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THE M-1 ROCKET ENGINE PROJECT

by Walter F. Dankhoff;
Lewis Research Center,
Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION · WASHINGTON, D. C. · OCTOBER 1963

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NASA, Lewis Research Center
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E-2335

THE M-1 ROCKET ENGINE PROJECT

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INTRODUCTION

The M-1 rocket engine is a 1.5-million-pound thrust system that employs a liquid-oxygen - liquid-hydrogen propellant combination. As such, it represents the most ambitious high-energy rocket engine ever committed to development in the United States. It is being developed to provide an optimum, high-performance engine of an appropriate size for post-Saturn launch vehicles, on a time scale consistent with requirements of having a qualified man-rated system available in the 1970's.

Originally, the M-1 was conceived as a 1.2-million-pound thrust engine, which when clustered, would provide the second-stage propulsion for a launch vehicle known as NOVA. The original NOVA concept was a state-of-the-art launch vehicle capable of the direct lunar landing. NOVA had an earth orbit payload capability of about 400,000 pounds. With this description, the M-1 engine development program was initiated and formally assigned to the Aerojet-General Corporation, Sacramento, California early in 1962. At that time the project was managed by the NASA George C. Marshall Space Flight Center.

In October 1962 responsibility for the M-1 was transferred to the Lewis Research Center, Cleveland, Ohio. During the interim, the payload requirements of the NOVA vehicle had increased considerably. Apollo program decisions to utilize lunar orbit rendezvous obviated the lunar payload requirement and delayed the funding availability of the NOVA vehicle. The primary application now for the NOVA vehicle is the manned planetary expedition. This role of the NOVA vehicle has increased its payload, launch rate, operational lifetime, and cost effectiveness requirements. An earth orbital capability of approximately 1,000,000 pounds is now the design point for NOVA.

In view of the changes in timing and large increase in the NOVA (post-Saturn) payload requirements and size, a "hard look" at the definition and design of the M-1 engine was taken after the transfer of the project to the Lewis Research Center. This study resulted in an increase in the nominal

*Manager, M-1 Engine Project.

vacuum thrust to 1.5 million pounds and incorporation of design features that would allow an additional uprating, at a later date, to at least 1.8 million pounds. The timing of the vehicle application made it possible to improve the basic engine design through the employment of the most current hydrogen-oxygen engine technology. The increased time also made it possible to lay out a more thorough and careful development program that included timely subcomponent and scale component tests.

Events subsequent to October 1962 have shown that the present configuration of the M-1 is ideal for the post-Saturn vehicle. Vehicle design and operational studies have been continued both in house and by contractors. The results of these studies made on the basis of development effort, cost, operational use, and timing show the M-1 to be ideal for the second stage of post-Saturn vehicles and well suited for the first-stage application. The first-stage application would be in either a conventional cluster of engines or cluster of engine modules in a single altitude compensating type nozzle.

ENGINE SYSTEM

The basic engine system, shown in figure 1 and described in table I, consists of separate hydrogen and oxygen turbopump assemblies mounted on a single thrust chamber that is hydrogen-cooled to a nozzle area ratio of 14:1. Beyond this area ratio, the nozzle extension is cooled to 40:1 area ratio with turbine exhaust gas. A gas generator, utilizing main engine propellants, energizes the hydrogen and oxygen turbopumps in series. The control system includes thrust chamber and gas generator valves, a helium start system, and a propellant utilization valve. Thrust vector control can be provided by the engine through a gimbaling system.

TABLE I. - ENGINE PERFORMANCE AND DESCRIPTIVE FEATURES

Nominal thrust at altitude of 200,000 feet	1,500,000±45,000 lb
Nominal specific impulse	428 sec
Thrust chamber pressure	1,000 psia
Nozzle area ratio	40:1
Oxidizer flow rate	2921 lb/sec
Fuel flow rate	584 lb/sec
O/F weight ratio	5.0±0.125
Rated duration	500 sec
Gimbal angle	$7\frac{1}{2}^{\circ}$ square pattern
Gimbal rate	15°/sec
Length-gimbal block to nozzle exit	321 in.
Diameter at nozzle exit	208 in.
Weight, dry	20,000 lb

Although development of a 1.5-million-pound thrust H_2-O_2 engine is the immediate program goal, a design philosophy is maintained to permit eventual engine growth to the 1.8-million-pound level or higher. In this respect, special emphasis is being placed on the turbopumps. They will be capable of discharge pressure growth commensurate with a chamber pressure of 1200 pounds per square inch absolute in a 1.8-million-pound thrust engine. De-rated operation at 1.2-million-pound thrust also will be possible. Conversion of the M-1 engine for optimum performance at sea level could be accomplished with a modified nozzle extension. Flexibility of thrust rating and the adaptability to clustered configurations will allow use of the engine in propulsion systems ranging from 1.2 to approximately 30 million pounds in first-stage as well as in upper-stage applications.

Figure 2 describes the M-1 engine fluid-flow system. An electrical signal begins the starting sequence by actuating the helium start valve. In addition, it energizes the thrust chamber and gas-generator spark ignitors. High pressure helium spins the fuel and oxidizer turbines. Thrust-chamber valves are opened by a rising pump discharge pressure and admit propellants to the thrust chamber where they are ignited. Gas-generator valves also are opened by a rising pump discharge pressure, thus supplying the gas generator with flow from the turbopumps. Combustion of propellants in the gas generator sustains the fuel and oxygen turbines during steady-state operation. Finally, gas-generator pressure closes check valves that terminate helium flow.

Engine shutdown is initiated by venting the gas-generator propellant valve actuators; the gas-generator valve-spring forces then close the valves and stop propellant flow to the gas generator. Consequently, the turbopumps are deprived of drive gases and their output decays, permitting spring forces to close the thrust-chamber valves.

ENGINE DEVELOPMENT

The M-1 rocket engine is a machine of impressive proportions. The overall length is 321 inches or approximately 27 feet. A $3\frac{1}{2}$-foot thrust-chamber diameter, coupled with a 17-foot-nozzle-exit diameter, are further evidence of the size of the engine. At design-point conditions, the M-1 fuel turbopump alone develops 75,000 shaft horsepower as compared with the 60,000 horsepower combination oxidizer and fuel turbopump on the F-1 engine. The M-1 gas generator consumes three times as much propellant per second as the entire RL-10 engine. This large size is a prime source of design, fabrication and development problems. Cost, lead time, and handling difficulty increase directly with size. As an example, procurement lead time of M-1 combustion-chamber injectors is scheduled at 48 to 63 weeks. The high unit cost of large components limits the number of alternate designs that can be pursued and intensifies the importance of hardware conservation. It is imperative that the first component design choices be good ones. Budget and schedule restrictions severely limit design redirection. To achieve good design within these restrictions, emphasis is placed on design verification by means of scale testing and subcomponent testing early in the development program.

Thrust chamber. - The overall M-1 thrust-chamber design concept is conventional; however, several unique details are incorporated. Novel fabrication techniques are being developed and special consideration is being given to possible combustion instability.

Experience has shown that combustion instability is related to size. The history of oxygen-kerosene propulsion systems justifies this conclusion. A similar trend appears to be developing in LO_2-LH_2 equipment. The development of the 15,000-pound-thrust RL-10 rocket engine has been relatively free of combustion instability. Some instability has been experienced in the 200,000-pound-thrust J-2 engine. It is probable that more serious combustion problems will arise in the much larger M-1 unless specific measures are introduced to forestall them. The injector is one area that influences this phenomenon. A coaxial-type injector has performed very well throughout the RL-10 program. Significantly, one has been incorporated in the J-2 engine to alleviate combustion instabilities that were experienced with a concentric ring design. Furthermore, combustion research at the Lewis Research Center supports the use of coaxial-injector elements in LO_2-LH_2 systems. In this design, each oxygen orifice is circumscribed by a coaxial hydrogen orifice. Consequently, excellent fuel and oxidizer mixing can be achieved as high-velocity hydrogen breaks up the oxygen to small droplet size through a momentum exchange. The coaxial configuration allows convenient LH_2 manifolding and cooling of the entire injector face.

Emphasis is being placed, therefore, on a coaxial injector for the M-1. Figure 3 illustrates a typical element and indicates the position of swirlers that will help mix the propellants. The first design calls for approximately 1200 elements to be located on the 42-inch-injector face. Testing may show that as many as 3000 elements are needed to meet performance specifications. (Injectors are not yet amenable to precise design procedures.) Each element is welded onto the 347 stainless-steel injector body. Transpiration cooling of the porous mesh faceplate is achieved with LH_2 at a 2- to 4-percent flow rate. Another deterrent to combustion oscillation is the incorporation of film-cooled baffles.

The combustion chamber wall, to the 14:1 area ratio, is fabricated by furnace brazing chemically milled tubes of 347 stainless steel (fig. 4). These tubes are designed with variable contour, variable wetted perimeter, and tapered wall thickness to cope with variations in the heat flux and internal pressure within the chamber. Two hundred tubes extend from the 8:1 location of the fuel inlet manifold to the 14:1 area ratio, while 300 tubes comprise the return pass to the injector. A roll-formed Inconel 718 bolt-on jacket is used to help contain the hoop stress in the chamber and to stiffen the chamber against gimbaling loads. (Inconel 718 is employed extensively throughout the M-1 engine for its excellent physical properties at cryogenic temperatures to 1300° F. It maintains high tensile and notch strength to weight ratio, impact resistance, and ductility over this range.) The fuel torus (fig. 5), which is mounted downstream of the combustion chamber throat, is built up from welded Inconel 718 forgings.

Main chamber ignition will be accomplished with small spark-initiated secondary combustors similar to those employed on the J-2 and RL-10. The spark ignition system consists of an exciter, a cable, and an igniter.

Regenerative cooling with fuel is provided only to the 14:1 area ratio. Complete fuel cooling of the nozzle extension (fig. 6) would require a 40-foot cooling path that would be too expensive in terms of pressure drop. Instead, turbine exhaust gas is passed through 347 stainless-steel tubes from the 14:1 area ratio. The M-1 is the first large-scale engine to use this form of convective cooling. Turbine exhaust gas enters the nozzle-extension inlet manifold at nominally 100 pounds per square inch absolute and 700° F. After increasing in temperature to 1100° F, it leaves through small sonic nozzles, at the main nozzle exit plane, producing 28,000 pounds of thrust.

The size of the nozzle extension presents an imposing fabrication problem. In order to develop fabrication techniques on a piece of hardware smaller than a full nozzle extension, a decision was made to divide the nozzle into eight separate segments. These segments will be furnace brazed as individual segments and later assembled into a complete nozzle extension. No furnace is available currently to braze the entire extension in the proper atmosphere. However, there are a few furnaces capable of handling the individual segments. Strong consideration is being given to the use of blanket brazing for the individual segments. Progress in the thrust-chamber area has followed a carefully prepared development plan. Designs for chamber, injector, and nozzle configurations were completed in late 1962 and early 1963. The first injector has been fabricated (fig. 7). Uncooled chamber tests will be conducted in early 1964. They will be helpful for injector evaluation and will offer an insight into possible combustion oscillation problems. The first cooled combustion chamber is scheduled for testing in June 1964.

<u>Turbopump</u>. - Size is also the major source of M-1 turbomachinery problems. If the oxidizer-pump thrust-compensating device were 95 percent effective, approximately 13,000 pounds of axial thrust would still be imposed on the thrust bearings. Often neglected, gyroscopic forces alone account for radial bearing loads as high as 1490 pounds in the fuel pump.

The low density of hydrogen is misleading in terms of pumping requirements. Rather than being an asset, low density necessitates a high head rise (ft) to achieve a given pressure level (psi). The M-1 hydrogen turbopump (fig. 8) consists of a ten-stage axial flow pump driven by a two-stage turbine. In general, the axial flow configuration lends itself to better design procedures than the centrifugal variety. Hence, the achievement of calculated efficiency is more probable. Furthermore, its inherent facility for growth made it the choice for the fuel pump. The pump includes an inducer stage, a transition stage, and eight main work stages. The inducer rotor has radial blade elements of constant blade thickness and a cut-back trailing edge that provides a uniform radial head distribution at the rotor discharge. The inducer stator incorporates 11 blades of the British C-4 series airfoil design. (The C-4 series demonstrates superior stress properties. In addition, cascade data for calculating incidence and deviation angles is available.) The inducer assembly is designed for a suction specific speed of 38,000, a net positive suction pressure of 10 pounds per square inch and a head rise of 7000 feet. The transition stage is characterized by light hydrodynamic loading and a relatively low head rise (2335 ft) and also uses the C-4 airfoil design. The stator serves as a return passage for the balance piston and bearing coolant flow of 40 pounds per second. Total flow is 624 pounds per second through the main stages so that 584 pounds per second can be supplied to the thrust chamber and gas generator after balance piston and bearing cooling demands are met. The head rise for each main stage is 6340 feet, and the eight main stage elements of C-4 shape are identical, thereby offering minimum cost. The loading on each work stage is purposefully low to enhance the operating range. A 25-percent increase in loading could be obtained for engine growth. The predicted fuel pump performance is shown in figure 9. Inconel 718 will be used in the rotor and stator blading as well as in the rotor spool. A program has been established to evaluate methods for welding in the pump rotor. Tungsten inert gas (TIG) and electron-beam welding are the prime candidates. The turbopump housing material is established presently as cast 304 stainless steel to minimize initial fabrication problems; however, lighter weight aluminum and fabricated Inconel 718 alternates are being studied.

A two-stage, velocity compounded turbine has been designed to drive the fuel pump. This design is a compromise between the efficiency gains of multi-staging and the attendant problems of increased weight and over-hung bearing loading. The first-stage nozzle is supersonic convergent-divergent. The first-stage rotor is a zero-reaction type, while the reversing vanes and second rotor have slight degrees of reaction. The shrouded blades and the turbine disks are fabricated from Inconel 718.

Gas-generator products enter the turbine at 900 pounds per square inch absolute and 1000° F and leave at 234 pounds per square inch absolute and 710° F through Inconel 718 manifolds.

A self-compensating balance piston, actuated with hydrogen and tapped from the pump discharge, will adjust to oppose axial pump loads. Its response has been studied on an analog computer and was found to be stable over the entire range of anticipated conditions. The balance piston was designed to allow some unidirectional bearing loading during steady-state operation, thus avoiding high-speed rolling-element skidding. The bearings are cooled and lubricated with liquid hydrogen. Provision has been made for separate axial and radial load distribution among ball and roller bearings, respectively. The pump end bearing assembly consists of one roller bearing and three thrust-sharing ball bearings. A single roller bearing is located at the turbine end. All of the bearings incorporate 440C stainless-steel races and rolling elements and glass-filled Teflon cages.

