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TECHNICAL MEMORANDUM

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EXPERIMENTAL PERFORMANCE OF AREA RATIO 200, 25 AND 8 NOZZLES ON JP-4 FUEL AND LIQUID-OXYGEN ROCKET ENGINE

By J. Calvin Lovell, Nick E. Samanich
and Donald O. Barnett

Lewis Research Center
Cleveland, Ohio

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SUMMARY

The performance of an area ratio 200 bell-shaped nozzle, an area ratio 25 bell-shaped nozzle, and an area ratio 8 conic nozzle on a JP-4 fuel and liquid-oxygen rocket engine has been determined. Tests were conducted using a nominal 4000-pound-thrust rocket in the Lewis 10- by 10-foot supersonic tunnel, which provided the altitude environment needed for fully expanded nozzle flow. The area ratio 200 nozzle had a vacuum thrust coefficient of 1.96, compared with 1.82 and 1.70 for the area ratio 25 and 8 nozzles, respectively. These values are approximately equal to those for theoretical frozen expansion. The measured value of vacuum specific impulse for the area ratio 200 nozzle was 317 seconds for a combustion-chamber characteristic velocity of 5200 feet per second. The vacuum-specific-impulse increase for the area-ratio increase from 8 to 200 was 46 seconds.

INTRODUCTION

Theoretical calculations for JP-4 fuel and liquid-oxygen rockets have indicated performance gains from high-area-ratio nozzles for upper-stage or space vehicles. To date, experimental data are available for a maximum nozzle area ratio of 48.

An experimental study of high-area-ratio nozzles is being conducted in the Lewis 10- by 10-foot supersonic wind tunnel using a nominal 4000-pound-thrust rocket. The program includes tests of a series of 15° half-angle conical nozzles and bell-shaped nozzles with area ratios of 200 and 25. The area ratio 200 bell-shaped nozzle has rapid initial expansion and is only 58 percent as long as the equivalent-area-ratio 15° conic nozzle. To date, only tests of the area ratio 200 and 25 bell-shaped nozzles and an area ratio 8 conic nozzle are complete. The low-area-ratio nozzles were included for comparative data.

*Title, Unclassified.

SYMBOLS

A nozzle flow cross-sectional area

C_F thrust coefficient, F/P_cA_t

c* chamber characteristic velocity, $P_cA_tg/\dot{w}$, ft/sec

F thrust, lb

f fuel weight-flow rate

g gravitational constant, 32.2 ft/sec^2

I specific impulse, $F/\dot{w}$, sec

o oxidant weight-flow rate

P pressure

R nozzle radius

r_t nozzle radius of curvature at throat

$\dot{w}$ total propellant weight-flow rate, lb/sec

x nozzle axial distance from throat

Subscripts:

c combustion chamber

n nozzle

t nozzle throat

APPARATUS

Rocket Engine

The engine was designed to develop a thrust of 4000 pounds at a combustion-chamber pressure of 600 pounds per square inch absolute and a nozzle area ratio of 8. The combustion chamber had a characteristic length of 50 inches, a chamber contraction ratio of 6.0, and a throat diameter of 2.2 inches. Coordinates of the nozzles are given in table I, and the nozzle shapes are shown in the drawings of figure 1. The area ratio 200 bell-shaped nozzle was designed assuming a specific-heat ratio of 1.125 and no boundary-layer displacement effects. The flow was initially expanded assuming source flow along a 54° half-angle cone to a Mach number of 1.75. It was then turned back, by the method of characteristics, to the axial direction at an area ratio of 560, assuming isentropic expansion. The nozzle was then cut off at an area ratio of 200, where the wall angle was 15°.

The injector (fig. 2) had 120 sets of 0.026-inch-diameter like-on-like fuel impinging doublets and 121 0.049-inch-diameter showerhead oxidant orifices. An additional 24 0.026-inch-diameter fuel orifices were located near the chamber walls for cooling. Spiral inserts $1\frac{5}{8}$ inches long, with 4 turns per inch, were used in the oxidant orifices to impart rotation and thereby improve atomization of the liquid oxygen.

Test Facility

The rocket test installation in the Lewis 10- by 10-foot supersonic tunnel is illustrated in figure 3. The engine was housed in a cone-cylinder shroud and mounted in the center of the tunnel test section on an electric strain-gage force balance. The shroud completely shielded the engine from the airstream, and the tunnel test section provided the altitude environment for the rocket exhaust. Propellant and cooling-water lines were guided through the shroud and the large strut giving the engine the required freedom to move fore and aft on the balance and also allowing the piping to expand and contract without measurable tare force (fig. 3(a)).

The tunnel was operated at a Mach number of 3.5. To assure fully expanded flow in the nozzles, the test-section static pressure was maintained at approximately 34 pounds per square foot absolute. This gave a model base pressure of about 10 pounds per square foot absolute, compared with a minimum nozzle-exit pressure of 36 pounds per square foot absolute for the area ratio 200 nozzle.

Instrumentation

The firing time of the rocket was approximately 8 seconds, during which the propellant flows, thrust, and chamber pressure were recorded every 0.9 second. The temperatures, nozzle pressures, and coolant flows were recorded every $2\frac{1}{2}$ seconds. The propellant and coolant-flow rates were measured by rotating-vane type flowmeters which were calibrated with water. Oxidant temperature was measured and the data of reference 1 were used to determine the oxidant density. No correction was made for the compressibility of liquid oxygen at the pressure existing at the flowmeter (1000 to 1200 lb/sq in. abs). Fuel density was determined from the measured temperature. Engine thrust was measured by an electric strain-gage balance. The balance was calibrated with the model completely assembled, both before and after the tests. Combustion-chamber and nozzle pressures were measured by electric strain-gage pressure transducers. Chamber pressure was measured at the injector face and corrected for the pressure drop across the chamber, due to chamber area ratio, using the momentum relation. This placed the experimental and theoretical chamber pressures on a comparable basis.

