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LIQUID-HYDROGEN ROCKET ENGINE DEVELOPMENT AT AEROJET, 1944-1950+

George H. Osborn, Robert Gordon, and Herman L. Coplen with George S. James (USA)++

I. INTRODUCTION

In the early 20th century, many early rocket pioneers favored liquid hydrogen; however, none of them used hydrogen as a rocket fuel because of its extreme physical properties and scarcity. Illustrative of the properties that make practical handling difficult is a boiling point of -426° at one atmosphere, and a density about one-seventh that of water. Interest in the methods and apparatus used in hydrogen gas liquefaction increased significantly in the mid-1940s when handling methods were developed to supply liquid hydrogen for the steadily increasing requirements of basic research. The authors had the good fortune to participate in one of the earliest programs in the United States to systematically investigate hydrogen-oxygen propellants for high-energy rocket engine application.

From late in 1944 to the cession of tests in August 1949, the hydrogen-oxygen programs at the Aerojet General Corporation, under the sponsorship of the Navy Bureau of Aeronautics, advanced these propellants from theoretical performance studies to practical sources of high specific impulse. Specifically, this work tested transpiration-cooled thrust chambers, investigated the concept of ablative-cooled thrust chambers, developed the first successful 1,000-lb-thrust gaseous-propellant rocket engine, conducted the first tests of the effect of jet overexpansion and separation on performance of rocket

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++G.H. Osborn, Assistant to the Vice President, Engineering Operations, Aerojet Liquid Rocket Company; Robert Gordon, President, Structural Composites Industries, Azusa, Calif. 91702; H.L. Coplen, Manager, LOFT Program, Idaho Nuclear Corporation; and G.S. James, Program Manager, Intergovernmental Science and Public Technology Division, National Science Foundation, Washington, D.C., 20550.

thrust chambers, constructed and operated the first continuous 24-hour-operation hydrogen liquefaction plant specifically devoted to rocket engine use, conducted the first liquid-hydrogen tests of the coaxial injector, developed a 3,000-lb-thrust liquid-hydrogen thrust chamber, and tested the first pump to successfully produce high pressures in pumping liquid hydrogen, demonstrating that pumping liquid hydrogen in a turbo-rocket engine was perfectly feasible and could be accomplished with a single-stage centrifugal pump.

II. EARLY HYDROGEN-OXYGEN THRUST CHAMBER TESTS

In December 1944, Fritz Zwicky, then Director of Research for Aerojet, wrote a final report of research and development activities for the Bureau of Aeronautics, Navy Department, (Contract NOa(s)-3055). Zwicky surveyed the work being conducted at various institutions associated with the Navy program to produce chemicals commercially that would allow actual jet velocities between 9,000 feet/sec and 10,000 feet/sec.

These chemicals were mostly boron compounds of a nature not then commercially available.[1] The value of boron hydride compounds as a combined source of hydrogen and high chemical energy had been indicated early in 1944 by James M. Carter of Aerojet.[2] Subsequently, under Contract NOa(s)-5350, Donald L. Armstrong calculated that specific impulse of 311 at 600 psi chamber pressure was theoretically possible for an aluminum borohydride/water reaction.[3] In a separate report under the same contract Paul W. Webster calculated that the performance of hydrogen-oxygen as gaseous propellants would be substantially higher than that of the boron compounds.[4]

With this theoretical background, the first gaseous-hydrogen-oxygen thrust chamber tests were conducted at the Azusa proving grounds on October 15, 1945.[5] During the first test the uncooled thrust chamber burned out completely (injector, chamber, and nozzle) in 15 seconds. During the brief period of equilibruim, engineers measured a thrust of 45 lb at 375 psia chamber pressure with an estimated exhaust velocity of 8470 ft/sec. For the next test a specially designed water-cooled injector and a regulator water-cooled nitromethane type nozzle and chamber were used. Performance was 100 lb thrust at 295 psia chamber pressure with an estimated exhaust velocity of 7280 ft/sec. With the water-cooled thrust chamber an average heat flow density of 3-3 1/2 Btu/sec in^2 was measured, although the chamber eroded slightly in the region adjacent to the injector.[6]

By February 1946, the test facilities had been enlarged to allow the testing of thrust chambers of up to 500-lb thrust. Provision for water cooling the thrust chambers consisted of a 1,000-gallon water tank and a centrifugal pump which supplied 50 gallons per minute at 150 psi. Propellant was drawn from trailer trucks furnished by the National

Cylinder Gas Company. The oxygen trucks had a capacity of 33,000 cubic feet at standard conditions; the hydrogen trucks had a capacity of 28,000 cubic feet at standard conditions. With the given fuel capabilities it was possible to run a 500-lb-thrust chamber for five and one-half minutes before the hydrogen was depleted. A second series of tests with the 100-lb-thrust chamber was conducted during March 1946. The first test, using a mixing chamber injector burned out the injector after seven seconds; the second test with a modified injector was successful; and the third test ran for 60 seconds without damage. Starting, operating, and stopping the thrust chamber presented no unusual hazards. Starts and stops were very smooth and without any explosions.

> The motor was started by opening the propellant valves (an electric spark was on at all times) while holding the throttles closed. The throttles were then advanced maintaining a rich mixture (excess hydrogen) until the correct injection pressure was reached; both throttles could then be advanced or retarded together to vary the thrust while holding constant mixture ratio as in normal multi-engine airplane operation. However both throttles could be jockeyed separately in order to vary the mixture ratios. Control was positive and smooth throughout. The motor was stopped by closing the throttles to reduce fuel flow to a low value. The propellant valves were then closed.[7]

Development of 100-to 400-lb gaseous-hydrogen-oxygen thrust chambers continued for the Bureau of Aeronautics, under Task 6 of Contract NOa(s)-7968. Various techniques of cooling were explored including transpiration cooling (liquid and gas), convection cooling, radiation cooling, and heat capacity cooling. The most successful water-transpiration-cooled thrust chamber, shown in Figure 1, operated for 60 seconds without damage.

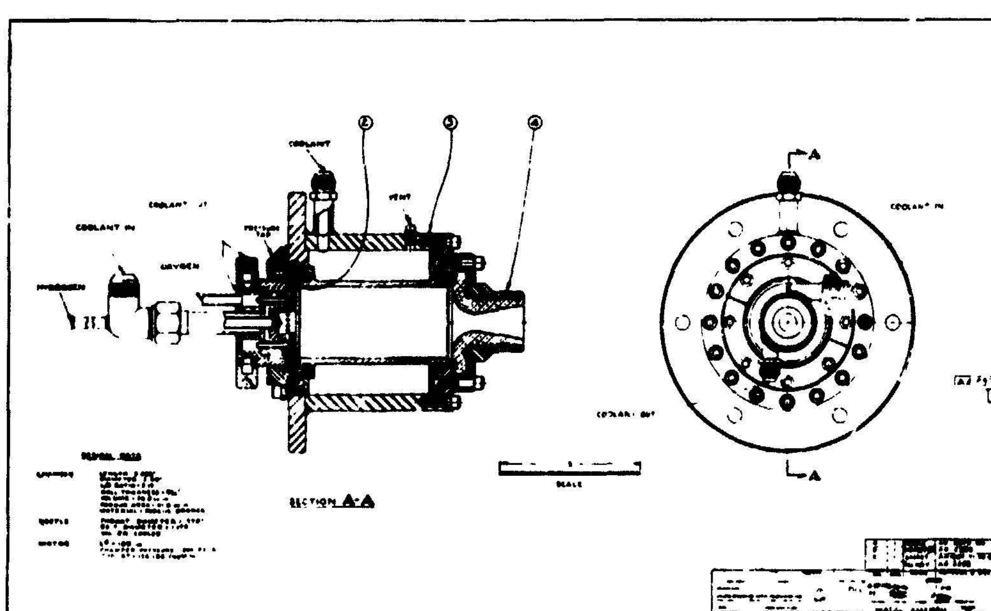


Fig. 1
Drawing of Porous Metal Thrust Chamber Assembly

The combustion chamber wall was made of porous bronze (Oilite). In subsequent tests the combustion chamber wall was cooled with gaseous hydrogen while the injector and nozzle were water cooled. An all convection-cooled thrust chamber of conventional design with a De Laval type nozzle was developed for comparison purposes. It successfully operated for 1 minute with high performance for such an early test (see Table I), and confirmed the performance of the hydrogen-oxygen propellants as predicted by Paul Webster.[8] This thrust chamber, shown in Figure 2, used externally circulated water as the cooling liquid to obtain basic heat transfer data for design of future motors and to develop a "work horse" motor that could be used to test experimental motor parts of any design.[9] Teflon chamber liners were tried as ablative liners to reduce the heat flow to the chamber. This they did effectively. However, the strength of Teflon when heated proved to be very low, and the material did not have sufficient structural strength to stay in the chamber for more than 15 seconds.

The rate of reaction for gaseous hydrogen-oxygen proved to be extremely rapid compared to liquid phase propellants. Consequently, a marked decrease in chamber volume (or L*) appeared possible. In considering designs of small L* thrust chambers, it seemed probable that a thrust chamber configuration consisting of a cylindrical chamber discharging into the divergent portion of a nozzle should give good performance. This configuration, called a "flared tube" type, appeared desirable from the nozzle cooling standpoint for small thrust chambers. It also appeared to be advantageous for large-thrust, high-altitude thrust chambers because of the rapid increase in nozzle size relative to the

TABLE I

PERFORMANCE OF 100-POUND-THRUST, CONVECTION-WATER-COOLED,

GASEOUS OXYGEN/GASEOUS HYDROGEN THRUST CHAMBER ASSEMBLY, 1946

Chamber Pressures	300 psia	500 psia
Thrust	100 lbs	100 lbs
H_2/O_2 Molar Ratio	3:1	3:1
I_{sp}	311 sec	336 sec
C	10,000 ft/sec	10,850 ft/sec
C*	7440 ft/sec	7970 ft/sec
C_F	1.341	1.360
L*	110 in.	188 in.

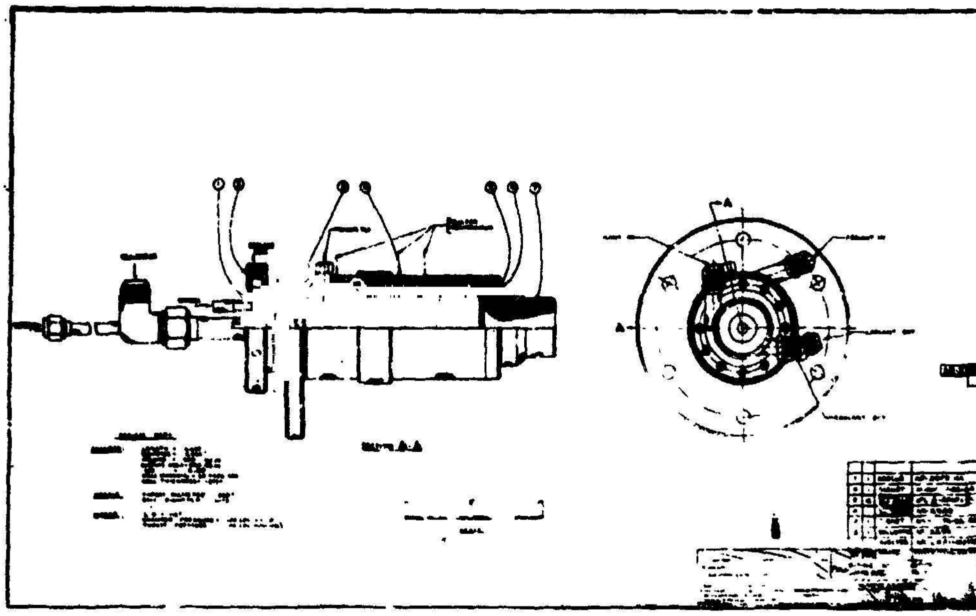


Fig. 2
Drawing of Convection-Cooled Oxygen/Hydrogen Thrust Chamber Assembly

combustion chamber. Consequently, engineers decided to make a flared tube thrust chamber with an L* of 10 from cast electrolytic copper. Results from the test of this thrust chamber led to the fabrication and testing of additional flared tube heat capacitance chambers with L*'s of 2 and 5. Maximum values of specific impulse from the four tests conducted during June 1946 were obtained over a range of L* values from 4.5 to 7 inches. For L* values of 2.0 inches or greater, Isp was not less than 300 seconds. These initial tests indicated that the flared tube thrust chamber configuration was efficient and could result in savings in weight and could simplify thrust chamber design.

As a result of the above tests, two water-transpiration-cooled 300-lb-thrust chambers were constructed from Oilite (Figure 3). However, the coolant flow rates were found to be very low and the porosity of the particular pieces of Oilite much less than that of previous pieces. In tests, the thrust chamber suffered erosion due to the lack of adequate cooling. The final report concluded that the operation of thrust chambers at exhaust velocities above 10,000 ft/sec (310 sec Isp) had become commonplace using gaseous hydrogen and gaseous oxygen. The report recommended that hydrogen, especially liquified hydrogen, offered real and immediate benefits for long-range, large-scale rocket propulsion.[10]

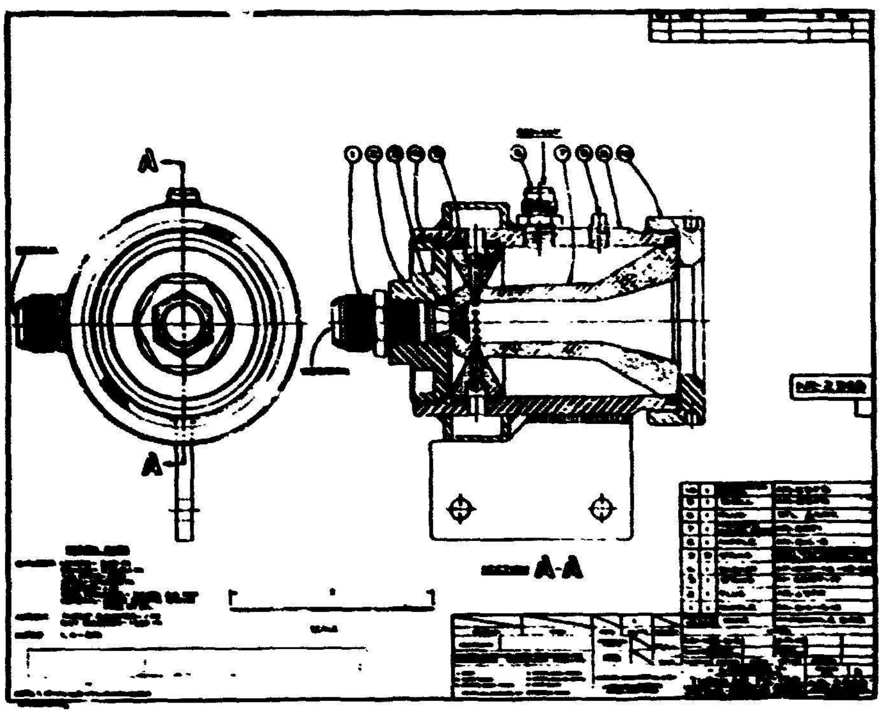


Fig. 3
Drawing of Oilite Flared Tube Thrust Chamber Assembly

III. 300,000-LB-THRUST HYDROGEN-OXYGEN ENGINE DESIGN STUDY

The requirements of Navy Contract NOa(s)-8496 issued in July 1946 called for the design study of a 300,000-lb-thrust rocket engine using liquid hydrogen and oxygen as propellants, and the development of a gaseous-hydrogen-oxygen rocket thrust chamber of 1,000-lb thrust capable of three minutes operation at 300 seconds Isp. The design study would provide a liquid-hydrogen-oxygen rocket engine suitable for use in a high-altitude test vehicle, such as the single-stage satellite vehicle, under study by the Glenn L. Martin Company under Contract NOa(s)-8376[11] and North American Aviation under Contract NOa(s)-8349.[12]

The target specifications included a rated thrust of 300,000 pounds in a vacuum, a specific impulse of 310 seconds at sea level and 425 seconds in a vacuum, a weight less than 4,000 pounds, and a duration greater than 300 seconds.[13] Tables II and III present additional details. Activities devoted to a 1,000-lb-thrust gaseous engine supplied experimental data for the design study, as discussed in Section IV. Various thrust chamber configurations were investigated with special reference to the variation of performance and pressure distribution in tubular and flared tube thrust chambers.