Design of the initial fuel turbopump is nearing completion, and long lead items (impeller forgings, turbine disk forgings, pump housings, etc.) have been ordered and/or received. Full-scale fuel-pump testing will be initiated in September 1964.

The oxidizer turbopump (fig. 10) consists of a single-stage backward-swept centrifugal pump driven by a two-stage axial-flow turbine. The LO_2 pump is large in relation to the flow rate it delivers; however, it is designed to operate at a NPSH of 30 feet compared to 65 feet for the F-1 LO_2 pump. Thus, the weight penalty of the pump can be more than compensated by savings in reduced tank-pressure requirements. The backward-swept centrifugal configuration provides the necessary discharge pressure at a conservative tip speed, together with a stable operating curve (fig. 11). Advantage was taken, therefore, of its relatively simple mechanical construction. The pump will generate an axial thrust in excess of 250,000 pounds during full-rated operation. Thirty-three impeller back vanes are provided to oppose this force. An oxygen flow rate of 3000 pounds per second, including recirculated bearing lubricant flow, is delivered at a suction specific speed of 36,700. The noncavitating head rise is 3400 feet.

Oxidizer pump and turbine material selections resemble those made for the fuel turbopump. The pump impeller is forged from 7079-T652 aluminum. Cast 304 stainless-steel has been specified for the Model I housing but will be replaced by a fabricated high-strength steel or cast-aluminum part. Inconel 718 will be used for the turbine housing, rotor and stator blades, disks, and drive shaft.

Two 11.5-inch crossover ducts supply the turbine inlet manifold with the gas-generator products bypassed around the fuel turbine for pressure utilization control, and the fuel turbine exhaust. The mixture arrives at 760° F and 200 pounds per square inch absolute. Oxygen turbine exhaust leaves through a spherical manifold and is carried by two ducts to the combustion-chamber nozzle extension.

A set of two tandem ball bearings carries the turbopump axial thrust while single roller bearings at the pump and turbine ends of the drive shaft carry the radial load. These bearings are also fabricated from 440C stainless steel and include glass-filled Teflon cages.

The first oxidizer turbopump will be assembled in March 1964. Full-scale testing will be initiated in July 1964. An intensive effort is being exerted in the field of turbopump bearings. Special testers have been built to evaluate individual bearings, bearing sets, and complete power transmission assemblies. These units make it possible to duplicate the rigorous bearing-operating conditions for both the fuel and oxidizer turbopumps. Axial thrust loads in excess of 30,000 pounds and radial loads in excess of 10,000 pounds are anticipated during start and stop transients in both turbopumps. In addition, speed excursions of the turbomachinery will likely result in DN values of 1.6 and 0.4 million for the hydrogen and oxygen bearings, respectively. High acceleration rates in the fuel turbopump further complicate the problem. Internal geometry, race material, rolling-element material, cage material, and application of lubricant represent some variables that are being analyzed. To date, individual, 110-mm ball bearings, lubricated with hydrogen, have been operated successfully under thrust loads of 15,000 pounds and at speeds up to 15,000 rpm (1.65 million DN). Oxygen-lubricated 110-millimeter bearings were tested successfully with thrust loads up to 35,000 pounds while operating at 4000 rpm (0.44 million DN). A parallel development program is being conducted on dynamic seals for both turbopumps. Configurations and materials are being tested over a wide range of conditions designed to simulate turbopump operation.

The energy expended in the component development program is in keeping with a desire to eliminate unverified machine elements from large system tests and to conserve hardware. Extensive instrumentation is another manifestation of this purpose.

A scale model LH_2 and LO_2 pump program will be conducted at the water tunnel test facility of Aerojet-General Corporation. A 0.645-scale-model cold-air turbine tester will be operated at the Lewis Research Center to obtain fuel turbine aerodynamic performance data. A 0.450-scale test will be conducted with the oxidant turbine. Results from the scale pump and turbine programs will be applied to later models of the full-size turbopumps.

Gas Generator. - The M-1 gas generator (fig. 12) burns main chamber propellants at the rate of 110 pounds per second at an oxidant- to fuel-flow-rate ratio of 0.8. The hydrogen-rich steam that results leaves the gas generator at 1100 pounds per square inch and 1000° F. The initial gas generator version consists of an uncooled chamber and concentric ring, multiorifice injector. Tests of this design (fig. 13) have indicated erosion of the ignitor, injector face, and uncooled baffles. Consequently, an alternate approach that uses coaxial injection and a porous injector face is being expedited.

Thrust-Vector Control. - An ability to provide a thrust-vector angle of $7\frac{1}{2}^{\circ}$ is a vehicle requirement that the M-1 engine must meet. Several approaches to thrust-vector control have been studied to ascertain the best choice for the M-1. The possible use of jet vanes, jet tabs, jetavators, and other mechanical obstructions within the nozzle exhaust stream was dismissed early because of the large drag losses and cooling problems they introduce. Gimbaling and secondary fluid injection were the two most promising alternatives. Gimbaling, of course, has been used widely in rocket engines and enjoys a more advanced state of development. On the other hand, secondary fluid injection eliminates the need for flexible suction and interconnecting lines, the gimbal block and bearing, and the gimbal actuators. A detailed design study of the following types of secondary injection systems was made:

(1) Turbine Exhaust

(2) Four Auxiliary Gas Generators

(3) Main Gas Generator Tapoff

(4) Thrust Chamber Tapoff

(5) Turbine Exhaust Plus Gas-Generator Booster

(6) Heated or Cold Hydrogen

The study showed that the Four-Auxiliary-Gas-Generator system was the best of the fluid-injection systems. Weight and vehicle trade-off studies indicated little or no payload gains with fluid injection; hence, system performance is not enhanced with fluid injection. Sufficient engineering and development could make the fluid-injection system reliable, but its performance characteristics presently remain uncertain. The gas-injection system would also extend the program length since additional altitude-simulated engine tests would be required. Since no significant gains were indicated and a fluid-injection thrust vector control approach introduced additional risks, an engine gimbal system was adopted.

FACILITIES

A number of new facilities as well as modifications to existing facilities are required to supplement those currently available within the test complex of the contractor in Sacramento, California. These new facilities in many instances constitute a development program in themselves due to their complexity and physical size resulting from the propellant flow rates, total firing duration, and size of the M-1 engine. The installation of special test equipment for component development also broadens the scope of facility operations.

Laboratory Facilities. - The Cryogenics Laboratory has been expanded to permit testing of M-1 engine components. A total of 11 test bays are available to evaluate bearings, power transmissions, seals, valves, lines, and fittings. In addition, a spin pit has been constructed to spin-test turbine wheels, pump impellers, and other rotating parts. Structural, vibration, and altitude-simulation testing is also being conducted there. Liquid hydrogen, oxygen, and nitrogen can be directed into the test bays remotely from consoles within the laboratory. The Cryogenics Laboratory is provided with integrated data acquisition and tape recording equipment.

The Hydraulics Laboratory is being equipped to perform leak tests on individual thrust-chamber coolant tubes and tube bundles with high-pressure water. Waterproof and leak tests will verify the structural integrity of the thrust chamber, injector, and turbopump manifolds and housings. Gimbal assemblies and system valves also will be proof-tested in the Hydraulics Laboratory.

Other laboratories, including those devoted to vibration analysis, materials evaluation, chemical analysis and instrument calibration, are being equipped to support the special needs of the M-1 engine program. All modifications and additions to these facilities will be completed before the end of 1963.

Thrust Chamber and Gas-Generator Test Stand C-9. - A stand has been constructed to provide a combined capability for thrust chamber start and gas-generator testing (fig. 14). Short duration (2 sec) thrust-chamber operation at the 1.5-million-pound level and 50-second gas-generator operation can be accommodated. Common run vessels are used for both thrust-chamber and gas-generator assembly testing. The LH_2 vessel is vacuum jacketed and has a 6000-gallon capacity at 1800 pounds per square inch. The LO_2 vessel has a 2000-gallon capacity at 2160 pounds per square inch. Gas-generator testing will start in October 1963, on C-9, and activation for thrust-chamber testing will be completed during January 1964.

Turbopump Test Stands E-1 and E-3. - An existing facility is being modified to provide two turbopump test positions, as shown in figure 15. Both stands E-1 and E-3 have the capability of testing either the fuel or oxidizer turbopumps separately or together in series. Each of the stand positions is equipped with overhead tankage providing a transient turbomachinery test duration of 15 seconds. In addition, off-stand propellant run-tankage (fig. 16) (370,000 gal for LH_2 and 115,000 gal for LO_2) supplement the on-stand tankage to provide a test run duration of 320 seconds. Catch vessels of the same size will recover the turbopump discharges. The hydrogen vessels are double-walled vacuum-insulated stainless-steel Dewar types. The E-1 and E-3 existing superstructures have been modified and all propellant tankage is in place. Propellant line and controls fabrication is proceeding rapidly to permit the first oxidizer turbopump test in July 1964.

Thrust Chamber and Gas-Generator Stand H-8. - An existing test stand, designated H-8 (fig. 17), is being modified to extend the capabilities for thrust chamber testing to a 20-second duration. Gas-generator testing for durations up to 210 seconds will also be possible. A 25,000-gallon LH_2 run vessel and a 7750-gallon LO_2 run vessel will provide propellants at 1800 pounds per square inch from an off-stand location. A thrust-measuring system will be provided for 1.5-million-pound testing. The test stand and thrust-measuring system will be capable of accepting thrust-chamber assembly gimbaling up to 2°. This facility will be activated for gas-generator testing in April 1964 and thrust chamber testing in June 1964.

Engine-System Testing K Zone. - A new complex, designated "Test Zone K" will be located on presently undeveloped property adjacent to the existing liquid rocket test facilities. A schematic layout of this complex is presented in figure 18. It consists of four engine test stands, complete with all necessary support facilities including two control buildings. The construction of four test stands, K-1, K-2, K-3, and K-4, will be programed incrementally to match the requirements of the M-1 development schedule.

Test Stand K-1. - Test Stand K-1 is designed to be a short-duration vertical-firing single-position full-scale engine facility, capable of ignition tests at sea level and altitude conditions (fig. 19).

The ignition tests at a simulated altitude of 90,000 feet will be achieved through the use of a test enclosure evacuated to the simulated altitude with a mechanical pumping system. The test enclosure will be sized to accommodate the full-scale engine with a 25:1 area ratio nozzle extension and with an ullage volume of 80,000 cubic feet to accept the discharge gases without voiding the simulated altitude condition during the engine start condition. A system of "blow-doors" or diaphragms will be incorporated into the test enclosure for smooth transition from altitude start to sea-level test conditions. Environmental equipment will be provided in the test enclosure for cryogenic conditioning of the engine at simulated altitude as well as ambient temperature, rain, and humidity conditioning for a simulated prelaunch condition. The test stand will also be capable of demonstrating sea-level start and shutdown characteristics as well as $7\frac{1}{2}^{\circ}$ engine gimbaling with sections of the test enclosure removed.

On-stand tankage will provide propellants for 15-second test runs at the 1.5-million-pound thrust level. The facility consists of a structural steel frame superstructure mounted on a concrete base and has a 3.0-million-pound thrust potential. K-1 will be completed during the latter half of 1965. At present, 40 percent of the design phase has been completed and excavation for the substructure has been started.

Test Stand K-2. - Test Stand K-2 (fig. 20) will be a two-position complex capable of simulated altitude and sea-level engine testing at the optimum rated 1.5-million-pound thrust level in the horizontal positions. One of the two horizontal test positions will provide pressure and temperature conditioning for engine starts along with a 335-second duration sustained run capability at a simulated altitude of 70,000 feet with the complete engine, including the 40:1 expansion ratio nozzle extension. Altitude simulation will be achieved through the use of an environmental engine enclosure and a single supersonic diffuser system. The diffuser and environmental enclosure will be removable to provide a capability for 335-second engine run tests at sea-level conditions. The diffuser system will be capable of eventual usage in the second test position through the use of diffuser handling equipment.

A single-component thrust-measuring system (axial) rated at 1.5 million will be provided in each of the test positions. Provisions for a multicomponent thrust-measuring system will be built into each of the test positions.

The horizontal test positions will consist of adjacent thrust blocks of reinforced concrete designed to accept horizontal thrust of 3 million pounds.

Test Stands K-3 and K-4. - Test Stands K-3 and K-4 (fig. 21) are identical single-position facilities for full-scale long-duration testing in a vertical attitude. On-stand propellant run vessels will allow full-duration (500 sec) engine firings at maximum sea-level thrust. K-3 and K-4 will accommodate engine gimbal testing of $7\frac{1}{2}^{\circ}$ in a square pattern. Structural design has been directed toward a thrust capacity of 3.0 million pounds. These stands are still in the preliminary design stage and are timed for participation in the engine PFRT program.

PROJECT MANAGEMENT

The overall M-1 engine project coordination and funding originate within the Office of Manned Space Flight at NASA Headquarters. Detailed project management and technical responsibility have been delegated to the M-1 Project Office within the Lewis Research Center. Figure 22 describes the M-1 project management structure, emphasizing the lines of coordination among Headquarters, Lewis, and the Marshall Space Flight Centers. The M-1 Project Office is organized about technical disciplines that roughly parallel the contractor's engineering establishment (fig. 23). It incorporates an integrated technical, administrative and procurement staff that is able to perform most management functions independently of outside support. The main body of the management group is housed at Lewis where advantage can be taken of the propulsion specialists who are actively pursuing in-house research. A plant representative office has been established at the contractor's plant to expedite management resolutions, promote cooperation, and safeguard NASA interests. Authority to make on-the-spot decisions has been delegated to the Resident Manager, Mr. L. E. Baughman.

SUMMARY

The M-1 Rocket Engine Development Program represents an effort to provide this nation with a flexible high-energy propulsion system for post-Saturn vehicles. Its growth capability and clustering potential make it well suited to meet the requirements of various launch vehicle applications.

A thorough component development program is being incorporated to achieve maximum reliability at minimum cost. To date, the most outstanding progress has been made in the areas of design, analysis, and facility construction which must precede hardware fabrication and evaluation; however, 1964 will be characterized by a rapid increase in the pace of testing, especially of large component assemblies. These will follow the advances already made in gas generator, bearing, seal, and ignitor testing.