RESULTS AND DISCUSSION

In the performance comparisons, the theoretical curves are from references 2 and 3, which give results for conditions of equal nozzle-exit and ambient pressure. These data and the experimental data have been converted to vacuum conditions by correcting for the force resulting from the difference between the ambient pressure and the nozzle-exit pressure acting over the nozzle-exit area. No correction to specific impulse for the heat loss to cooling water was made because the effect was less than 0.5 percent. The area ratio 200 and 25 nozzles burned out before complete data could be obtained covering the range of oxidant-fuel ratio at chamber pressures up to 600 pounds per square inch absolute.

Performance Comparisons

All the experimental performance data obtained are summarized in figure 4, which gives results for chamber pressures from 395 to 590 pounds per square inch absolute. The chamber characteristic velocity c^* was approximately 5200 feet per second, which is 90 percent of the maximum for theoretical frozen expansion. This value is low primarily because of poor atomization of liquid oxygen by the showerhead oxidant injector. No significant effects of o/f on thrust coefficient or specific impulse for the range of o/f covered were noted.

The average experimental value of C_F for each nozzle from figure 4 and the theoretical values of C_F at maximum specific impulse are shown in figure 5. A value of C_F of 1.96 was measured for the area ratio 200 nozzle, compared with 1.82 for the area ratio 25 nozzle, and 1.70 for the area ratio 8 conic nozzle. The values of C_F from reference 4 at area ratios 5.5 and 48 fall near the faired curve through these data. In order to put all the data on a comparable basis, the data of reference 4 have been corrected for the pressure drop across the combustion chamber using the momentum relation. All the measured values of C_F are fairly close to the values for theoretical frozen expansion. It cannot be concluded from this fact, however, that the expansion process itself is of frozen composition, because the flow in the nozzles is not isentropic or axially discharged, as is assumed for the ideal nozzles of the theoretical curves. Figure 6 shows that the measured pressure distribution for the area ratio 200 nozzle differed considerably from the theoretical distributions calculated for isentropic flow. The measured pressure ratios in the vicinity of the throat are appreciably less than the theoretical values. This is attributed to the very high rate of turning at the throat (wall radius/throat radius = 0.169, see table I) which gives expansions near the wall which are greater than those for the conical flow assumed in the nozzle design. These overexpansions are usually followed by compression waves as the flow is turned back toward the axial direction. The combination of low pressure ratios near the throat followed by compression shocks results in lowered nozzle performance.

The average measured values of specific impulse from figure 4 are compared with the maximum theoretical values and those of reference 4 in figure 7. The measured vacuum specific impulse for the area ratio 200 nozzle is 317 seconds, which is low compared with the value for frozen expansion because of the low chamber c^* of 5200 feet per second. The increases in specific impulse with nozzle-area-ratio increase, however, are appreciable. For the area-ratio increase from 8 to 200 the vacuum-specific-impulse increase is 46 seconds, which is slightly larger than the increase for theoretical frozen expansion. If it is assumed that, for a given nozzle, C_F does not change with c^*, the specific impulse is proportional to c^*, and the effect of a more efficient chamber can be predicted. For a rocket having the area ratio 200 nozzle and a c^* equal to the value for frozen expansion (5800 ft/sec), the predicted vacuum specific impulse is approximately 347 seconds.

SUMMARY OF RESULTS

The performance of an area ratio 200 bell-shaped nozzle is compared with that for an area ratio 25 bell-shaped nozzle and an area ratio 8 conic nozzle. The experiments were conducted using a nominal 4000-pound-thrust JP-4 fuel and liquid-oxygen rocket in the Lewis 10- by 10-foot supersonic wind tunnel. The following results were obtained:

1. The area ratio 200 nozzle had a vacuum thrust coefficient of 1.96, compared with 1.82 and 1.70 for the area ratio 25 and 8 nozzles, respectively. These values are fairly close to those for theoretical frozen expansion.

2. The measured value of vacuum specific impulse for the area ratio 200 nozzle was 317 seconds for a combustion-chamber characteristic velocity of 5200 feet per second. For the nozzle-area-ratio increase from 8 to 200, the vacuum-specific-impulse increase is 46 seconds, which is slightly larger than the increase for theoretical frozen expansion. A vacuum specific impulse of approximately 347 seconds is predicted for a rocket having the area ratio 200 nozzle tested and a characteristic velocity of 5800 feet per second.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, May 11, 1960

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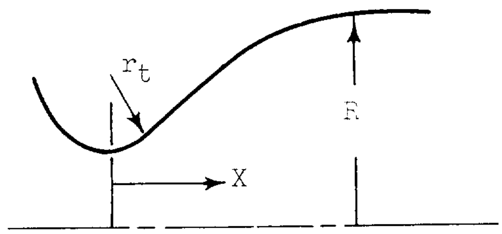
2. Huff, Vearl N., and Fortini, Anthony: Theoretical Performance of JP-4 Fuel and Liquid Oxygen as a Rocket Propellant. I - Frozen Composition. NACA RM E56A27, 1956.

3. Huff, Vearl N., Fortini, Anthony, and Gordon, Sanford: Theoretical Performance of JP-4 Fuel and Liquid Oxygen as a Rocket Propellant. II - Equilibrium Composition. NACA RM E56D23, 1956.

