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DESIGN AND DEVELOPMENT OF A 1,500,000-POUND-THRUST SPACE BOOSTER ENGINE

By

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F-1 ENGINE DESIGN

The F-1 engine (Fig. 1) is a 1,500,000-pound-thrust rocket engine using liquid oxygen and RP-1 (kerosene) as propellants.

In clusters of 5 (Fig. 2) this engine will provide the booster power for the Advanced Saturn V ~~[illegible]~~ vehicle to boost the Apollo capsule into its moon flight-path. ~~Studies are also being [illegible] by NASA to use clusters of [illegible] 12 F-1 engines as the booster for the proposed Nova vehicle (Fig. 3)~~.

The principal objectives in the design of the F-1 engine were reliability, simplicity, and the degree of ruggedness required to provide a high margin of safety for the manned applications.

Assuming that high reliability is designed into each part of a rocket, then the highest over-all engine system reliability would be achieved by using the fewest parts. With this concept in mind, the design was directed toward utilizing a single component for each required function.

The achievement of this goal is demonstrated in the engine presently being developed for the first major milestone of the preliminary flight rating tests (PFRT). This engine consists of one thrust chamber, one injector and dome, one direct-drive turbopump, one gas generator, two fuel valves, two oxidizer valves, a start valve, and a gas generator ignition exciter. The latter two components are the only engine parts requiring electrical energy. Instrumentation to monitor engine operation during flight also requires electrical energy but is independent of engine operation.

To illustrate how this engine design was achieved, a review must be made of the developed and proved concepts used, as well as some of the new concepts, and the manner in which these concepts were combined with advanced metallurgical concepts.

Feasibility studies of large thrust chambers firing at high chamber pressures began in 1958 under an Air Force-sponsored contract. This effort included, Rocketdyne tests of an uncooled thrust chamber, with the approximate combustion chamber dimensions of today's F-1 engine, at the 1,000,000-pound-thrust level. Therefore, at the time of the award of the NASA F-1 contract to Rocketdyne on 9 January 1959, the use of the single-chamber concept appeared feasible.

The final F-1 thrust chamber assembly design (Fig. 4) provides for a tubular wall, regeneratively cooled thrust chamber with an extension, a double-inlet oxidizer dome, and a flat-face injector.

The chamber is designed to flow 2 tons/sec of liquid oxygen and 1 ton/sec of fuel, and to burn these propellants at a pressure of approximately 1150 psi at the injector face. The fuel-cooled portion of the thrust chamber extends to a 10:1 expansion area ratio. Its approximate measurements are: a 40-inch combustion chamber diameter, a 9-1/2-foot nozzle exit diameter, and a length of 11 feet. To meet requirements of expansion area ratios up to 16:1, a skirt is attached to the thrust chamber.

Experience had demonstrated that the injector presented the major development hurdle. Thus as a starting point, Rocketdyne immediately started an F-1 injector test program, utilizing uncooled, heavy-duty thrust chamber hardware which was fabricated easily and inexpensively. The successes and failures encountered in past injector programs were re-analyzed to determine the experimental approach to design and development of the F-1 injector. Advanced design criteria had to be established for an injector operating at a much higher injector density and chamber pressure than any previously known. Copper rings were found to be structurally adequate while affording a greater degree of protection from face overheating or local burning.

At this point, injector design (Fig. 3) and the problem of combustion stability will be discussed. The initial injector designs for the F-1 engine were similar to, and based upon, the design concepts developed for earlier Rocketdyne engines. Stable injector configurations were quickly evolved; one of these is still a basic Rocketdyne configuration. However, a new requirement has been added in line with the Rocketdyne manrated safety concept. The requirement is that the injector, as well as the entire engine system, must be dynamically stable. This means that if the system is disturbed from any source, it will quickly damp out the resulting oscillation.

This is a severe requirement but Rocketdyne has made good progress and is confident of success by the end of the program.

A large share of Rocketdyne's total development effort is concentrated on this task. This effort is in three areas: analysis, model research testing, and full-scale design and testing. The analysis team has re-examined all earlier test records, and there were thousands of them, looking for significant trends. The team also carefully analyzed design concepts for sources of disturbances.

The design team examined results and made a list of characteristics and requirements for a good injector. This list is continually re-examined and revised as the program progresses.

Two-dimensional models (Fig. 6) are used to examine particular features in detail by means of high-speed photography and high-response instrumentation.

The real proof, however, lies in building full-scale hardware. This is, of necessity, a slow and costly process. Rocketdyne is following many parallel design paths and has achieved encouraging results from injector systems with good dynamic stability characteristics.