LIST OF FIGURES

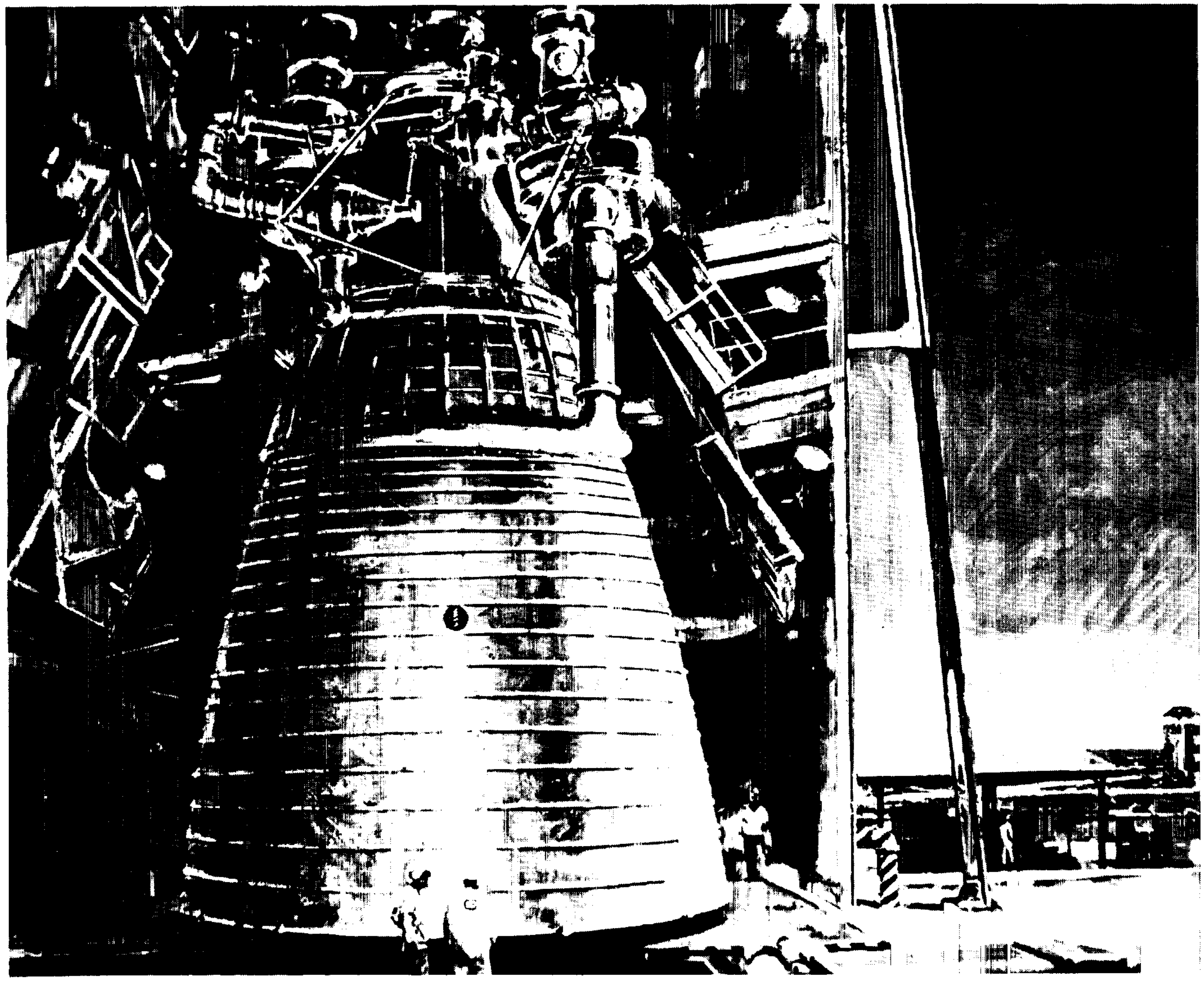

Figure 1. - M-1 Rocket Engine Project - Basic Engine System.

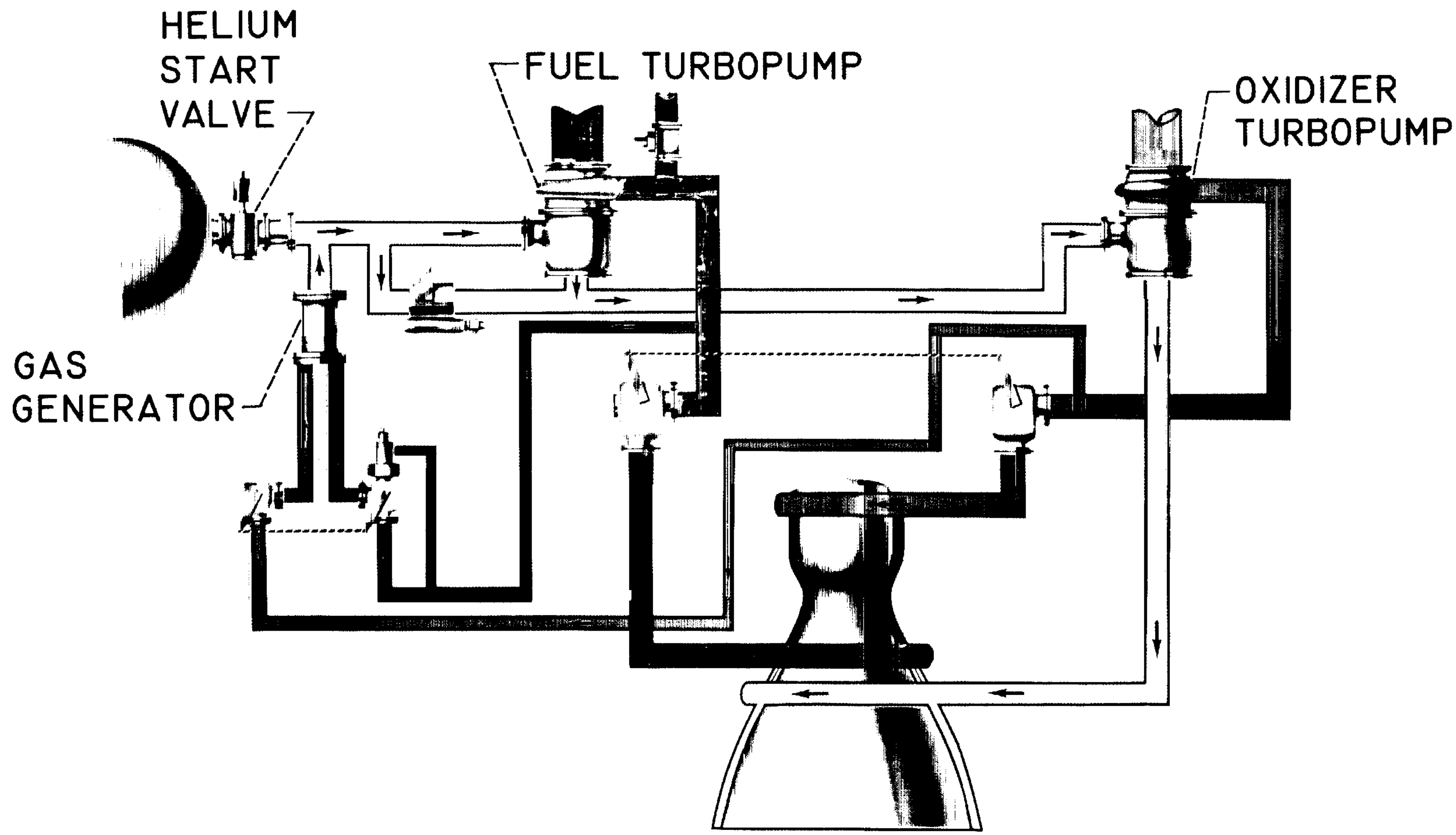

Figure 2. - M-1 Rocket Engine Project - Engine Fluid-Flow System.

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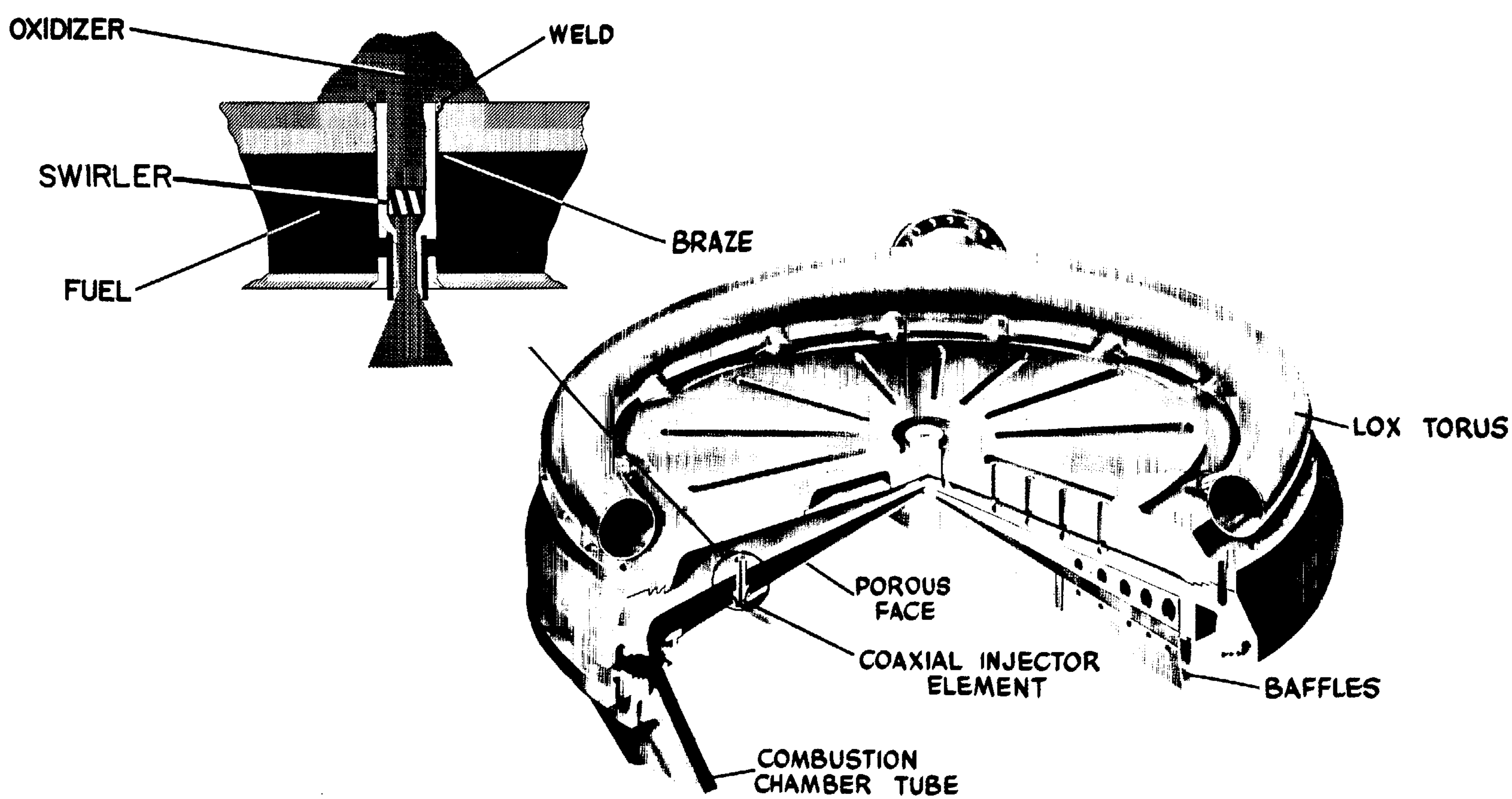

Figure 3. - M-1 Rocket Engine Project - Thrust-Chamber Coaxial Injector.

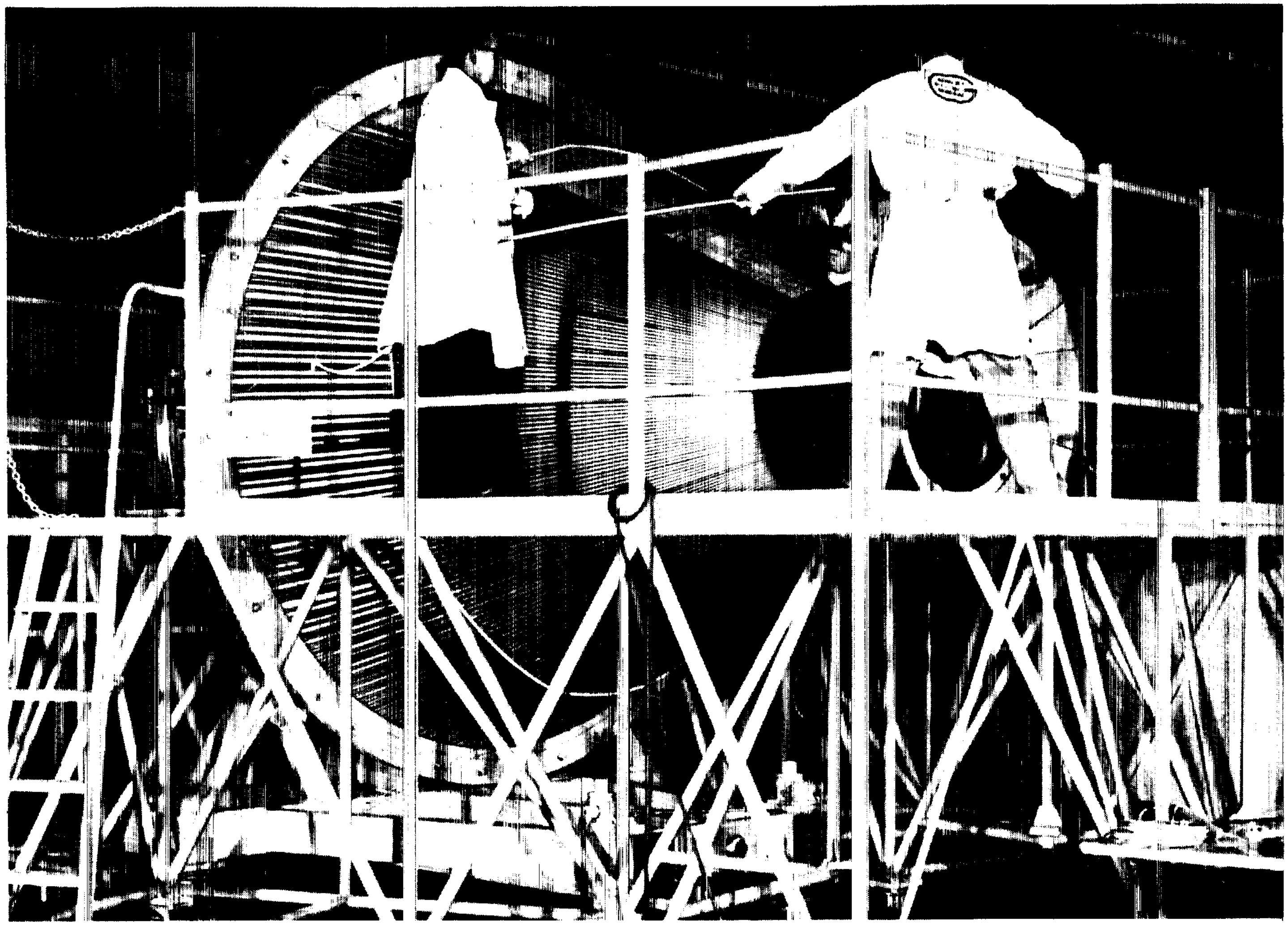

Figure 4. - M-1 Rocket Engine Project - Full-Scale (14 to 1) Combustion-Chamber Mandrel During Application of Tubing.