4. 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. - NOZZLE COORDINATES

[All dimensions in inches.]



Area ratio (nominal), 200; R_t/r_t, 0.169.

x	R
0	1.12
.380	1.43
2.13	3.25
4.58	5.10
7.42	6.80
10.42	8.30
13.42	9.64
16.42	10.73
19.42	11.89
23.67	13.33
27.17	14.35
31.47	15.59

Area ratio, 25; R_t/r_t, 2.05.

x	R
0	1.10
Linear	Linear
2.32	2.00
4.37	2.96
6.19	3.74
8.19	4.55
10.98	5.40
11.23	5.50

Area ratio, 8; R_t/r_t, 2.04.

x	R
0	1.10
Linear	Linear
7.76	3.11

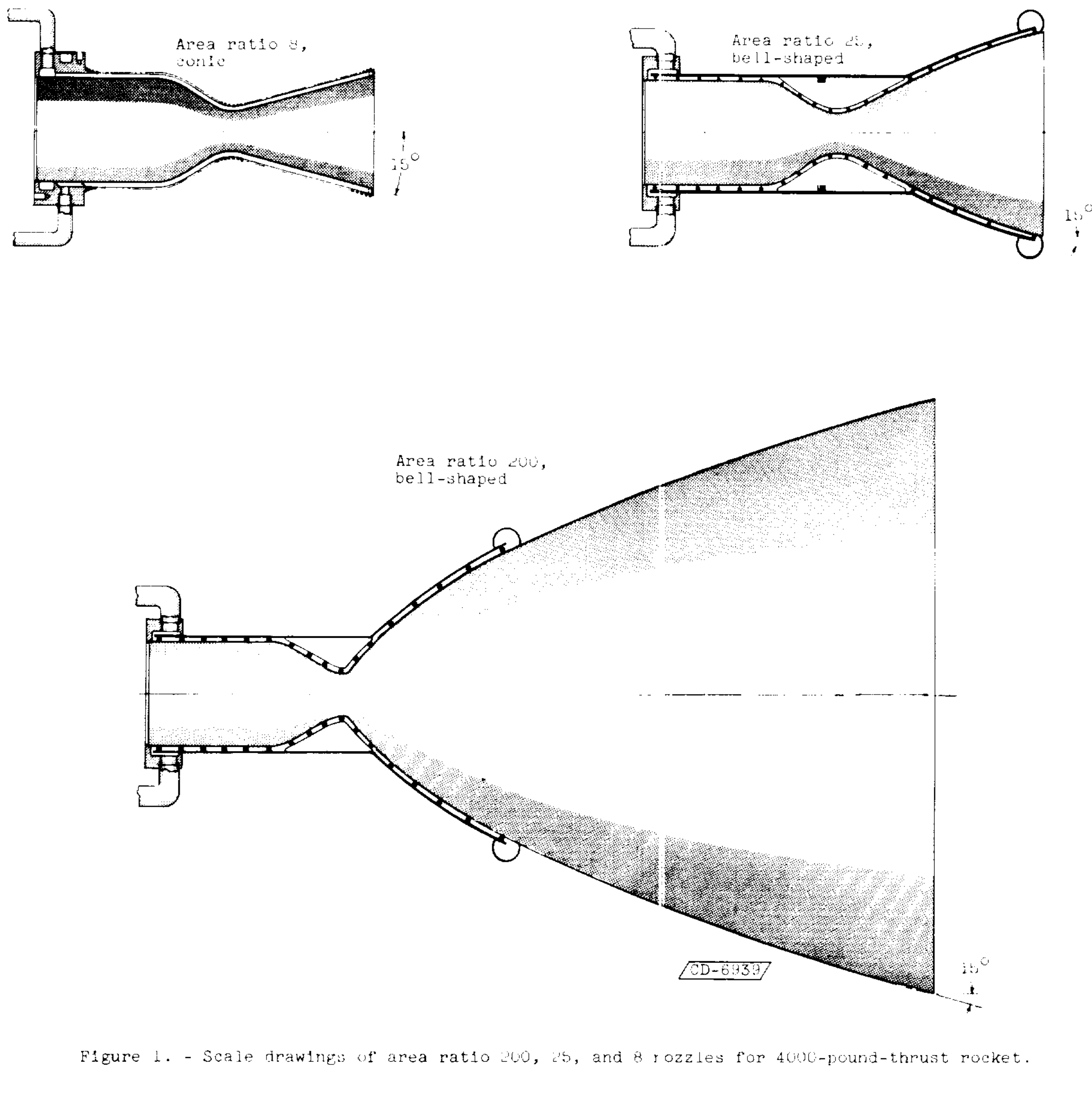


Figure 1. - Scale drawings of area ratio 200, 25, and 8 nozzles for 4000-pound-thrust rocket.

