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A PRELIMINARY INVESTIGATION OF OXIDIZER-RICH OXYGEN-HYDROGEN COMBUSTION CHARACTERISTICS

by Curtis R. Bailey

George C. Marshall Space Flight Center
Huntsville, Ala.

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DEFINITION OF SYMBOLS

Symbol	Definition
A_t	Nozzle Throat Area
C^*	Characteristic Velocity
F	Thrust
g	Acceleration due to Gravity
L^*	Characteristic Chamber Length
P_c	Combustion Chamber Pressure
ΔP_f	Differential Fuel Injection Pressure
ΔP_o	Differential Oxidizer Injection Pressure
Q	Nozzle Coolant Heating Rate
t	Test Duration
W_f	Fuel Flowrate
W_o	Oxidizer Flowrate
η	Efficiency

CONVERSION FACTORS

Pound force = 4.4482 newton

Pound force/inch^2 = 6890 newton/meter^2

Pound mass = .4536 kilogram

Inch = 2.54×10^{-2} meter

Gallon (U. S. Liq.) = 3785.4×10^{-6} meter^3

Foot/second = .3048 meter/second

Btu/second = 1056 joule/second

A PRELIMINARY INVESTIGATION OF OXIDIZER-RICH OXYGEN-HYDROGEN COMBUSTION CHARACTERISTICS

SUMMARY

The operating characteristics of oxygen-hydrogen combustion were investigated over a propellant mixture ratio (O/F) band of 20 to 150. Firings were conducted in a 3600 pound thrust combustor at a chamber pressure of 1000 psia. Wedges fabricated from Inconel-X, Rene-41, and Waspalloy were placed in the exhaust of the combustor and subjected to the hot gases ranging in mixture ratio from 75 to 150. There were no significant heating problems with any of the combustor components. There was no erosion or melting of the test wedges except during a mixture ratio shift through stoichiometric in the start transient of one firing. Maximum characteristic velocity efficiencies of approximately 85 percent were achieved at propellant mixture ratios of 23 and 144. A minimum value of approximately 70 percent was observed at a mixture ratio of 70.

INTRODUCTION

Recent studies indicate that advanced rocket engine systems using oxygen and hydrogen propellants will operate at combustion chamber pressures of approximately 3000 psia. One engine concept being considered is the dual preburner cycle. The components for this engine consist of a main thrust chamber, fuel and oxidizer pumps, and two preburners that furnish the working fluids to drive the turbines. The preburners for the oxidizer and fuel pump turbines are operated oxidizer-rich and fuel-rich respectively. After passing through the turbines, the hot gases are ducted to the main combustion chamber and are burned at near stoichiometric mixture ratio. Several advantages are claimed for this cycle: Injection of the main combustor propellants as gases contributes to combustion stability; specific impulse is increased because the turbine exhaust gases pass through the main chamber instead of being ducted overboard; and no positive seal is required between fuel-rich turbine gases and the oxidizer pump.

Serious consideration of the dual preburner cycle required the demonstration of oxidizer-rich combustion and verification of the ability of turbine blade materials to withstand the resulting environment. This investigation was conducted to provide a preliminary demonstration of these effects and to identify possible problems. Due to facility limitations, the nominal chamber pressure was limited to 1000 psia. The range of propellant mixture ratios (O/F) was approximately 20 to 150.

The author acknowledges the efforts of the Advanced Technology Test Section, Test Laboratory, in conducting this program.

TEST CONFIGURATION

Preburner and Injector

The model preburner used in all firings had an uncooled copper combustion chamber and a water-cooled copper nozzle. The chamber was equipped with removable 7.6-cm (3.0-in.) sections to allow variation of characteristic length (L^*) between runs. The assembly is shown in Figure 1. Nominal thrust rating of the chamber at 6.89 MN/m^2 (1000 psia) chamber pressure was 16 000 newtons (3600 lb).

The chamber diameter was 9.49 cm (3.73 in.) and the nozzle throat diameter was 4.45 cm (1.75 in.), with a corresponding contraction ratio of 4.5. Characteristic length (L^*) was 1.20 meters (47.5 in.) with one 7.6-cm (3.0-in.) chamber section installed and 1.55 meters (61.0 in.) with two sections installed. Nozzle exit diameter was 13.1 cm (5.15 in.) providing an expansion ratio of 8.65.

A typical injector used for all tests is shown in Figure 2. The concentric tube injector was operated with oxidizer surrounding the fuel. The design used 69 stainless steel tubes welded to the injector center body using the TIG(tungsten-inert gas) welding process. The injector face was copper and the remainder of the injector was 300-series stainless steel.

Except for the asbestos gaskets used to seal the chamber spacers, rubber O-rings were used throughout the chamber assembly. The injector assembly used a teflon fuel cavity seal, and a metal O-ring hot gas seal. Rubber O-rings, initially used for the oxidizer manifold were incompatible with LOX, and the O-ring material was changed to silicon.

Turbine Test Materials

Wedge-shaped specimens of materials that could be used in LOX-rich preburner turbines were fabricated of Inconel-X, Waspalloy, and Rene 41. A typical wedge is shown in Figure 3. The specimens were mounted singly in the exhaust stream of the combustor with the leading edge of the wedge 3.8 cm (1.5 in.) downstream of the nozzle exit.

TEST INSTALLATION AND OPERATION

The test installation used for all firings was located at Vehicle Components and Sub-System Branch Test Facilities, George C. Marshall Space Flight Center. The thrust stand installation is pictured in Figure 4, and Figure 5 shows the turbine material test installation.

A schematic of the cooling water and propellant systems is shown in Figure 6. Liquid oxygen was supplied from a 1.89 m^3 (500 gal) tank that was pressurized with gaseous nitrogen. Gaseous hydrogen, the fuel used in most of the tests, was supplied from a 34.4 MN/m^2 (5000 psi), 2550 scm (90 000 scf) storage battery. Liquid hydrogen, used for the fuel in later tests, was supplied from a 7.57 m^3 (2000 gal) tank that was pressurized with the gaseous hydrogen storage battery.

Ignition was initiated by an electric spark harness that was inserted through the nozzle throat into the combustion chamber approximately 2.5 cm (1 in.) from the injector face. The harness installation, shown in Figure 1, consisted of two wires supported in a 0.64-cm (0.25-in.) copper tube that was insecurely fastened to the nozzle. A spark between the uninsulated wire ends ignited the propellant mixture. The harness was then blown out of the chamber by the combustion gases.

To minimize the potential start problems associated with LOX-rich combustion, a rather elaborate start sequence was devised. At firing command, the electric spark was energized and the propellant start valves were sequenced open with a 0.070-second fuel lead. The start valve lines were orificed to allow a chamber pressure buildup of approximately 1.03 MN/m^2 (150 psia) during the ignition phase. If a chamber pressure of 0.69 MN/m^2 (100 psia) was not achieved within 0.75 second after firing command, automatic cutoff was initiated. If 0.69 MN/m^2 (100 psia) chamber pressure was achieved, the main propellant shutoff valves were signaled to open with a 0.30-second fuel lead. If a chamber pressure of 4.13 MN/m^2 (600 psia) was not achieved in 1.90 seconds after firing command, a high chamber pressure switch initiated cutoff. This ignition sequence proved satisfactory during checkout and performance tests, but was not satisfactory for the turbine material tests. The main propellant fuel lead created a mixture ratio transient through the stoichiometric range that caused severe burning of a CRES checkout test wedge. The sequence was, therefore, modified to give a 0.060-second oxidizer lead, and no further problems were encountered.

For the last four firings, the start sequence was simplified to check the sensitivity of oxidizer-rich ignition. The start valves were eliminated and ignition was accomplished by activating the spark and opening the main propellant valves with a 0.060-second oxidizer lead. Ignition by this method proved to be reliable although not quite as smooth as when the start valves were used. Preburner cutoff was accomplished by closing the main propellant valves in sequence to provide an oxidizer-rich condition. Oscillograms of both the ignition and cutoff sequences are shown in Figure 7.

INSTRUMENTATION AND DATA REDUCTION

Instrumentation was provided for the measurement of thrust, oxidizer and fuel flowrates, oxidizer and fuel injection pressures, chamber pressure, nozzle cooling water flowrate and differential temperature. In addition, two thermocouples were installed in the apex of each test material wedge to measure metal temperature and subsequently determine the resistance of the material to oxidation. A single thermocouple was installed in the chamber spacer approximately 0.127 cm (0.050 in.) from the inner wall to measure chamber skin temperature. Chromel-alumel thermocouples, with an accuracy of approximately ± 1.5 percent, were used for these measurements.

Thrust measurements were made with a Revere Corporation load cell that restrained the movement of a ball bearing supported thrust parallelogram. The load cell was electrically calibrated using an integral shunt. Measurement accuracy was within ± 3.0 percent.

Liquid propellant flow measurements were made with water-calibrated Potter turbine flowmeters that provided a volumetric flow measurement. Mass flowrate was then calculated by multiplying volumetric flowrate by propellant density. Oxidizer density was determined from a temperature measurement taken by a copper-constantan thermocouple. Liquid hydrogen density was determined from measurements taken by a Rosemount resistance thermometer and a Dynisco pressure transducer. Overall accuracy of the liquid flow measurements was within ± 1.5 percent. The gaseous hydrogen flowmeter was a Potter turbine flowmeter that was calibrated with gaseous hydrogen at the University of Colorado Engineering Experimental Station. Gaseous hydrogen density was determined from measurements taken by a chromel-alumel thermocouple and a Dynisco pressure transducer. A gradual shift in calibration occurred with this flowmeter during the test series, and the meter was recalibrated at George C. Marshall Space Flight Center. It was then necessary to apply correction factors to the measured flowrates. Because of the uncertainties involved, the overall accuracy of gaseous hydrogen flow measurement was only ± 5 percent.