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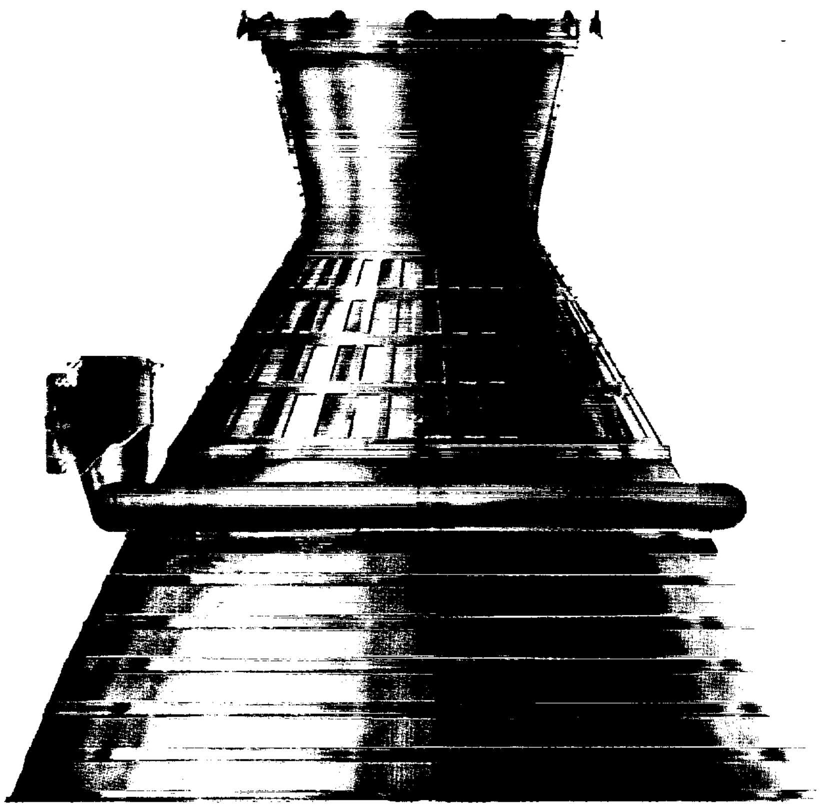

Figure 5. - M-1 Rocket Engine Project - Fuel Torus Mounted on Combustion Chamber.

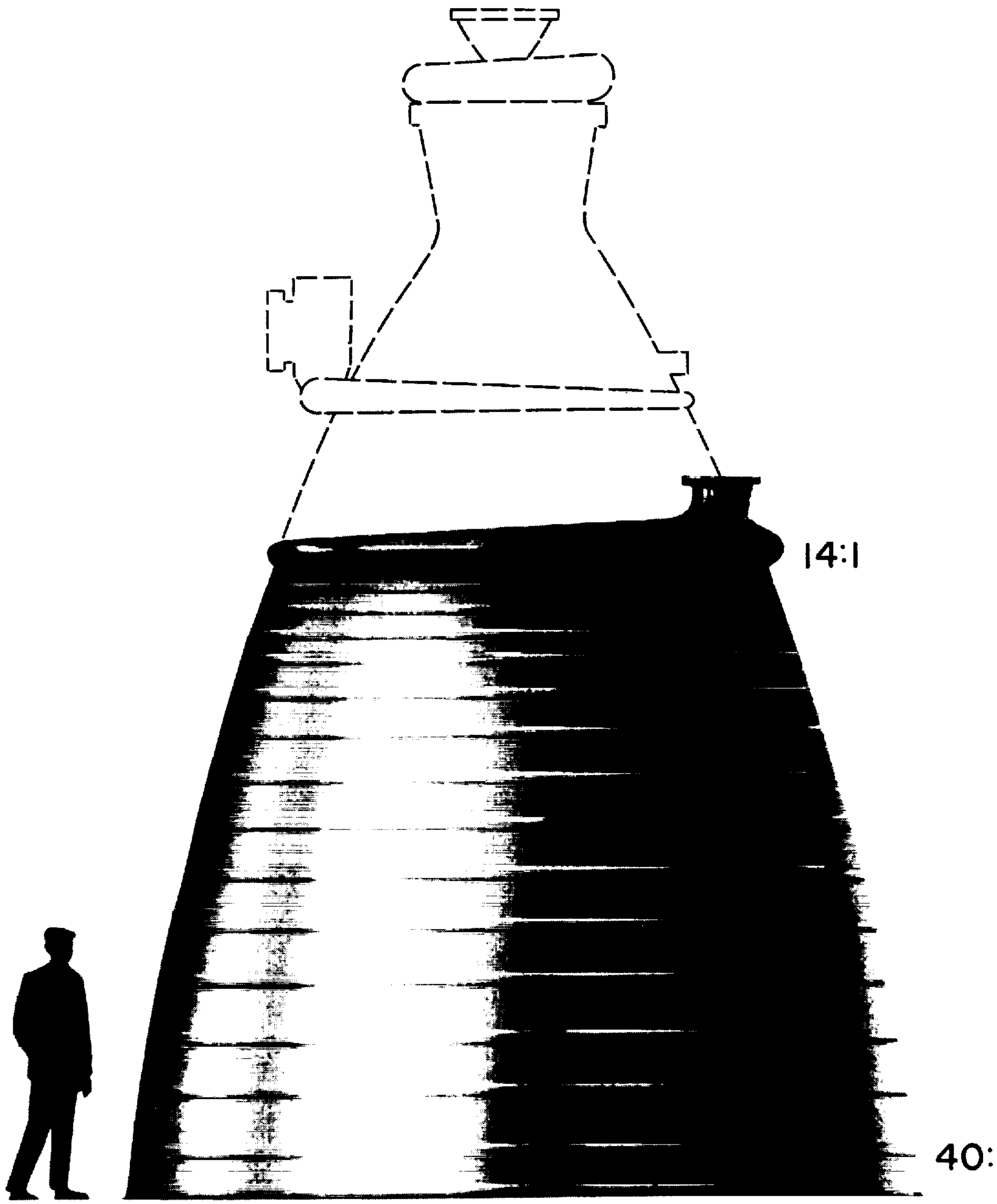

Figure 6. - M-1 Rocket Engine Project - Nozzle Extension (Area Ratio 14:1 to 40:1).

[redacted]

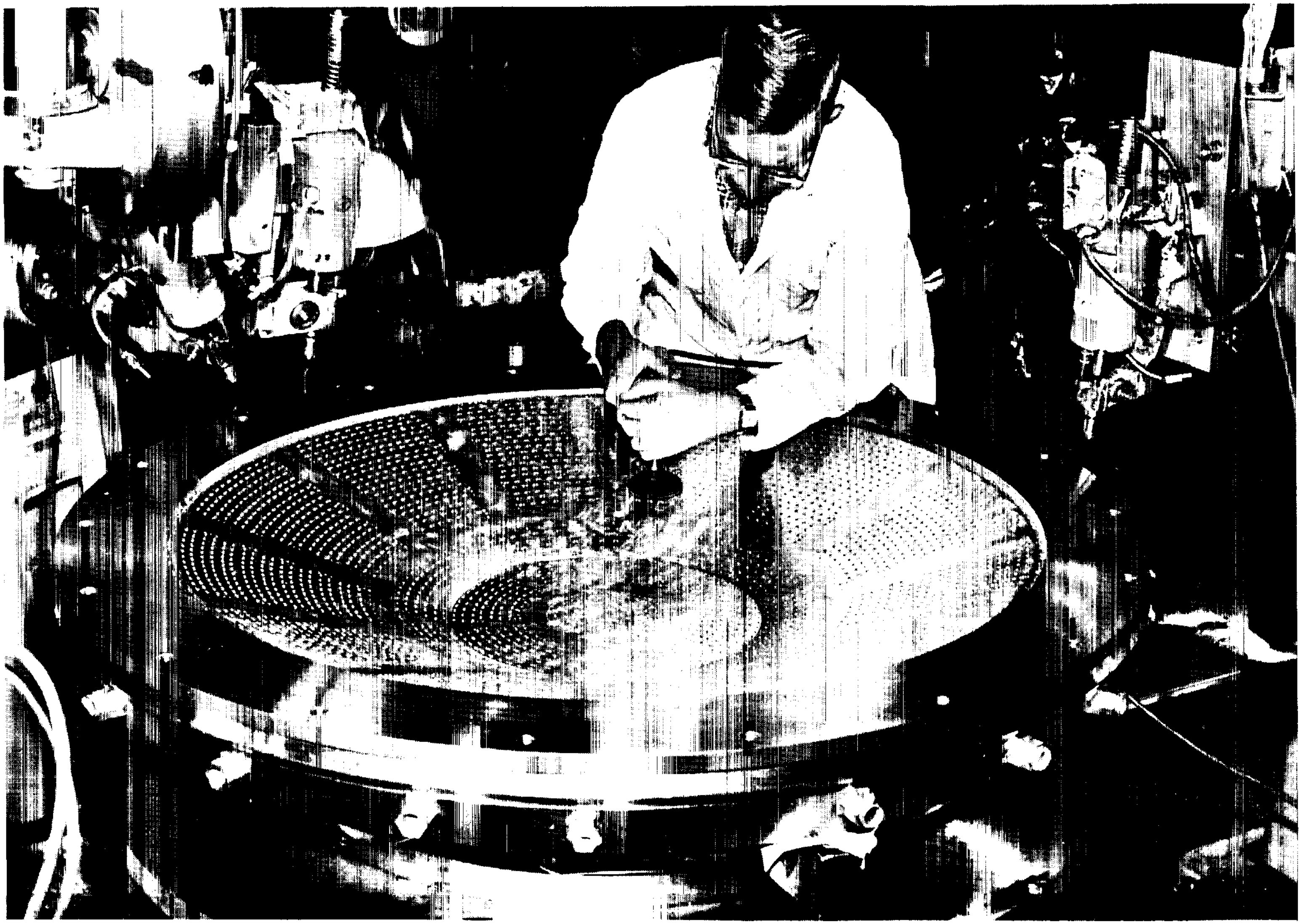

Figure 7. - M-1 Rocket Engine Project - Final Deburring of Thrust-Chamber Injector.

[redacted]

[redacted]

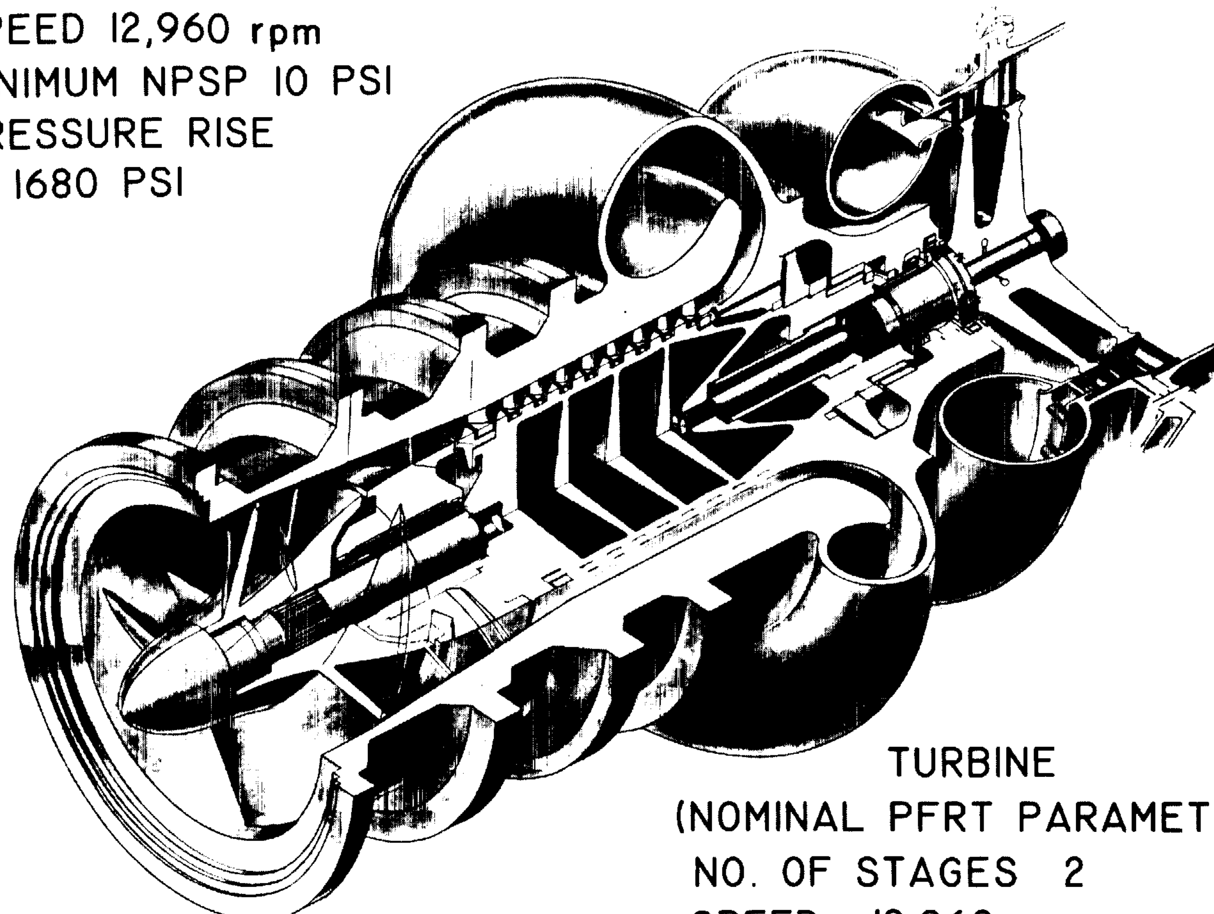

Figure 8. - M-1 Rocket Engine Project - Liquid Hydrogen Turbopump.

