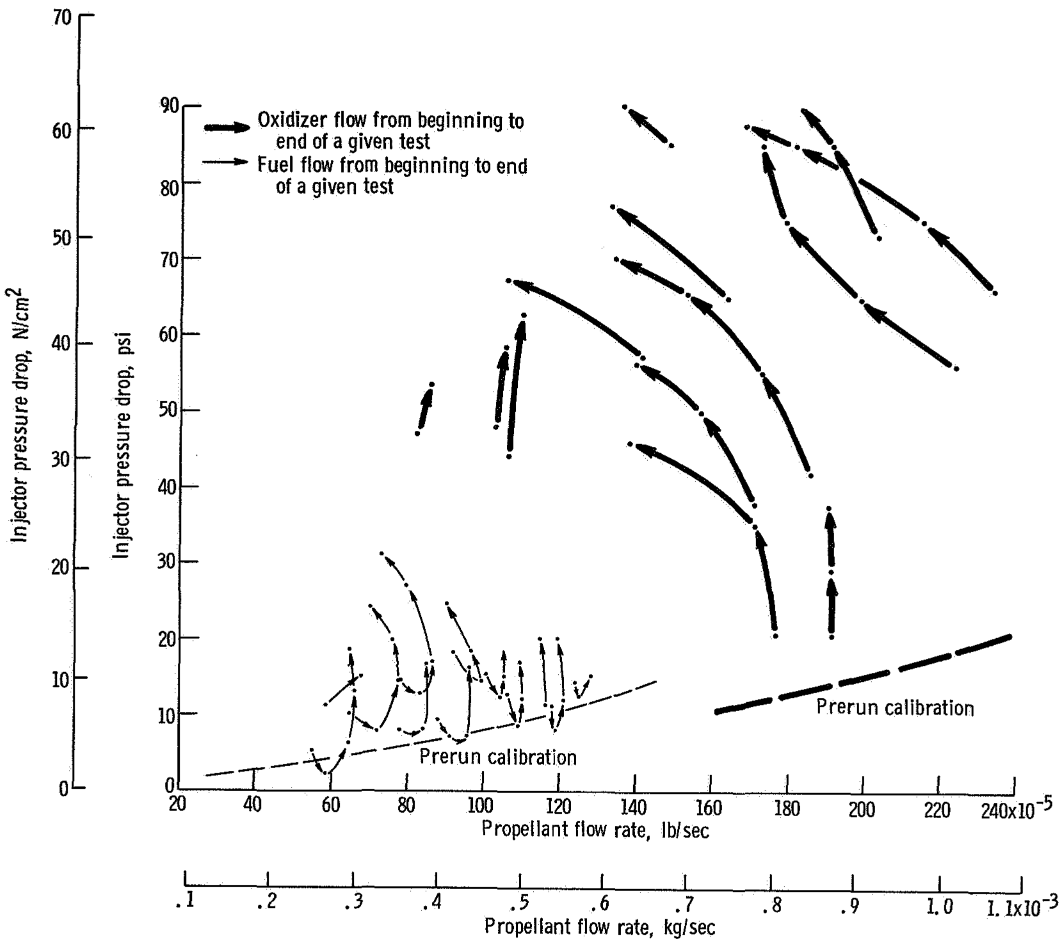

Figure 7. - Triplet-jet thruster propellant injection pressure drop during series of performance tests.

Each time the injector passages became plugged, they were cleaned and backflushed such that water calibrations indicated the same pressure drop to flow rate relation as shown in figure 6. However, whenever the thruster was retested with the propellants, the propellant injector passages again started plugging. The blockage material appeared to be mostly chemical salts which formed most heavily around the oxidizer jet holes; and for oxidant-rich runs, the deposits were heavy in and below the thruster nozzle. Similar blockages have occurred in other investigations (information obtained from R. J. Salvinski of TRW).

Propellant Ignition Characteristics

Ignition was not a problem, providing injector blockage did not prevent admission of propellants into the combustor. The ignition delay time (5 to 20 msec) was close to that reported in reference 1.

Within the limits of the ability of the instrumentation to detect pressure oscillations, the starts did not have any thrust surges or "spikes". Previous investigators (ref. 3) have suggested that the formation of various propellant salts in the combustion chamber of large-size thrusters has resulted in subsequent ignition pressure spikes. In this investigation, the O/F was varied to such an extent that many reaction residues were formed but apparently none promoted unstable ignition.