Propellant injection and chamber pressure measurements were made with Dynisco strain-gage transducers. The injection pressure transducers were flush mounted in the propellant feed lines and had a frequency response of approximately 10 000 Hz (cps). The chamber pressure was monitored through a 38-cm (15-in.) length of 0.64-cm (0.25-in.) tubing limiting the frequency response to approximately 200 Hz (cps). A Wianko pressure calibration system was used. The over-all measurement accuracy was witnin ± 1.5 percent.

All data were recorded on a Systems Engineering Laboratories digital instrumentation system, and the pressures and flowrates were also recorded on a Consolidated Electronics Corporation oscillograph.

The preburner performance was based on calculated values of characteristic velocity (C*). Characteristic velocity is computed as follows:

$$C^* = \frac{P_c A_t g}{W}$$

where: C^* = Characteristic Velocity
P_c = Chamber Pressure
A_t = Nozzle Throat Area
g = Gravitational Acceleration
W = Propellant Flowrate

The chamber pressure term that appears in this equation is the isentropic stagnation pressure at the throat of the nozzle. The pressure measurements used for characteristic velocity calculations were the chamber static pressure measurements taken near the injector. The ratio of these two pressures approaches 1.0 as nozzle contraction ratio is increased. Since the contraction ratio of the nozzle used was large (4.55), the two pressures were assumed to be equal.

For tests that included the measurement of nozzle stagnation temperature (turbine material wedge tests), characteristic velocity was also calculated as a function of temperature. This was computed as follows:

$$C* = \frac{\sqrt{RT_c}}{\Gamma}$$

where: R = Gas Constant

T_c = Combustion Temperature

$$\Gamma = \gamma\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma + 1}{2(\gamma - 1)}}$$

γ = Specific Heat Ratio

DISCUSSION OF TEST RESULTS

LOX/GH_2 Ignition and Performance Tests

Initial tests were conducted to determine the performance and operating characteristics of oxidizer-rich LOX/GH_2 mixtures and to determine if the spark harness method was adequate for ignition. This phase consisted of runs 172-9 through 172-17. Test durations ranged from 3 to 10 seconds and O/F ratios varied from 22 to 85. Characteristic chamber length for these firings was 1.20 meters (47.5 in.).

There were no problems with the ignition system, and the staged propellant valve sequence provided a smooth chamber pressure buildup. Neither the chamber pressure nor the injection pressure traces showed any indication of instability. Tabulated performance data for these and subsequent firings are presented in Tables I and II. Theoretical performance and thermodynamic data calculated by the method described in the Reference 1 are shown in Figure 8.

Post-test examination for firing No. 172-10 revealed that the injector LOX O-ring grooves were severely burned. It was discovered that BUNA-N O-rings, which are not compatible with LOX, had been inadvertently used.

Silicon O-rings were installed, and no further problems of this type were encountered.

Test No. 172-24 was conducted to check a modified start sequence. As previously discussed, the propellant start valves were eliminated and ignition was obtained by turning on the spark and opening the main propellant valves with a 0.060-second oxidizer lead.

Turbine Material Tests

This phase of the program consisted of tests No. 172-18 through 172-23 and was conducted to determine the effects of oxidizer-rich combustion products on typical turbine materials. The firings were conducted with a characteristic chamber length of 1.55 m (61.0 in.) and mixture ratios varying from 76 to 193.

Test No. 172-18 was conducted using an uninstrumented stainless steel wedge. The wedge was severely burned shortly after ignition, and the test was aborted at 3.4 seconds. The overheating was probably caused by a mixture ratio transient through stoichiometric during the start sequence. The sequence was modified to give an oxidizer-rich start and was successfully tested in run No. 172-19. No turbine test material was used.

Test No. 172-20 was conducted using an Inconel-X wedge and a propellant mixture ratio of 184. There was no damage to the wedge after a test duration of 10 seconds. The thermocouple leads were broken after approximately six seconds due to exhaust gas turbulence behind the wedge.

Since overheating did not appear to be a problem, it was decided to extend test duration to thirty seconds and to operate at three different mixture ratios during each run. This procedure was used in tests No. 172-21 through 172-23, and it was accomplished by increasing the tank pressure every ten seconds. The turbine material specimens tested were Rene 41, Waspalloy and Inconel-X. There was no test damage during any of these firings (except for broken thermocouple leads during run No. 172-21). Post-test inspection for these firings revealed a thin layer of ice inside the combustion chamber approximately 2.5 cm (1 in.) downstream from the injector face. This was attributed to the rapid chilldown resulting from the oxidizer-rich cutoff. The preburner exhaust was very cloudy due to condensation of steam during expansion in the nozzle.

LOX/LH_2 Ignition and Performance Tests

Test runs No. 172-25 through 172-27 were conducted to determine if the use of liquid hydrogen would give the same results that had been obtained using gaseous hydrogen. The start sequence used in test No. 172-24 was also used in these firings.

Because of difficulties with the liquid hydrogen flowmeter, the fuel flowrate measurements were generally erratic. There were no problems with either ignition or overheating. The test results were similar to those obtained using gaseous hydrogen.

Combustion and Heat Transfer Characteristics

Characteristic velocity data points and the theoretical equilibrium curve are shown in Figure 9. Since the accuracy of the liquid hydrogen flowrate measurements was questionable, the data for the runs which used liquid hydrogen were omitted.

The change in chamber length had no appreciable effect on characteristic velocity, and the experimental data points were approximately asymptotic to the theoretical as mixture ratio increased. The trend in characteristic velocity efficiency is clearly shown in Figure 10. As indicated by the legend, most of the data points were calculated by means of chamber pressure. Characteristic velocity efficiencies were also calculated using turbine material wedge temperatures. As shown, the temperature efficiencies were even higher than the efficiencies based on pressure, but served to verify the general magnitude and slope of the efficiency curve.

Combustor and turbine material skin temperatures for runs 172-22 and 172-23 are presented in Figures 11 and 12, respectively. Although the wedge temperatures did not stabilize during the ten seconds allotted for each mixture ratio, the thirty-second runs were sufficient to indicate that oxidation was not a problem at near theoretical combustion temperatures.

Heating rate of the nozzle cooling water as a function of propellant mixture ratio is presented in Figure 13. Considerable data scatter is seen, but the general trend of decreasing heating rate with increasing mixture ratio is evident.

CONCLUSIONS

This program demonstrated that oxidizer-rich combustion using oxygen and hydrogen propellants is feasible over a propellant mixture (O/F) range of approximately 20 to 150.

Electric spark ignition was reliable with and without the use of propellant prevalves.

Neither the turbine test materials nor the preburner assembly overheated or oxidized during oxidizer-rich ignition or main stage. Mixture ratio transients through stoichiometric during ignition caused burning of the turbine test material.

Maximum characteristic velocity efficiencies of approximately 85 percent were achieved at propellant mixture ratios of 23 and 144. A minimum value of approximately 70 percent was observed at a mixture ratio of 70.

TABLE I. PERFORMANCE DATA (ENGLISH SYSTEM)

RUN NO.	P_c psia	W_f lb/sec	ΔP_f psid	W_0 lb/sec	ΔP_0 psid	W_0/W_f	F lb.	t sec	C* ft/sec	η_c* %	Q btu/sec	COMMENTS
172-9	945	0.71	344	16.57	74	23.3	3286	3.0	4159	82.8	472.9	Ignition and Performance Test
172-10	961	0.66	257	19.55	102	29.3	3498	3.0	3647	79.6	465.5	Injector Burned - O-Ring and LOX Incompatible
172-11		NOT APPLICABLE TO THIS REPORT										
172-12	989	0.63	176	21.03	36	33.4	3490	3.0	3502	80.2	335.2	LOX/ GH_2 Performance Test
172-13	934	0.63	164	26.89	125	42.5	3462	2.7	2602	66.1	280.7	LOX/ GH_2 Performance Test
172-14	961	0.48	91	32.74	115	68.3	3464	5.0	2222	70.5	223.5	LOX/ GH_2 Performance Test
172-15	898	0.42	136	34.58	203	82.3	3407	5.0	1968	68.0	214.2	LOX/ GH_2 Performance Test
172-16		ENGINE DID NOT REACH MAINSTAGE										LOX Valve Malfunction
172-17	1012	0.47	118	35.08	173	74.7	3445	10.0	2152	71.0	136.8	LOX/ GH_2 Performance Test
172-18	979	0.43	74	38.12	155	88.7	3578	3.4	1948	70.0	142.8	CRES Material Test - Specimen Burned
172-19	939	0.29	64	39.40	148	136.0	3492	5.0	1815	81.1	128.5	Ignition Sequence Test - LOX Rich Start
172-20	959	0.28	66	38.92	86	139.0	3627	10.0	1877	84.9	129.5	INCONEL-X Material Test - No Damage
172-21A	955	0.40	81	37.38	168	93.5	3615	0 to 10	1930	70.7	78.5	RENE-41 Material Test - No Damage
172-21B	927	0.45	152	36.19	221	80.5	3711	10 to 20	1940	66.0	91.6	RENE-41 Material Test - No Damage
172-21C	1006	0.48	105	35.37	164	73.6	3792	20 to 30	2150	70.5	105.0	
172-22A	959	0.32	67	38.11	98	119.0	3561	0 to 10	1915	79.8	65.5	WASPALLOY Material Test - No Damage
172-22B	955	0.41	134	36.59	190	89.2	3744	10 to 20	1979	70.7	78.5	WASPALLOY Material Test - No Damage
172-22C	953	0.43	174	35.78	212	83.1	3838	20 to 30	2018	69.8	105.0	" " " " "
172-23A	958	0.27	77	38.98	159	144.0	3469	0 to 10	1871	86.3	79.8	INCONEL - X Material Test - No Damage
172-23B	952	0.29	129	38.82	197	134.0	3575	10 to 20	1868	83.0	92.5	INCONEL-X Material Test - No Damage
172-23C	948	0.34	139	38.62	198	113.0	3614	20 to 30	1867	75.3	92.5	LOX/ GH_2 Ignition Test - No Engine Start Valves
172-24	957	0.29	73	39.08	148	135.0	3316	12	1864	82.8	54.5	LOX/ GH_2 Ignition Test - No Engine Start Valves
172-25	999	—	114	30.68	112	≈102	3477	17	—	—	284.0	LOX/ LH_2 Performance Test - LH_2 Flowmeter Malfunction
172-26	1039	—	73	32.66	116	≈115	3537	10	—	—	188.0	LOX/ LH_2 Performance Test - LH_2 Flowmeter Malfunction
172-27	1132	≈.361	80	32.68	111	≈ 91	3996	10	2633	≈95.4	238.0	LOX/ LH_2 Performance Test