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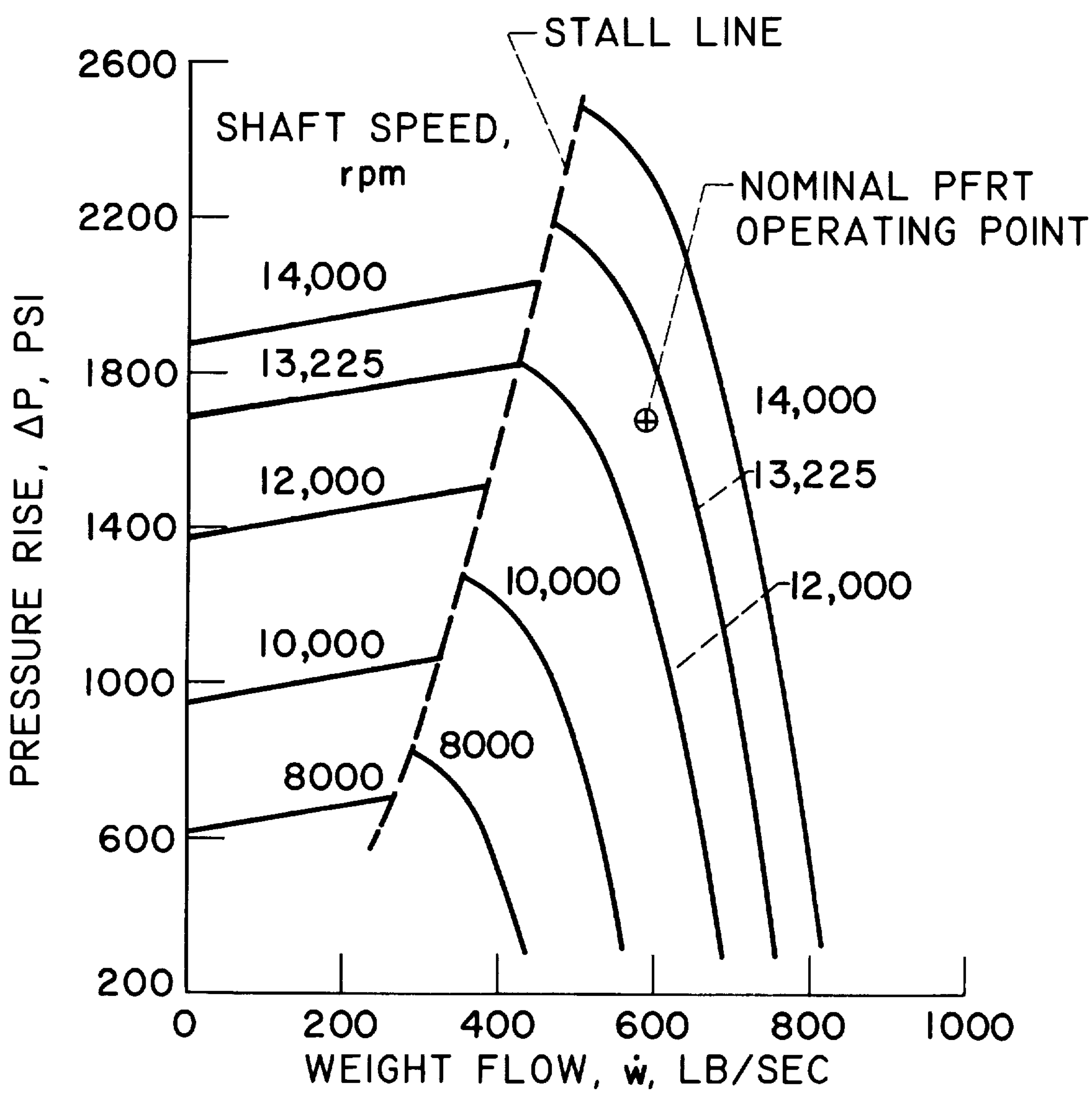

Figure 9. - M-1 Rocket Engine Project - Predicted Fuel Pump Performance.

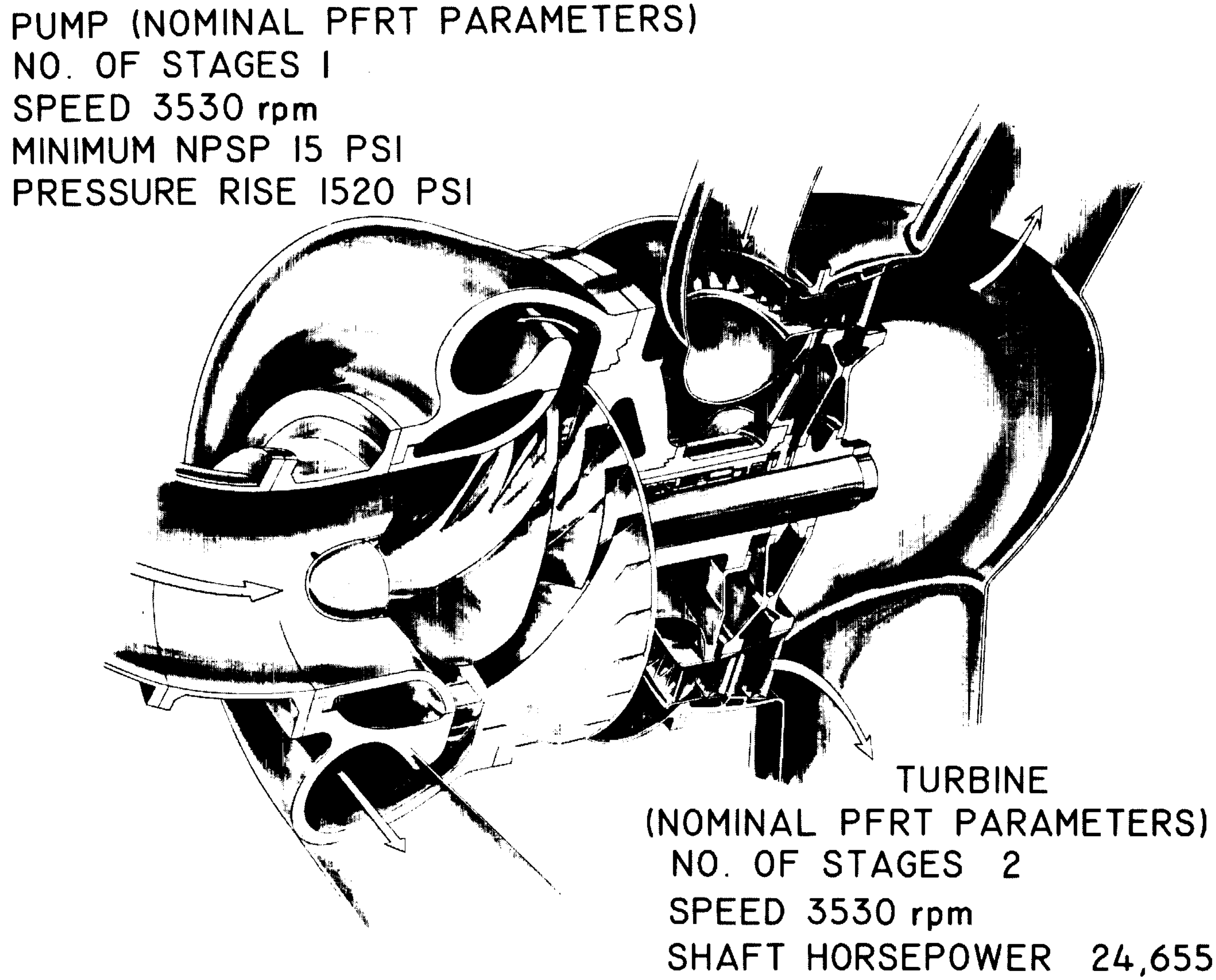

Figure 10. - M-1 Rocket Engine Project - Liquid Oxygen Turbopump.

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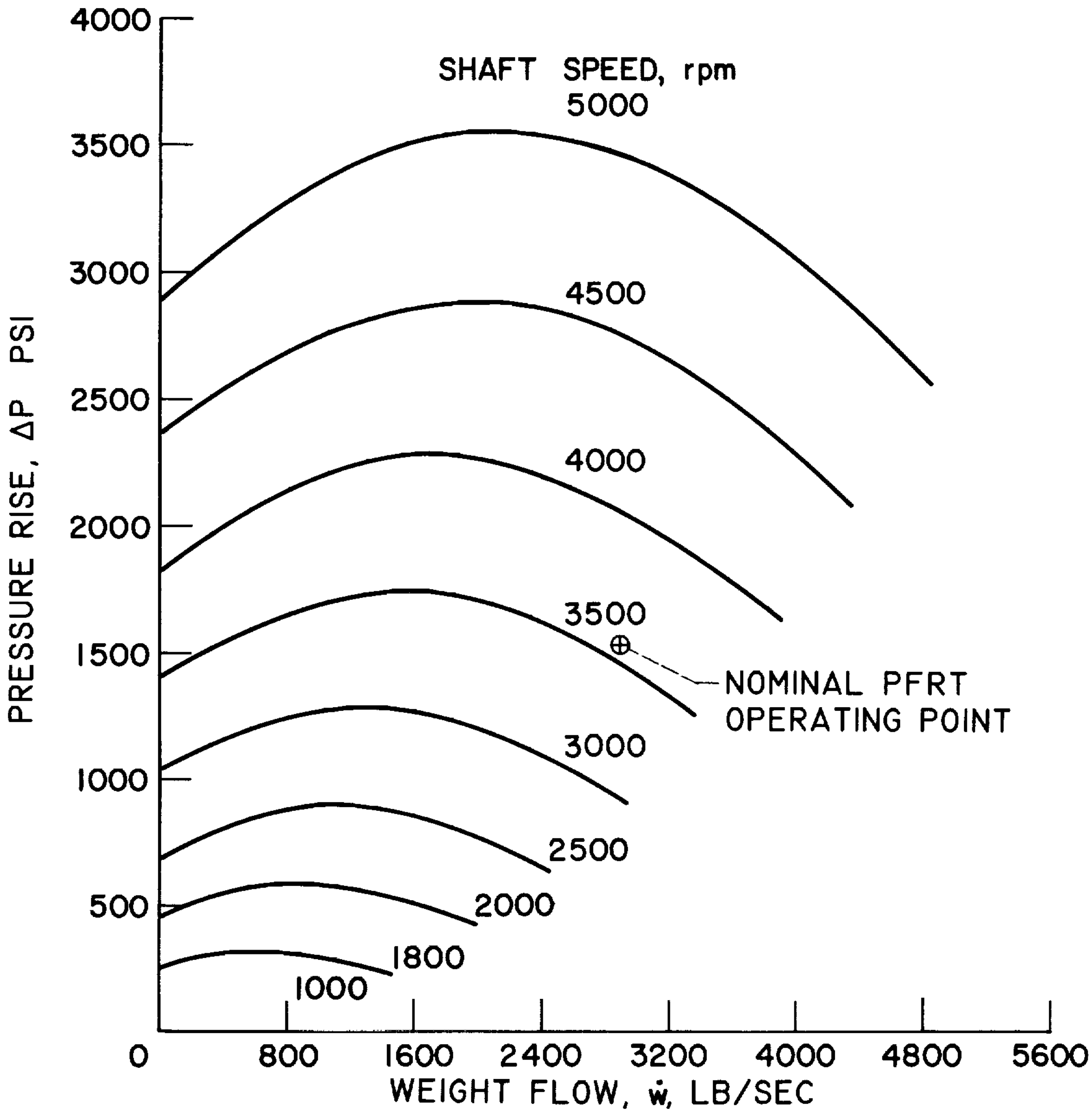

Figure 11. - M-1 Rocket Engine Project - Predicted Oxidizer Pump Performance.

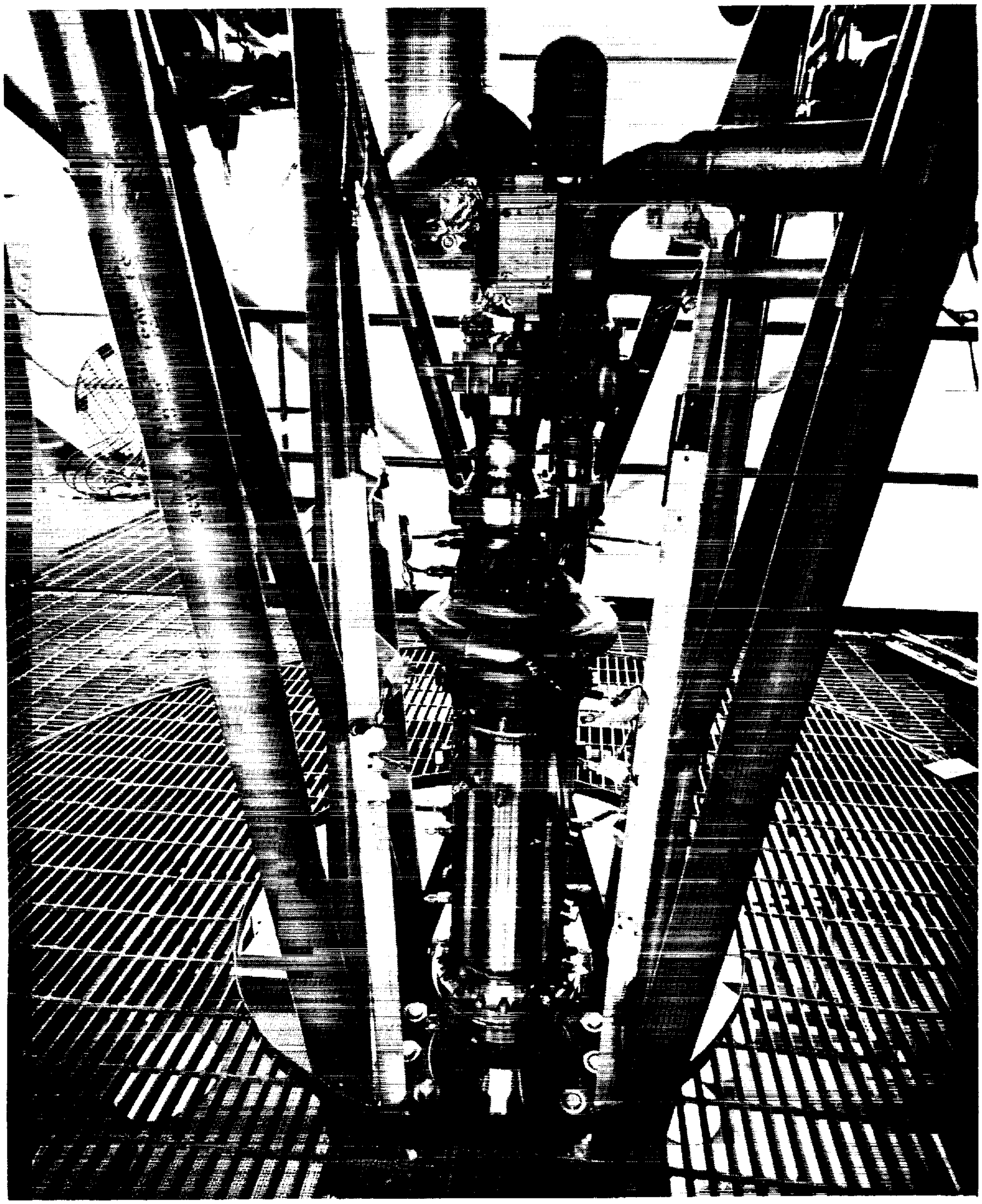

Figure 12. - M-1 Rocket Engine Project - Gas-Generator Assembly Installed in Test Stand C-6.

