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COMPARISON OF ROCKET PERFORMANCE USING EXHAUST DIFFUSER AND CONVENTIONAL TECHNIQUES FOR ALTITUDE SIMULATION

By Joseph N. Sivo and Daniel J. Peters

Lewis Research Center
Cleveland, Ohio

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SUMMARY

A rocket engine with an exhaust-nozzle area ratio of 25 was operated at a constant chamber pressure of 600 pounds per square inch absolute over a range of oxidant-fuel ratios at an altitude pressure corresponding to approximately 47,000 feet. At this condition, the nozzle flow is slightly underexpanded as it leaves the nozzle. The altitude simulation was obtained first through the use of an exhaust diffuser coupled with the rocket engine and secondly, in an altitude test chamber where separate exhauster equipment provided the altitude pressure. A comparison of performance data from these two tests has established that a diffuser used with a rocket engine operating at near-design nozzle pressure ratio can be a valid means of obtaining altitude performance data for rocket engines.

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INTRODUCTION

For the upper stages of multistage space vehicles, rocket engines incorporating large-area-ratio exhaust nozzles may be used to obtain higher specific impulse. The performance of such engines cannot be evaluated at sea level unless some means of reducing exhaust-nozzle back pressure is provided. Because of the cost of additional altitude facilities and on-site availability considerations, it has been proposed that exhaust diffusers be used and thus utilize the energy of the rocket exhaust gases to reduce the pressure surrounding the exit of the exhaust nozzle. Exhaust diffusers have been used in the past to extend the useful altitude range of altitude facilities in tests of turbojet and ramjet engines (ref. 1). Reference 2 gives rocket performance measurements made on an engine with an area-ratio-48 exhaust nozzle while using the diffuser technique. There has been some question, however, as to the validity of this technique in rocket applications because of the possibility of boundary-layer feedback affecting the pressures in the exhaust nozzle and thus affecting the performance measurements.

*Title, Unclassified.

As part of a general investigation into the design criteria of exhaust diffusers for use with rocket engines, a brief program has been conducted at the NASA Lewis Research Center to check the validity of the diffuser technique. Performance measurements from a 1000-pound-thrust rocket engine with an area-ratio-25 exhaust nozzle operating with slightly underexpanded flow were obtained while (1) using an exhaust-diffuser to provide the reduced exhaust-nozzle back pressure and (2) allowing the engine to exhaust to a reduced ambient pressure in a conventional altitude facility; these performance measurements are compared herein.

The effects of several significant design variables on the performance of exhaust diffusers are not included in this report.

DESCRIPTION OF APPARATUS

Engine

The rocket engine used for this investigation provided a nominal thrust of 1000 pounds at a combustion chamber pressure of 600 pounds per square inch absolute; JP-4 fuel and liquid oxygen were used as the propellant combination. The engine had a combustion chamber diameter of 2.35 inches, a throat diameter of 1.20 inches, a nozzle exit diameter of 6.00 inches, and thus an exhaust-nozzle area ratio of 25. The divergent portion of the exhaust nozzle was a 15° half-angle conical section. The outer shell of the engine consisted of hydrostatically formed helical water-cooling passages as shown by the cutaway view of the engine in figure 1. A photograph of the engine installed on the rocket stand is presented in figure 2.

The injector used in this engine, illustrated in detail in figure 3, consisted of a flat face made of nickel and a body of stainless steel. It employed 82 units of like-on-like impinging jets for fuel and 70 for the oxidant. The fuel and oxidant jets were arranged in alternate rows to provide uniform propellant distribution throughout the combustion chamber. Engine ignition was accomplished by means of a gaseous oxygen - propane internal torch, which ignited the rocket propellants at the injector face.

Rocket Installation

As illustrated in the sketch of figure 4, the rocket was mounted on a thrust plate, which was suspended from a mounting stand by four 0.0125-inch-thick flexure plates. The thrust of the rocket was absorbed by a strain-gage-type load cell and the system was preloaded by means of a spring to ensure positive contact between the thrust plate and the load cell at all times. The main propellant supply lines were brought up through the mounting stand using flexible joints.

Fuel and oxidant were stored in 4-cubic-foot supply tanks, which were pressurized to provide propellant flow. Nitrogen was the pressurizing gas for the fuel tank and gaseous oxygen for the oxidant tank. The oxidant tank and supply lines were insulated to minimize heat addition to the oxidant thus maintaining nearly constant oxidant density during a series of rocket firings.

Capsule and Diffuser

The rocket engine assembly, including the thrust system, was totally enclosed in a capsule, which consisted of a semicylindrical cover that was clamped to the engine mounting stand as shown in figures 4 and 5. The capsule isolated the rocket assembly from sea-level pressure forces and prevented any secondary flow into the system. The exhaust diffuser, which was bolted to the capsule, had an inside diameter of 6.4 inches and a length of 42 inches. The inlet of the diffuser was approximately 1/8 inch downstream of the rocket nozzle exit. The diffuser was made up of two concentric tubes with the annulus thus formed used as a cooling-water passage. The kinetic energy of the rocket exhaust is utilized to produce and maintain a static-pressure rise across the diffuser analogous to that obtained in the diffuser of a supersonic tunnel, thereby providing a reduced rocket-nozzle-exit ambient pressure.

Altitude Test Chamber Installation

The entire rocket assembly was installed in a 10-foot-diameter altitude test chamber (see fig. 6). The hot rocket exhaust gases were removed by exhausting the rocket into the facility exhaust system through which airflow was maintained to ensure complete combustion of the rocket exhaust products. This facility was capable of altitude operation during the rocket firings from near sea level to 60,000 feet.