TABLE II. PERFORMANCE DATA (INTERNATIONAL SYSTEM)

RUN NO.	P_c N/m² x 10^{-6}	W_f kg/sec	ΔP_f N/m² x 10^{-6}	W_o kg/sec	ΔP_o N/m² x 10^{-6}	W_o/W_f	F Newton x 10^{-4}	t sec	C* m/sec	η_c* %	Q Joule/sec x 10^{-5}	COMMENTS
172-9	6.511	0.322	2.370	7.50	0.510	23.3	1.462	3.0	1280	82.8	4.986	Ignition and Performance Test
172-10	6.621	0.299	1.771	8.86	0.703	29.3	1.556	3.0	1110	79.6	4.908	Injector Burned - O-Ring and LOX Incompatible
172-11		NOT APPLICABLE TO THIS REPORT										
172-12	6.821	0.287	1.213	9.52	0.248	33.4	1.552	3.0	1070	80.2	3.534	LOX/GH_2 Performance Test
172-13	6.442	0.287	1.130	12.20	0.861	42.5	1.540	2.7	794	66.1	2.960	LOX/GH_2 Performance Test
172-14	6.621	0.217	0.627	14.80	0.792	68.3	1.541	5.0	678	70.5	2.356	LOX/GH_2 Performance Test
172-15	6.187	0.190	0.937	15.70	1.399	82.3	1.516	5.0	600	68.0	2.258	LOX/GH_2 Performance Test
172-16		ENGINE DID NOT REACH MAINSTAGE										LOX Valve Malfunction
172-17	6.973	0.213	0.813	15.90	1.192	74.7	1.532	10.0	656	71.0	1.442	LOX/GH_2 Performance Test
172-18	6.745	0.195	0.510	17.30	1.068	88.7	1.592	3.4	594	70.0	1.506	CRES Material Test - Specimen Burned
172-19	6.470	0.131	0.441	17.80	1.020	136.0	1.553	5.0	554	81.1	1.355	Ignition Sequence Test - LOX Rich Start
172-20	6.607	0.127	0.455	17.60	0.593	139.0	1.613	10.0	572	84.9	1.365	INCONEL-X Material Test - No Damage
172.21A	6.580	0.181	0.558	16.90	1.158	93.5	1.608	0 to 10	589	70.7	0.840	RENE-41 Material Test - No Damage
172-21B	6.387	0.204	1.047	16.40	1.523	80.5	1.651	10 to 20	592	66.0	0.966	RENE-41 Material Test - No Damage
172-21C	6.931	0.217	0.723	16.00	1.130	73.6	1.687	20 to 30	656	70.5	1.110	RENE-41 Material Test - No Damage
172-22A	6.608	0.145	0.462	17.30	0.675	119.0	1.584	0 to 10	585	79.8	0.691	WASPALLOY Material Test - No Damage
172-22B	6.580	0.186	0.923	16.60	1.309	89.2	1.665	10 to 20	604	70.7	0.828	WASPALLOY Material Test - No Damage
172-22C	6.566	0.195	1.199	16.20	1.461	83.1	1.707	20 to 30	615	69.8	1.109	WASPALLOY Material Test - No Damage
172-23A	6.601	0.122	0.531	17.70	1.096	144.0	1.543	0 to 10	571	86.3	0.842	INCONEL-X Material Test - No Damage
172-23B	6.559	0.131	0.889	17.60	1.357	134.0	1.590	10 to 20	570	83.0	0.976	INCONEL-X Material Test - No Damage
172-23C	6.532	0.154	0.958	17.50	1.364	113.0	1.607	20 to 30	569	75.3	0.976	LOX/GH_2 Ignition Test - No Engine Start Valves
172-24	6.594	0.131	0.503	17.70	1.020	135.0	1.475	12	568	82.0	0.575	LOX/GH_2 Ignition Test - No Engine Start Valves
172-25	6.883	—	0.785	13.92	0.772	≈102	1.547	17	—	—	3.000	LOX/LH_2 Performance Test - LH_2 Flowmeter Malfunction
172-26	7.159	—	0.503	14.81	0.799	≈105	1.573	10	—	—	1.987	LOX/LH_2 Performance Test - LH_2 Flowmeter Malfunction
172-27	7.799	≈.164	0.551	14.82	0.765	≈ 91	1.777	10	811	≈95.4	2.510	LOX/LH_2 Performance Test

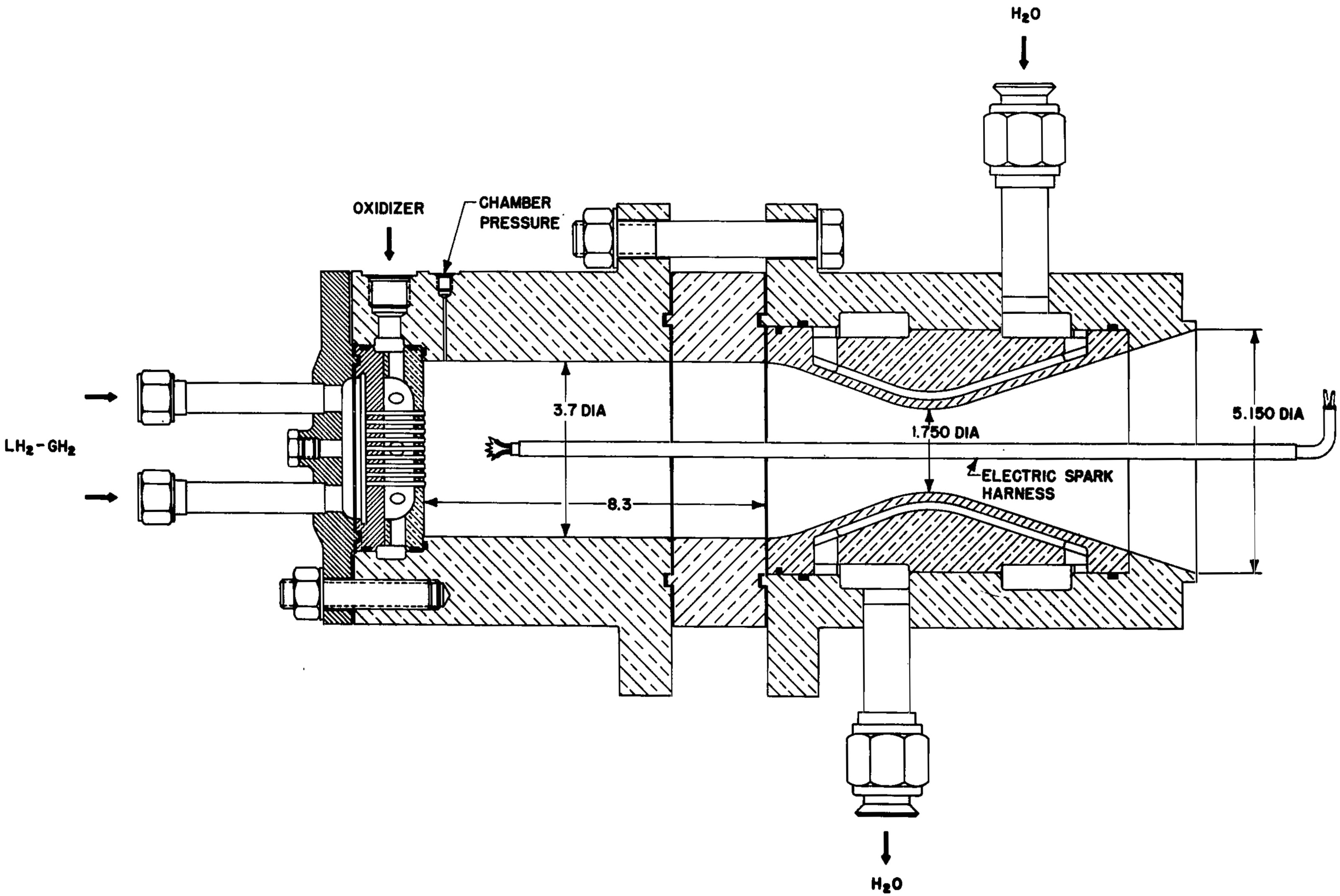


FIGURE 1. PREBURNER ASSEMBLY
(English System Dimensions)