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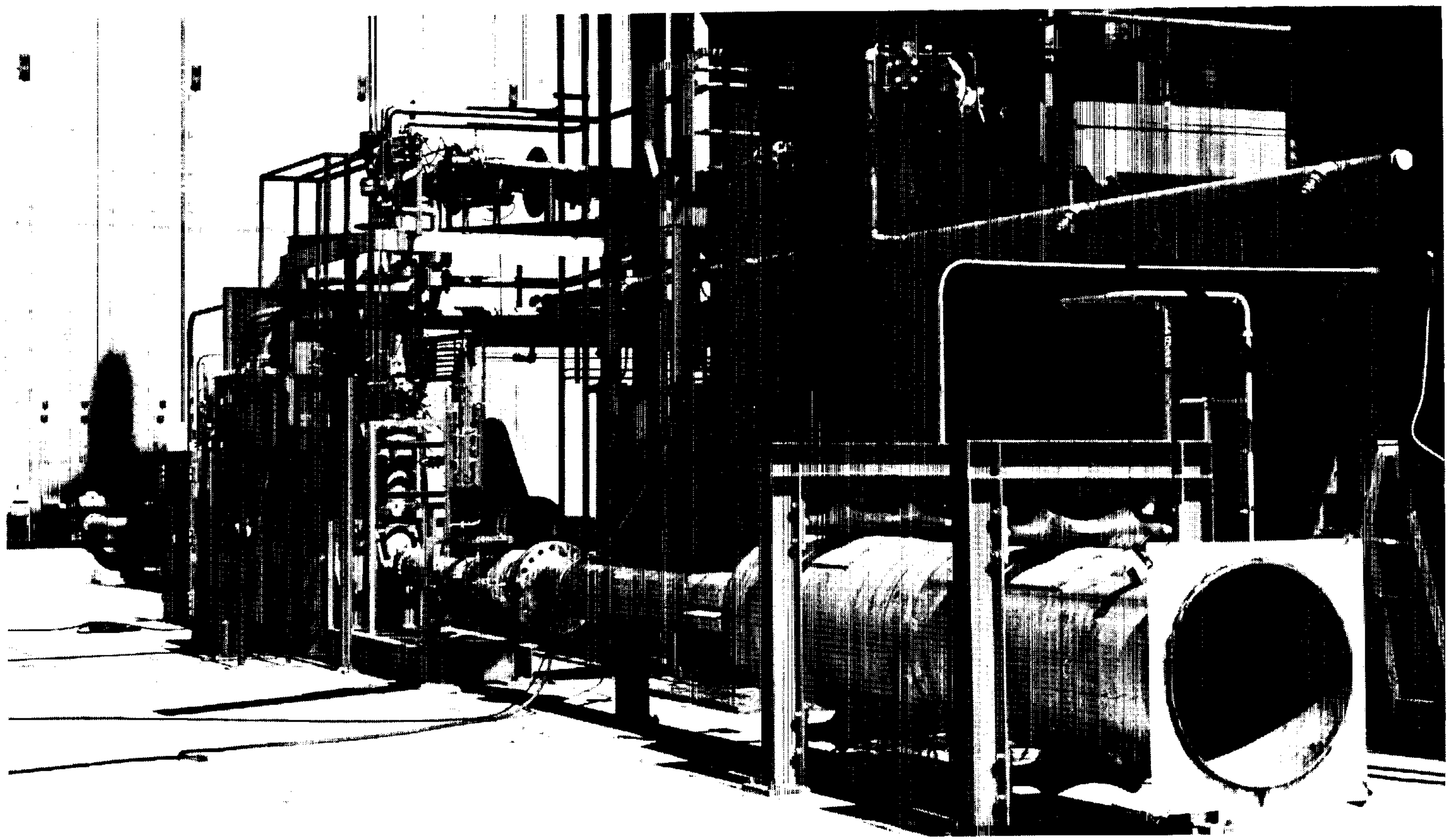

Figure 13. - M-1 Rocket Engine Project - Gas-Generator Assembly Installed in Test Stand C-9.

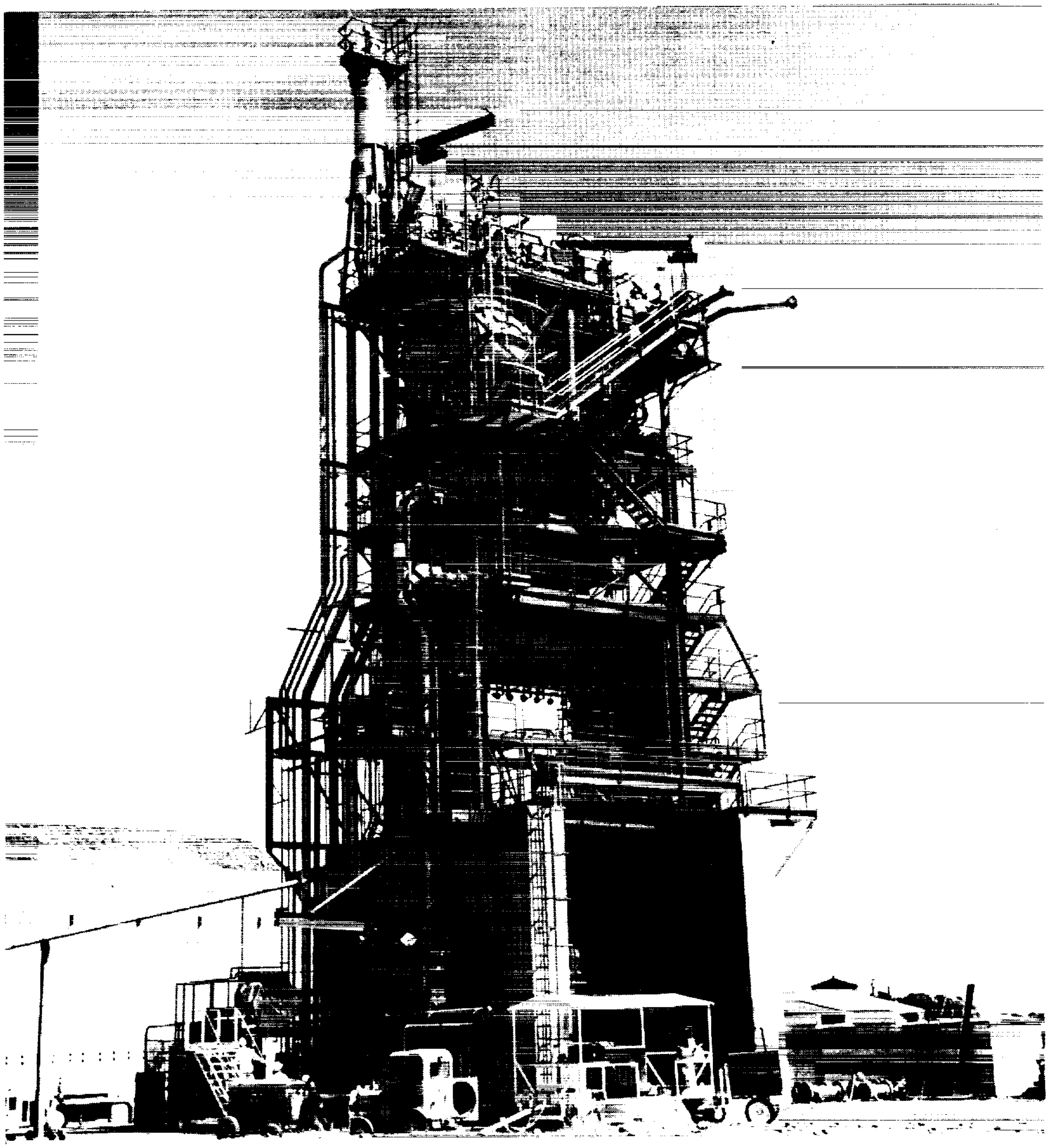

Figure 14. - M-1 Rocket Engine Project - Thrust Chamber and Gas-Generator Test Stand C-9.

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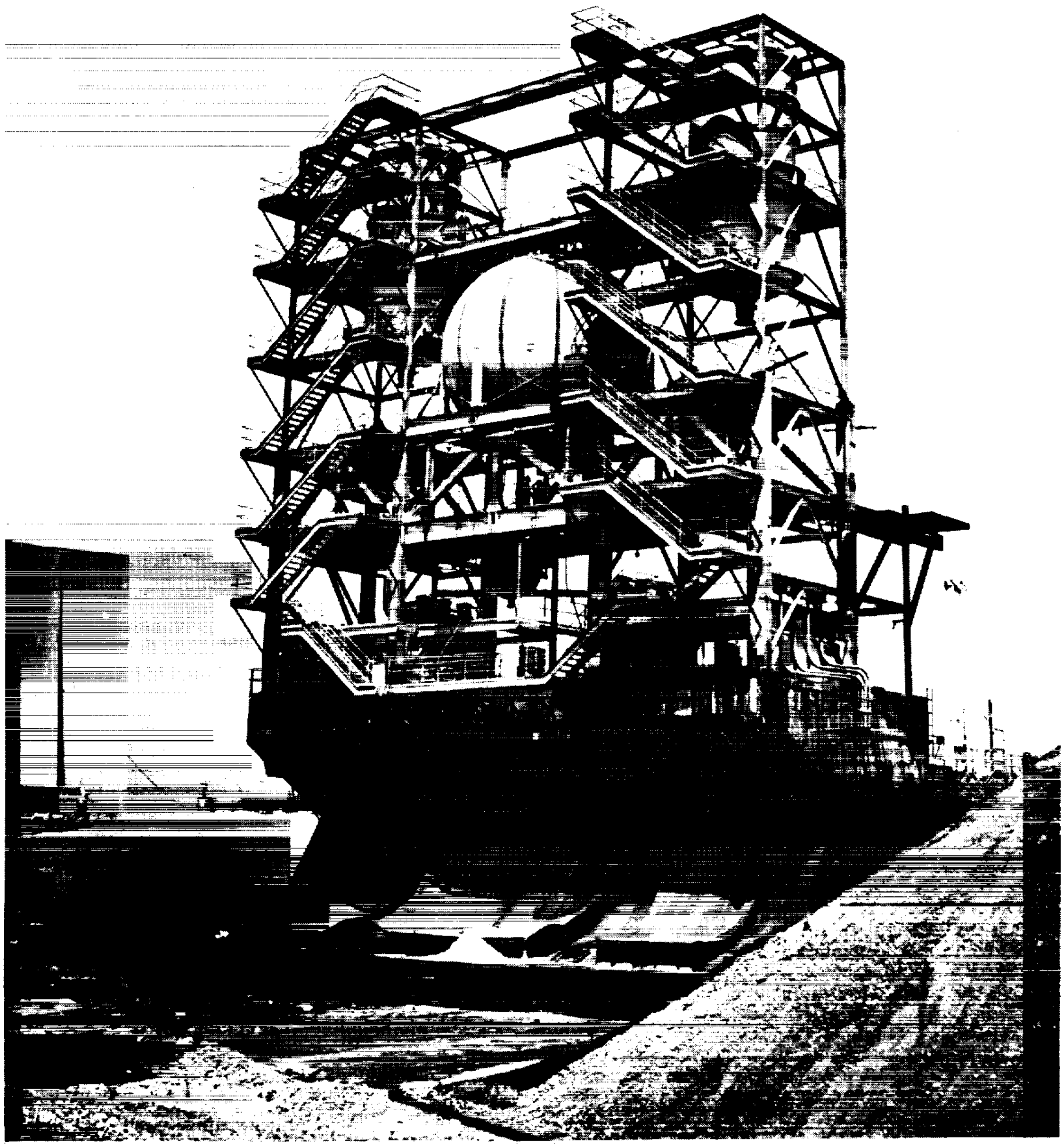

Figure 15. - M-1 Rocket Engine Project - Turbopump Test Stands E-1 and E-3.