Instrumentation

In order to determine the performance of the rocket engine being tested, the following variables were measured and recorded:

(1) Jet thrust

(2) Fuel flow

(3) Oxidant flow

(4) Fuel and oxidant temperatures

(5) Chamber pressure

(6) Nozzle-exit wall static pressure

(7) Nozzle-exit ambient pressure (capsule pressure with diffuser installed, test chamber pressure with diffuser removed)

(8) Altitude test chamber ambient pressure

(9) Fuel and oxidant tank pressures

The output of the instruments used in the measurement of these preceding variables was recorded by two separate systems. The source of the data presented herein was a multichannel, high-speed digital voltmeter, which provided several complete data scans every second. The second method of data recording was a 36-channel, direct-writing oscillograph. This device helped to establish the time at which steady operation of the rocket occurred and provided verification of the data obtained on the digital voltmeter.

Thrust was measured with a strain-gage compression-type load cell through a thrust bearing, which prevented error in thrust measurement due to nonaxial loads. Propellant flows were measured with rotating-vane-type flowmeters. All pressure measurements were obtained with strain-gage-type pressure transducers. Locations of static-pressure taps are shown in figures 2 and 4. The oxidant temperature was measured with a copper-constantan thermocouple and fuel temperature with an iron-constantan thermocouple, both using a 100° F oven as a reference temperature.

PROCEDURE

Steady-state rocket altitude performance data were obtained at a constant combustion-chamber pressure of 600 pounds per square inch absolute over a range of oxidant-fuel ratios from 1.6 to 3.6 by two methods of altitude simulation. For the first method, an exhaust diffuser was used as shown in figure 4 with approximately sea-level ambient pressure at the diffuser exit. In the second method, the capsule enclosing the engine and the diffuser was removed and the ambient pressure in the altitude test chamber was reduced to the value previously observed within the capsule. The rocket performance data, both with and without the diffuser installed, were obtained with the same engine and instrumentation during a 1-day test period, thereby improving the consistency of the relative comparison of performance data. The instrumentation used during this test period was calibrated before and after the day's rocket firings.

Some small variations in rocket exhaust-nozzle back pressure occurred as the oxidant-fuel ratio was varied during firings with the diffuser installed and as altitude test chamber ambient pressure drifted during firings with the diffuser removed. Therefore, the data presented herein have been adjusted to a constant exhaust pressure to permit performance comparisons more readily. The maximum adjustment applied did not exceed 1/2 percent in thrust on any given data point. The symbols used herein are listed in appendix A and the methods of calculations are presented in appendix B.

RESULTS AND DISCUSSION

A plot of a typical oscillograph trace taken during a rocket firing with the exhaust diffuser removed is shown in figure 7. Rocket ignition occurred during a period of low propellant flow (propellant valves partially open), the duration of which was approximately 3 seconds. When ignition was established, the propellant valves were opened fully and the engine was allowed to run at design propellant flow. Although the rocket engine start and shutdown transients are shown in figure 7, they do not represent the actual transient accurately because of the relatively slow response of the pressure instrumentation. Valid data were obtained only during steady-state operation at design propellant flow.

The rocket engine altitude performance data, obtained with and without the diffuser installed, are presented in table I and figure 8. The rocket nozzle exhausted to a pressure of approximately 2.0 pounds per square inch absolute, which corresponds to an altitude of 47,000 feet. The rocket performance parameters such as pressure ratio, characteristic exhaust velocity c^*, thrust coefficient C_F, specific impulse I, and jet thrust are shown as a function of the oxidant-fuel ratio. A comparison of the measured pressure ratio (fig. 8(a)) shows no indication that any boundary-layer feedback might have occurred when the exhaust diffuser was used. Examination of the other performance parameters clearly shows that the use of the diffuser for altitude simulation had no effect on the measured performance of the rocket engine. The results of this investigation have established that properly designed diffusers used with rocket engines operating at a nozzle pressure ratio near design can be a valid means of obtaining altitude performance data on rocket engines.

SUMMARY OF RESULTS

A rocket engine of 1000 pounds thrust at a chamber pressure of 600 pounds per square inch absolute, using JP-4 fuel and liquid oxygen as the propellant, was operated over a range of oxidant-fuel ratios at a simulated altitude of 47,000 feet. A 15° half-angle conical exhaust nozzle with an area ratio of 25 was used and, at the altitude simulated, flow from the nozzle was slightly underexpanded. The altitude simulation was obtained in one case through the use of an exhaust diffuser. In the second case, the altitude was accomplished with the engine installed in an altitude test chamber. A comparison of performance data from these tests indicated that there was no difference in the measured rocket performance for the case with the diffuser installed as compared with the case without the diffuser. These data establish that exhaust diffusers can be used in conjunction with rocket engines operating at near-design nozzle pressure ratio to provide valid altitude simulation.

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Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, June 10, 1959

APPENDIX A

SYMBOLS

A	nozzle flow cross-sectional area, sq in.
C_F	thrust coefficient
c^*	characteristic exhaust velocity, ft/sec
F	thrust, lb
g	gravitational constant, 32.17 ft/sec^2
I	specific impulse, lb-sec/lb
O/F	oxidant-fuel ratio
P	total pressure, lb/sq in. abs
p	static pressure, lb/sq in. abs
v	exhaust velocity, ft/sec
W	flow rate, lb/sec

Subscripts:

a	altitude test chamber
adj	adjusted
c	combustion chamber
e	nozzle exit
f	fuel
l	liquid oxygen
o	ambient, external of nozzle at its exit
T	total
t	throat

APPENDIX B

METHODS OF CALCULATION

Jet thrust:

Measured jet thrust, $F = \frac{W_T v}{g} + A_e(p_e - p_o)$, was adjusted to a constant pressure p_o of 2.0 pounds per square inch absolute. The following equation was used:

$$F_{adj} = F_{measured} + A_e(p_o - 2.0)$$

where p_o is the measured pressure obtained during the rocket firing.