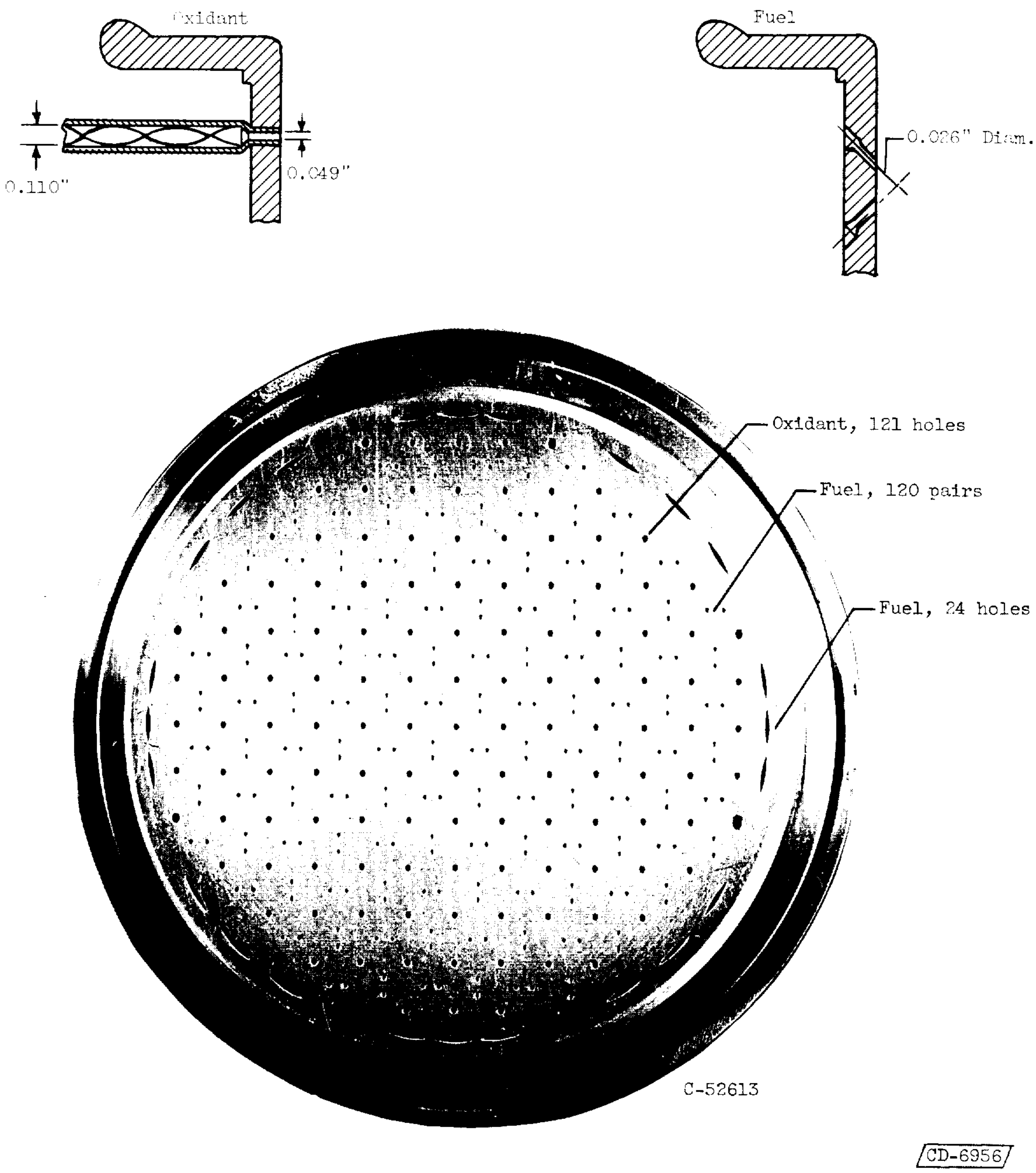
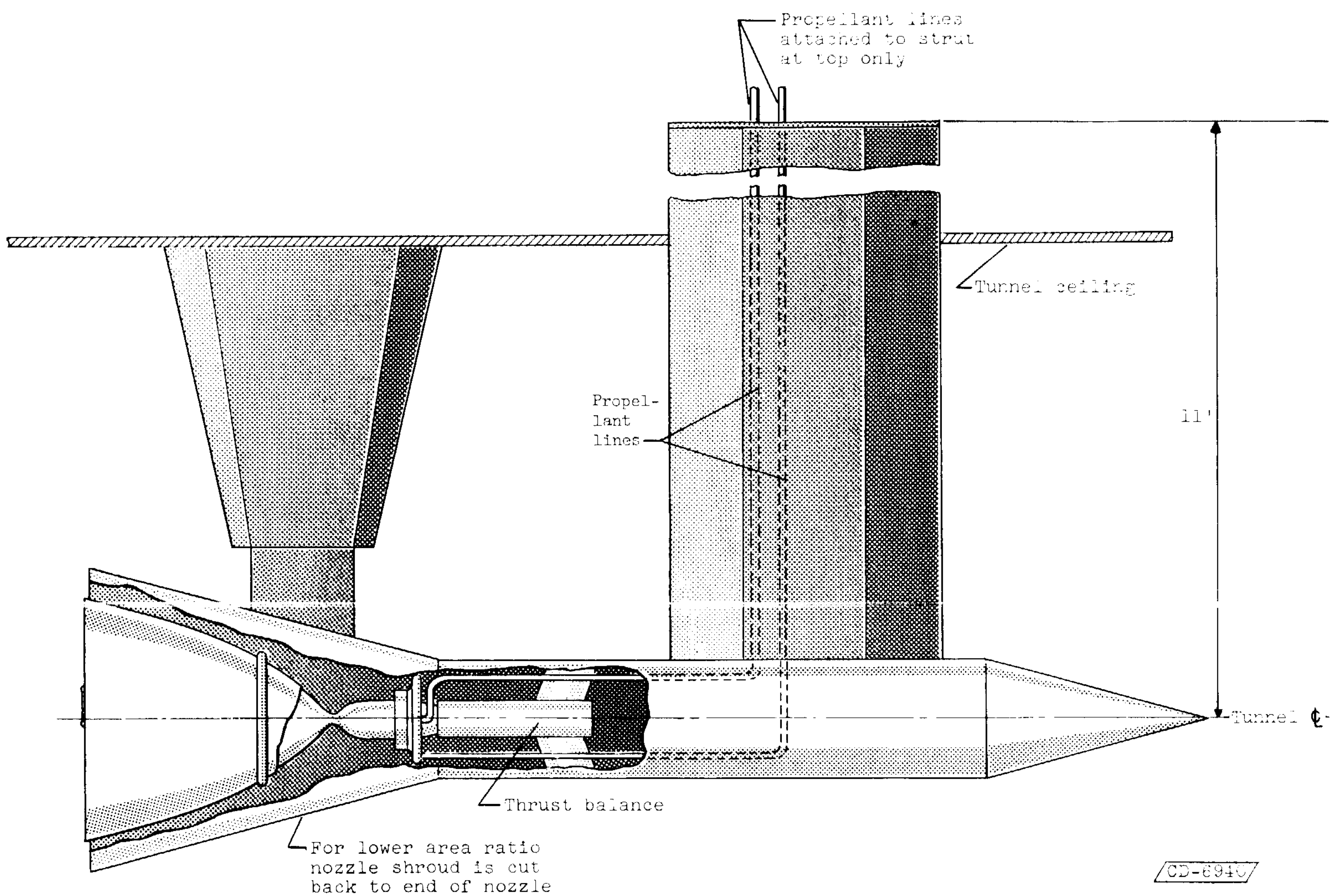