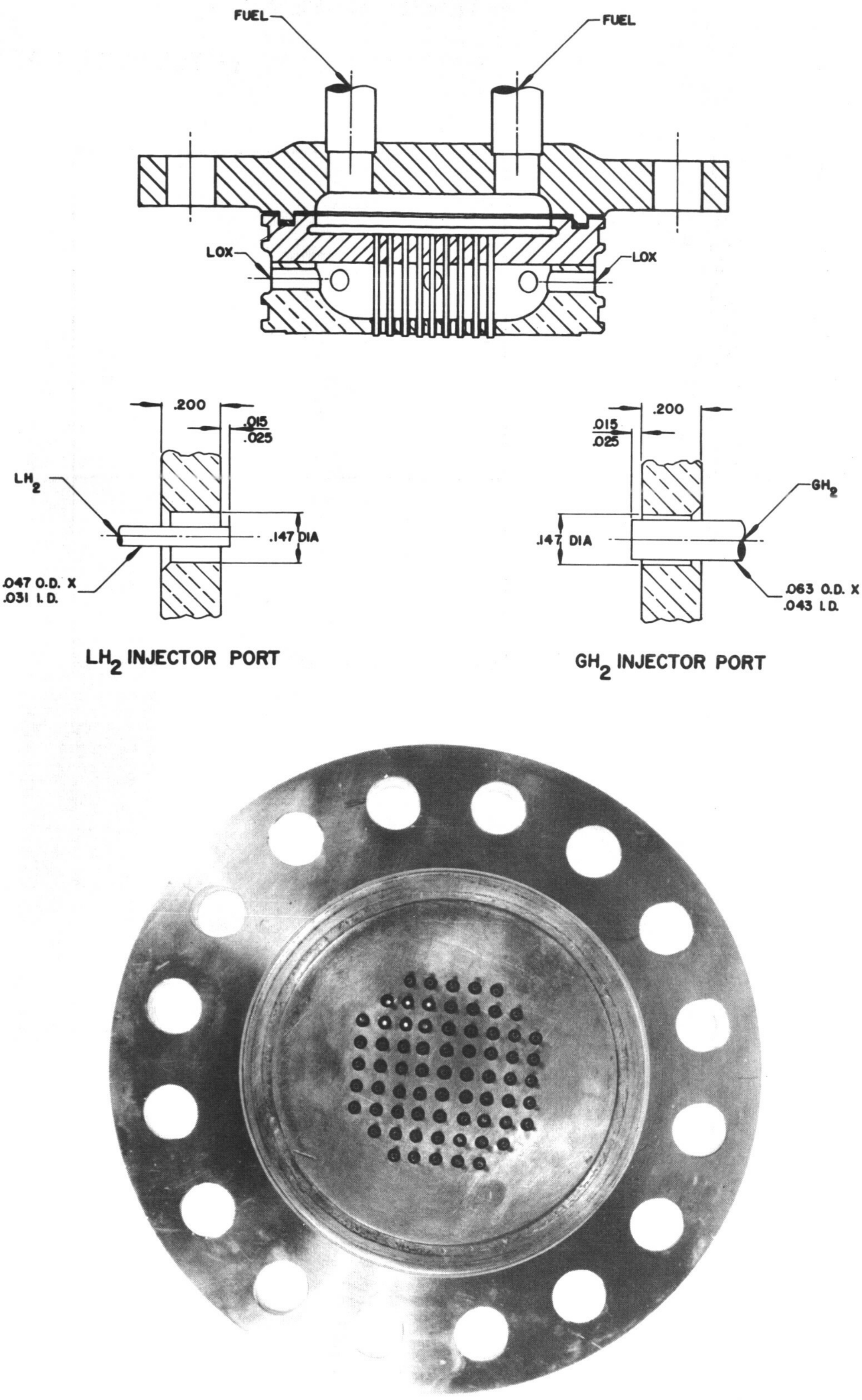
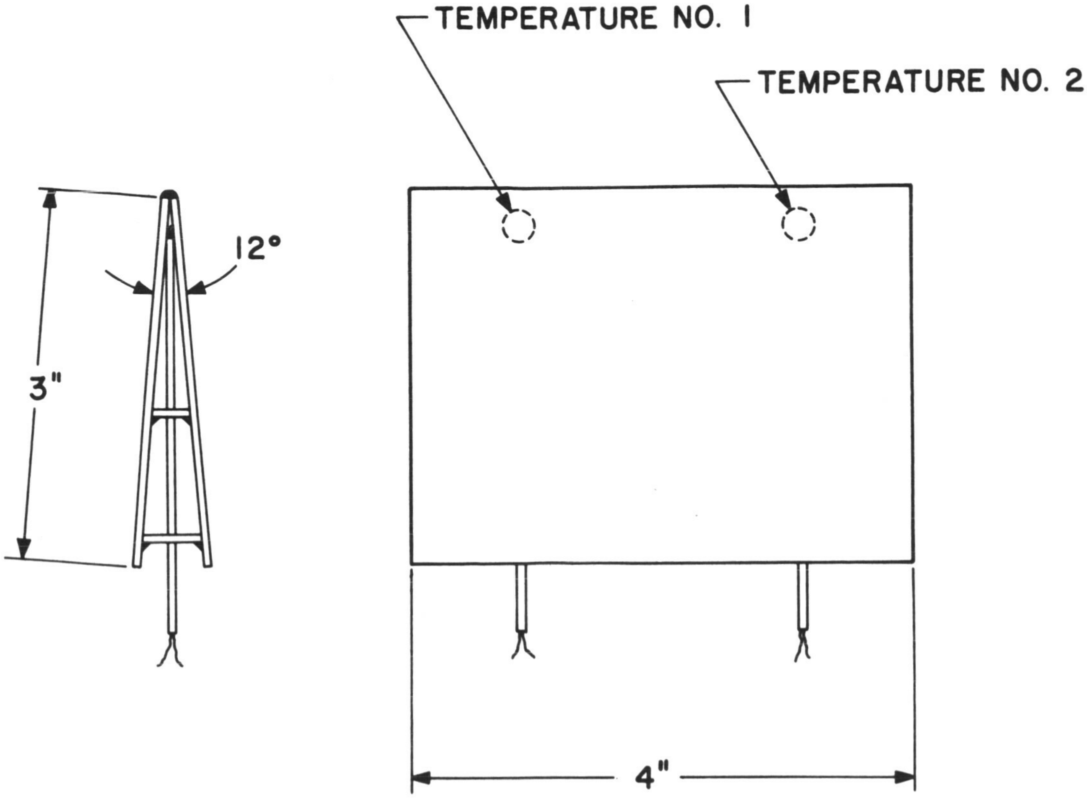


FIGURE 2. PREBURNER INJECTOR
(English System Dimensions)