Specific impulse:

$$I = \frac{F_{adj}}{W_T}, \frac{lb}{lb/sec}$$

Thrust coefficient:

$$C_F = \frac{F_{adj}}{A_t P_c}$$

Characteristic velocity:

$$c^* = \frac{Ig}{C_F}, \frac{ft}{sec}$$

REFERENCES

1. Povolny, John H.: Use of Choked Nozzle Technique and Exhaust Jet Diffuser for Extending Operable Range of Jet-Engine Research Facilities. NACA RM E52E12, 1952.

2. Fortini, Anthony, Hendrix, Charles D., and Huff, Vearl N.: Experimental Altitude Performance of JP-4 Fuel and Liquid-Oxygen Rocket Engine with an Area Ratio of 48. NASA MEMO 5-14-59E, 1959.

TABLE I. - ROCKET PERFORMANCE DATA WITH AND WITHOUT DIFFUSER

Oxidant-fuel ratio, O/F	Total propellant flow, W_T, lb/sec	Thrust, F, lb	Chamber pressure, P_c, $\frac{lb}{sq\ in.}$ abs	Nozzle-exit wall pressure, p_e, $\frac{lb}{sq\ in.}$ abs	Nozzle back pressure, p_o, $\frac{lb}{sq\ in.}$ abs	Test chamber ambient pressure, p_a, $\frac{lb}{sq\ in.}$ abs	Adjusted thrust, F_{adj}, lb	Characteristic exhaust velocity, c*, ft/sec	Thrust coefficient, C_F	Specific impulse, I, sec	Nozzle pressure ratio, P_c/p_e
Exhaust diffuser installed											
1.648	4.066	1160	600	2.24	1.85	13.82	1156	5369	1.705	284	268
1.841	4.006	1168	600	2.37	1.88	13.85	1165	5447	1.719	291	253
1.856	4.010	1170	600	2.40	1.87	13.81	1166	5443	1.720	291	250
1.874	4.015	1166	600	2.36	1.85	13.72	1162	5439	1.714	289	254
1.943	3.991	1176	600	2.43	1.98	13.92	1175	5473	1.733	294	247
2.142	3.990	1190	606	2.43	2.01	13.75	1190	5528	1.738	298	249
2.424	4.006	1201	604	2.45	1.96	13.62	1200	5482	1.758	300	247
2.572	4.019	1205	606	2.53	2.07	13.62	1207	5487	1.763	300	240
2.830	4.033	1212	602	2.49	1.99	13.62	1212	5429	1.782	301	242
2.959	4.019	1208	600	2.44	2.10	13.51	1211	5432	1.786	301	246
3.381	4.073	1212	600	2.44	1.97	13.62	1211	5360	1.786	297	246
3.492	4.097	1215	604	2.58	2.03	13.60	1216	5360	1.782	297	234
3.620	4.115	1213	600	2.43	1.96	13.53	1212	5306	1.788	294	247
Exhaust diffuser removed											
1.475	4.131	1142	593	2.14	1.94	1.94	1140	5223	1.701	276	277
1.638	4.097	1160	602	2.31	1.95	1.95	1159	5343	1.704	283	261
1.923	4.019	1185	598	2.22	1.84	1.84	1180	5451	1.746	294	269
2.046	4.006	1190	602	2.38	1.85	1.85	1186	5465	1.744	296	253
2.461	3.984	1203	604	2.43	1.80	1.80	1197	5511	1.754	301	248
2.746	4.001	1207	601	2.39	1.82	1.82	1202	5464	1.770	300	251
2.749	4.012	1213	606	2.48	1.79	1.79	1207	5497	1.763	301	244
2.771	4.020	1214	604	2.55	1.87	1.87	1210	5463	1.773	301	237
3.082	3.995	1207	597	2.43	1.78	1.78	1201	5440	1.781	301	246
3.095	4.005	1204	597	2.38	1.87	1.87	1200	5426	1.779	300	251
3.382	4.038	1212	599	2.39	1.74	1.74	1205	5400	1.780	298	251