Figure 2. - Like-on-like fuel and showerhead oxidant injector for 4000-pound-thrust rocket.

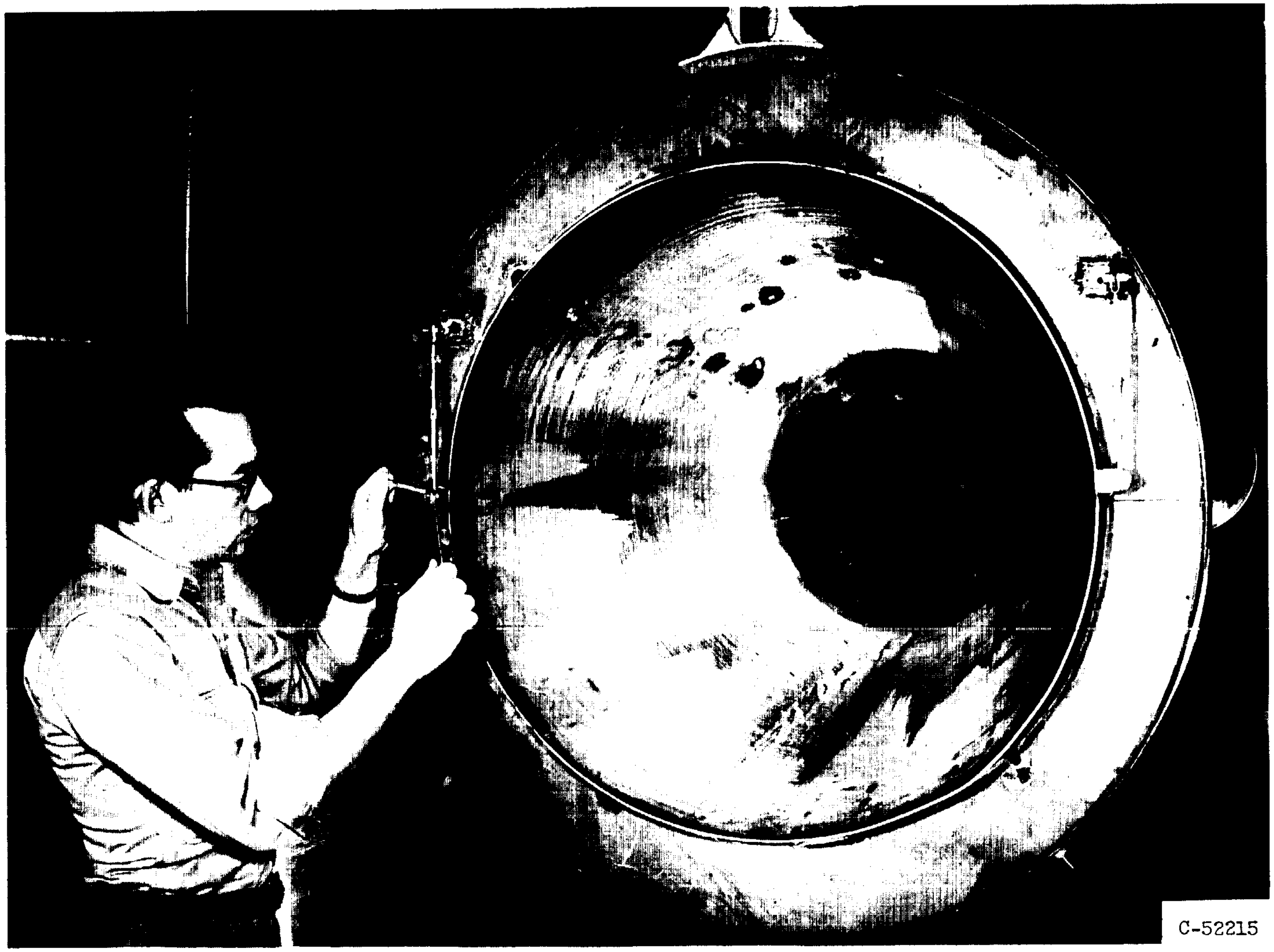


(a) Schematic drawing.

Figure 3. - Installation of 4000-pound-thrust rocket with area ratio 200 nozzle in Lewis 10- by 10-foot supersonic wind tunnel.