Thruster Combustion Performance

The experimental characteristic exhaust velocity (C*), which is a measure of thruster combustion performance, is plotted as a function of the O/F for the triplet-jet, vortex, and spray-jet thrusters in figure 8. Presentable performance data were not obtained while testing the micro-orifice thruster. The C* peak values for the triplet-jet, vortex, and spray-jet thrusters were all within 5 to 10 percent of each other, but they occurred at different O/F's. The values of the O/F for peak C* were 1.1 to 1.3 for the triplet-jet thruster, 1.9 to 2.1 for the vortex thruster, and 2.0 to 2.2 for the spray-jet thruster. Above or below these values the C* decreased rapidly with changing O/F.

The overall characteristic exhaust velocity of all three thrusters was very low. For the thrusters of reference 1 the C* values ranged from 4000 to 5000 feet per second

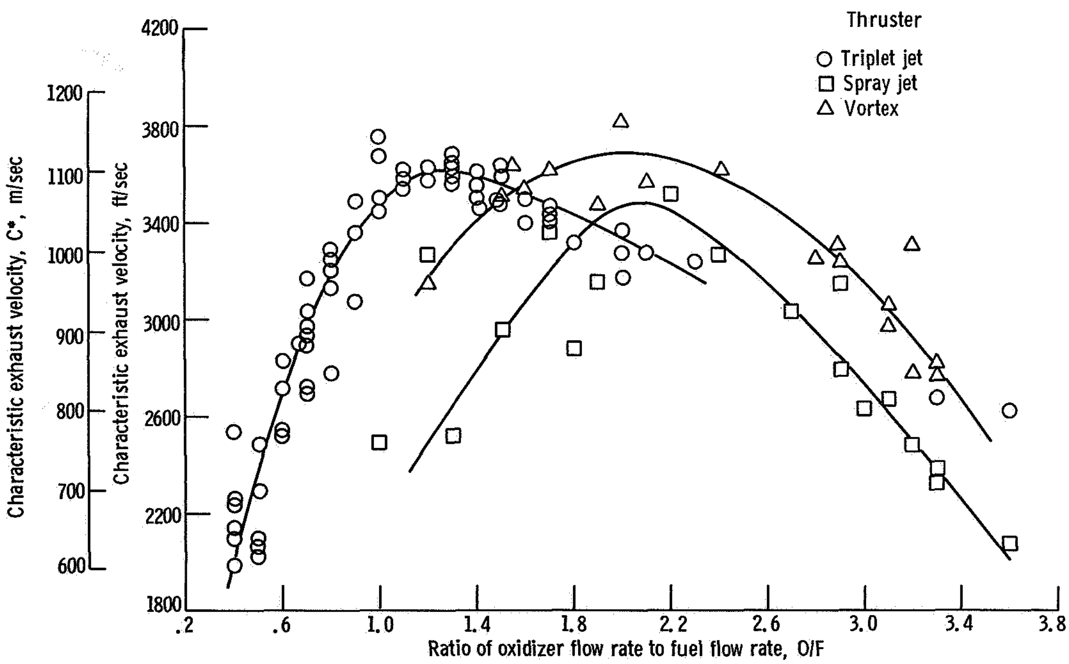

Figure 8. - Thruster characteristic exhaust velocity as function of oxidizer to fuel flow rate ratio.

(1220 to 1525 m/sec), but in this program the peak C^* was only 3400 to 3700 feet per second (1038 to 1130 m/sec). This is approximately 65 percent of theoretical characteristic exhaust velocity (frozen).

Thruster Nozzle Performance

The thruster nozzle performance is generally expressed in terms of a nozzle thrust coefficient, C_f.

$$C_f = \frac{F}{A_t P_c}$$

Since A_t is a constant value for a given thruster, the thrust coefficient is proportional to the thrust to combustion-pressure ratio. These two variables are plotted as functions of each other for the triplet-jet, vortex, and spray-jet thrusters in figure 9. Within the data accuracy, the thrust and combustion pressure for each of the three thrusters have a linear relation. This is similar to the thrust to combustion-pressure data reported in reference 1.

The triplet-jet thruster had the largest F/P_c of the three thrusters tested. This thruster is similar to the number 4 thruster reported in reference 1. However, the triplet-jet thruster operated at a slightly higher F/P_c (0.0204) than did thruster number 4 of reference 1 (0.0158). This resulted in a C_f of 1.75 for the triplet-jet thruster compared with a C_f of 1.52 for the number 4 thruster. Most of the difference in the obtained C_f was probably due to environment pressure. The C_f is strongly influenced by slight changes in environment pressure when the thruster is operating at a fixed nozzle-area ratio. Thruster number 4 of reference 1 was tested in a 5- to 10-torr environment, while the triplet-jet thruster was tested in a 0.5- to 3-torr environment.