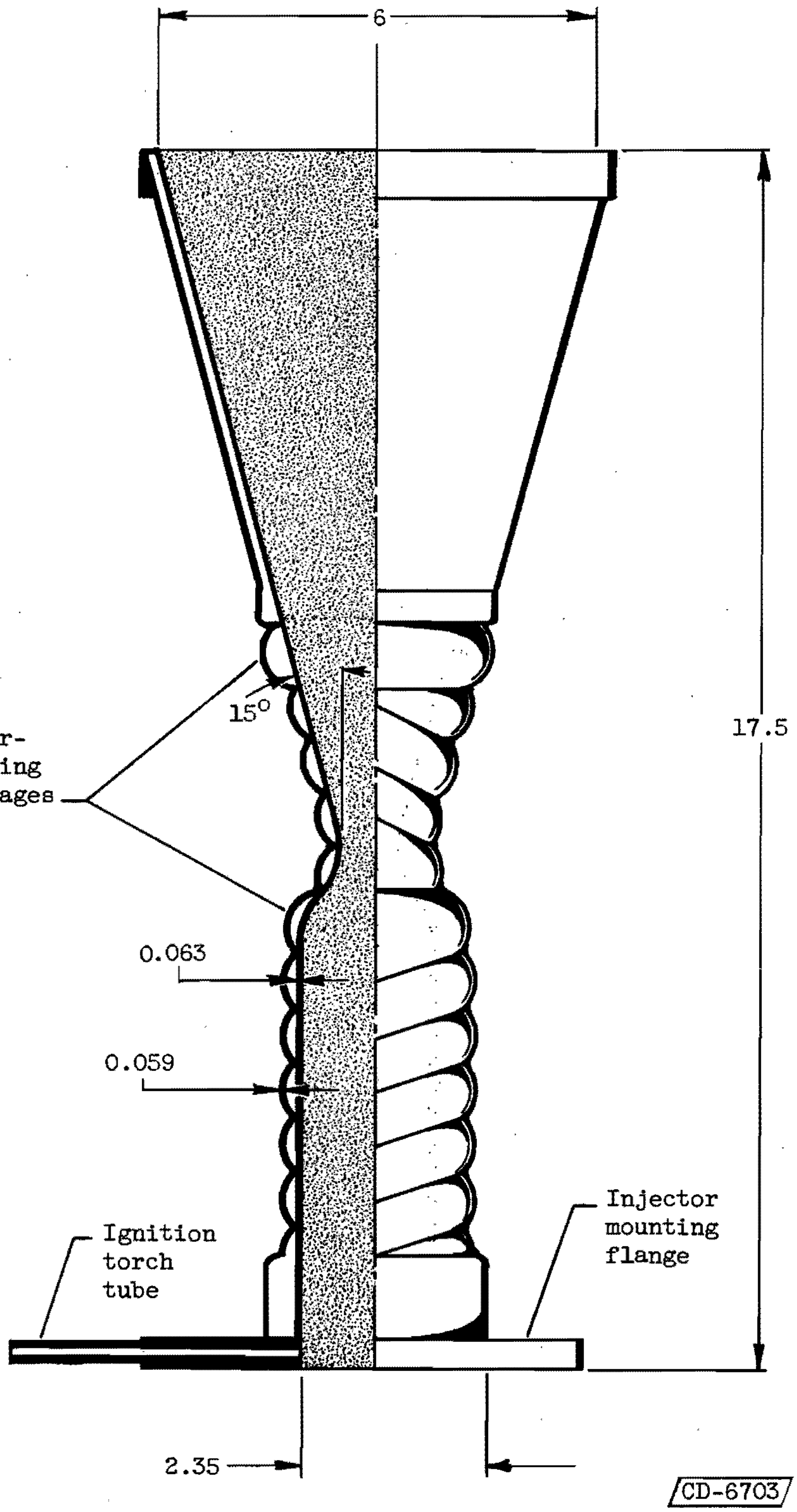

Figure 1. - Section of rocket engine. (All dimensions in inches.)

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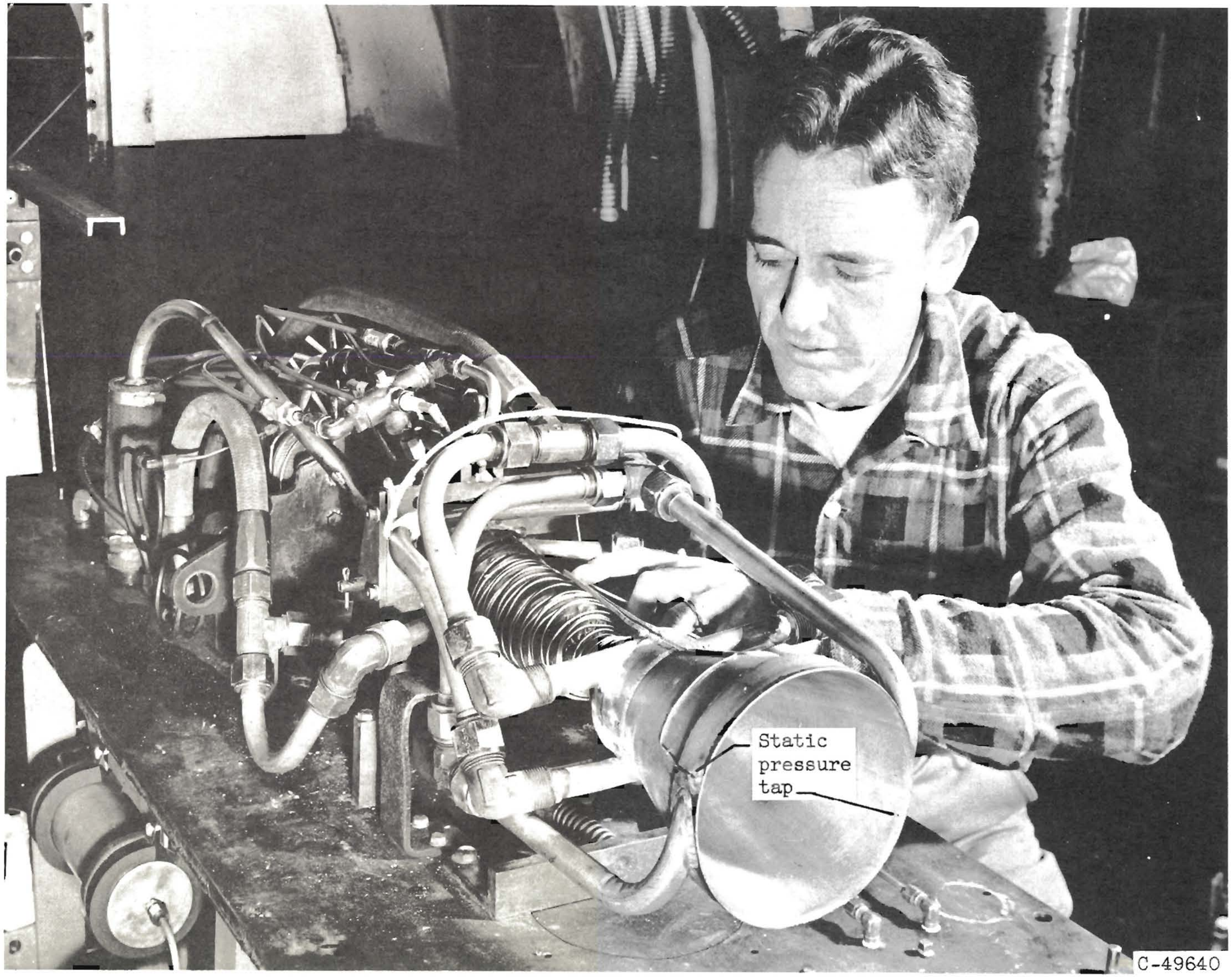

Figure 2. - Rocket installation with engine with area-ratio-25 nozzle.