(b) Front view.

Figure 3. - Continued. Installation of 4000-pound-thrust rocket with area ratio 200 nozzle in Lewis 10- by 10-foot supersonic wind tunnel.



(c) Rear view.

Figure 3. - Concluded. Installation of 4000-pound-thrust rocket with area ratio 200 nozzle in Lewis 10- by 10-foot supersonic wind tunnel.

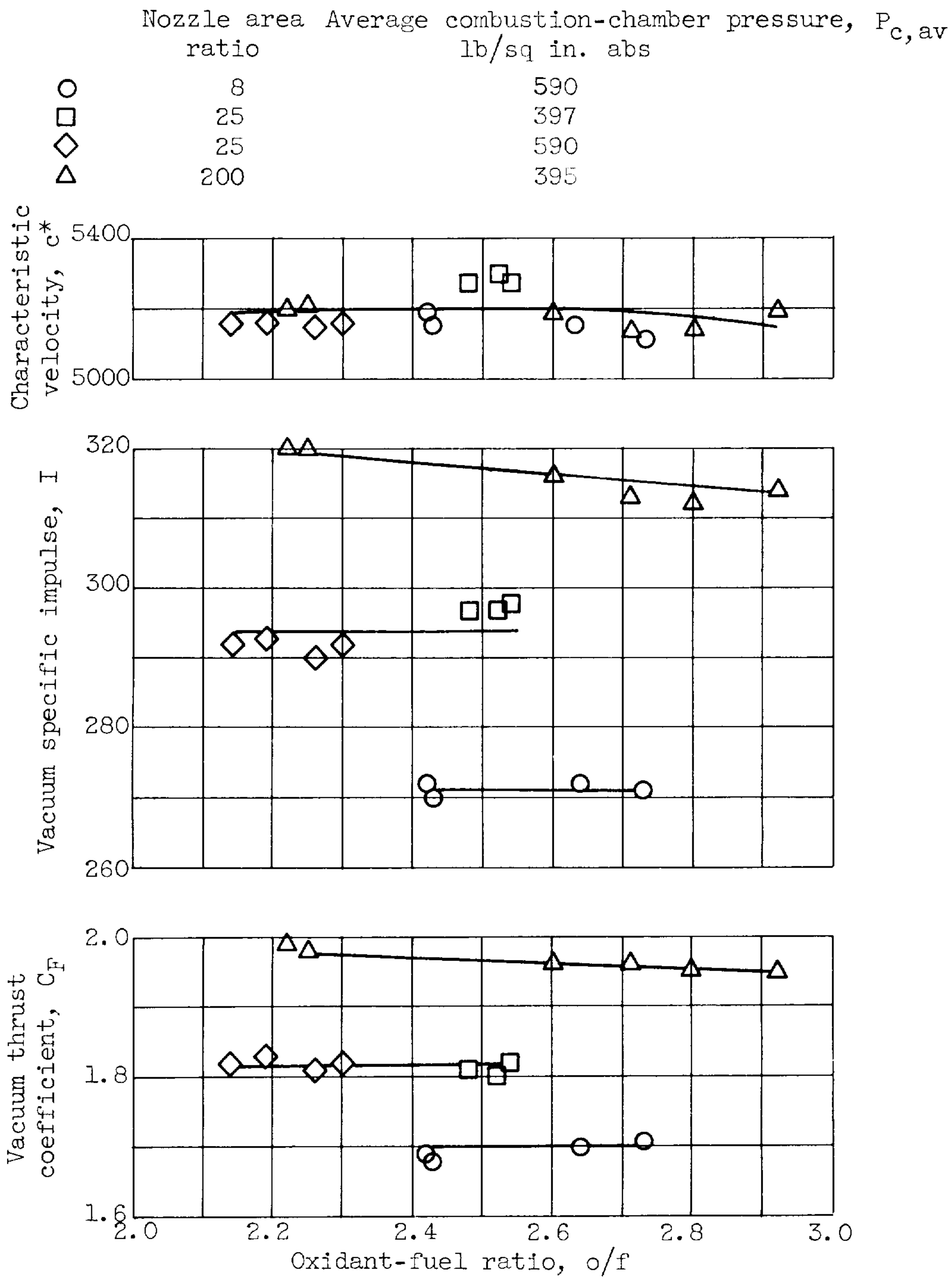


Figure 4. - Experimental and theoretical vacuum performance of JP-4 fuel and liquid-oxygen rocket for nozzle area ratios of 200, 25, and 8.