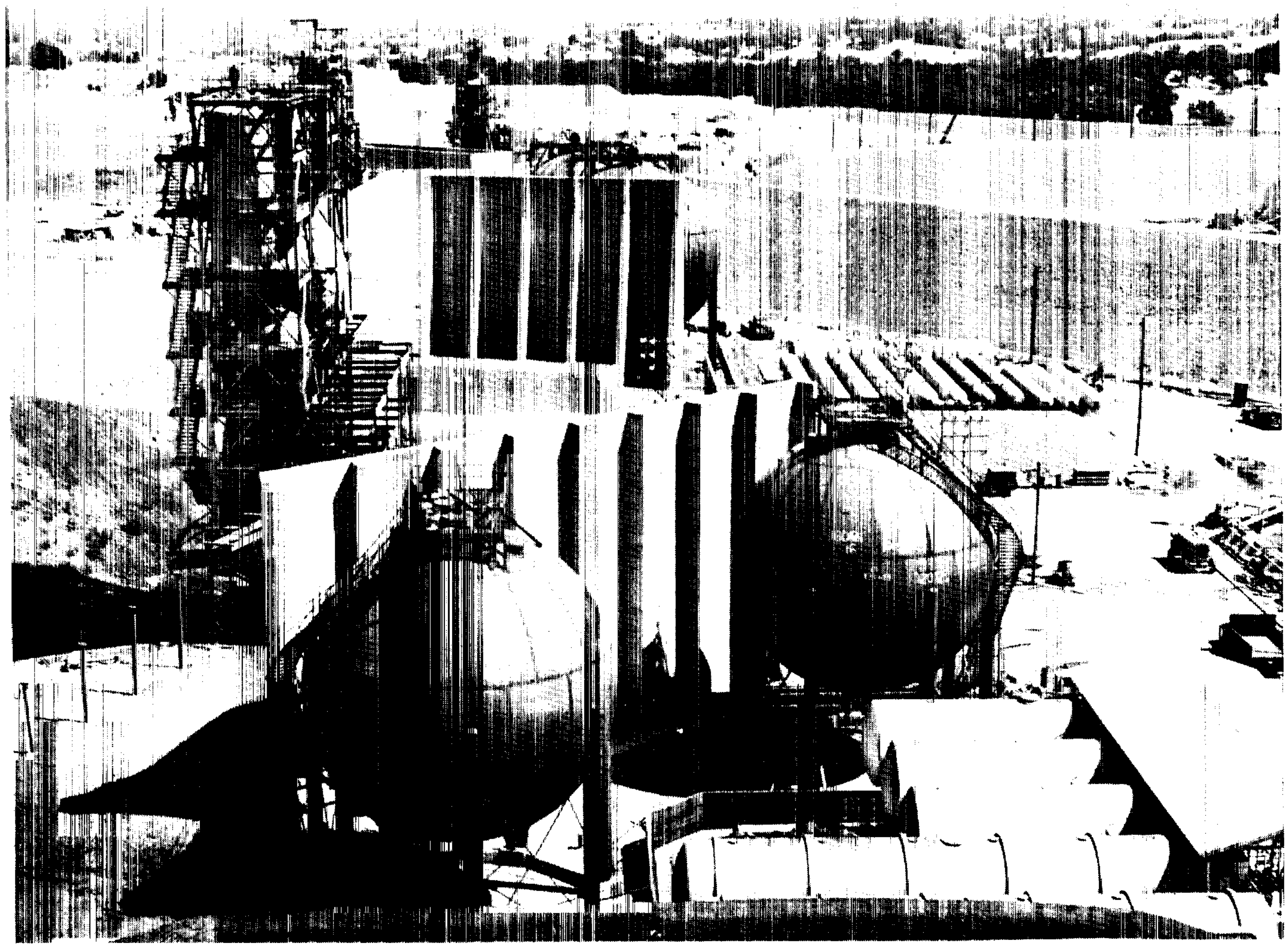

Figure 16. - M-1 Rocket Engine Project - Off-Stand Propellant Run-Tankage for Test Stands E-1 and E-3.

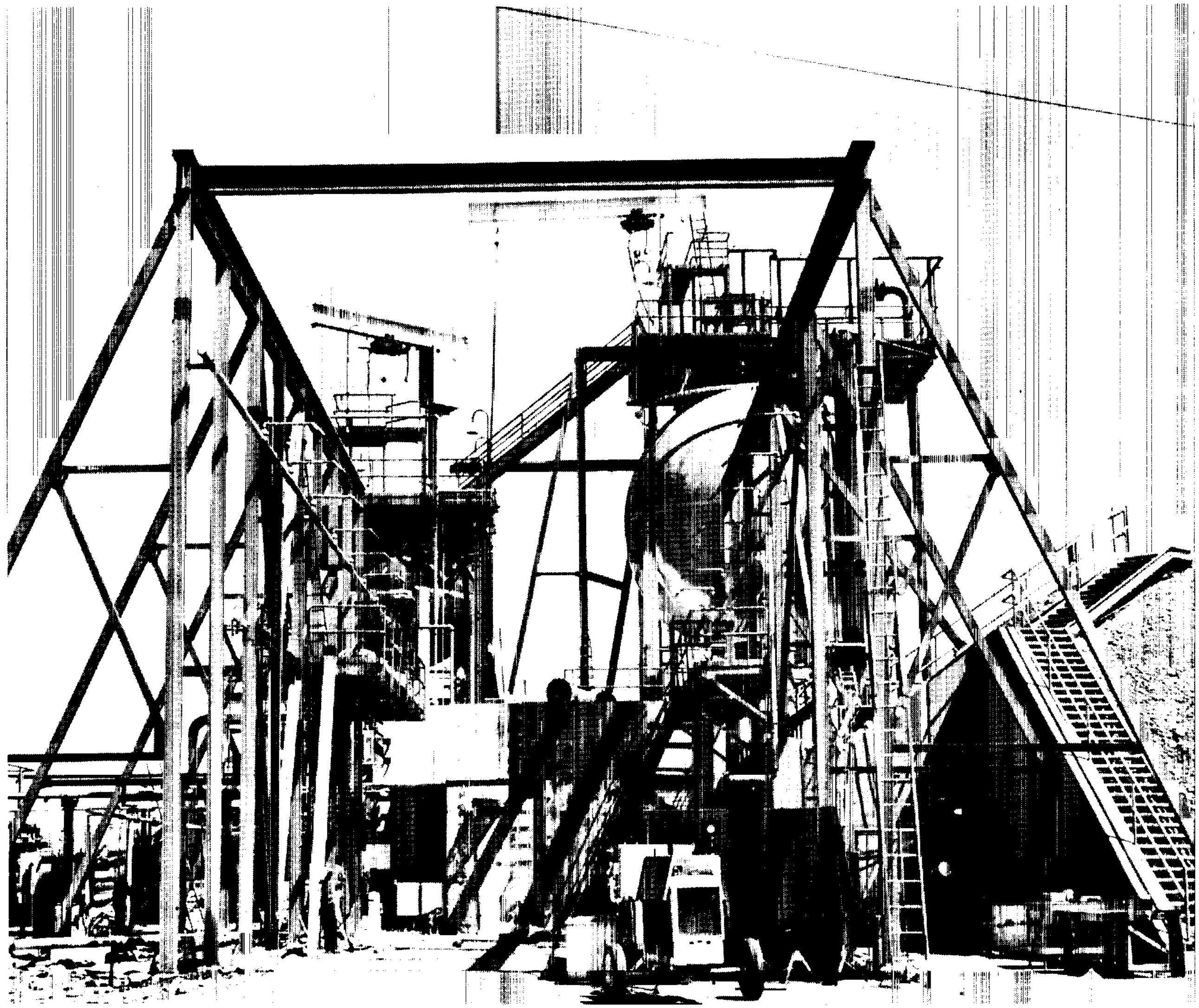

Figure 17. - M-1 Rocket Engine Project - Thrust-Chamber and Gas-Generator Test Stand H-8.

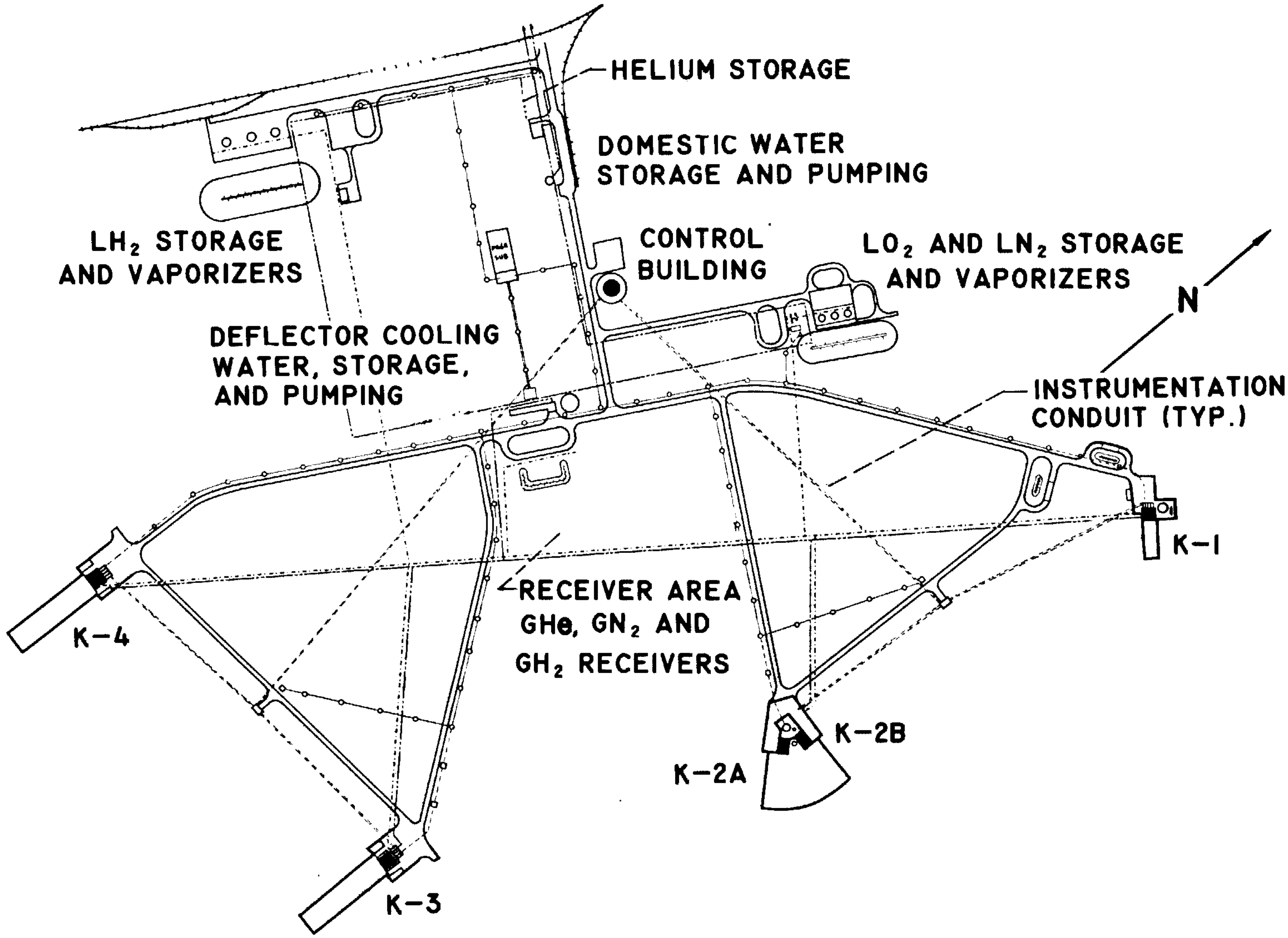

Figure 18. - M-1 Rocket Engine Project - Engine-System Testing K Zone.

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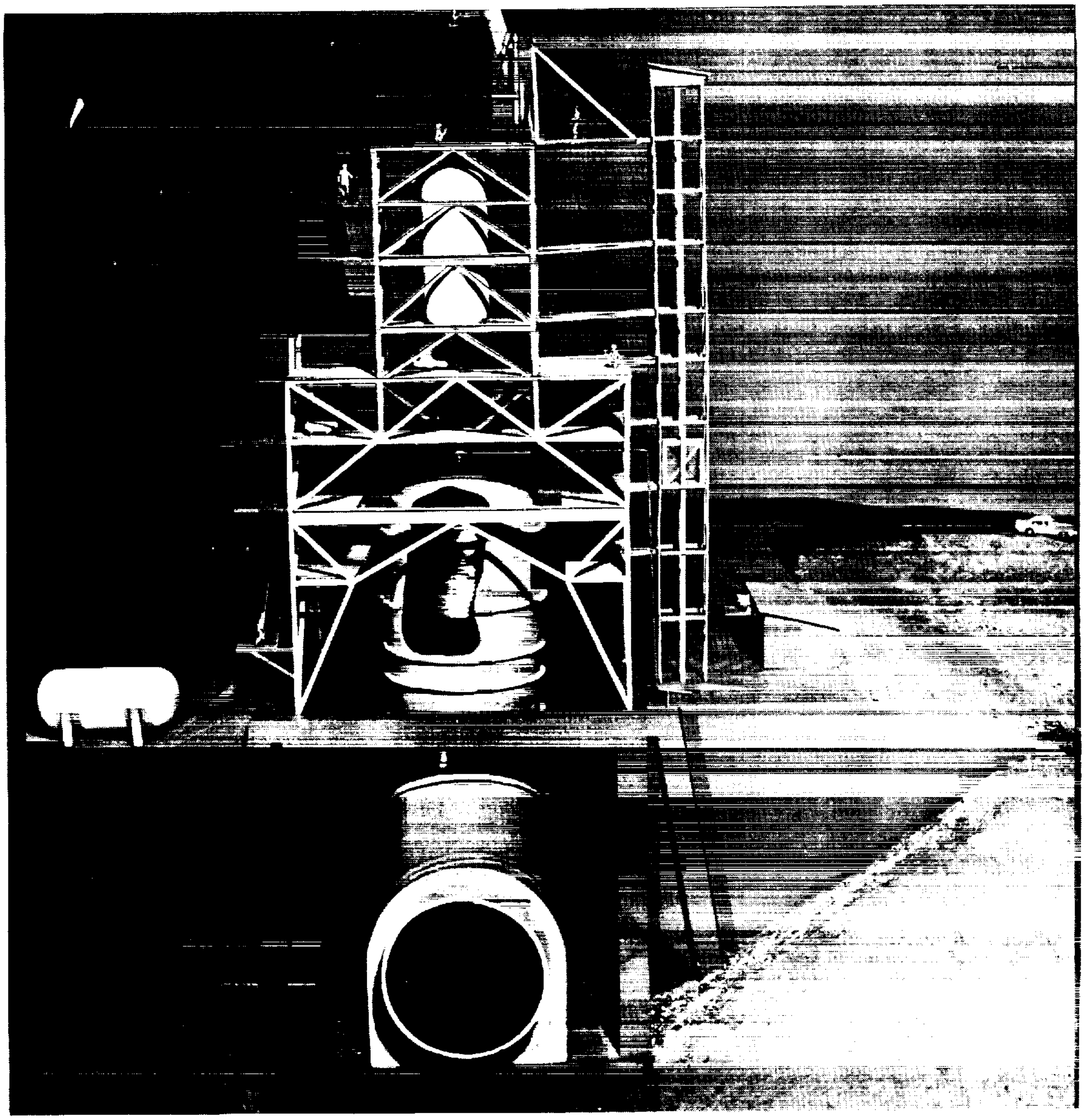

Figure 19. - M-1 Rocket Engine Project - Test Stand K-1 Concept.