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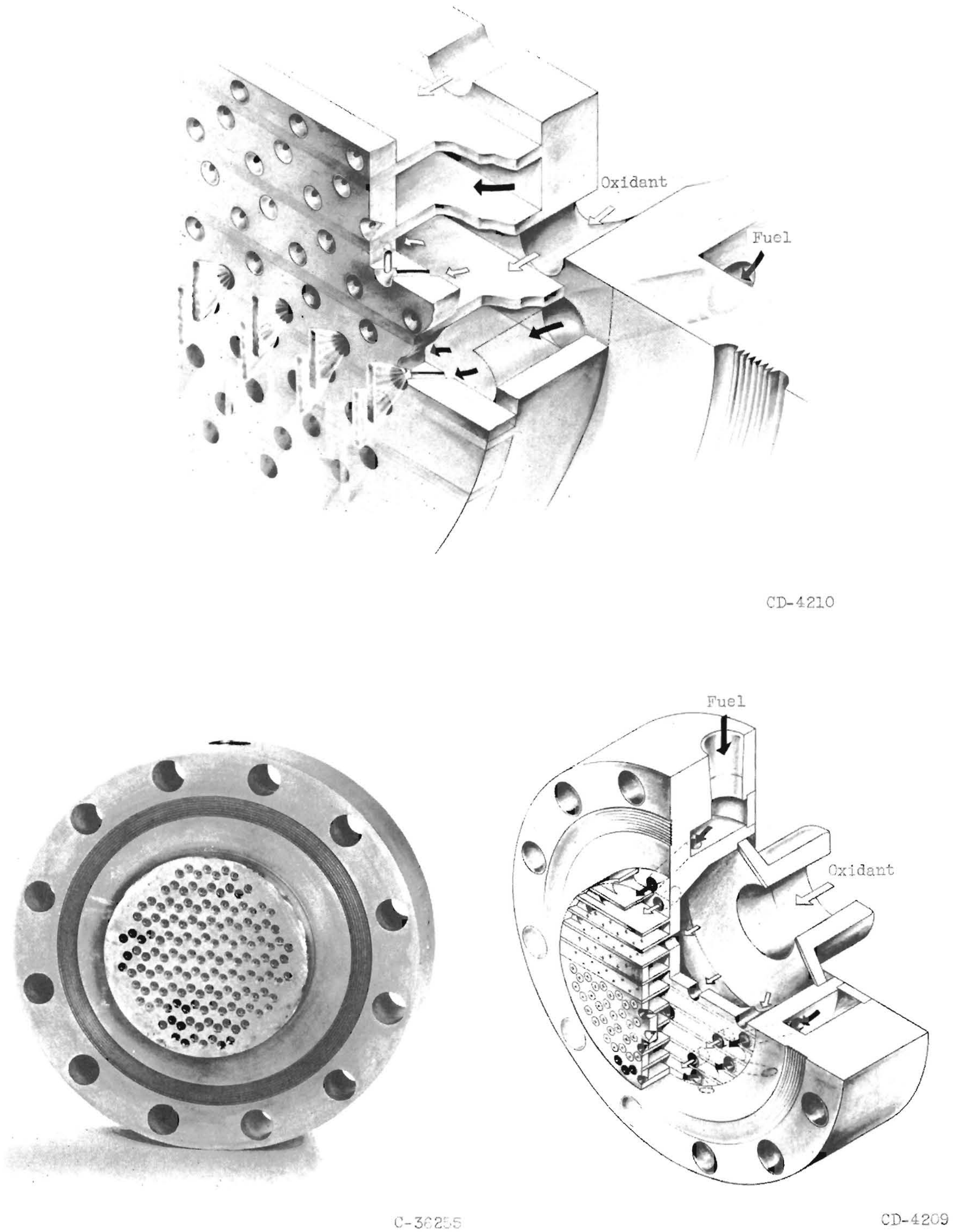

Figure 3. - Like-on-like propellant injector with holes arranged in rows on flat face. Fuel jets,   pairs; oxidant jets, 70 pairs.