General observations were made in an effort to determine the thermodynamic behavior of the propellant gases. Transpiration cooling was investigated with special reference to the development of sources of supply of the various types of porous materials and to the determination of the flow of coolant required for the protection of the thrust chamber walls.

TABLE II

SUMMARY OF TARGET SPECIFICATIONS FOR 300,000-POUND-THRUST LIQUID-OXYGEN/LIQUID-HYDROGEN ROCKET ENGINE

ENGINE DATA:		
F	=	300,000 lbs
I_{sp}	=	425 seconds
P_c	=	500 psia
w	=	689 lbs propellant per sec
M.R.	=	3 mols H_2 to 1 mole O_2 (6 lbs H_2 + 32 lbs O_2/38 lbs mixture)
D_f	=	4.40 lbs/ft^3 (density)
D_o	=	71.2 lbs/ft^3 (density)
P_p	=	750 psia (pump discharge pressure)
VEHICLE DATA:		
89,000 lbs propellant		
14,100 lbs hydrogen		
74,900 lbs oxygen		
3,220 ft^3 hydrogen		
1,050 ft^3 oxygen		
40-psia gas pressure over propellants in tanks		
15-psia vapor pressure of propellants in tanks		

TABLE III

SUMMARY OF PUMP DESIGN DATA FOR 300,000-POUND-THRUST

LIQUID-OXYGEN/LIQUID-HYDROGEN ROCKET ENGINE

	LH_2	LO_2
Capacity (Q gpm)	11,100	3,660
Propellant Vapor Pressure Heads (H_v ft) (15 psia)	490	30.3
Suction Head (H_s ft)	1,353	117
Suction Head Above Vapor Pressure (H_{sv} ft)	863	86.7
Discharge Head (H_d ft)	24,550	1,517
Total Head (H ft)	23,200	1,400
Cavitation Constant (σ)	.0372	.0621
Shaft Speed (N rpm)	11,000	11,000
No. Impeller per Stage	1	6
No. Stages	4	1
Impeller Diameter (d in.)	13	6.25
Specific Speed (N_s rpm)	1,860	1,180
Water Horsepower ($H.P._w$)	4,610	1,480
Efficiency (%) (Estimated)	75	75
Shaft Horsepower ($H.P._s$)	6,150	1,970
Total Shaft Horsepower Required	8120	

During the 300,000-lb-thrust chamber design study the following assumptions were made: (a) Propellants would be injected as liquid hydrogen and liquid oxygen, (b) the performance of the propellants calculated on a basis of non-dissociation of the combustion products, (c) two percent of the total propellant flow would be diverted for the turbine, and (d) the hydrogen required to cool the motor would be available from the extra mole of hydrogen in the 3:1 molar ratio.[14] The requirements led to the choice of a transpiration cooled flared tube thrust chamber with an extremely high expansion ratio.

It was believed that the flared tube thrust chamber would help keep the weight of the propulsion system down, while the high expansion ratio and the propellants chosen would give the required performance. The large expansion ratio decreased the relative size of the combustion chamber and resulted in a thrust chamber which appeared to be almost entirely an expanding section of a normal de Laval nozzle.

Press and furnace capacity limited the available size of porous metal sheets. Consequently, a metal spray technique under development by J. Wulff of the Massachusetts Institute of Technology was proposed for construction of the 33-percent porous stainless steel combustion chamber inner liner. The material chosen for the outer shell of the thrust chamber was Stainless W, an experimental alloy produced by the United States Steel Company.[15]

Pump capacities, determined by the thrust and specific impulse of the thrust chamber, the mixture ratio and the propellant densities, were 11,100 gallons/minute of liquid hydrogen and 3,660 gallons/minute of liquid oxygen. Because pump vapor pressure head at the entrance of the pump affects pump cavitation, efforts were made to minimize this factor. It was assumed that the propellant would be carried in special tank trucks under atmospheric pressure until being delivered to the test vehicle propellant tanks. It was also assumed that the tanks, pipes, valves, and pumps would be cooled by evaporation of a small quantity of propellant so that when the bulk of the propellant supply had been delivered to the test vehicle propellant tanks it would have practically the same temperature and hence the same initial vapor pressure as it had in the tank trucks. This would result in a hydrogen vapor pressure head of 490 feet of liquid hydrogen and an oxygen vapor pressure head of 30.3 feet of liquid oxygen (see Table III above).

The design of the gas turbine was undertaken with the realization that only a basic design could be formulated without extensive experience in the use of hydrogen and oxygen as gas turbine propellants. It was decided that the propellant combination used for the main thrust chamber would also be used for the turbine because of its high specific impulse, and because it helped keep the test vehicle mass ratio as high as possible. The turbine design data is summarized in Table IV.[16] The final report on the engine design study, issued on 31 March 1947,[17] concluded that a 300,000-lb-thrust engine was entirely feasible. Though many detail problems remained there were no fundamental reasons why such a propulsion system could not be built to propel a single stage satellite vehicle as shown in Figure 4.

The 300,000-lb-thrust chamber was designed to give the performance of 425 seconds specific impulse as specified. A chamber pressure of 500 psia was assumed. Calculations indicate that the total power plant weight would remain essentially constant between chamber pressures of 300 and 500 psia, so the latter figure was chosen. Performance would be affected by the mixture ratio of the propellants. Increasing the hydrogen

TABLE IV

SUMMARY OF TURBINE DESIGN DATA FOR 300,000-POUND-THRUST

LIQUID-OXYGEN/LIQUID-HYDROGEN ROCKET ENGINE

Pump Horsepower ($H.P._p$)	8,120
Accessory Horsepower ($H.P._{acc}$)	80
Turbine Horsepower ($H.P._t$)	8,200
Turbine Fuel	Hydrogen
Turbine Oxidizer	Oxygen
Bucket Velocity (U ft/sec)	1,200
Turbine Wheel Working Stress (S_t lbs/in^2)	30,000
Bucket Temperature (T_b)	1,500
Chamber Temperature (T_c °F) (T_c °F)	1,580 1,130
Mixture Ratio (M.R. mols hydrogen: mols oxygen)	14:1
Chamber Pressure (P_c lbs/in.^{2})	500
Discharge Pressure (P_e lbs/in.^{2})	18
Discharge Temperature (T_e °K)	460
Enthalpy of Products of Combustion in Chamber (h_c Kcal/gm)	1.89
Ratio of Specific Heats of Products of Combustion (Y)	1.37
Enthalpy of Products of Combustion at Exit (h_e Kcal/gm)	0.74
Change in Enthalpy of Products of Combustion (Δh Kcal/gm)	1.15
Discharge Velocity (C ft/sec)	10,180
Heat and Turbulence Loss Factor	0.90
Absolute Velocity of Gases Entering Buckets (V_1 ft/sec)	9,650

TABLE IV (Continued)

SUMMARY OF TURBINE DESIGN DATA FOR 300,000-POUND-THRUST

LIQUID-OXYGEN/LIQUID-HYDROGEN ROCKET ENGINE

Absolute Angle of Gases Entering Buckets (α_1, deg)	26.5
Relative Velocity of Gases Entering Buckets (V_2 ft/sec)	8,600
Relative Angle of Gases Entering Buckets (B_1 deg)	30
Bucket Loss Coefficient (Y)	0.645
Relative Velocity of Gases Leaving Buckets (V_3 ft/sec)	5,550
Relative Angle of Gases Leaving Buckets (B_2 deg)	30
Absolute Velocity of Gases Leaving Buckets (V_4 ft/sec)	4,560
Vectorial Difference in Tangential Components of Absolute Velocities of Gases Entering and Leaving Buckets (ΔV_t ft/sec)	11,000
Mechanical Efficiency of Turbine (Estimated)	0.94
Propellant Consumption (W lbs propellant/sec)	11.7
(W_f lbs hydrogen/sec)	5.46
(W_o lbs oxygen/sec)	6.24

above stoichiometric would reduce the combustion temperature and increase the performance. However increasing the hydrogen content would reduce the density impulse.[18] Since the optimum performance occurred at about 3:1, this mixture ratio was chosen. The calculated performance at sea level was 330 seconds specific impulse at 223,600-lb thrust. As designed, this thrust chamber had an exit diameter of 13 1/2 ft, a chamber diameter of 2 ft, and a length of 22 1/2 ft. It would be hydrogen transpiration cooled, with separate coolant control to 28 compartments. The permeable inner liner consisted

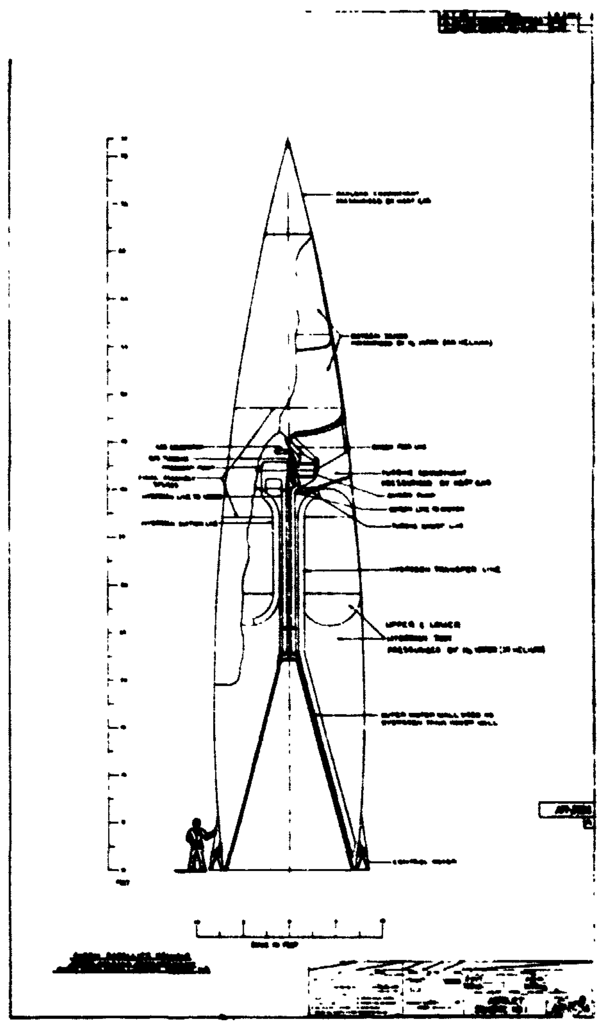


Fig. 4
Drawing of Earth Satellite Vehicle (Glenn L. Martin Scheme No. 5)

of 33% porous stainless steel, 1/8 to 3/16 inch thick. The outer motor wall, circular ribs, and longerons were of Stainless W. The calculated total thrust chamber weight was 2678 lb.

In the pump design studies the factors affecting shaft speed, number of impellers per stage, number of stages and impeller diameter were analyzed. Pump layouts were prepared and weight calculated. In the turbine design study the factors affecting propellant consumption and turbine wheel stresses were analyzed. A turbine layout was prepared and the weight calculated. As a result of these studies it was concluded that a turbopump of the following description (Figure 5), would be suitable for pressurizing the liquid-hydrogen and liquid-oxygen propellants for a 300,000-lb-thrust high-altitude rocket engine:

Weight of turbopump	850 pounds
Length of turbopump	70 inches
Maximum diameter of turbopump	30 1/4 inches

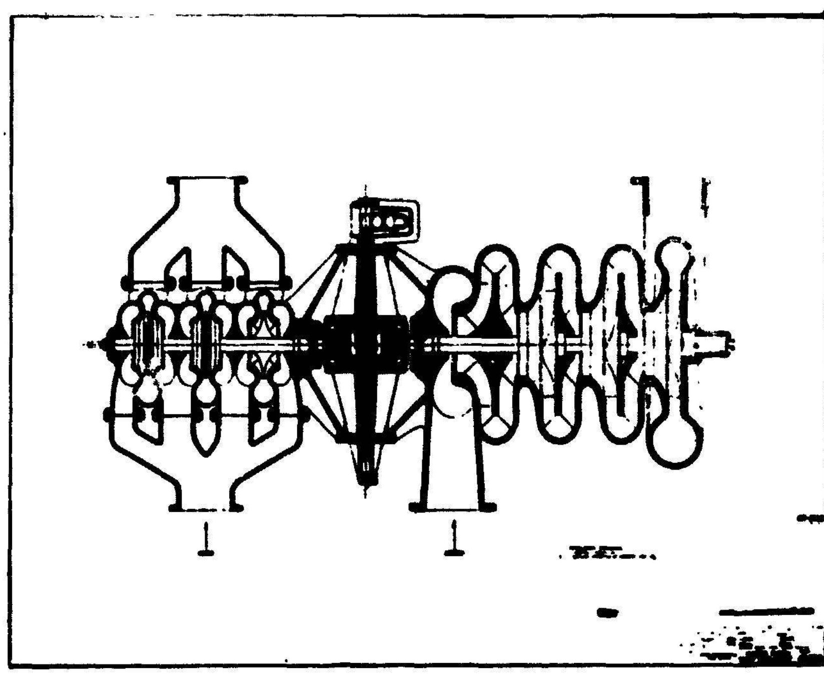


Fig. 5
Drawing of Turbopump Assembly, 300,000-Pound-Thrust Engine

Four small engines mounted parallel to the center line appeared the best method of controlling roll, pitch, yaw, and for changing the trajectory of the satellite vehicle. These engines would operate continuously and would change the direction of their thrust by pivoting their thrust chambers. The size of these control engines could not be established without a fi[illegible] determination on the amount of turning moment required to control the vehicle. Turning the small thrust chambers with servo mechanisms presented no special problems.