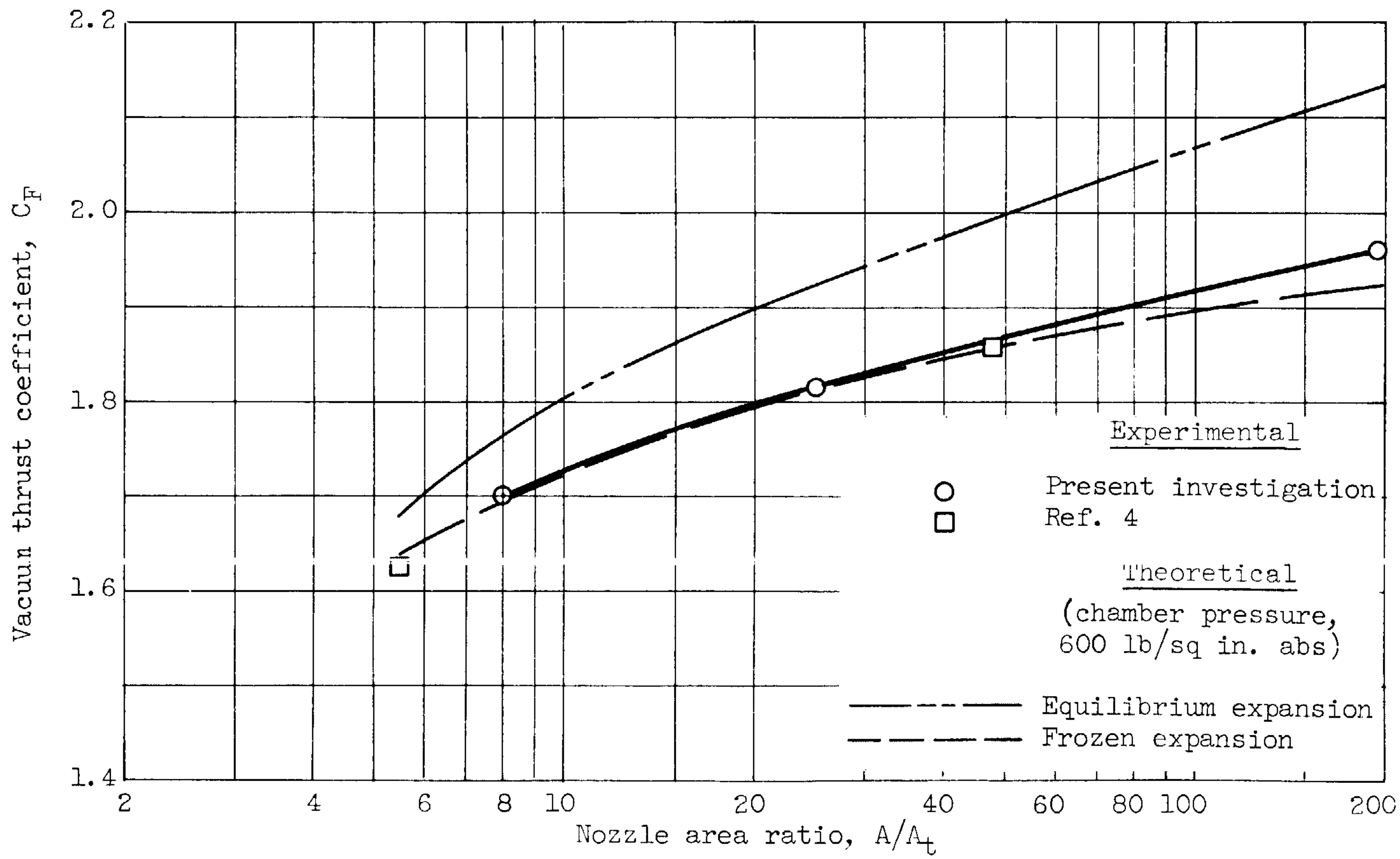


Figure 5. - Experimental and theoretical vacuum thrust coefficients for JP-4 fuel and liquid-oxygen rocket.