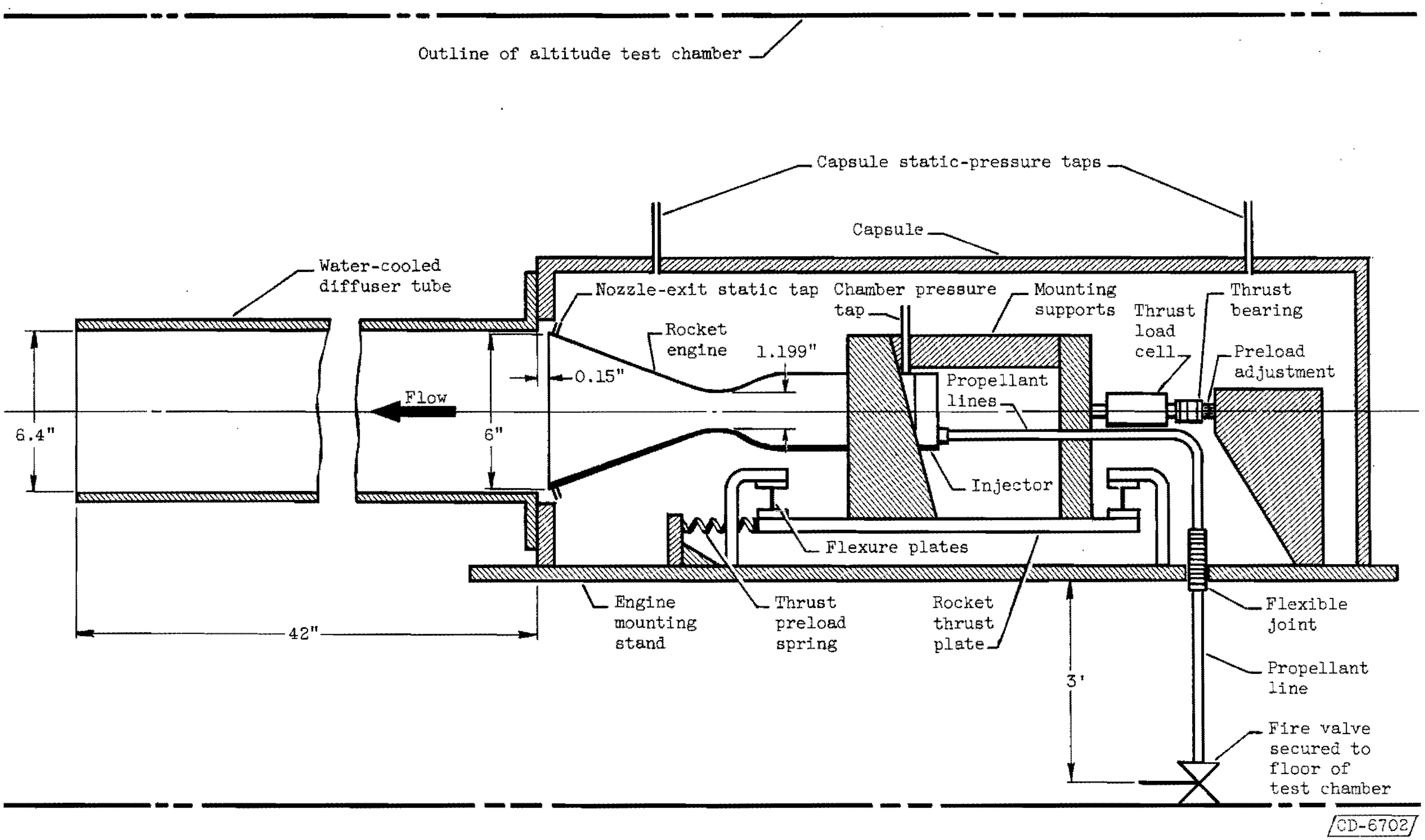

Figure 4. - Schematic diagram of rocket installation showing diffuser and thrust system.

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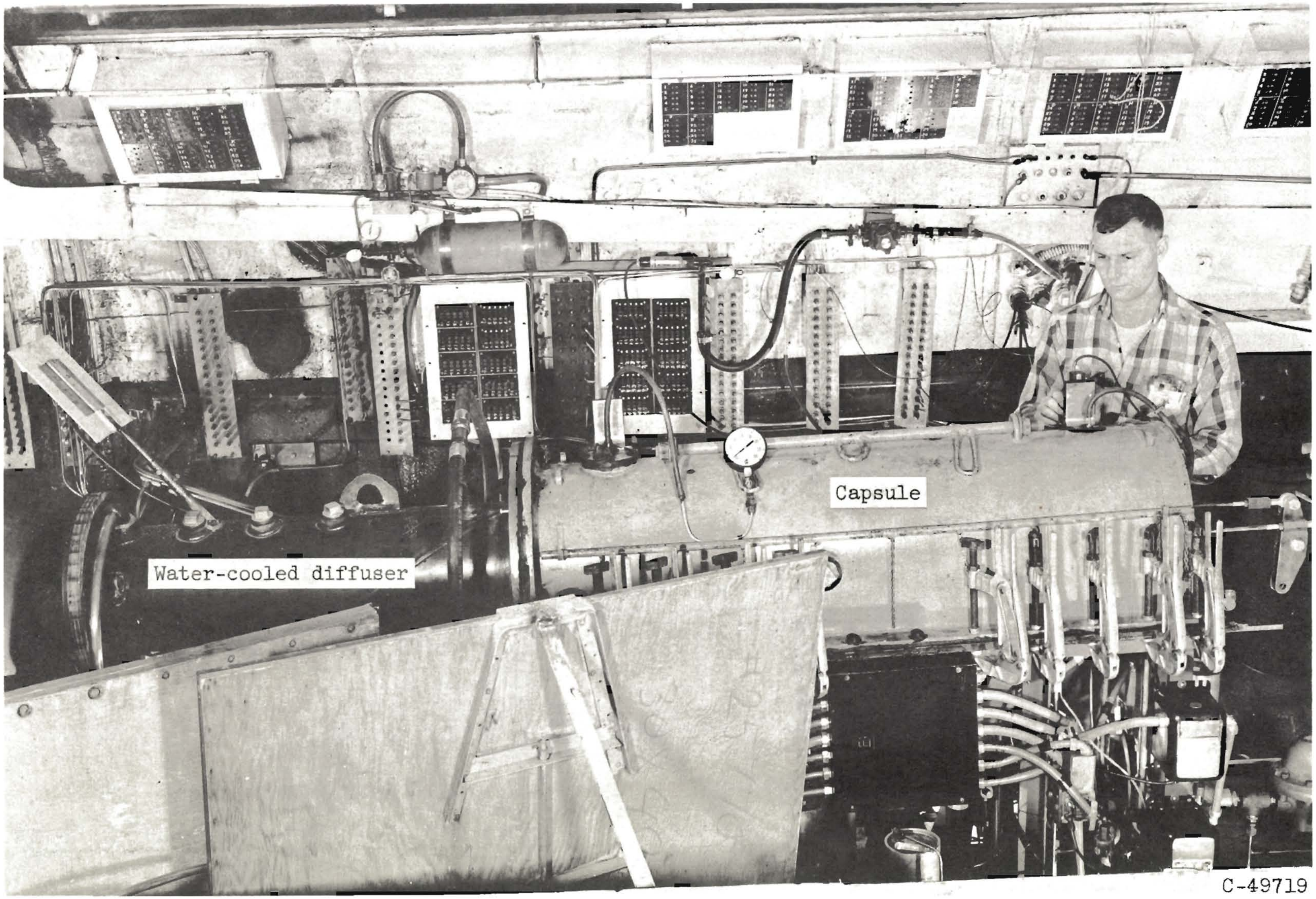

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Figure 5. - Rocket installation with diffuser and capsule in place.