A schematic diagram showing the major elements of the main engine controls, the small engines, and the small engine controls is presented in Figure 6. The valves and

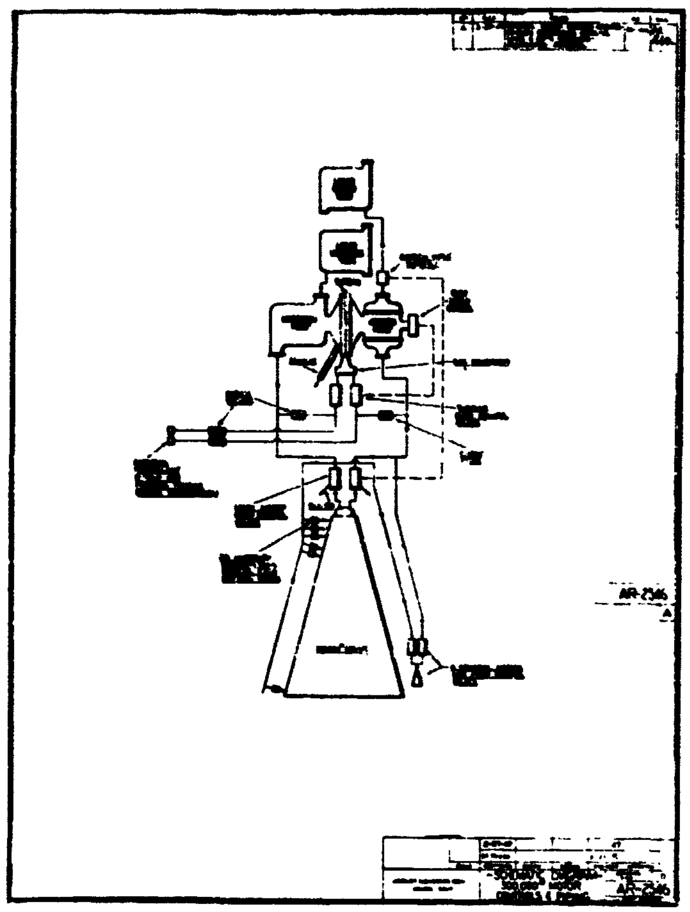


Fig. 6
Schematic Diagram, 300,000-Pound-Thrust Engine, Controls

plumbing, because of the very large size and low temperature, offered problems well beyond any available equipment. The weight of these elements including the necessary insulation was estimated at 355 pounds.

The dry weight of the engine, not including the control engines, was as follows:

Thrust Chamber	2678 pounds
Turbopump unit	850 pounds
Valves and plumbing	355 pounds
	3883 pounds

The total weight given above was within the target weight. Aerojet recommended that a development program be instigated in order to find solutions to the fabrication and propellant handling problems brought out by this design study.

IV. 350-LB-AND 1000-LB-THRUST GASEOUS-HYDROGEN-OXYGEN THRUST CHAMBERS

During the work to produce a 1,000-lb gaseous engine, engineers developed a 350-lb-thrust, 5 1/2-inch L*, flared tube, water-convection-cooled, thrust chamber that was operated undamaged for periods of more than 1 minute with gaseous hydrogen and gaseous oxygen as propellants.[19] This thrust chamber (Figure 7), operating with a 3 H_2/O_2 molar mixture ratio at 300-psia chamber pressure, delivered 330 seconds specific impulse (95% of theoretical Isp).

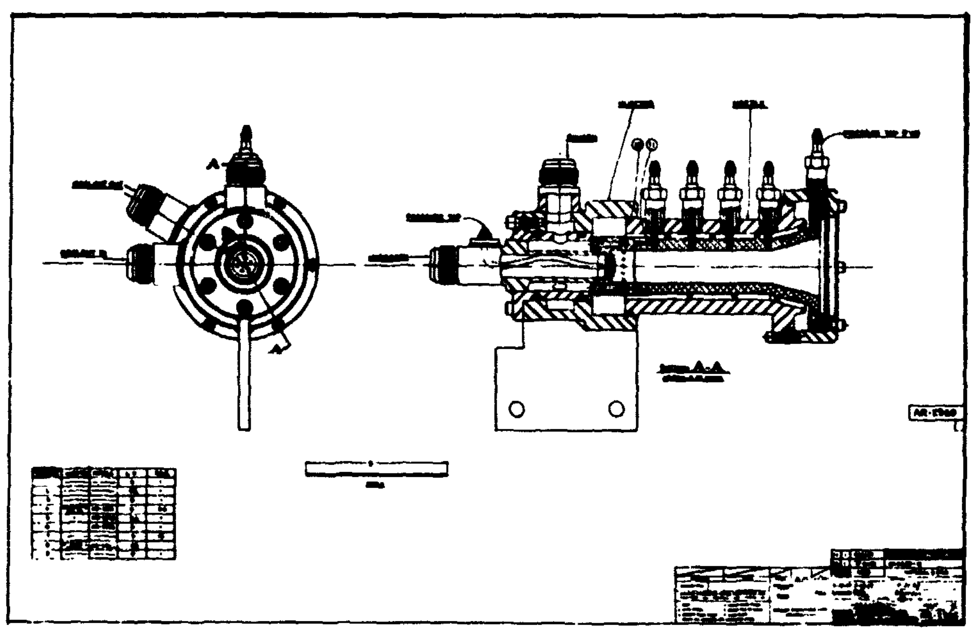


Fig. 7
Drawing of 400-Pound Liquid-Cooled Thrust Chamber Test Assembly

On June 26, 1947, a 1000-lb-thrust chamber which completely fulfilled the aims and specifications of Contract NOa(s)-8496 was successfully tested. The specification test was made on the thrust chamber (Figure 8) with a 19 1/2 percent coolant water flow. The configuration was a modified flared tube, with the nozzle portion of the motor water transpiration cooled, using porous nickel as the liner material. The injector was convection cooled with the water subsequently used for transpiration cooling. The test run was terminated by the operator after 190 seconds elapsed time, of which 183 seconds were at full performance.[20] The minimum performance maintained for the 183 seconds period appears below.

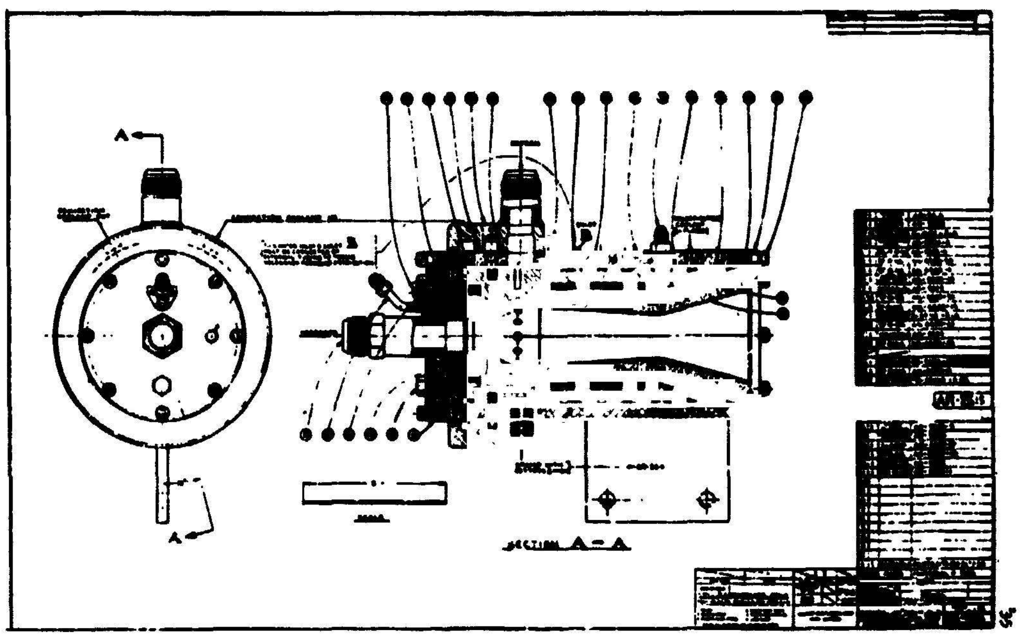


Fig. 8
Drawing of 1000-Pound-Thrust, Flared-Tube, Hybrid-Cooled Thrust Chamber

Thrust = 1230 lb
Isp = 309 seconds
P_c = 500 psia
H_2/O_2 = 4:1 molar ratio
Coolant = 19 1/2%

These experimental thrust chambers were designed on the same principals used in the design of the motor of the 300,000-lb-thrust rocket engine; in general, they confirmed the design data for the large thrust chamber.

V. LARGE-SCALE PRODUCTION AND HANDLING OF LIQUID HYDROGEN FOR THE XLR16-AJ-2 ENGINE AND PTV-N-3 VEHICLE REQUIREMENTS

During July 1947, Contract NOa(s)-8496 was amended to authorize the development of a liquid oxygen-liquid hydrogen engine, the XLR16-AJ-2, suitable for use in a small scale version of an earth satellite vehicle, the PTV-N-3, under study by the Glenn L. Martin Company.

The initial targ::: specifications of the engine were:[21]

Thrust	2000-3000 lb
Duration	60 seconds
Isp	303 seconds minimum
Weight	,5 lb maximum
Engine Inlet Pressure	35 psia maximum

This section of the paper describes the development of the Aerojet liquid hydrogen plant which supplied the original 4.7-pound (30 liter)-per-hour requirements, and, subsequently in August 1948, the requirements of 12 pounds (76 liters) per hour when the engine thrust was increased to 3000 lb, nominal, and its duration of burning extended to 180 seconds.[22] The revised XLR16-AJ-1 engine and PTV-N-3 vehicle specifications are shown in Tables V and VI. In 1947, the requirement for liquid hydrogen for the Aerojet contract had been initially estimated at 3000 to 6000 pounds depending on the cost. In addition, the Jet Propulsion Laboratory at the California Institute of Technology required from 1300 to 2000 pounds for an Army Ordinance sponsored program.[23] This combined demand could be met by a 25 liters/hour hydrogen liquefier similar to the installation at the Ohio State Cryogenic Laboratory, designed by Herrick L. Johnston.[24]

TABLE V

SUMMARY OF SPECIFICATIONS FOR THE XLR16-AJ-2 ROCKET ENGINE

Propellants:	Liquid hydrogen and liquid oxygen.
Motor:	Single cylinder, fully gimbaled. Maximum deflection ±15° with approximately 5 cycles/second response. Servo mechanism not a pc :ion of the motor.
Mixture Ratio:	4-1/2:1 hydrogen to oxygen molar ratio, overall. (Note this change from 4:1 is made to facilitate motor cooling and will require Bureau of Aeronautics approval).
Specific Impulse:	303 sec minimum.
Thrust:	3000 lb nominal at sea level. (Exact definition is 10 lb/sec of propellant consumption at above mixture ratio.)
Duration:	3 minutes nominal.
Propellant Pressurization:	Turbopump, same propellants.
Weight Breakdown:	Motor, pumps, and valves . . . . 85 lb Gimbal ring . . . . 18 lb
Suction Pressure (NPSH):	35 psia nominal, subject to experimental verification.

TABLE V (Continued)

SUMMARY OF SPECIFICATIONS FOR THE XLR16-AJ-2 ROCKET ENGINE

Dimensions:	To fit into a truncated cone approximately 2 ft dia. at motor end, 3 ft diameter at turbopump end (this diameter to be considered fixed) and approximately 4 1/2 ft long.
Electric Power:	24 volts dc from external source.

TABLE VI

SUMMARY OF SPECIFICATIONS FOR THE PTV-N-3 PROPULSIVE TEST VEHICLE

Propellants:	Liquid hydrogen and liquid oxygen.
Weight:	2000 - 2500 pounds gross.
Payload:	95-lb telemetering gear.
Mass Ratio:	0.65 - 0.7 propellant weight/gross weight nominal.
Weight:	1300 - 1750 pounds.
Structure:	Pressurized thin-skin integral-tank construction. (Stainless Steel estimated at .020-inch thick.)
Pressurization:	Evaporated propellants at 35 psia nominal.
Power Plant:	Aerojet XLR16-AJ-2 rated at 3000-lb thrust nominal.
Length:	25 - 29 ft depending on gross weight.
Diameter:	3 ft maximum.
Shape:	Ogive nose (65-ft radius) 13 ft long. Cylindrical body (3-ft dia.) 11 ft. Boat Tail Motor Corp. (1.96 ft minimum) 4 1/2 ft.
Control:	Fully gimbaled motor for pitch and yaw control. Turbine exhaust for roll control. Controls powered by hydraulic pressure.
Filling:	External support for structure allowing propellants to be held at 1 atm pressure (or subcooled) until immediately prior to take-off. Oxygen filling connections in space between tanks, hydrogen filling in motor compartment or in space between tanks.
Insulation:	a) Internal fuzz type. b) External drop-away blanket, helium filled.
Motor Compartment:	Helium filled prior to firing.
Electrical Supply:	24 volts dc.

Commercial producers were contacted but they either believed large-scale production of liquid hydrogen to be unfeasible, or quoted prices considerably higher than would be incurred if Aerojet constructed and operated a plant at Azusa. Additional checking on the cost of an Aerojet-built plant indicated that the total cost to build and operate it would be $100,000. In view of the considerable difference between the commercial and the Aerojet estimate in both cost and convenience, a request was made to the Bureau of Aeronautics for permission to erect and operate the plant at Azusa. Aerojet received verbal approval in September 1947, and action was initiated to obtain the consulting services of H. L. Johnston of Ohio State University.

Dr. Johnston spent October 13-16 at Aerojet in conferences on the detailed design of a plant utilizing commercial gaseous hydrogen and liquid nitrogen as a precoolant instead of liquid air. He returned to Ohio State to prepare drawings of the special units required for the liquefaction cycle and to supply certain design specifications to assist Aerojet's design of the more conventional special units. Nearly all commercial component parts were located and specified for purchase during October. Preliminary plant layouts established building requirements.

Following the receipt of formal authorization on December 16, 1947, purchase orders were placed for all major commercial units required except the freon refrigeration system for which complete specifications were not yet available.[25] Construction of the Aerojet building to house the liquefier also began in December. Transport of the liquid hydrogen pressure vessels to and from the liquefier building was planned to be done by a heavy duty lift truck.

The Aerojet cycle was a modified form of the cycle used at the Ohio State University Cryogenic Laboratory.[26] The flow of hydrogen from gas to liquid is shown in Figure 9. The Aerojet plant utilized the liquid nitrogen precooled Joule-Thomson (Onnes) cycle because suitable heat-exchanger cryostat designs were available for a plant of this size. The use of an established cycle was necessary because of the urgent need for a propellant in the test programs. Major modifications made in the purification portions of the cycle included:

1) Provision to utilize a commercial gaseous hydrogen supply,
2) Continuous catalytic removal of the oxygen impurity in the commercial gaseous hydrogen supply,
3) Adsorption purification at liquid nitrogen temperatures to remove any remaining gaseous nitrogen and oxygen from the hydrogen upstream of the hydrogen expansion valve,
4) Utilization of a commercial liquid nitrogen supply for the liquefier precooler interchanger, and
5) Use of parallel purification units where necessary to allow continuous operation over extended periods.