The spray-jet thruster had the lowest F/P_c because it was built to have a high combustion-pressure operating range while producing a low amount of thrust. The C_f for this thruster is approximately 1.64, which is higher than many of the larger thrusters reported in reference 1.

The vortex thruster had the lowest C_f of the three tested - 1.36. Its F/P_c is better than that of the spray-jet thruster, but when the ratio is divided by the A_t, which is twice that of the spray-jet thruster, the resulting C_f becomes less than that of the spray-jet thruster C_f. One reason for this poor nozzle efficiency is that the vortex thruster nozzle was not made like a venturi in the throat region, but had a sudden contraction throat, like an orifice. Instead of getting smooth gas flow and an associated high flow coefficient of

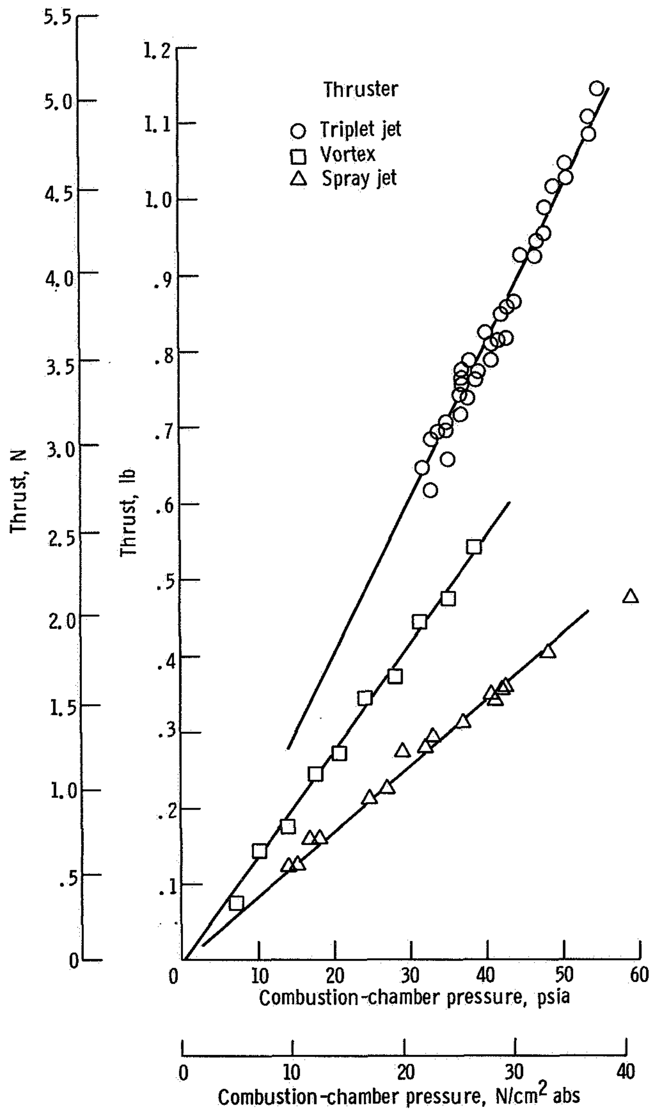

Figure 9. - Thrust as function of combustion-chamber pressure.

0.95 or better, the sudden change in the gas flow pattern resulted in a flow coefficient of 0.85 or less. There is a discussion on the effect of nozzle shapes on flow efficiency in reference 4. Thus, in addition to the usual rocket nozzle losses, another loss in efficiency due to nozzle contour must be included in the overall loss factor.

Specific Impulse Performance

The specific impulse (I_{sp}) is a measure of the overall performance of the thruster.