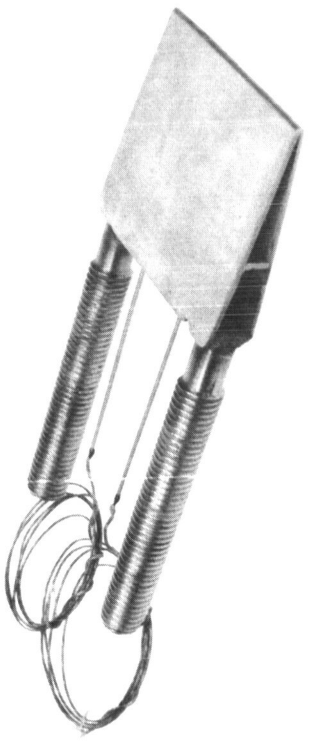


FIGURE 3. TURBINE TEST MATERIAL

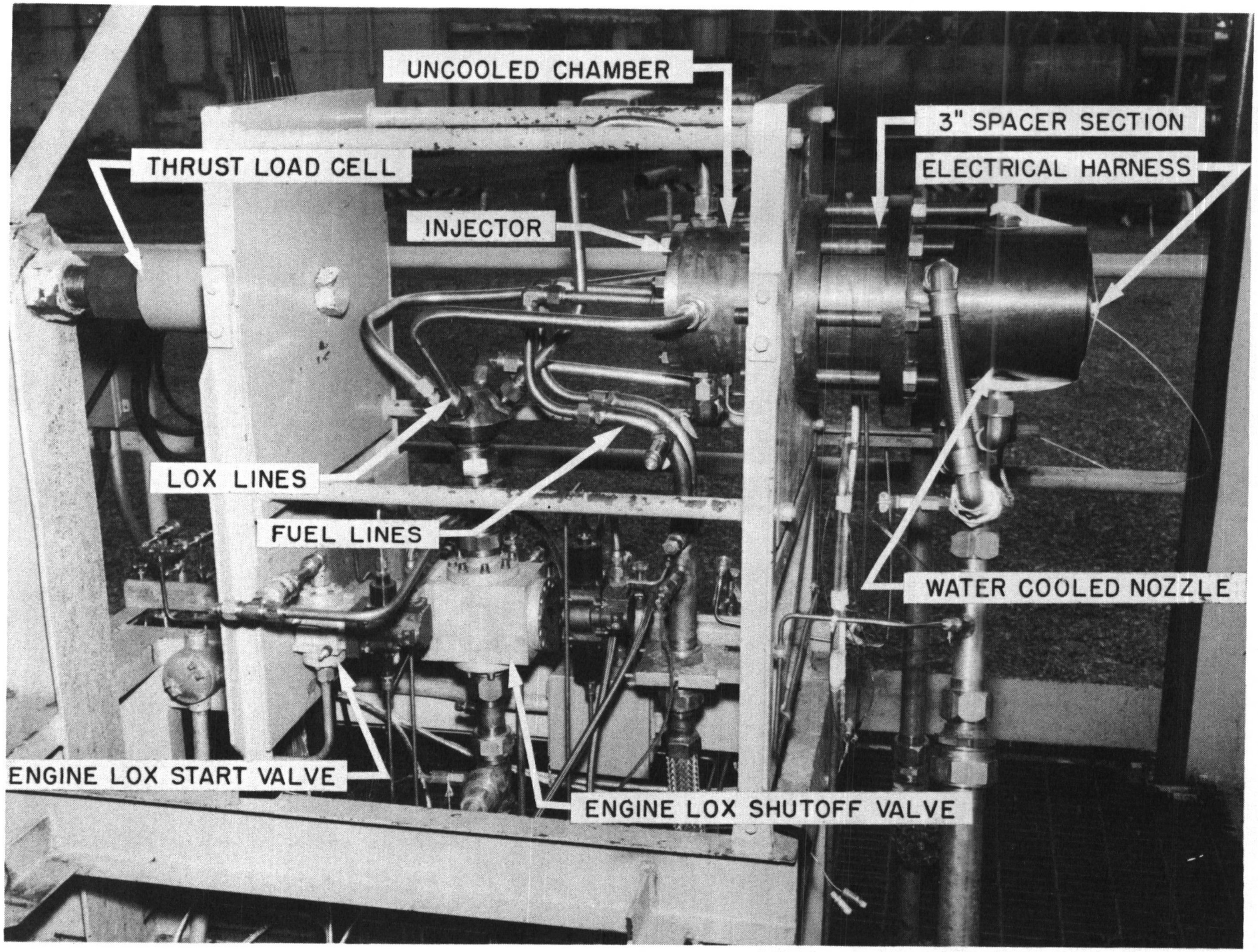


FIGURE 4. OXIDIZER-RICH PREBURNER TEST INSTALLATION