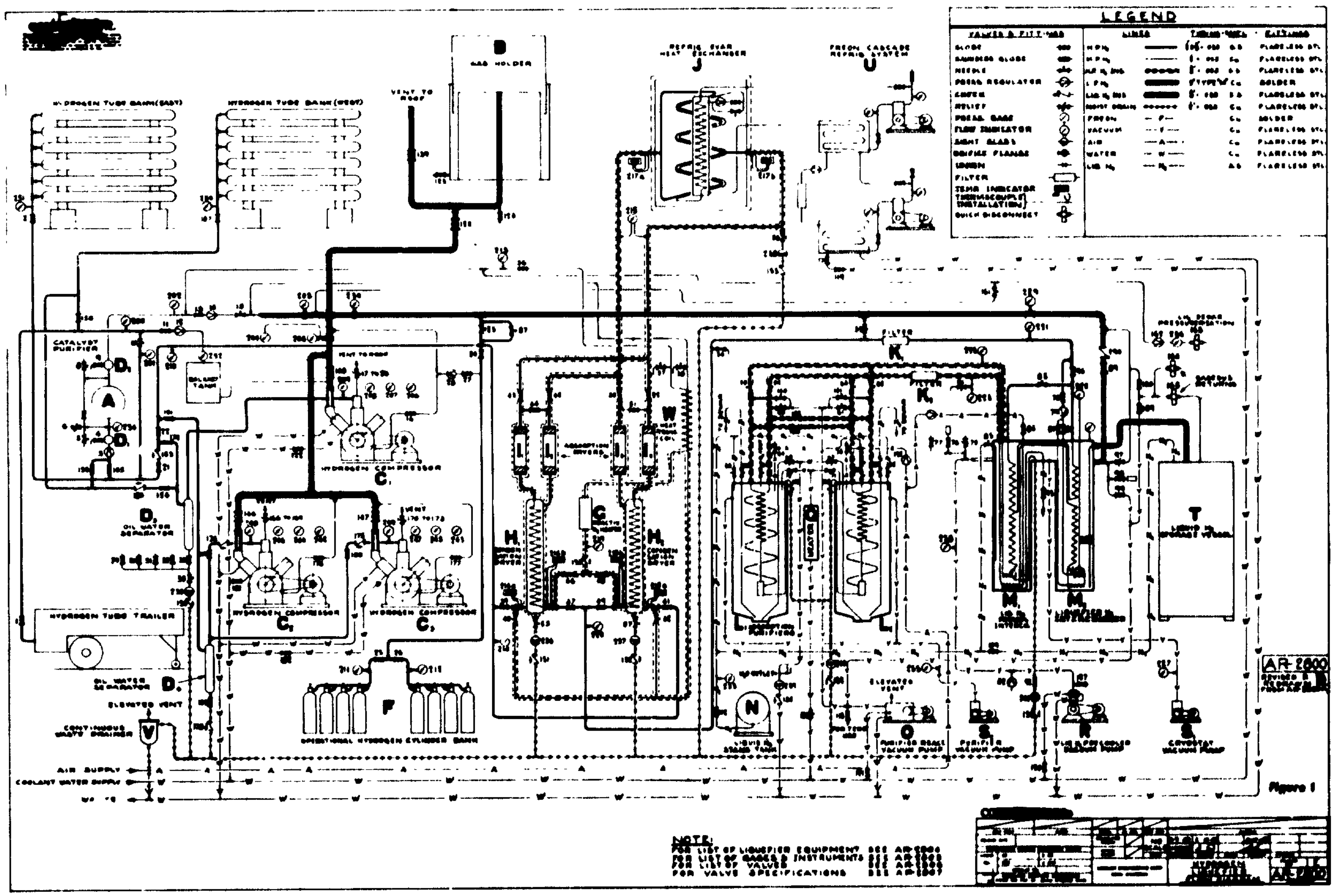


Fig. 9
Hydrogen Liquefier Flow Diagram

On May 21, 1948, a ten-hour operating test was made on the hydrogen liquefier cryostat assembly which had been completed and installed in the liquefaction plant system at the Ohio State University Cryogenic Laboratory. The test was supervised by H. L. Johnston. H. L. Coplen observed and assisted in the operation of the unit. The performance of the unit was very satisfactory, especially in its economy of liquid air for precooling.[27] The liquefier cryostat assembly was subsequently shipped and installed in the Azusa plant.

The first production of liquid hydrogen occurred on September 3, 1948, and furnished about 12 liters of liquid hydrogen. Subsequently, a number of improvements, adjustments, and repairs were made to the system to correct discovered faults. The plant was again operated on September 21, 22, and 23. On September 22, approximately 120 liters were produced of which a net of 75 liters (11.75 lbs) was supplied to the Jet Propulsion Laboratory of the California Institute of Technology for their rocket test program.

As originally designed, the plant had a production capacity of 4.7 pounds (30 liters) per hour and was so operated from September 1948 to March 1949. Because of the increase in propellant requirements resulting from the 3000-lb-thrust chamber tests, the plant shown in Figures 10, 11, and 12 was improved to increase the capacity to approximately 12 pounds (76 liters) per hour, and it was operated on a three-shift basis

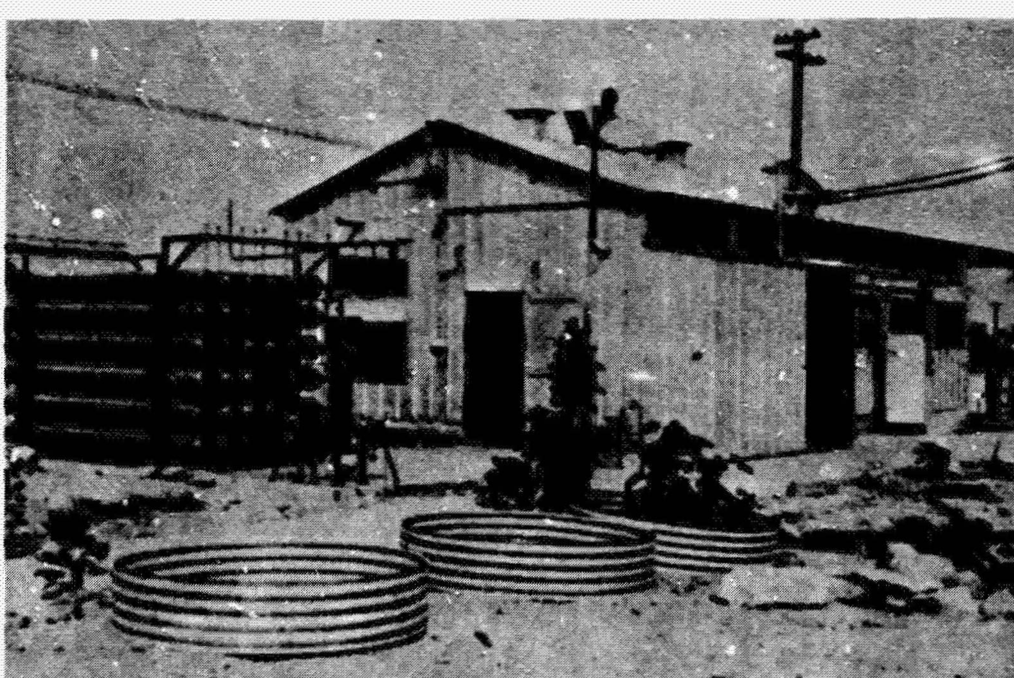


Fig. 10
Aerojet Liquid-Hydrogen Plant

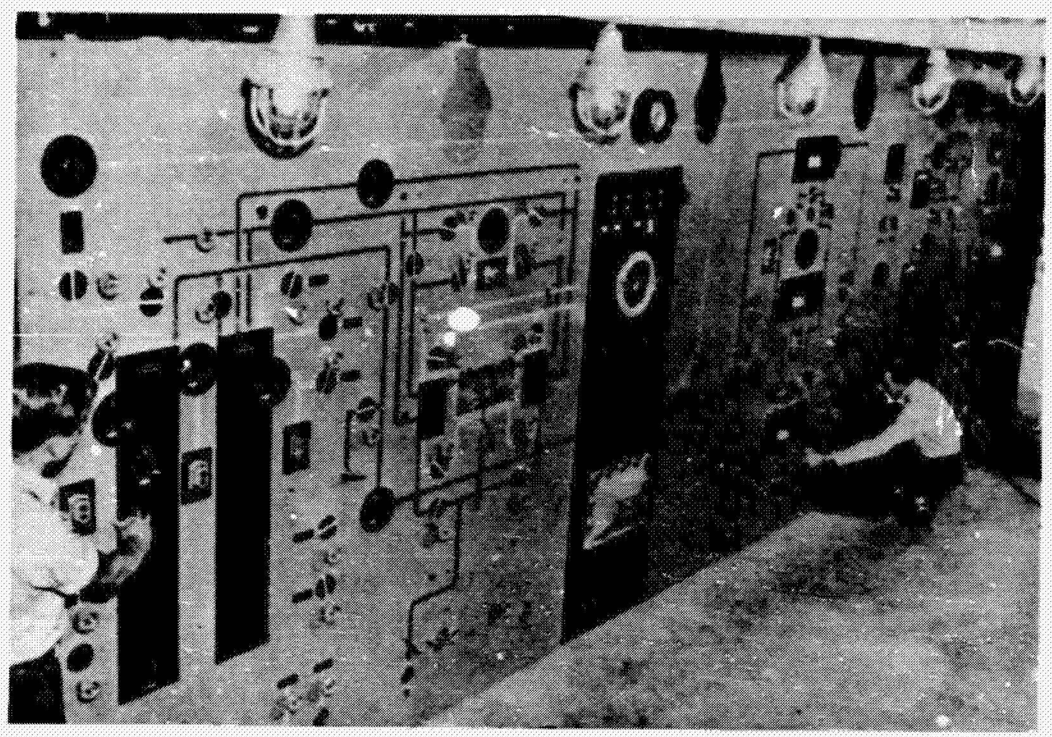


Fig. 11
Liquid-Hydrogen Plant Control Panel

at this capacity from March to June 1949. From September 1948 to June 1949 t gross plant production was 7400 pounds (47,000 liters) of liquid hydrogen, with about 5300 pounds (33,700 liters) being produced in the last four months of this period at a production cost of about $13.48 per pound. The use of commercial sources of gaseous hydrogen and liquid nitrogen is responsible for this comparatively high production cost.

Two 100-lb-capacity liquid-hydrogen storage vessels were designed, constructed, and used extensively.

VI. STUDY OF THE RELATIVE IMPORTANCE OF PROPELLANT DENSITY AND SPECIFIC IMPULSE ON SATELLITE AND VEHICLE PERFORMANCE

The study of the relative importance of propellant density and specific impulse on the performance of a satellite test vehicle was presented in a special report in May 194? [28] Because the contract NOa(s)-8496 was pointed toward the development of a single-stage satellite vehicle, the analysis of the relative influence of specific

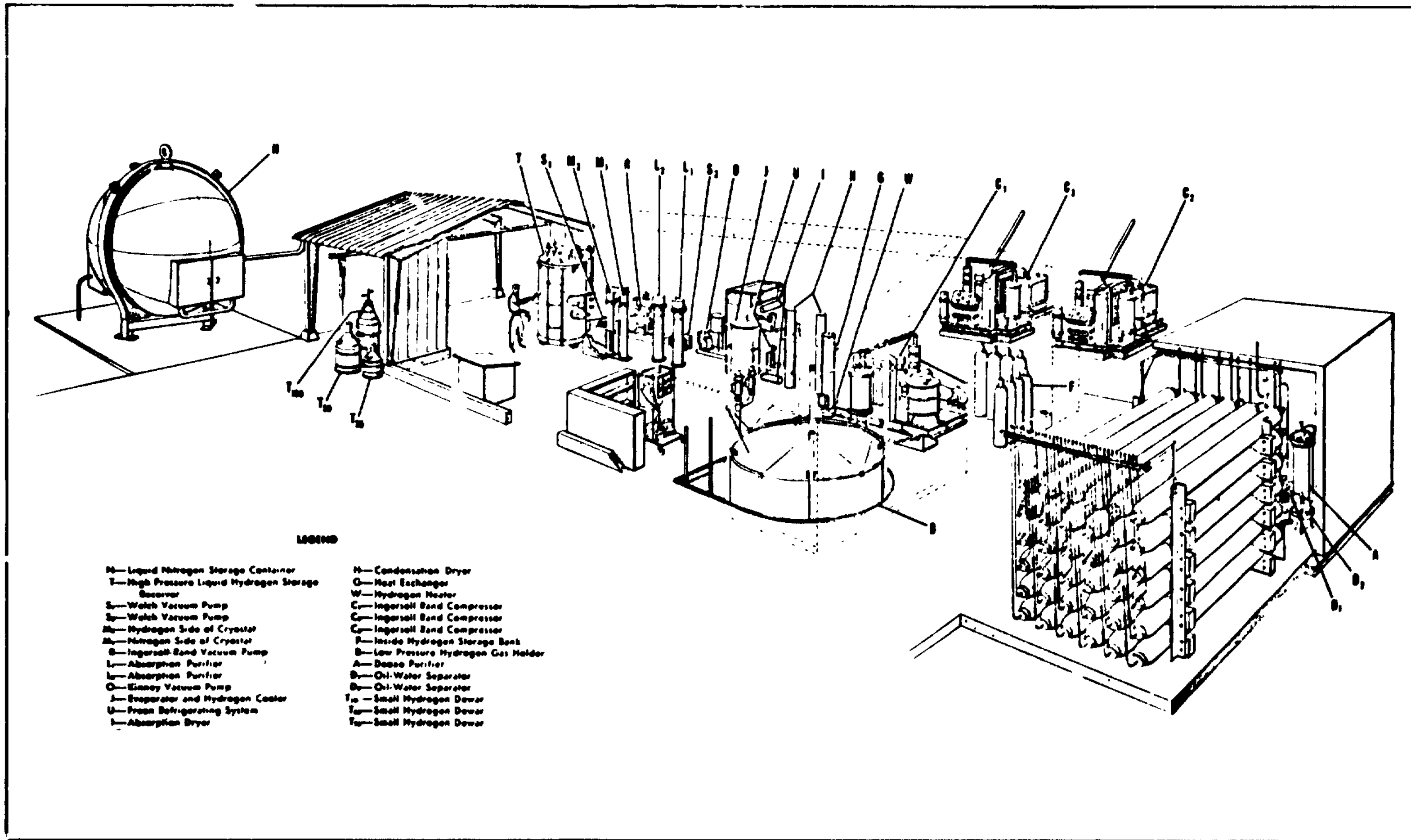


Fig. 12
Hydrogen Liquefier Plant Layout

impulse and propellant density on vehicle performance involved a vehicle with the following requirements:

1) 300-mile-altitude satellite orbit
2) 1000-lb payload
3) Single-stage vehicle+
4) Sea-level, non-boosted takeoff

After some consideration of the complexities of the actual trajectory required to reach a stable orbit, engineers sought the vertical altitude which would result from imparting orbital energy to the vehicle. In determining the energy of a satellite vehicle in a stable orbit the variation of gravity with altitude was taken into account. The energy of a satellite in an orbit at 300-mile altitude was assumed to be 10.12×10^6 ft-lb/lb (sea-level ft-lb of energy). Vertical trajectory calculations were performed by a Runge-Kutts stepwise integration of the basic equation of motion, using a perturbation technique including the effect of drag. The altitude and kinetic energy were added at the end of burning to obtain an equivalent total energy expressed as an energy altitude in feet. The results were presented as a series of curves giving this energy altitude as a function of the mass ratio for different sets of values of the standard specific impulse, vehicle density loading, and total initial acceleration.