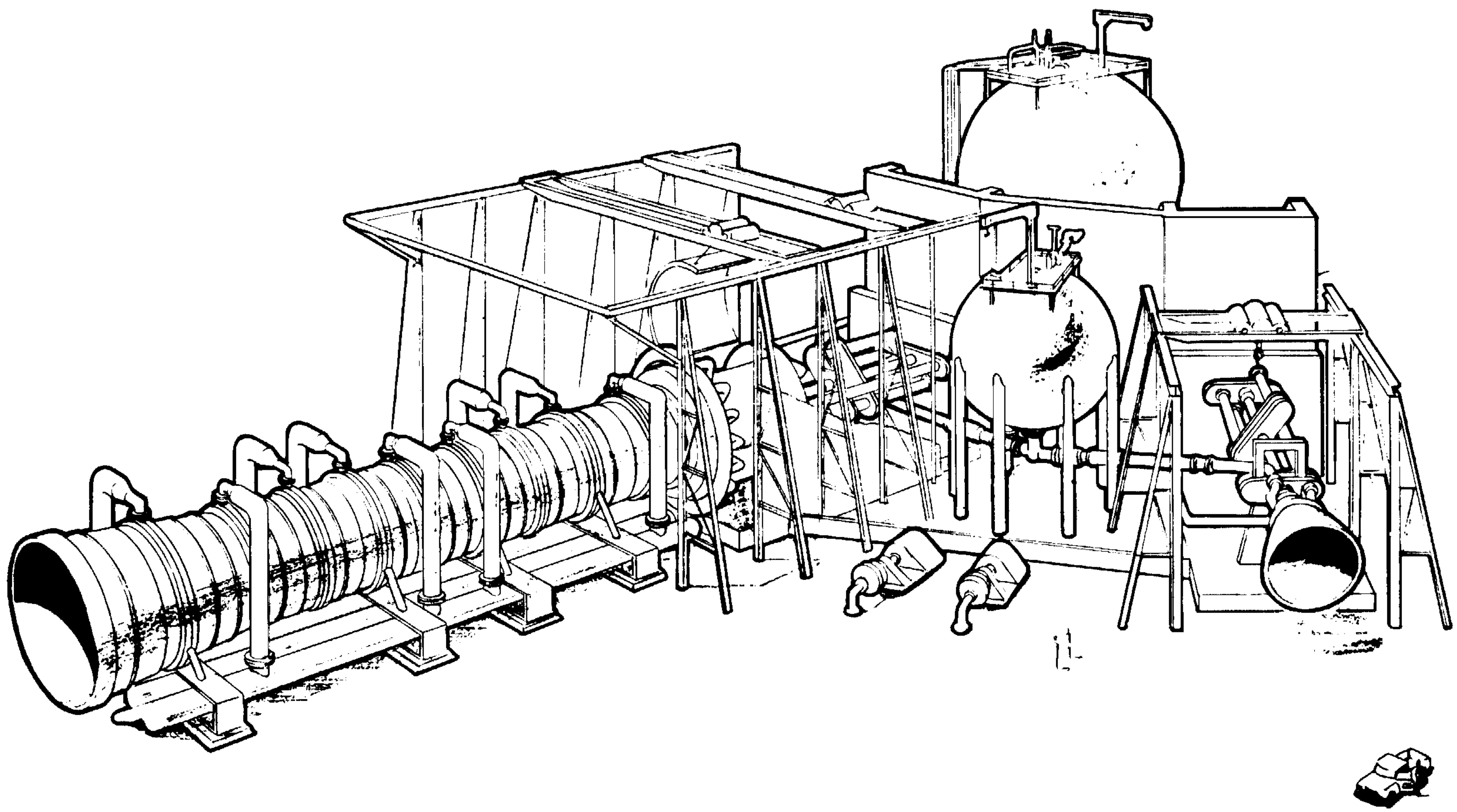

Figure 20. - M-1 Rocket Engine Project - Test Stand K-2 Concept.

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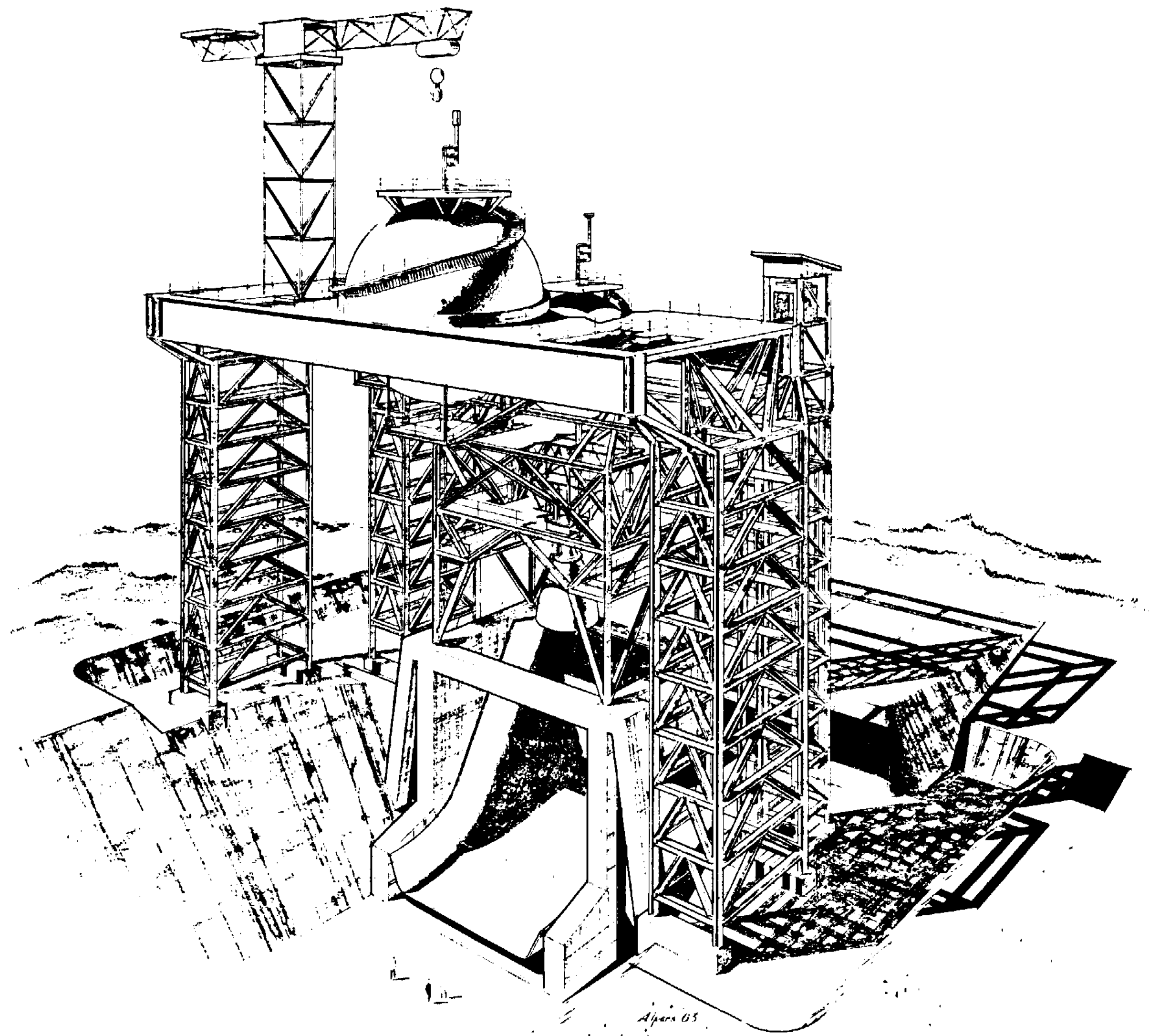

Figure 21. - M-1 Rocket Engine Project Test Stands K-3 and K-4 Concept.

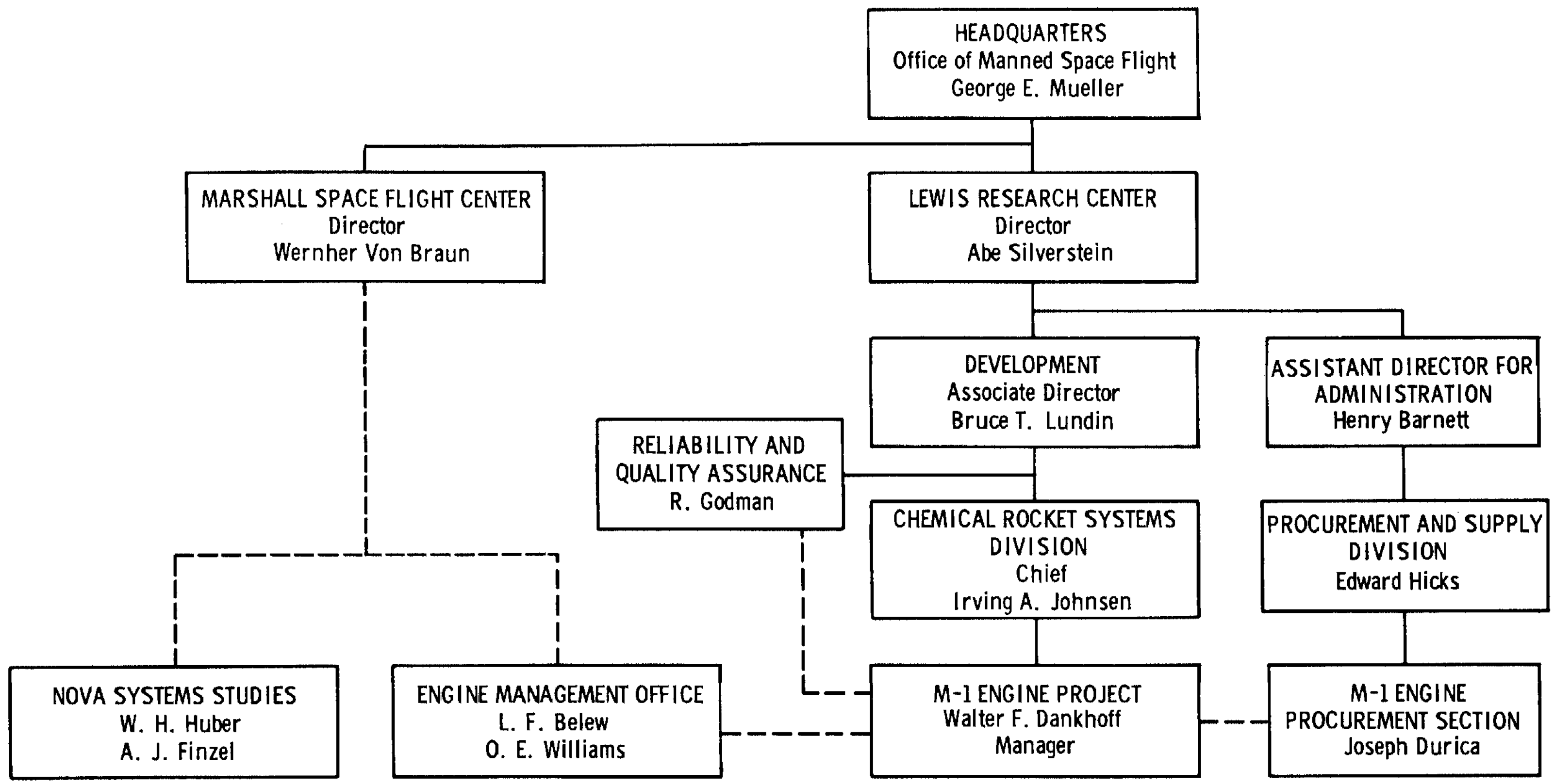

Figure 22. - M-1 Rocket Engine Project - Project Management Structure.

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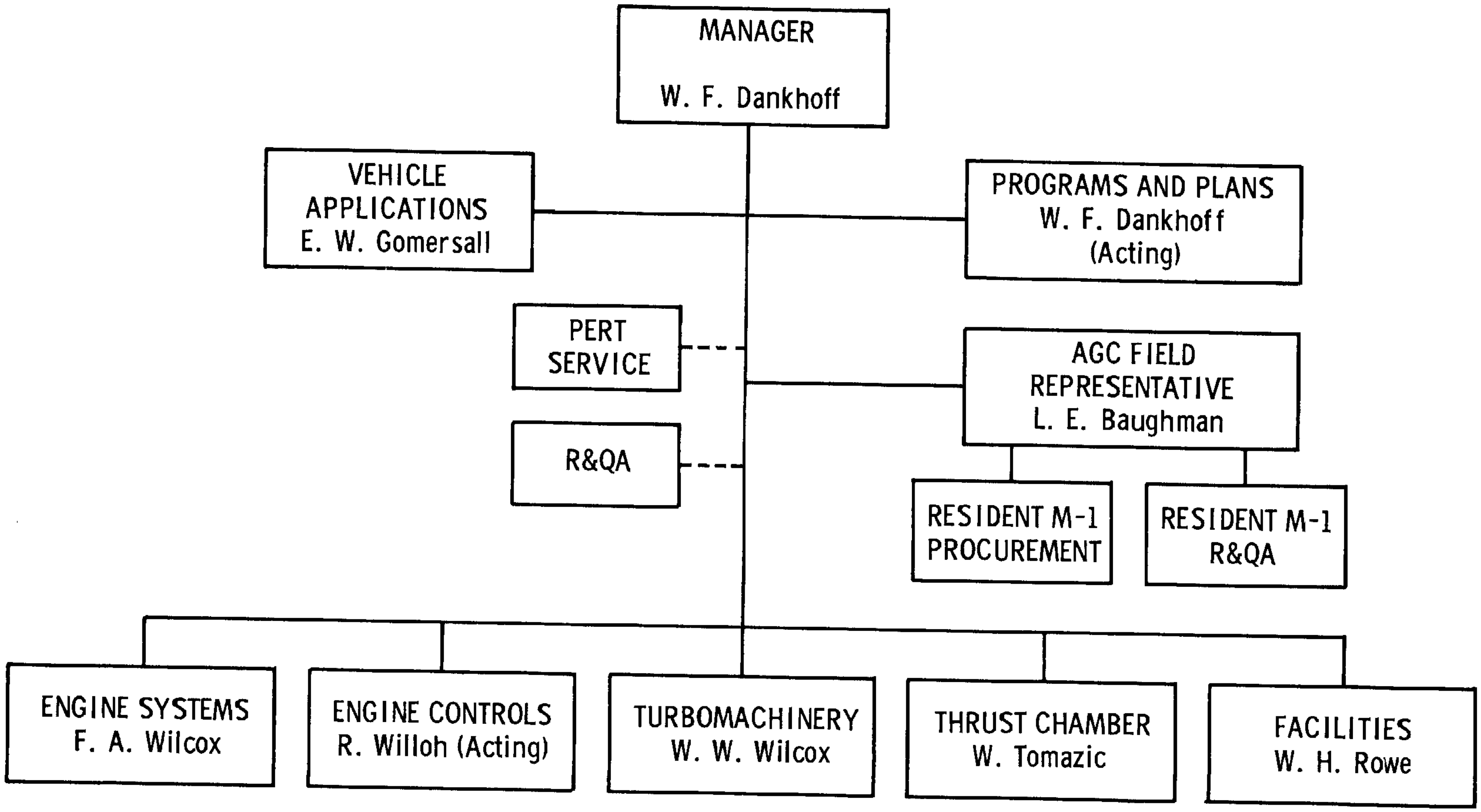

Figure 23. - M-1 Rocket Engine Project - Project Structure.

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