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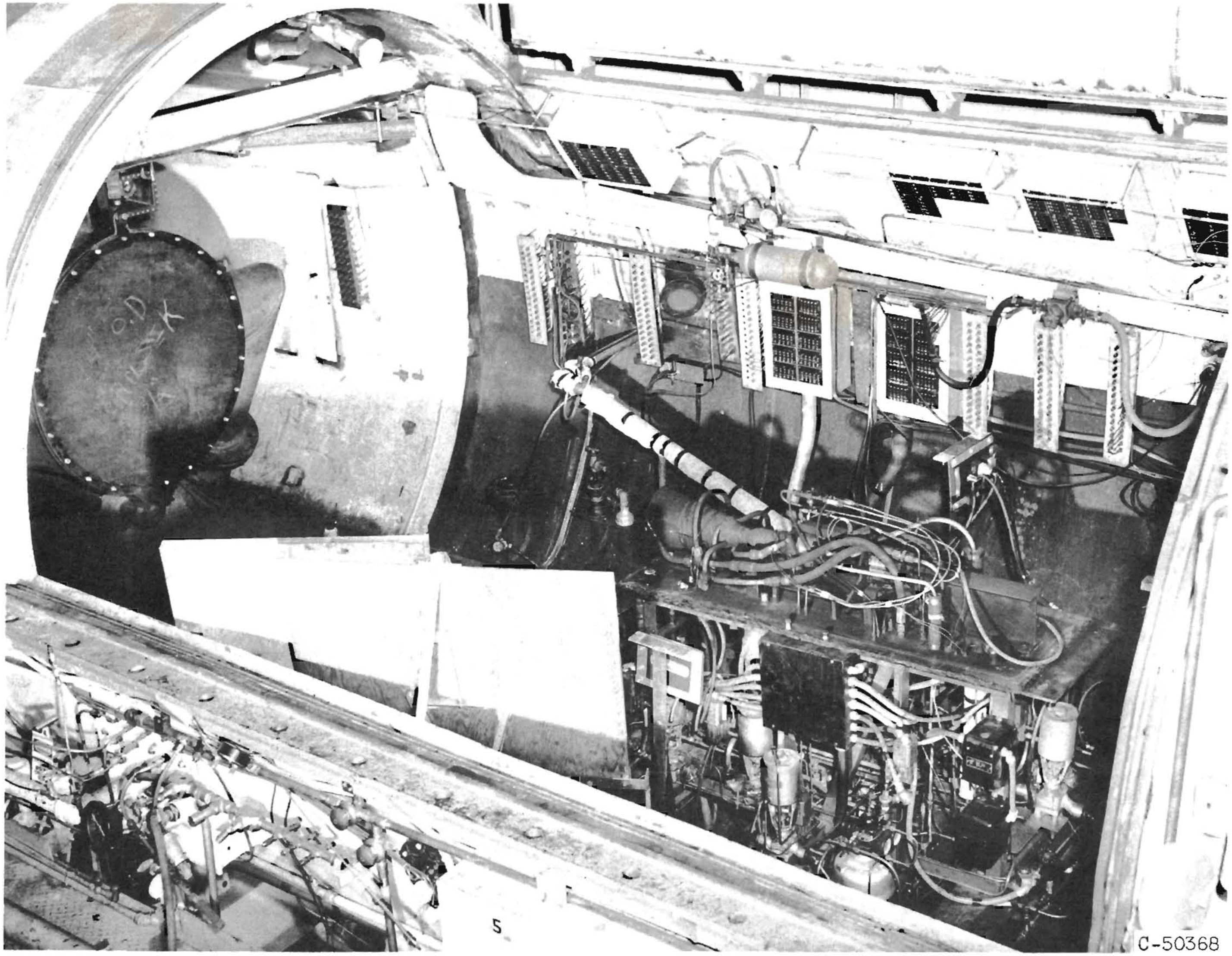

Figure 6. - Rocket installation in 10-foot altitude test chamber with test engine.

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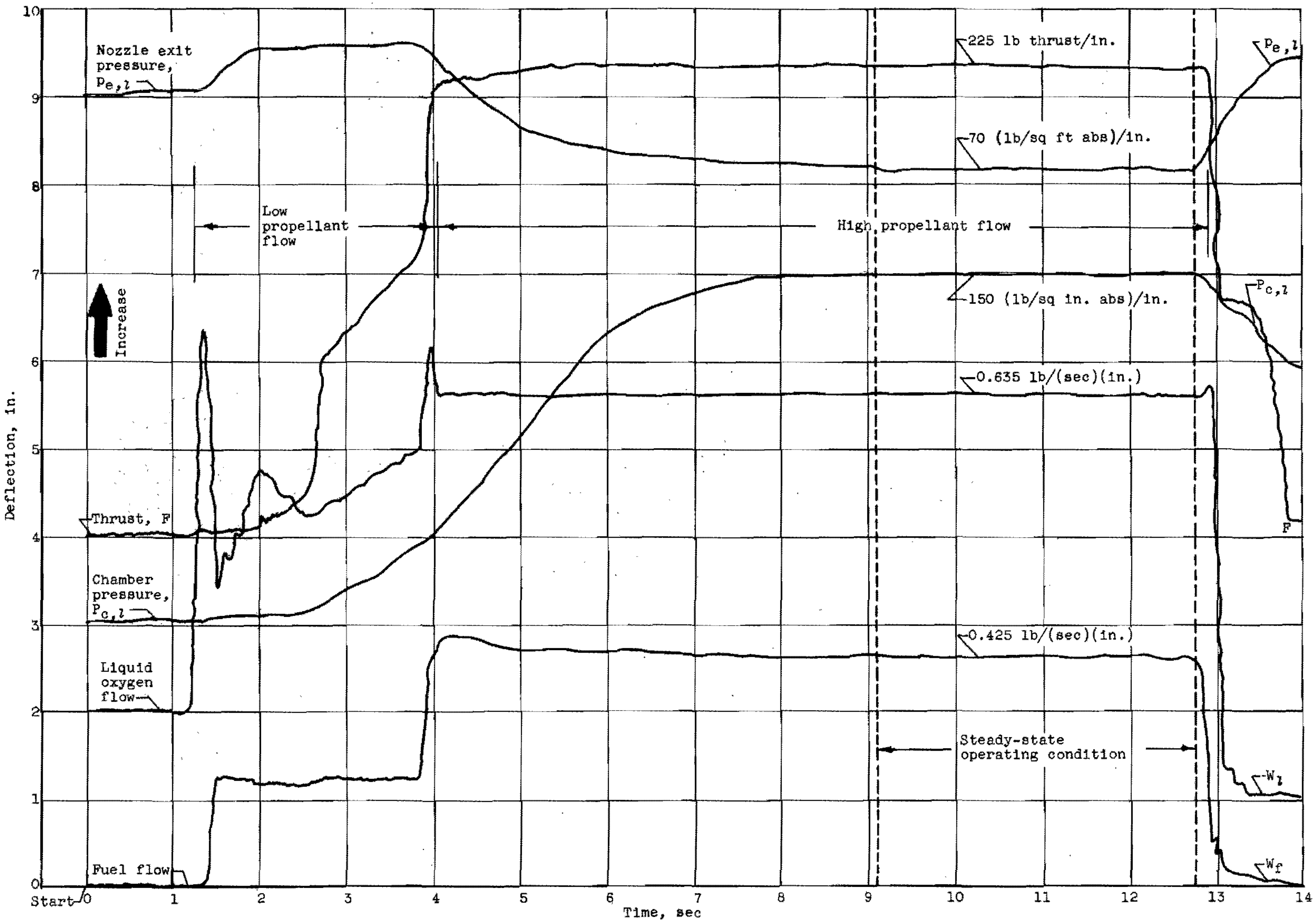