These calculations served to show the effect of specific impulse on vehicle performance. The effect of propellant density is concealed by the fact that vehicle density loading and mass ratio, both dependent on density, are not expressed as direct functions of density. The gross vehicle weight was determined as the sum of all the vehicle components divided into suitable groups, and expressed as functions of the pertinent vehicle and propellant properties. It was found possible to evaluate the motor, pumping plant, piping, valves, and controls by means of available data on existing vehicles and by detailed design studies.

The structural weight did not lend itself to the above-noted method of analysis. In order to achieve a mass ratio adequate for satellite performance, it was necessary to use unconventional fabrication techniques. The best method had been proposed by the Glenn L. Martin Company, Baltimore, Maryland, and North American Aviation Inc., Inglewood, California. It comprised the use of a highly stressed thin skin in a pressurized structure with integral propellant tanks. The pressure placed all the skin material under tension sufficient to prevent any compressive loads. The ribs and stringers required to carry compression loads in conventional structures could be omitted in this way, and the skin could be highly stressed.[29]

+See R. Cargill Hall, "Earth Satellites, A First Look By The United States Navy," in this volume - Ed.

The sum of all the vehicle components, propellant and payload weights was then used as the gross weight. The results were prepared in a series of curves in which mass ratio and density loading were shown as functions of gross weight for a given set of values of propellant density, total takeoff acceleration, and specific impulse at standard conditions. Almost none of the vehicle components were weight dependent on specific impulse. The assumptions and methods used in this work had been evaluated by H. S. Tsien of the Massachusetts Institute of Technology, a consultant for this program.[30] These calculations served to show the relation between propellant density and mass ratio, and between propellant density and weight loading for a number of vehicles suitable for satellite or extreme long-range performance.

The results shown in Figure 13 indicated that specific impulse is much more important than propellant density. This result contradicted the findings of many reputable groups and individuals in the field of rocketry at that time. The author, Robert Gordon, cautioned that the results of the report should not be indiscriminately applied to all rockets under all conditions. Such a practice would be as indefensible as was the practice of applying A-4 (V-2) data and performance to every new application no matter how far removed from the A-4.

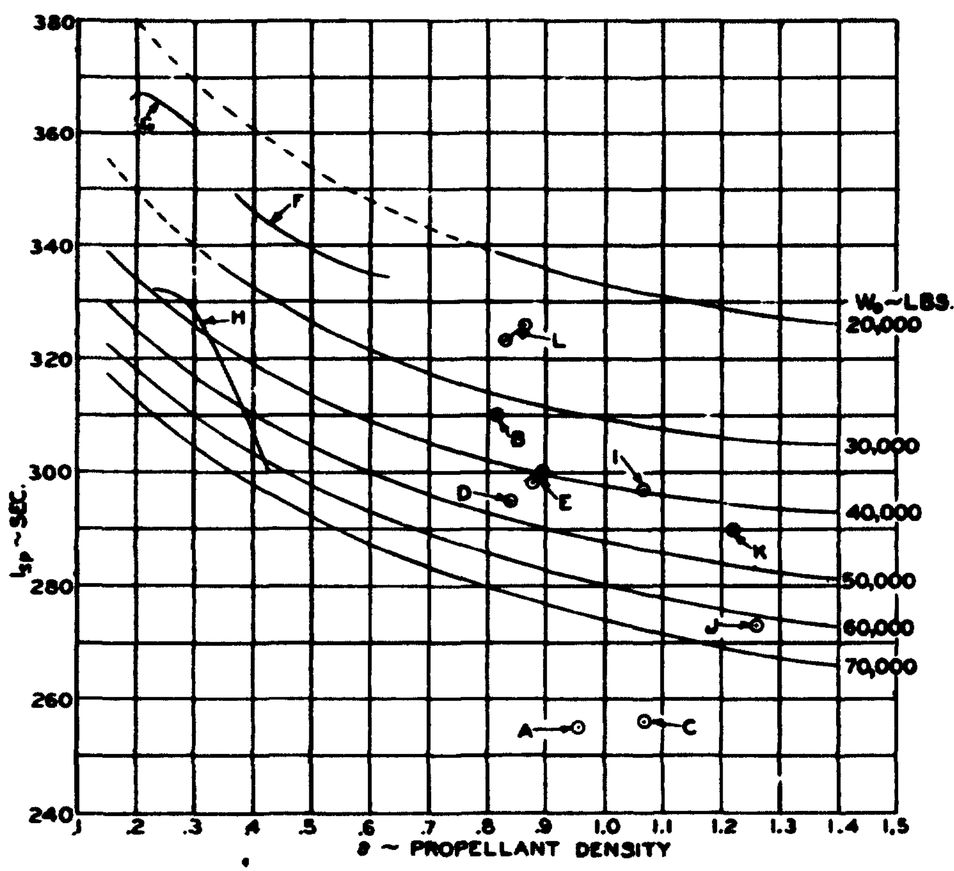
Among the report's conclusions:

1) If it were desired to build rockets for extremely long range, i.e., 15,000 miles to infinity, it would be necessary to achieve vehicles with high mass ratios. In this type vehicle, specific impulse is of greater importance than is propellant density.
2) The minimum size vehicle capable of satellite performance varied from about 25,000 to 60,000 pounds gross weight for known liquid propellants.
3) The best propellants for satellite vehicles based on minimum gross weights would be:

	Propellant	Minimum Gross Weight, Pounds
1.	O_3 and H_2	23,000
2.	F_2 and H_2	25,000
3.	F_2 and Li	25,000
4.	O_2 and Al $(BH_4)_3$	33,000
5.	O_2 and H_2	38,000
6.	F_2 and N_2H_4	40,000

VII. THE PUMPING OF LIQUID HYDROGEN

In 1946 Dietrich Singelmann, a German rocket-engine designer employed by the Air Material Command, revealed the unique characteristics of a centrifugal pump



ASSUMPTIONS:

1. $K_0 = 2$ g
2. DRAG DATA FROM A-4
3. MOTOR DESIGNED FOR 5J,000 FT. (P_c = 560 γ = 1.2)
4. ORBITAL ALTITUDE = 300 MILES
5. PAY LOAD = 1000 LBS.
6. THE EFFECT OF
   a. g
   b. ALTITUDE
   c. DRAG

   ON PERFORMANCE HAS BEEN INCLUDED.
7. W_0 GROSS TAKE OFF WEIGHT
8. DASHED PORTION OF W_0 CONTOURS REPRESENT PROPELLANT COMPARTMENT SKIN THICK. LESS THEN .010 INCH
9. I_{sp} VALUES AS PLOTTED ARE 95% OF THEOR. * I_{sp} AT 300 P.S.I.A. P_c
10. STRUCTURE OF VEHICLE IS THIN STAINLESS STEEL SHEET STABILIZED BY INTERNAL GAS PRESSURE.

* SHIFTING EQUILIBRIUM BASIS.

KEY	OXID.	FUEL	KEY	OXID.	FUEL
A	O_2	NH_3	G	O_3	H_2
B	O_2	$AL(BH_4)_3$	H	O_2	H_2
C	O_3	N_2H_4	I	F_2	N_2H_4
D	O_2	B_2H_6	J	O_3	N_2H_4
E	O_2	B_5H_9	K	F_2O	N_2H_4
F	F_2	H_2	L	F_2	LI

Fig. 13
The Relative Influence of Propellant Density and Specific Impulse on the Size of a Satellite Vehicle

consisting of a light-weight radial vane rotor with a concentric case that he had developed for the oxidizer pump of the BMW 109-718 Booster Rocket Engine manufactured by the Bayerische Moteren Werke. The rotor of this pump (Figures 14A and 14B), consisted of a hub with three or more radial thin metal vanes. A partial shroud extending to perhaps 1/3 of the radius, was used on the back side of the vanes. No forward side shroud was used. The discharge of the pump was taken through two or more round tangential orifices whose diameter determined the discharge characteristics of the pump. Dr. Singelmann described one pump in particular which operated at 25,000 rpm and developed 82 atmospheres pressure with nitric acid. The impeller was 75 millimeters diameter by 12 millimeters wide at the tip and pumped 7 liters or 25 lbs/sec at an efficiency of 45 percent. This same pump developed about 75 atmospheres pressure with water.[31]

Subsequently, the group at Ohio State University Research Foundation No. 264, constructed such a pump based on Singelmann's designs under Contract W33-038 ac-14794 (16243). The results were discussed during a visit by Aerojet personnel in November 1947.[32] Due to the extremely low temperature of liquid hydrogen, the Ohio State Group had found it necessary to provide some method of insulating the pump from room temperature. It appeared practical from the test standpoint to use a large vacuum jacket tank and submerge the pump in the particular fluid being handled. Results of pump tests with water and liquid nitrogen were good, but with liquid hydrogen the pump cavitated most of the time. This was caused by an excessive heat leak into the pump resulting from incomplete submersion. The rapid evaporation of the liquid hydrogen in the submersion tank made maintaining an adequate liquid level extremely difficult. Nevertheless some of the points seemed to fall on the anticipated head flow curve at higher flows.[33]

A subsequent interview with Singelmann after the Ohio State visit resulted in recommendations for pumps useful to the XLR16-AJ-2 engine program.[34] During the following month, it was learned from an interview with M. N. Nyborg of the Naval Air Missile Test Center, Point Mugu, that two pumps from the BMW 109-718 Jato Unit (for the ME 262 airplane) could be obtained to provide an early test on the basic characteristics of such a design.[35] Performance tests with water of this 3.3-inch-diameter oxidizer pump revealed that the characteristics of this pump were superior to those of conventional centrifugal pumps in the same speed range. Of greatest interest was the constant flow characteristics when operated at reduced head.[36] Subsequently performance and cavitation tests with water were conducted with the Aerojet designed, AR-2720, a 6-inch diameter, shroudless, radial vane, centrifugal pump with a conical discharge diffuser.[37]

The impeller, bearings, and seals of the AR-2720 pump were modified for testing with liquid hydrogen, and the pump reidentified as AR-2797. This pump incorporated an impeller machined from a single billet of 17ST aluminum alloy. The bearings used in the

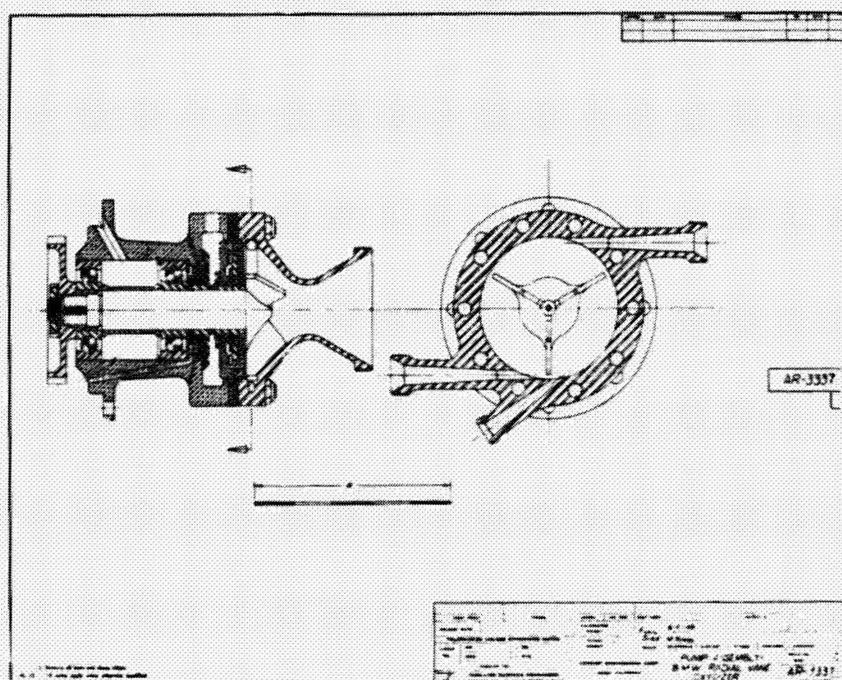


Fig. 14A
Drawing of BMW Radial Vane Oxidizer Pump Assembly

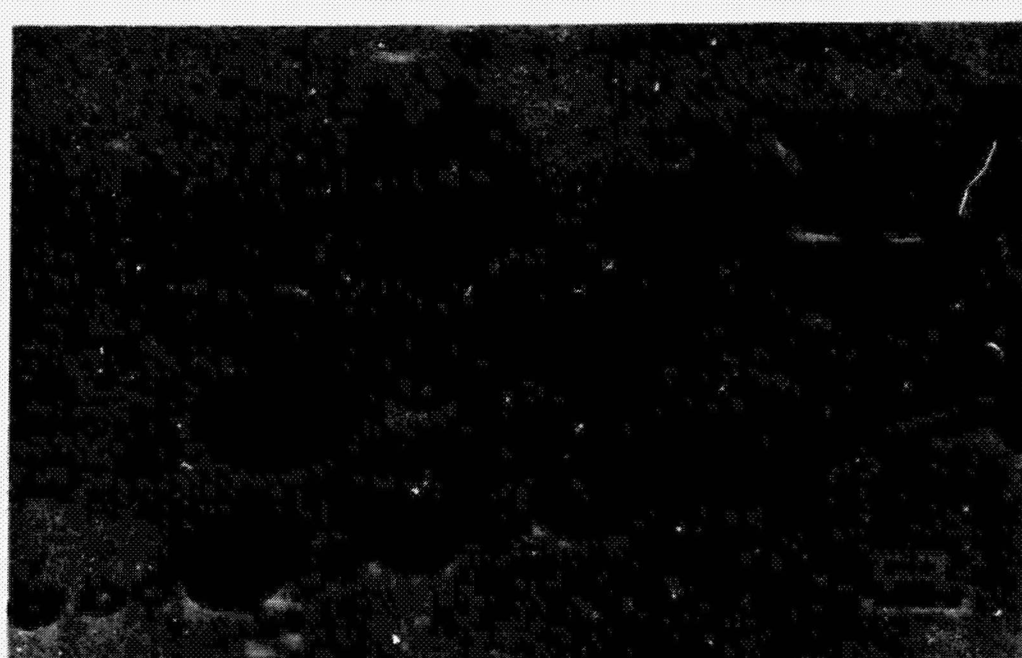


Fig. 14B
BMW Oxidizer Pump Disassembled

pump were deep-groove ball bearings having bronze cages. Micarta clearance seals were used in the pump for hydrogen testing. Provision was made for pressurizing the seal section with helium in order to minimize hydrogen leakage from the impeller housing. The construction of the pump is shown in Figure 15.[38] The pump was designed to the following specifications:

$$N = 40{,}000 \text{ revolutions per minute}$$
$$H_{max} = 23{,}300 \text{ feet}$$
$$Q_{max} = 87 \text{ gallons per minute}$$

Difficulty was experienced in cooling the pump to a temperature low enough to permit filling the pump case with liquid hydrogen. This was caused by the small size of the diffuser through which the gas generated during cooling had to escape. This trouble was eliminated by adding four 1/4-inch-diameter holes in the pump case, and connecting these to a bleed valve which was closed after cooling the pump and before starting the test run. With this difficulty overcome, liquid hydrogen was pumped successfully. A flow rate of 0.68-lb per second and a pressure rise across the pump of 400 psi were obtained. The head-vs-capacity curve is shown in Figure 16. This curve shows the unique property of this type of pump: that of operating at a constant capacity over a wide range of pressures, which in this case was from 40 percent or lower up to 85 percent of the maximum pump-pressure rise. This unique property offered advantages when used in a rocket engine pumping plant. The flow rates could be controlled by the pumps, eliminating the need for flow-regulating valves and resulting in a basically simpler and lighter engine.