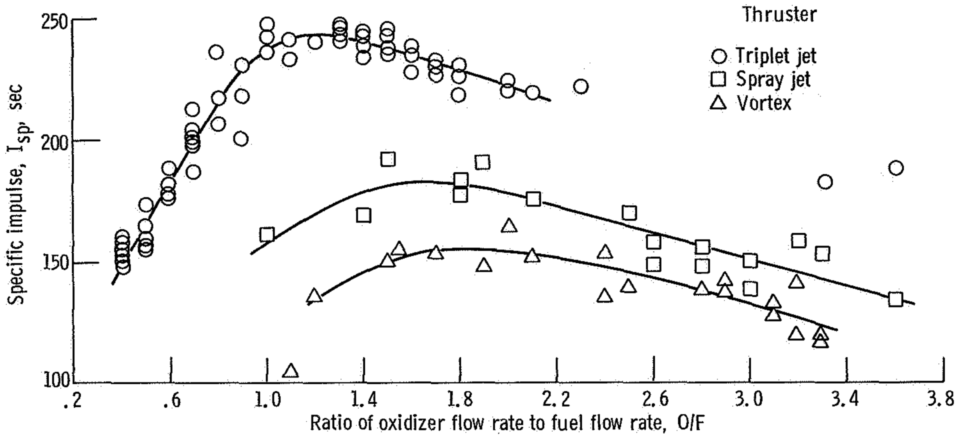

Figure 10. - Thruster specific impulse as function of oxidizer to fuel flow rate ratio.

It is a function of the thrust coefficient C_f and the characteristic exhaust velocity C^*:

$$I_{sp} = C_f C^*/g$$

The specific impulse at an environment pressure of 1 to 3 torr is plotted for the triplet-jet, vortex, and spray-jet thrusters in figure 10. The I_{sp} was obtained after the thruster had operated long enough in each test to give steady-state data. By the time the data were taken the environment pressure had slowly increased from 0.5 torr to 1 to 3 torr. The average I_{sp} curve for each thruster is plotted as a function of the O/F.

The triplet-jet thruster had the largest I_{sp} of the three thrusters, and the peak I_{sp} was 240 to 250 seconds at an O/F of about 1.2. The triplet-jet thruster is similar to thruster number 4 whose performance was reported in reference 1. Thruster number 4 had a peak I_{sp} of about 220 seconds. But thruster number 4 was operating in a higher pressure environment and had, consequently, a lower C_f.

The peak I_{sp} of the spray-jet thruster was about 185 seconds. This specific impulse was lower than that of the triplet-jet thruster because of lower C^* and C_f values. Also, the peak I_{sp} for the spray-jet thruster was at a more oxidizer-rich region of the O/F spectrum.

The specific impulse of the vortex thruster was the lowest of the three thrusters tested. The C^* performance in figure 8 for the vortex thruster peaked at a higher value than that of the triplet-jet thruster, but this is based on physical throat area which is probably not the true gas-flow throat area for the vortex thruster. Since the specific impulse is not a direct function of the nozzle throat area, it is more indicative of the thruster performance than the C^*.

The specific impulse efficiency of the triplet-jet thruster is about 78 percent of theoretical "frozen" performance; that of the spray-jet and vortex thrusters is about 50 to 60 percent of theoretical. A comparison of these results with those of reference 1 shows that the steady-state performance of these small-thrust units is equivalent to the average I_{sp} of the larger-thrust engines operating in a pulsing mode. Such low I_{sp} and C^* data suggest that a large percentage of the propellant is not reacting in the thruster but is behaving like a cold gas medium going through the thruster.

SUMMARY OF RESULTS

An experimental investigation of the performance capabilities of four fractional-pound-thrust chemical bipropellant rocket thrusters yielded the following results:

1. For thrusters operating in the thrust range of 3/4 to 1 pound (3.3 to 4.4 N), the specific impulse was 75 to 80 percent of theoretical compared with about 85 percent of theoretical for thrusters operating with the same propellants in the thrust range of 1 to 5 pound (4.4 to 22 N). Thrusters operating at thrust levels below 3/4 pound (3.3 N) had a specific impulse of less than 60 percent of theoretical.

2. The thruster combustion characteristic exhaust velocity was very much lower (only about 65 percent of theoretical) for less than 1 pound (4.4 N) thrust compared with the 5-pound (22-N) thrusters.

3. The thrusters with conventional venturi-type nozzles had high nozzle thrust coefficients of 90 percent, or more, of theoretical. When an unconventional (orifice type) nozzle was used, the nozzle thrust coefficient was only about 75 percent of theoretical.

4. These fractional-pound thrusters had very small propellant injection passages which became blocked when flowing propellants in a test environment. The blockage appeared to be worse in the oxidizer passages.

5. There was no problem in obtaining hypergolic ignition in a 0.5-torr vacuum environment. No ignition pressure instability was detected. Large quantities of exhaust products and semireacted propellants remained after these small thrusters were tested.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, July 12, 1968,
128-31-02-05-22.

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