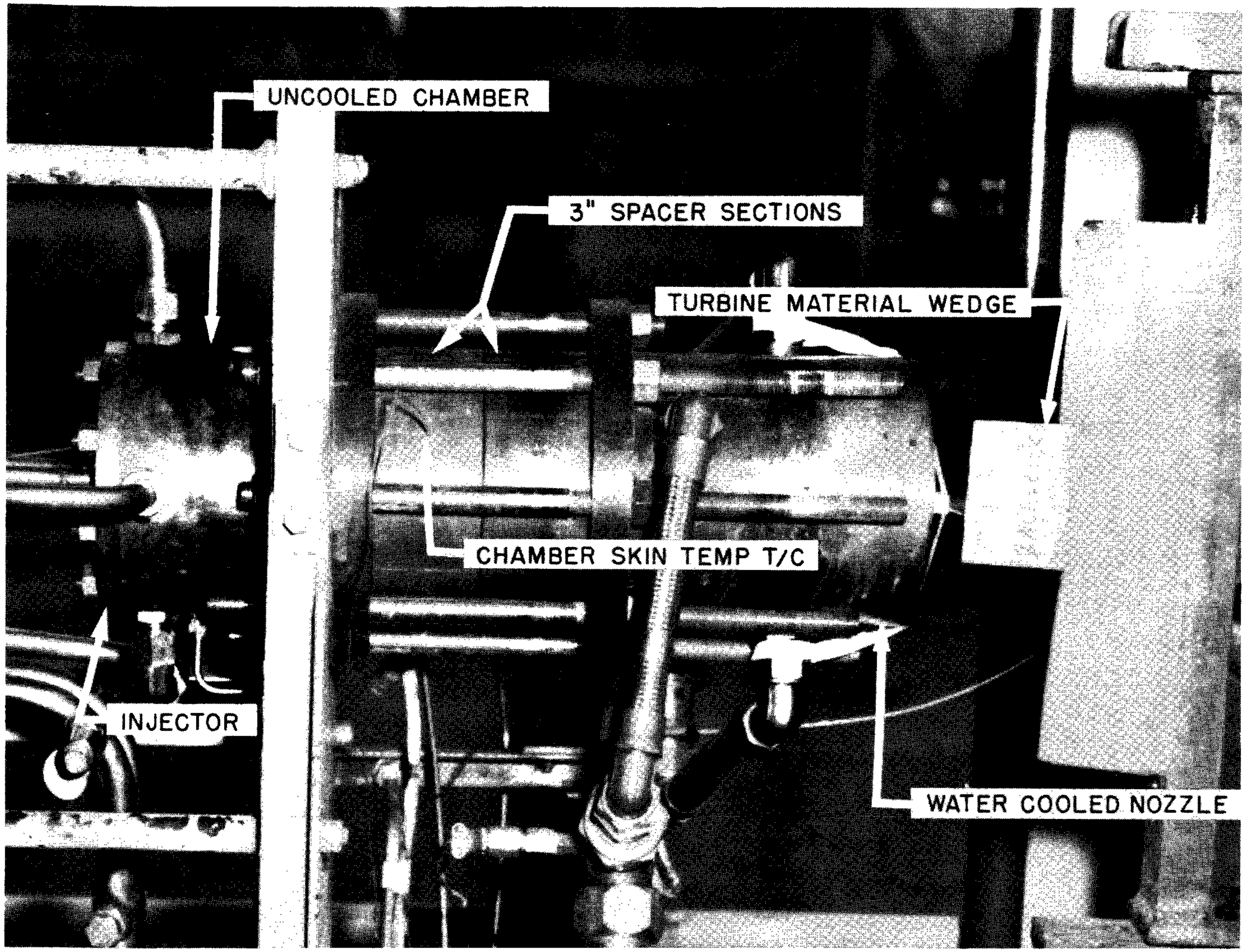


FIGURE 5. TURBINE MATERIAL TEST INSTALLATION

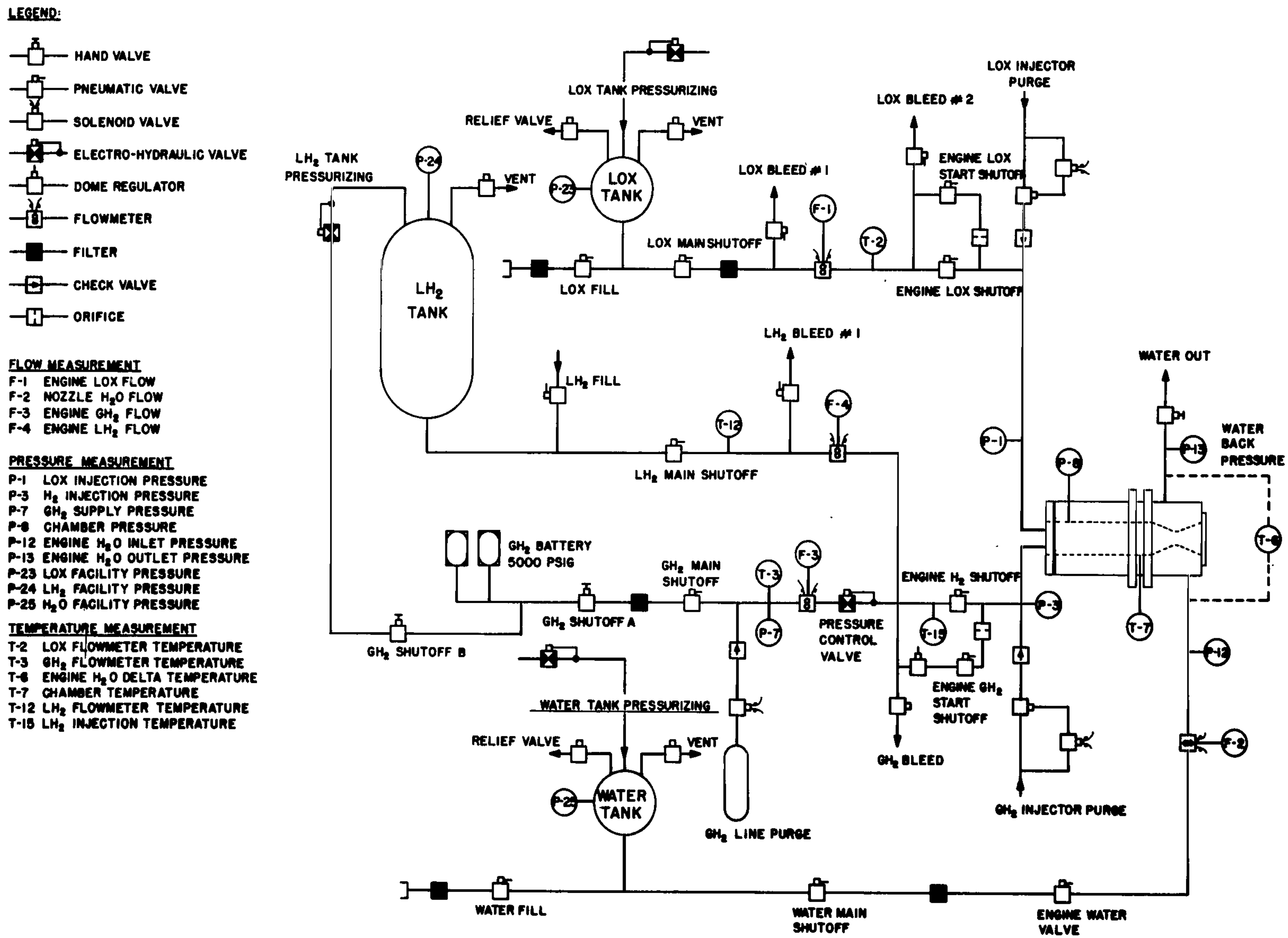
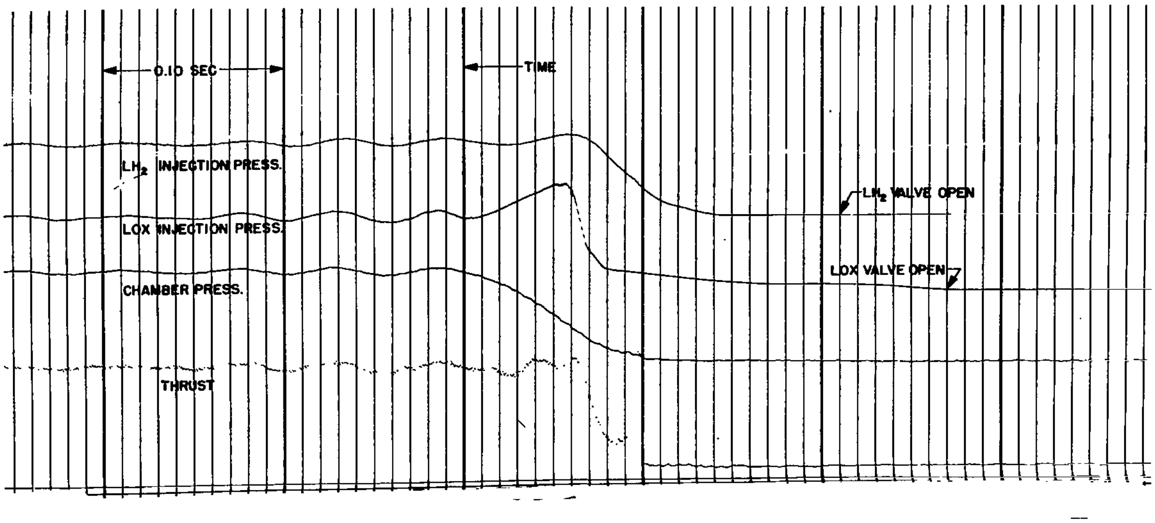
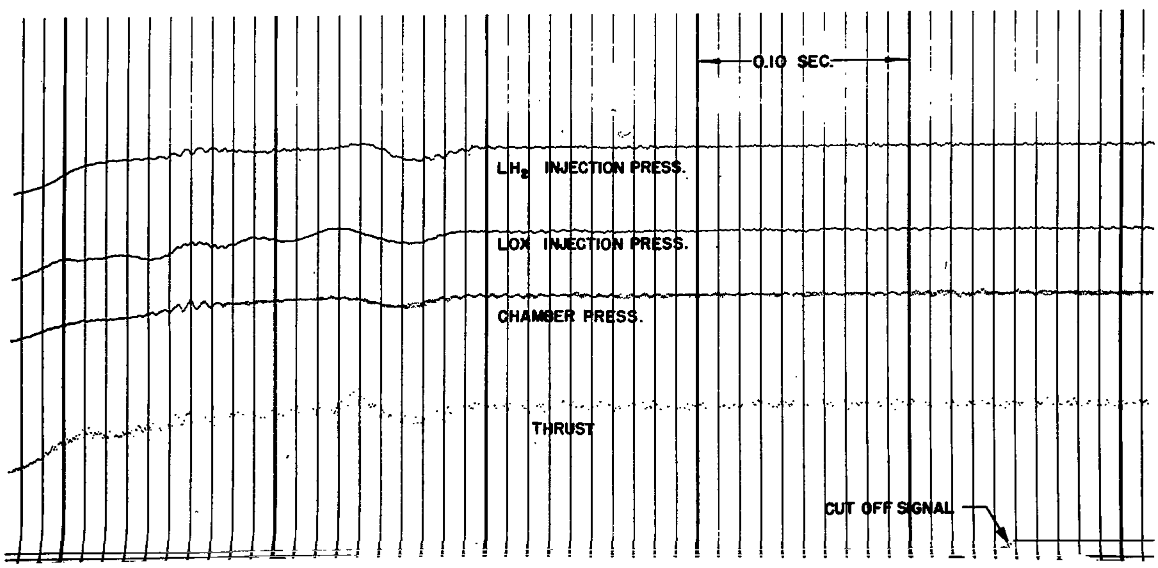