For the AR-2797-type pump, the head varied approximately as the square of the speed while the capacity varied directly as the speed. At the design speed of 40,000 rpm, and the corresponding peripheral velocity of 1,047 feet per second, the head would be in excess of 20,000 feet and the capacity would be approximately 86 gallons per minute. These results were in approximate agreement with the predicted values. The power limitation of the pump test stand prevented testing at 40,000 rpm, the design speed for the pump. The pump shown in Figure 17 was operated successfully with as little as 25 psi difference between this suction pressure and the vapor pressure. This indicated that a suction pressure as low as 40 psia could be used in a high-altitude test vehicle or missile in which the liquid hydrogen reached the pump inlet with a vapor pressure of approximately 15 psia.

The pumping of liquid hydrogen was demonstrated successfully for the first time with the lightweight, high-speed centrifugal pumps, in a series of tests conducted during March 1949.[39]

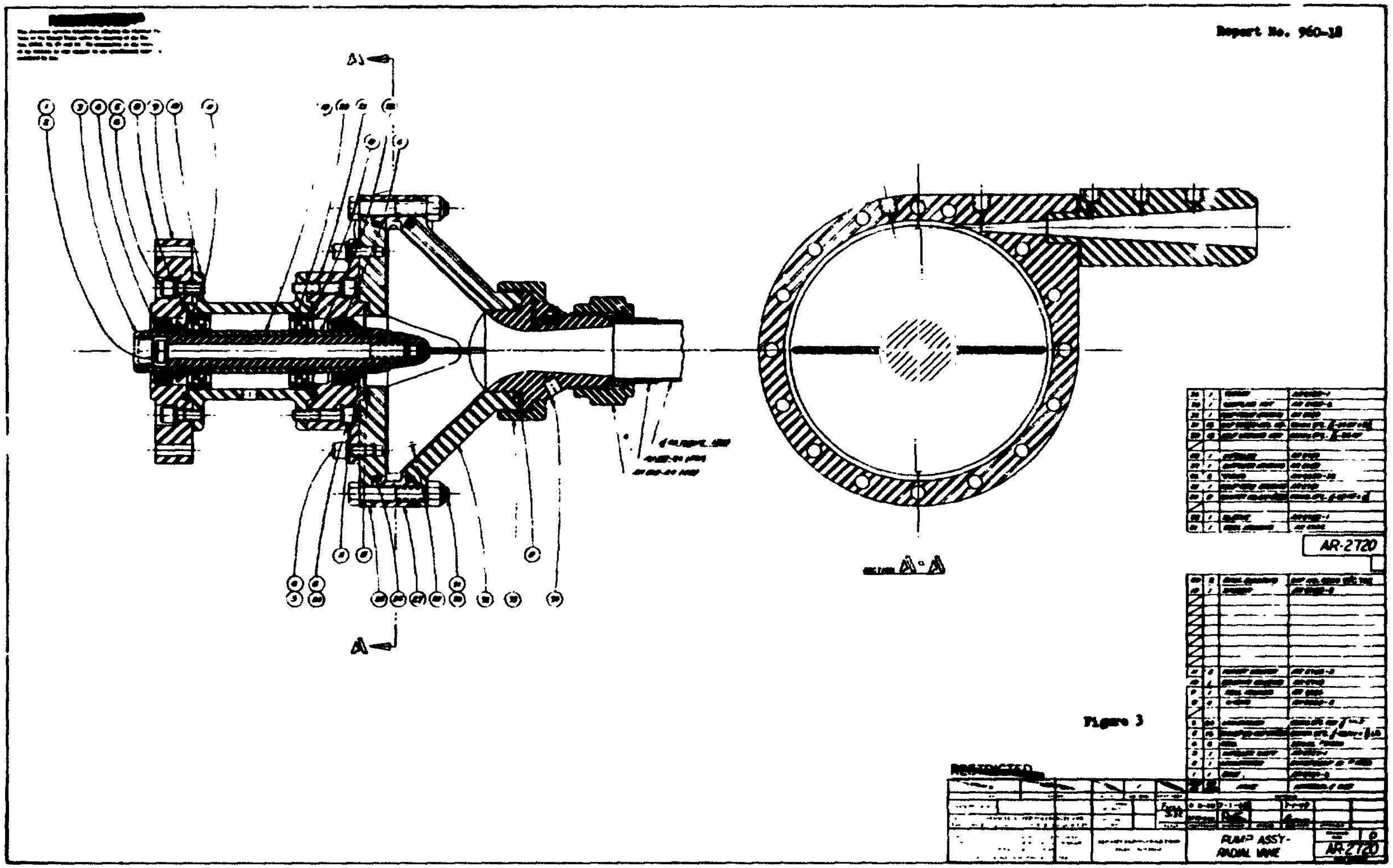


Fig. 15
Drawing of Radial Vane Pump Assembly

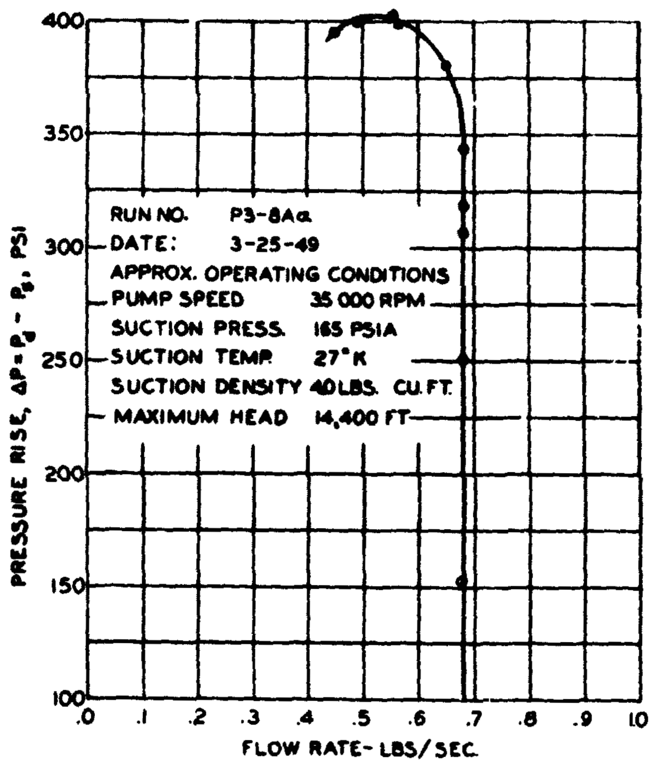


Fig. 16
Performance Curve of Liquid-Hydrogen Pump (Aerojet)

VIII. THE EFFECT OF JET OVEREXPANSION AND SEPARATION ON ROCKET THRUST CHAMBER PERFORMANCE

Early theoretical analysis had indicated that a rocket engine designed correctly r high altitudes (low ambient pressures) would suffer serious losses in performance when erated at low altitudes (high ambient pressures).[40] The results (Figure 18) indicated at although the performance of a sea-level motor was 20 percent less than that of a gh-altitude motor, the high-altitude motor would lose 95 percent of its performance if erated at sea level. This assumed complete overexpansion in the high-altitude rocket gine. The PTV vehicle would have to use a high-altitude version of the XLR16-AJ-2 gine in order to obtain high performance. However, this engine usually would be started sea level and would have to furnish sufficient thrust to lift the vehicle. If .e

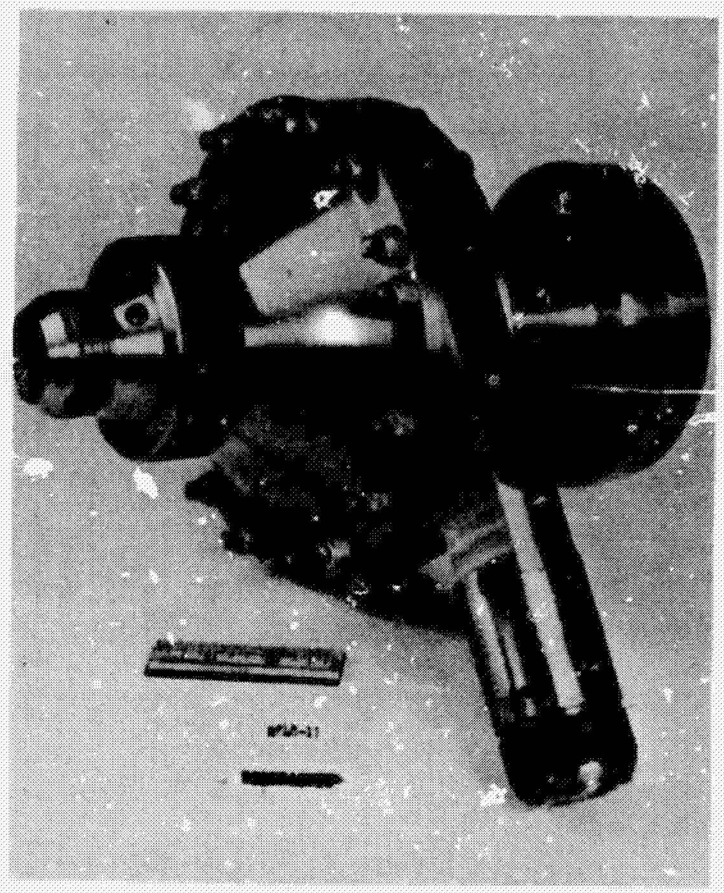


Fig. 17
Aerojet Liquid-Hydrogen Pump

losses were as great as calculated, it would be necessary to develop some means of controlling the excessive overexpansion which causes the losses in performance shown in Figure 18.[41]

A flared tubular thrust chamber with a 33:1 area ratio nozzle was constructed for tests to determine the loss in thrust chamber performance caused by overexpansion in the chamber pressure region from 150 to 350 psia. The thrust chamber appears in Figure 19. Twenty tests were made with the chamber. In addition, sixteen runs were made on an identical chamber except that a sea level (16:1) area ratio nozzle was employed. Performance of the overexpanding-nozzle test thrust chamber and the normal-area-ratio thrust chamber at mixture ratios of 4:1 and 3:1 under the same operating conditions were essentially the same, thus, the chamber parameter c* was not affected. Therefore, the nozzle parameter C_F was affected and the overexpansion losses occured only in the nozzle.

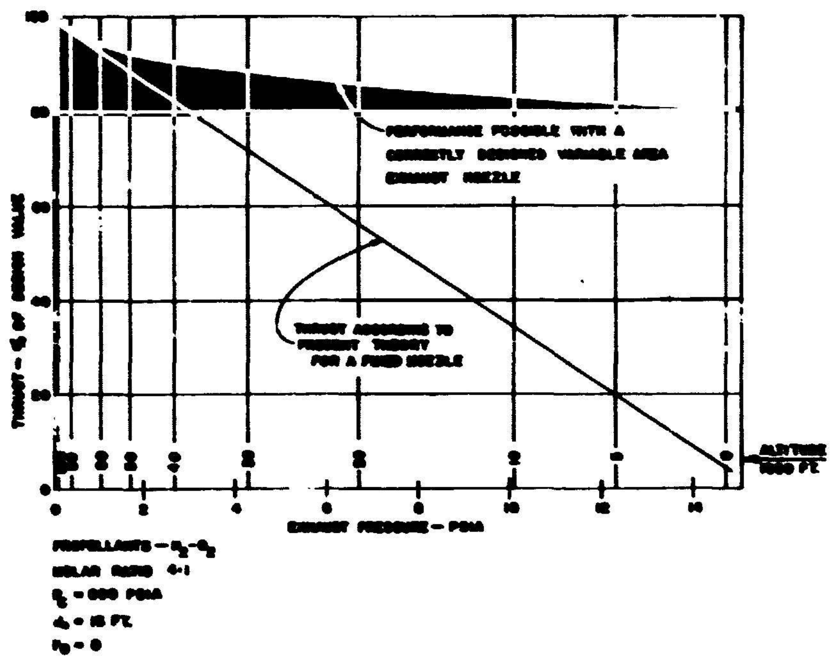


Fig. 18
Performance at Low Altitude of a Nozzle Designed for High Altitude

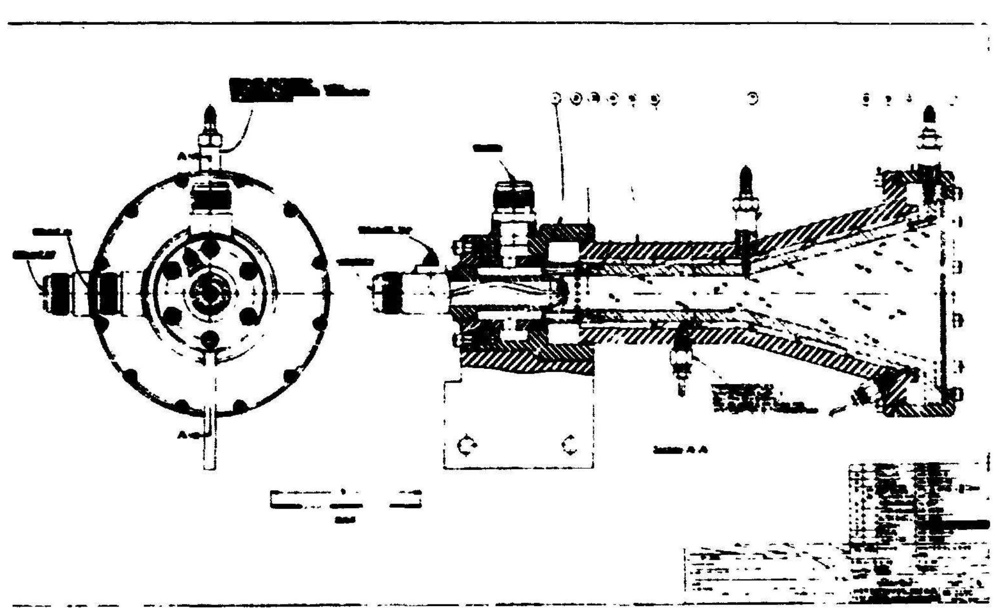


Fig. 19
Dr-wing of Overexpanding-Nozzle Thrus amb-r

It appeared from these tests that the losses in the overexpanding-nozzle thrust chamber were about 13 percent greater than in a normal expansion-nozzle thrust chamber.

It was apparent that the problem of minimizing losses would still remain to be solved for single-stage, long-range rocket missiles. The conceptual solutions were easily formulated, and involved the injection of gases into the side wall of the nozzle to cause separation at the correct area ratio. However, at this point in the contract, it was considered undesirable to continue to divert manpower for this investigation.