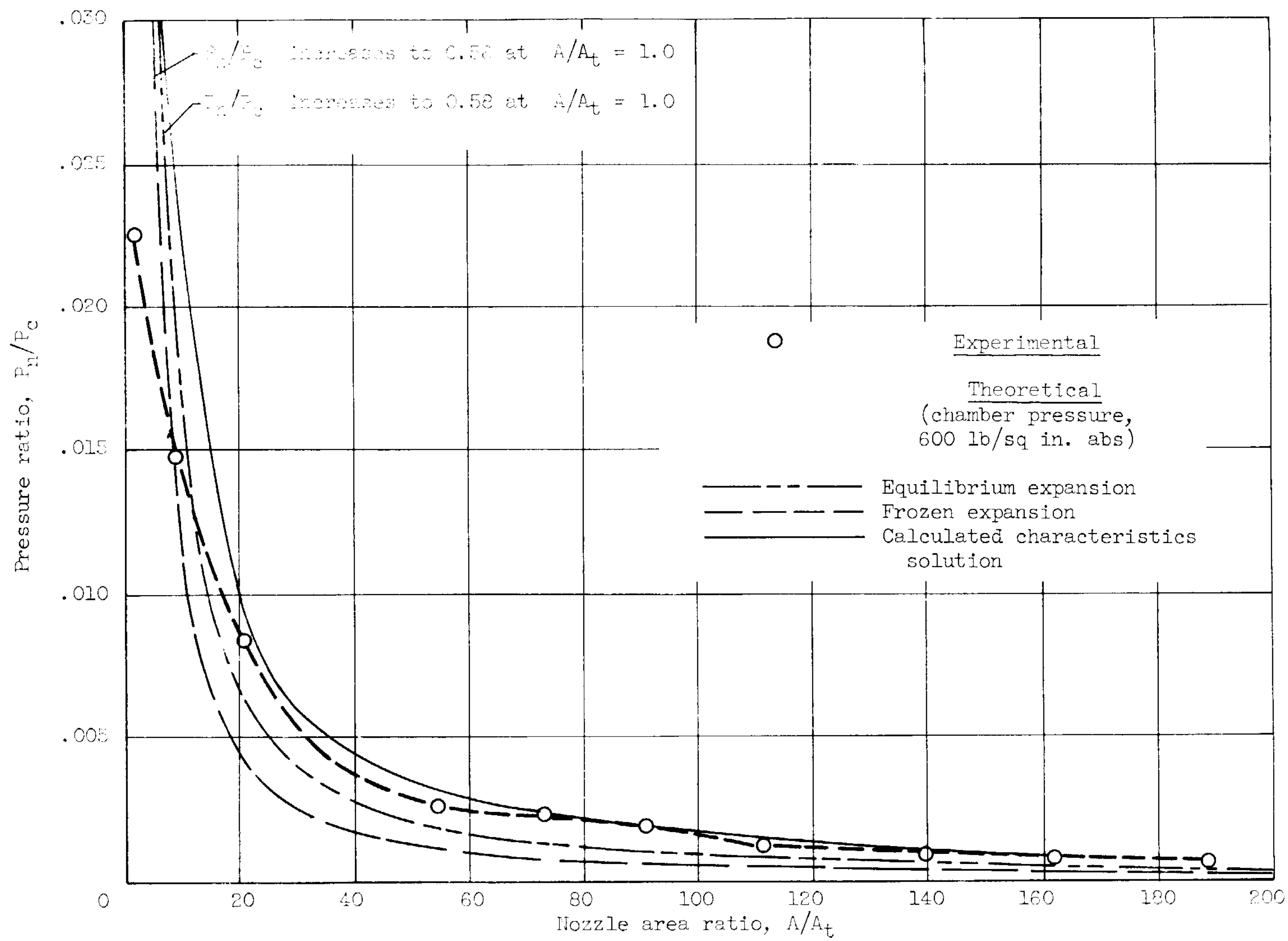


Figure 6. - Nozzle pressure distribution for area ratio 200 bell-shaped nozzle. Combustion-chamber pressure, P_c, 385 pounds per square inch absolute; oxidant-fuel ratio, o/f, 2.8.

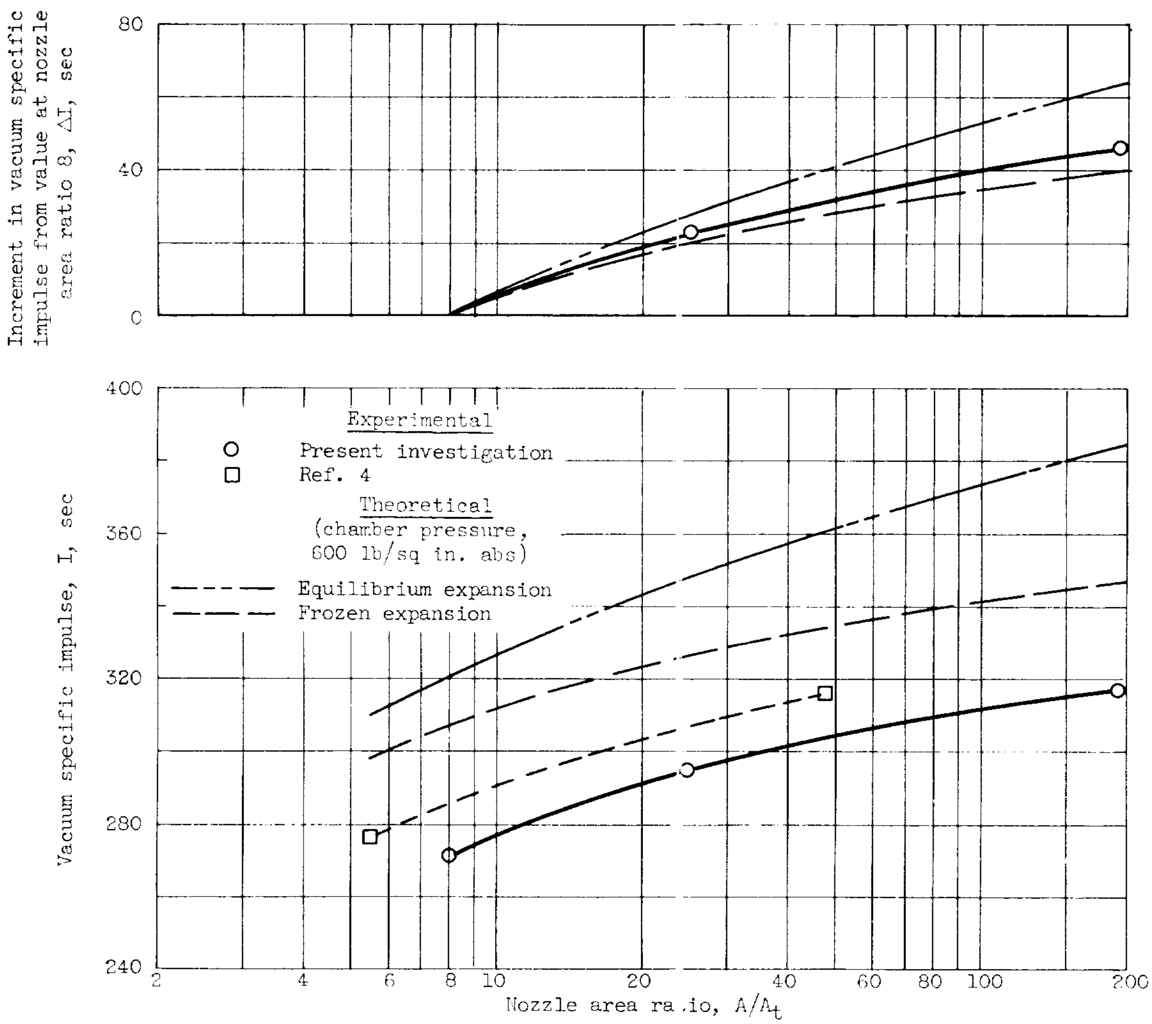


Figure 7. - Experimental and theoretical vacuum specific impulse for JP-4 fuel and liquid-oxygen rocket.