FIGURE 6. TEST INSTALLATION SCHEMATIC



IGNITION SEQUENCE



CUTOFF SEQUENCE

FIGURE 7. OSCILLOGRAM OF IGNITION AND CUTOFF SEQUENCES, TEST 172-26

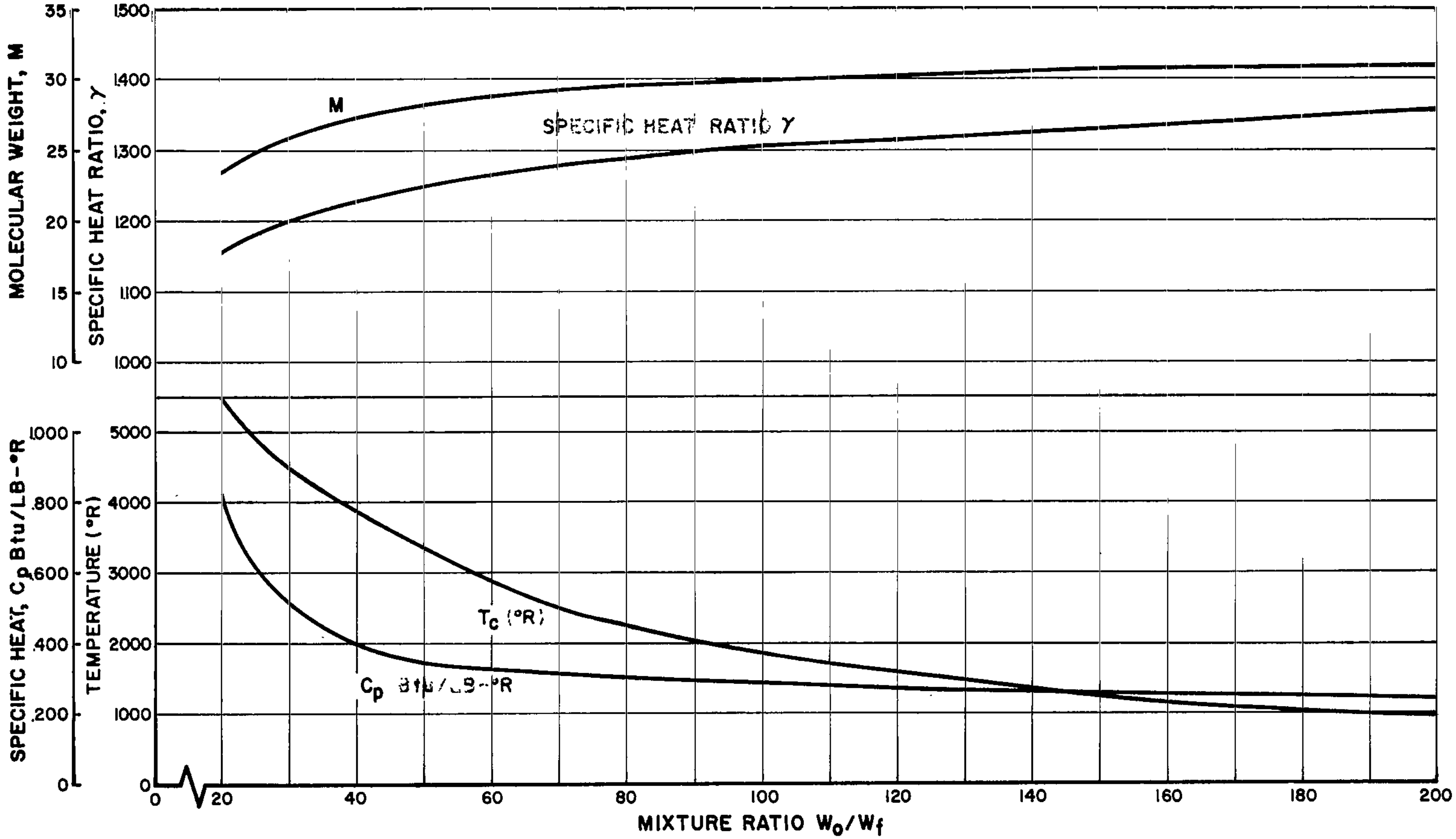


FIGURE 8. COMBUSTION CHARACTERISTICS OF OXYGEN/ HYDROGEN AT MIXTURE RATIOS FROM 20 TO 200

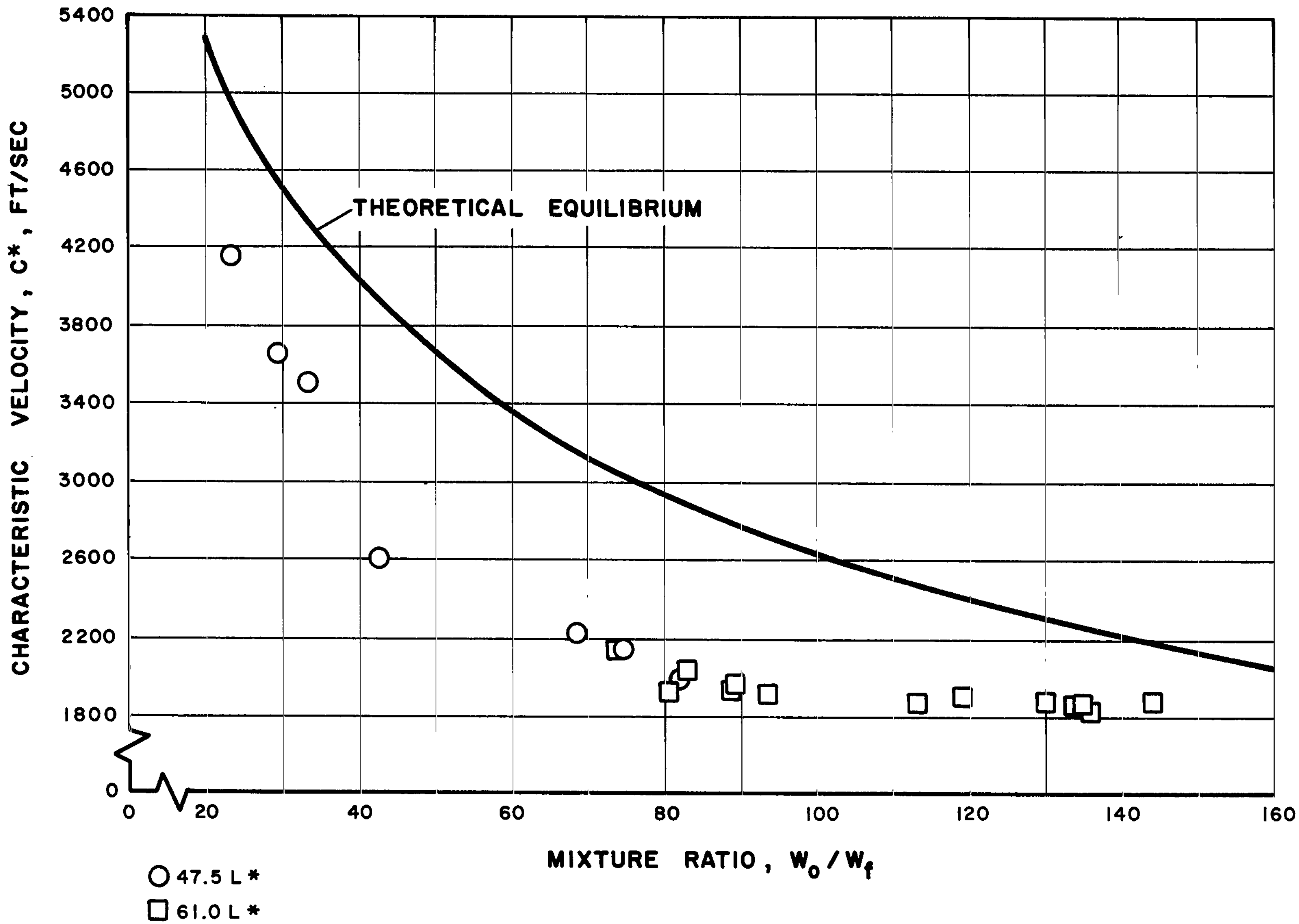


FIGURE 9. CHARACTERISTIC VELOCITY AS A FUNCTION OF MIXTURE RATIO

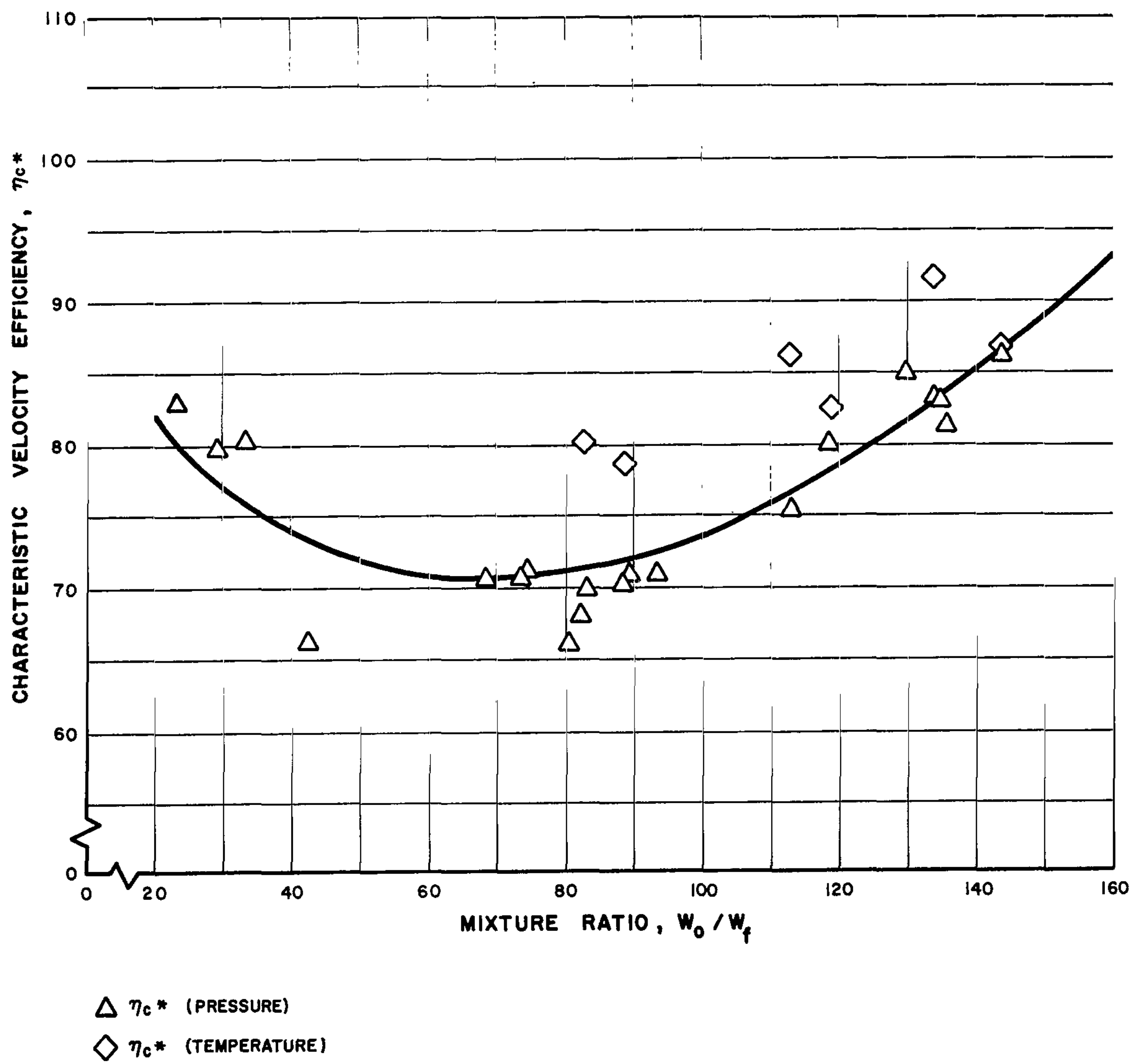


FIGURE 10. CHARACTERISTIC VELOCITY EFFICIENCY AS A FUNCTION OF MIXTURE RATIO

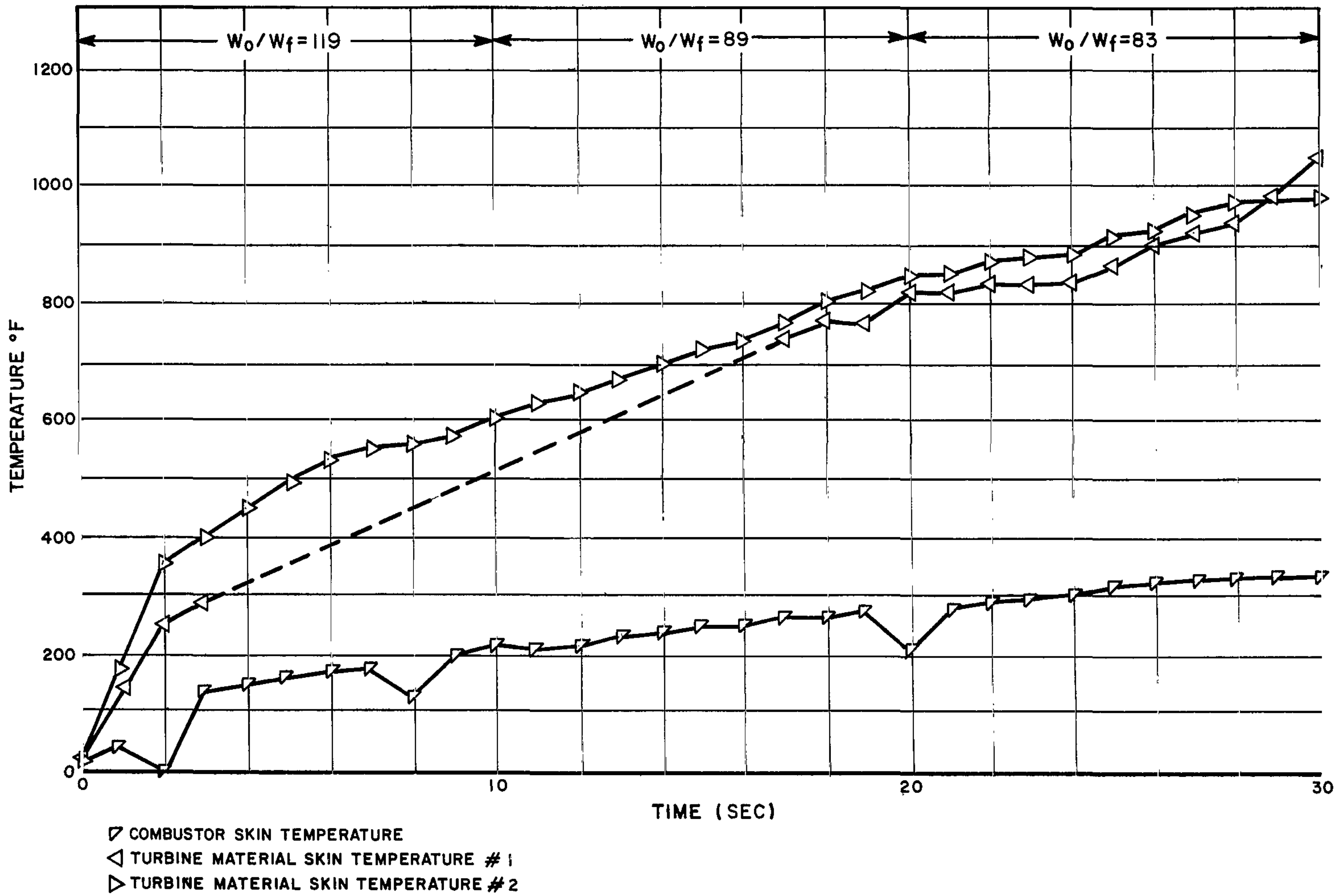


FIGURE 11. CHAMBER SKIN TEMPERATURE AND TURBINE MATERIAL TEMPERATURES AS FUNCTIONS OF RUN DURATION, TEST 172-22