IX. LIQUID HYDROGEN-LIQUID OXYGEN THRUST CHAMBER DEVELOPMENT

The object of this portion of Contract NOa(s)-8496 was to develop a 3000-lb-thrust regeneratively cooled rocket thrust chamber which would satisfy the target specifications for the rocket engine described in Section V. An over-all engine I_{sp} (specific impulse) of 303 seconds was required, with estimates that the turbopump would require six percent of the total propellant flow. Consequently, the thrust chamber would have to produce an I_{sp} of 322 seconds. A further requirement, based upon the tankage of the missile, called for the $H_2:O_2$ molar ratio for the thrust chamber to be only 4:1.[42]

The design of a regeneratively cooled 3000-lb liquid hydrogen-liquid oxygen rocket engine demanded that certain new design data be established because all earlier work had been conducted with gaseous propellants. This data was to be obtained from a 400-lb thrust chamber development program planned in two phases. Phase One would determine the effect of motor geometry and injection configuration upon heat-transfer and performance. These tests were to be made without the benefit of film cooling by liquid hydrogen. Having determined a suitable thrust chamber geometry and best injector configuration, Phase Two was to be initiated to determine the effects of film cooling by liquid hydrogen upon performance and heat transfer. The film coolant was to be injected down the chamber walls from the injector face and/or in the chamber just forward of the nozzle. Adequate performance, as noted above, was an I_{sp} of at least 322 seconds. To satisfy performance and weight requirements and to permit regenerative cooling with liquid oxygen, it was decided that motor testing should be at 500 psia chamber pressure.

The best injector and chamber configuration developed on the 400-lb-thrust program would be used as a basis for the design of the 3000-lb combustion chamber and injector. Development of the 3000-lb-thrust chamber was to continue until target specifications had been achieved. But time and funds did not permit the completion of the systematic program outlined above. Instead, an abbreviated program, described in the following paragraphs, transpired.

The first test of the 400-pound-thrust unit was conducted on January 21 1949. By May 5, 1949, sixteen tests had been completed at 400-lb thrust, using liquid-phase propellants and water-convection-cooled, liquid hydrogen/liquid oxygen combustion chambers with suitable nozzles to produce 400 pounds nominal thrust at 300, 400 and 500 psia chamber pressures. The general construction of these combustion chambers and nozzles are shown in Figure 20. The inner chambers and nozzles were made of copper to permit large heat-flux densities to be handled with safety. The above chambers were of different lengths and two different diameters; hence L*, length, and diameter could all be used as parameters. A typical thrust chamber mounted on the test stand is shown in Figure 21. Five injector configurations were tested with a water-cooled plenum chamber of 64 L*. The best performance was obtained with the multi-tube concentric-orifice injector (coaxial) designed by G. H. Osborn (Figure 22).

In this design, the liquid oxygen was injected through the annular orifices and the liquid hydrogen through the central tubes. Each pair of orifices supplied propellant corresponding to about 8 pounds of thrust. Film cooling was again supplied by 24 0.015-inch-diameter orifices, which sprayed liquid hydrogen down the wall of the combustion chamber. The first run with this injector was very successful. The test lasted for 40 seconds without damage to the thrust chamber. This injector, at 504 lb/in^2 chamber pressure, gave a specific impulse of 366 seconds and an average heat-flux density of 6.45 Btu/in^2 in the chamber, and 11.3 Btu/in^2 in the nozzle. The appearance of the jet during this run is shown in Figure 23.

The 3000-lb combustion chamber (Figure 24) was designed and assembled using some parts from another program. The inner chamber was machined from a cast-copper billet. Electrolytic copper was preferred, but could not be obtained in the required size. The cast copper that was used for the inner chamber has only approximately one-third the strength and one-half the thermal conductivity of electrolytic copper. The inner and outer chamber before assembly is shown in Figure 25. This chamber assembly of 49 L* was designed with a nozzle exit to throat area ratio of only 4:1 because of size limitations of the existing parts. The helical coolant passage was designed to accomodate a water flow of ten pounds per second with a 110 psi pressure drop.

Successful tests with the multi-tube concentric-orifice injector at 400-lb thrust resulted in the 3000-lb-thrust concentric-orifice injector design (Figure 26). Four hundred and eighty-nine concentric orifices were used in addition to 60 hydrogen-film-coolant orifices. Each pair of orifices supplied propellant corresponding to about six pounds of thrust. The trouble experienced with the burning on the face of some of these injectors led to the conclusion that the fuel should be injected through annular orifices. Accordingly, the 3,000-lb injector was designed with the liquid hydrogen flowing in the annulus and the liquid oxygen flowing through the central tubes. Figure 27 shows this injector after fabrication. Figure 28 shows a water-flow test.

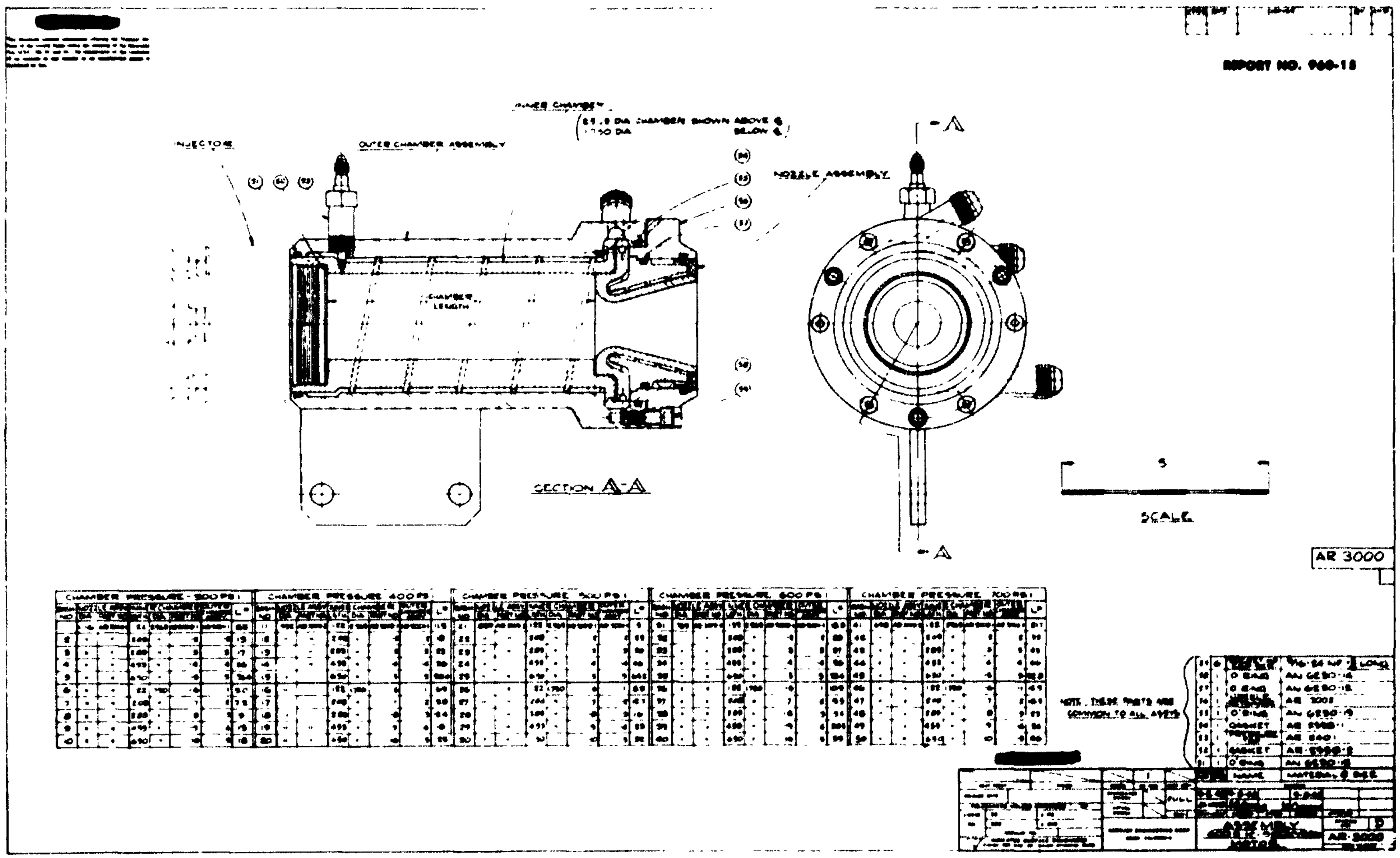


Fig. 20
Drawing of 400-Pound-Thrust Liquid Oxygen/Liquid Hydrogen Thrust Chamber

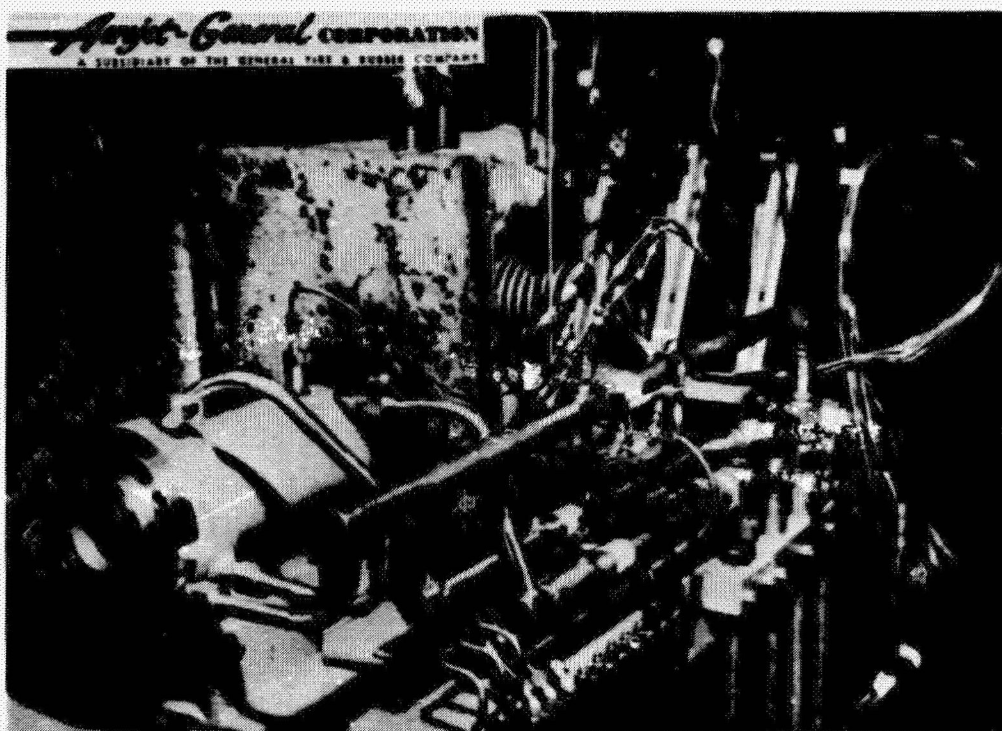


Fig. 21
View of Test Bay Showing 400-Pound-Thrust Chamber Installation

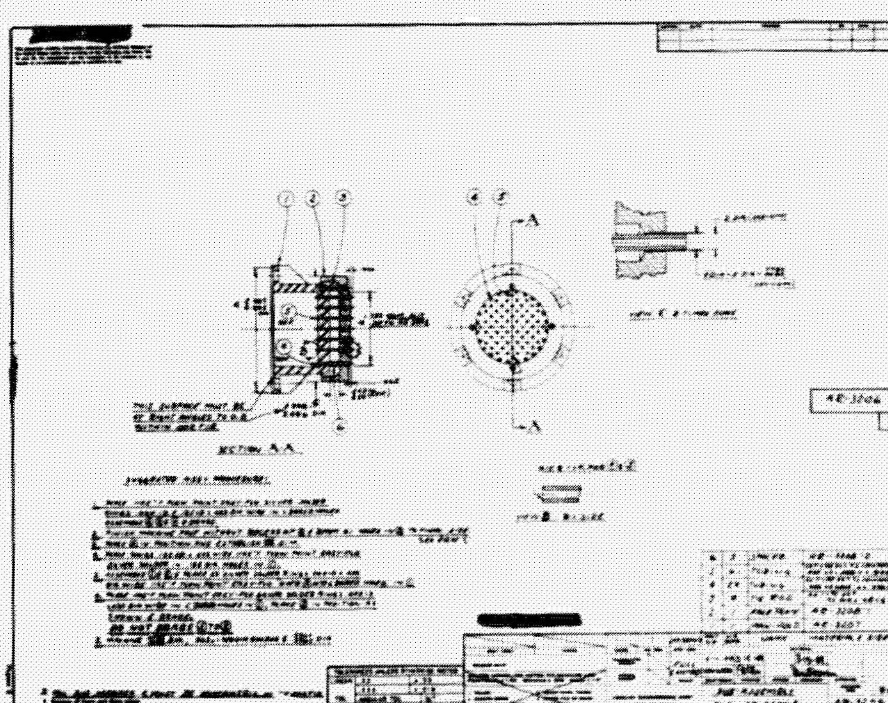


Fig. 22
Drawing of Coaxial Injector Design for 400-Pound-Thrust Chamber

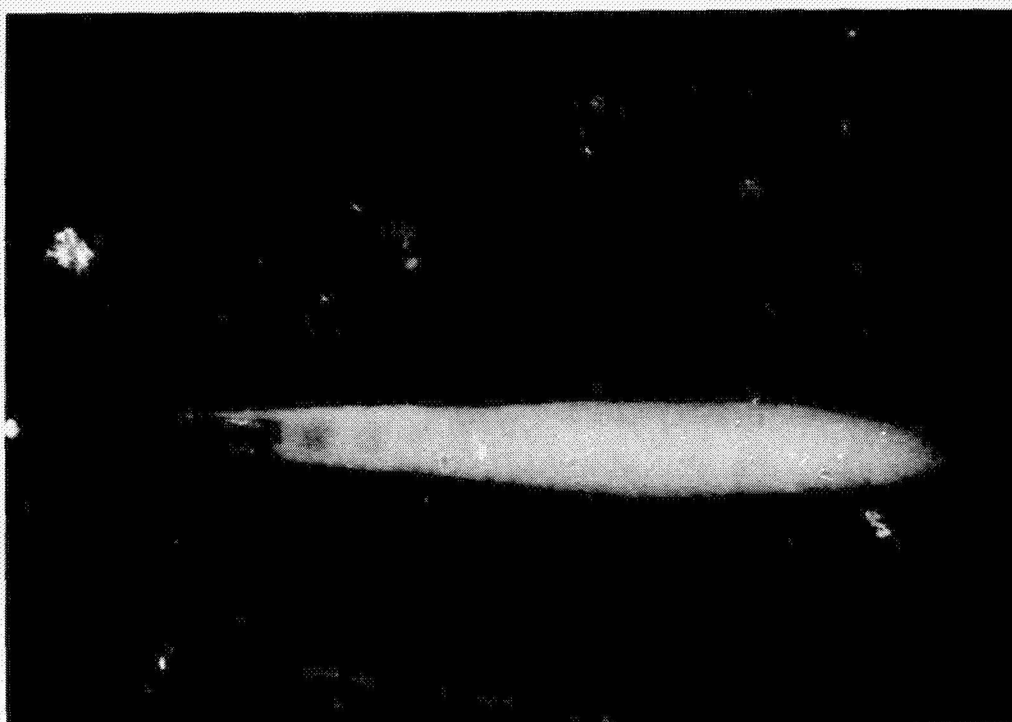


Fig. 23
Operation of 400-Pound-Thrust Liquid-Oxygen/Liquid-Hydrogen
Thrust Chamber Assembly with Coaxial Injector

Three tests were made using the water-cooled plenum chamber of 47 L* shown in Figure 26. Specific impulses of 345, 359, and 350 seconds were obtained. These values represent 93.4, 96.7, and 99.4 percent of theoretical impulse for the operating conditions for each test. The thrust chamber mounted on the vertical test stand firing into a flame deflector and producing 2655-lb thrust is shown in Figure 29. On March 2, 1949, a communication from the Bureau of Aeronautics directed a change of the fuel component from liquid hydrogen to anhydrous hydrazine; the oxidizer, liquid oxygen, was to remain the same. However, testing on the liquid hydrogen program was authorized to continue until the test results reported above were achieved.[43] The end of the program precluded further tests to adjust the rate of film coolant addition to balance a tolerable unit-heat-flux density against adequate performance.