Figure 7. - Sample trace of rocket firing without diffuser installed.

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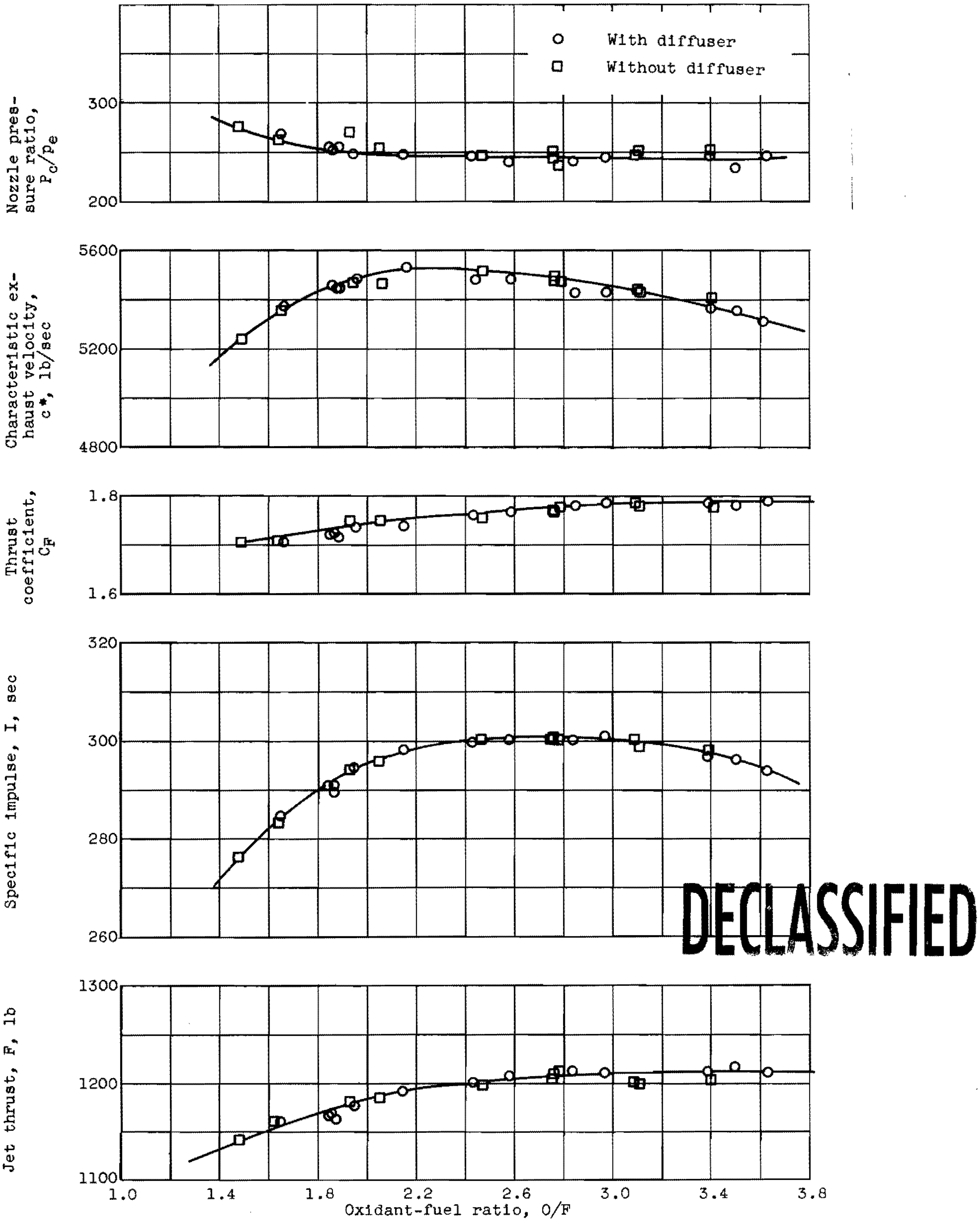

Figure 8. - Altitude performance of rocket engine with nozzle area ratio of 25. Simulated altitude, 47,000 feet.

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