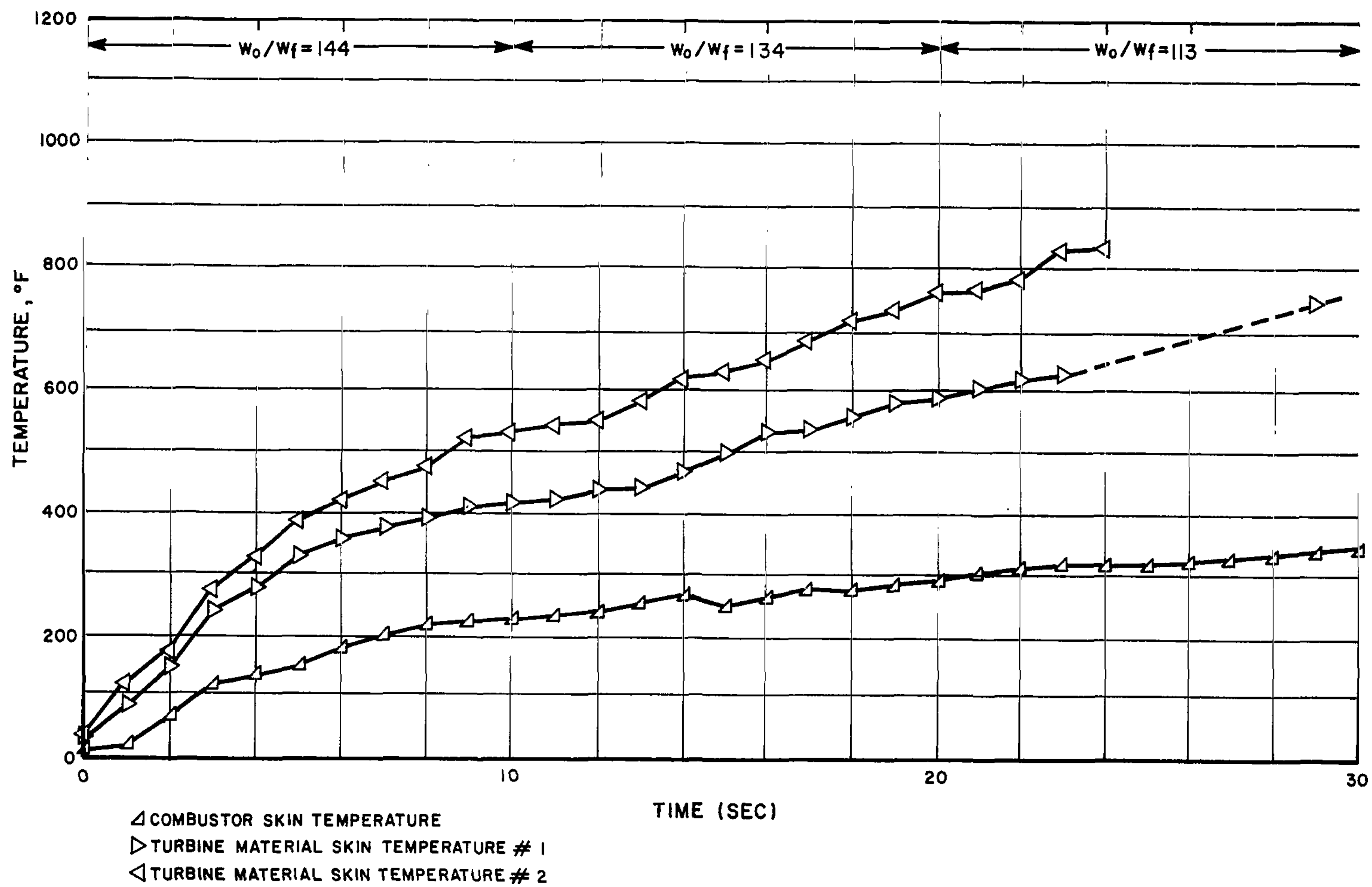


FIGURE 12. CHAMBER SKIN TEMPERATURE AND TURBINE MATERIAL TEMPERATURES AS FUNCTIONS OF RUN DURATION, TEST 172-23

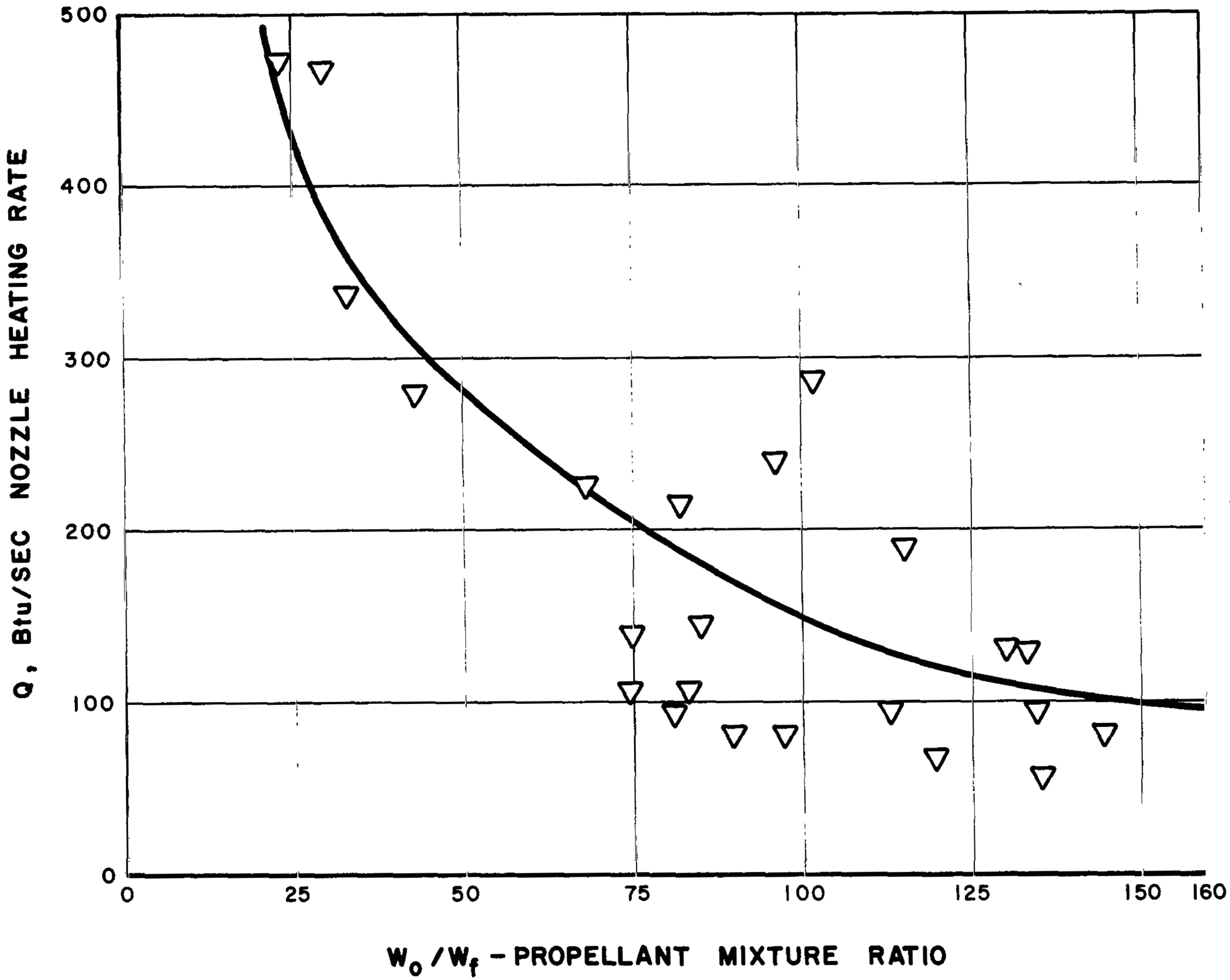


FIGURE 13. NOZZLE HEATING RATE AS A FUNCTION OF MIXTURE RATIO

REFERENCE

1. Zeleznik, Frank J.; and Gordon, Sanford: A General IBM 704 or 7090 Computer Program For Computation of Chemical Equilibrium Compositions, Rocket Performance, and Chapman-Jouguet Detonations. NASA TN D-1454, 1963.

NASA-Langley, 1966 M342

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