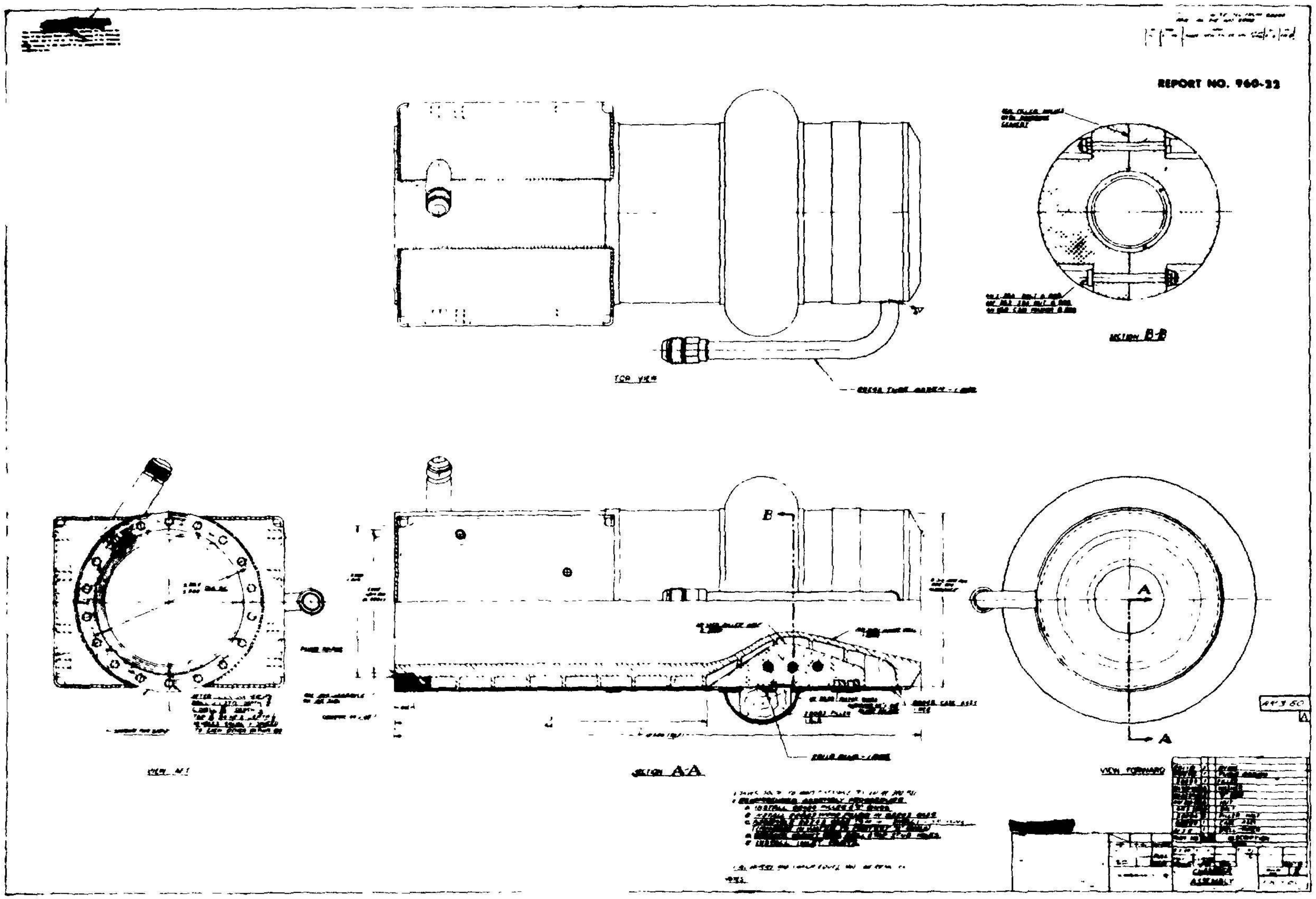


Fig. 24
Drawing of 3000-Pound-Thrust Liquid-Oxygen/Liquid-Hydrogen Thrust Chamber

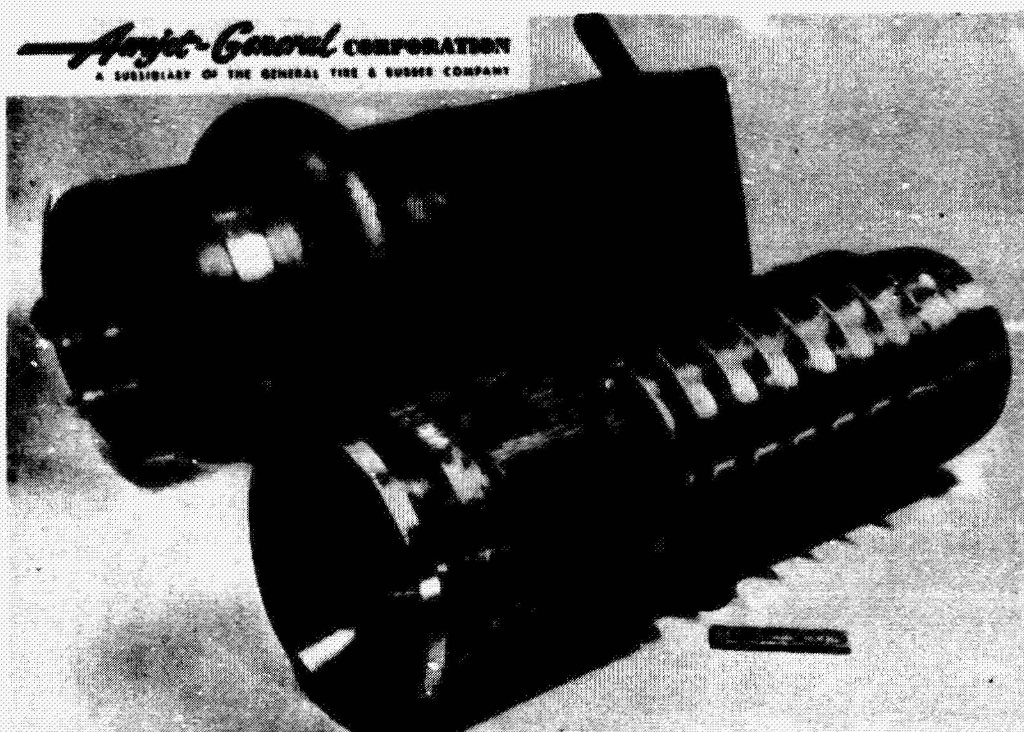


Fig. 25
3000-Pound-Thrust Chamber With Cast-Copper Inner Chamber

CONCLUSIONS

By August 1949, the liquid hydrogen-liquid oxygen program at Aerojet had demonstrated the feasibility of virtually all the components in present-day, high-energy, liquid-rocket engines. Transpiration and film-cooled thrust chambers had successfully operated. The first liquid-hydrogen tests of the coaxial injector had been conducted. The first pump to successfully produce high pressures in pumping liquid hydrogen had been tested. A 1000-lb-thrust gaseous propellant and a 3000-lb-thrust liquid-propellant thrust chamber had operated satisfactorily. And the first tests had been conducted to evaluate the effects of jet overexpansion and separation on performance of rocket thrust chambers with hydrogen-oxygen propellants. The production and handling of liquid hydrogen had been demonstrated to be practical and less hazardous than originally believed by industry spokesmen when the program started. These pioneering investigations established a valuable engineering background. However, the application of this technology had to wait for almost a decade before finding use in high-energy upper stages for the space program in the 1960s.

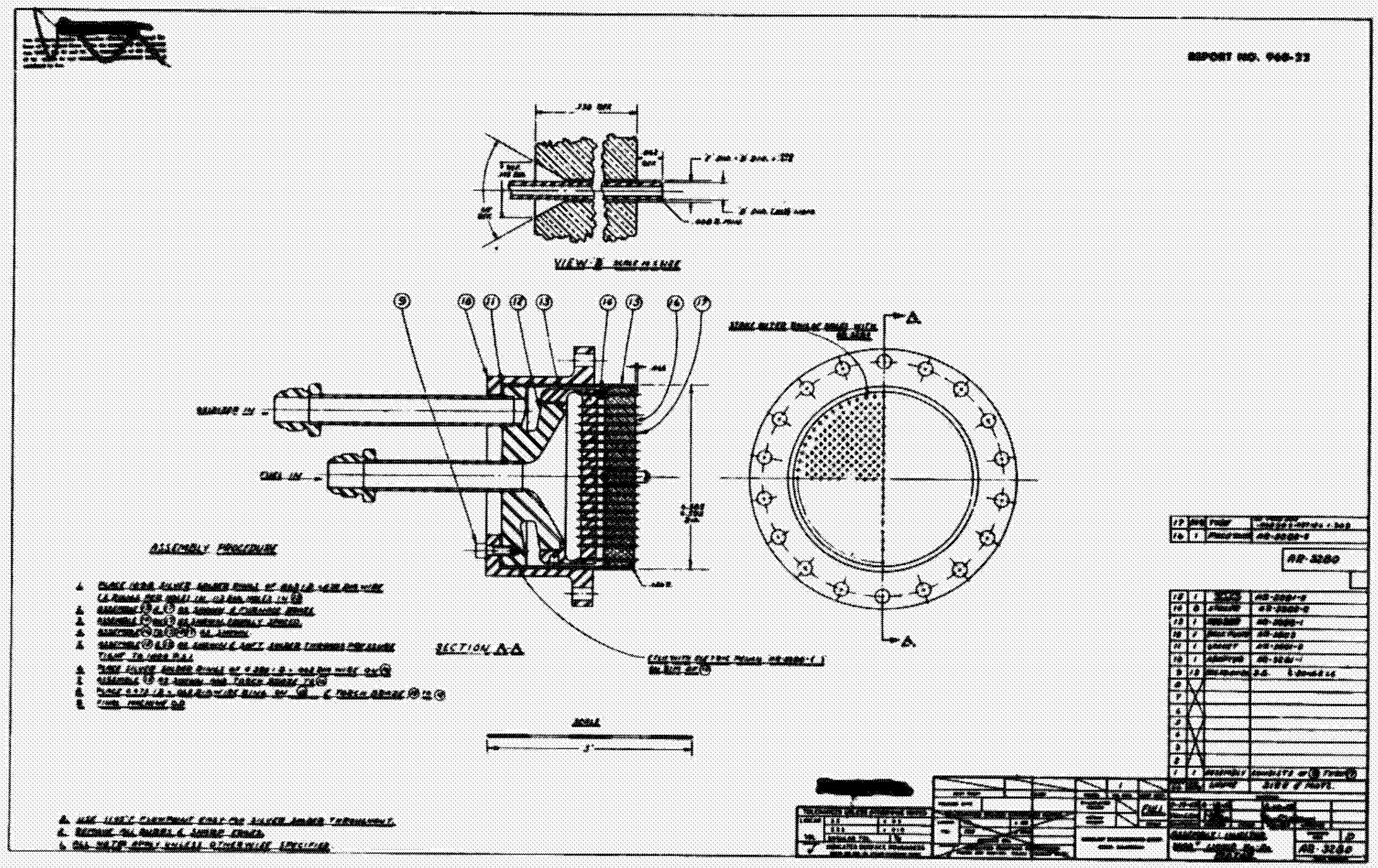


Fig. 26
Drawing of 3000-Pound-Thrust Liquid-Oxygen/Liquid-Hydrogen Coaxial Injector

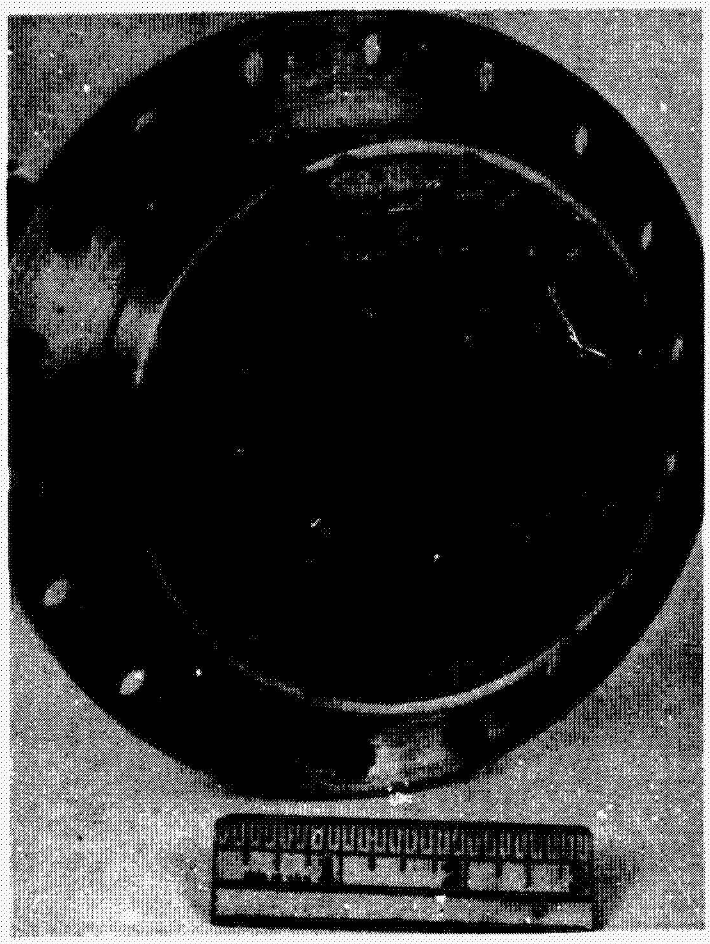


Fig. 27
Assembled 3000-Pound Thrust
Coaxial Injector

Fig. 28
Water Flow Test of 3000-Pound-
Thrust Coaxial Injector

Fig. 29
3000-Pound-Thrust Chamber
Assembly During Second Test

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