Some of the requirements for dynamic stability are:

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1. Feed system isolation, i.e., permitting no interaction of disturbances between feed systems supplying the injector elements with fuel and oxidizer
2. A system of baffles across the injector face to reduce radial and tangential vibration modes in the combustion chamber
3. Axial distribution of the propellants to break up the critical combustion zone
4. No joints or leakage sources within the injector assembly. The goal is to have parent metal between oxidizer and fuel all the way to the injector face.

The thrust chamber design required a structure for containing the combustion process and flowing high-pressure fuel in adequate amounts to provide proper cooling with a minimum pressure drop. Past Rocketdyne experience again indicated a tubular wall, regeneratively cooled thrust chamber would meet these requirements (Fig. 4). The search for a tube material included analysis of nickel alloy and steel in addition to various other alloys (Fig. 7). Model test firings, materials development research, and laboratory and shop investigations of fabrication techniques were conducted.

On the basis of this development and analysis, a high nickel alloy was selected as the tube material. The high-strength property of this alloy permitted design of thinner wall section tubes resulting in minimum weight. The thinner tubes also provided a lower gas side wall temperature even though their thermal conductivity was lower than that of some other materials considered. The design provided adequate chamber cooling with only approximately two-thirds of the total available fuel flow passing through the tubes. Therefore, a fuel side thrust chamber bypass arrangement was included in the design to reduce the system pressure drop.

Finally, a manufacturing process development program was conducted which demonstrated that the tubes could be hydraulically formed and furnace brazed.

Today's F-1 thrust chamber is a relatively lightweight unit, entirely furnace brazed (Fig. 8). Development testing has verified the adequacy of the design (Fig. 9).

The aforementioned thrust chamber extension was designed to provide expansion area ratios up to 16:1 by the attachment of an extension to the liquid-cooled thrust chamber. Analyses and other development test experience indicated that the extension would have an adiabatic wall temperature over 2796° C and, therefore, some cooling was required.

Cooling to the 649 to 816° C range appeared feasible by passing the turbine exhaust gas at temperatures of 482 to 649° C through a double-wall skirt. This design concept was based on the use of an alloy having high strength at elevated temperatures. Today, the skirt is undergoing development testing conducted to investigate the proper method of distributing the turbine exhaust gases through the skirt. The design is similar to that of a turbojet engine afterburner.

The thrust chamber liquid oxygen dome was provided with a dual inlet for improved liquid oxygen distribution to the injector. The dome material of high nickel alloy was selected for high strength, because the dome is the focal point for the transmission of the thrust through the gimbal bearing to the vehicle.

The primary consideration in selection of the turbopump design (Fig. 10) again was to attain reliability by using a minimum number of parts and proved design concepts. Ten turbopump design arrangements were investigated to determine performance characteristics of designs generally suitable for pumping propellants to a high-performance thrust chamber. The configurations investigated fell into two groups: the direct-drive nongear systems, and those requiring a gear transmission because of differential speeds between the pump and turbine or between pumping elements themselves.

The more important requirements considered included the net positive suction head (NPSH) requirement of 65 feet liquid oxygen and 80 feet fuel for the flight engine. The NPSH determines the maximum speeds at which the pump could be operated without cavitating. This is defined as the suction specific speed.

The turbine shaft horsepower (BHP) requirement was determined primarily by the pressure and flow needed by the thrust chamber.

On the basis of this evaluation and the development of a high suction performance inducer which permits the pumps to be connected directly to the normally high-speed turbine, the direct-drive system was selected.

The inducer or partial axial stage of the turbopump has the ability to operate at a high suction specific speed, thereby permitting the pumps to operate at approximately 6000 rmp without cavitation.

The pump elements selected were of centrifugal design because of speed, flow, and head requirements. Over-all pump diameter was reduced as much as possible by using double-volute rather than single-volute discharge. This design also minimizes radial thrust loads. The fuel pump was located between the oxidizer pump and turbine to separate the elements having the greatest temperature extremes. To reduce axial thrust loads, the pump impellers have balancing ribs on the back side. The main thrust bearing is a double row of split inner-race ball-bearing arrangement which can safely take high axial loads.

The turbine design is two-stage and velocity-compounded and consists of two rotating impulse wheels separated by a set of stationary impulse stators. In this design, the total pressure drop occurs across the inlet nozzles, which permits the rotating elements to be relatively free from any axial thrust loads and only the turbine inlet manifold to be subjected to the total gas pressure.

The selection of materials was carefully made with the thought of providing adequate safety margins wherever possible. A light, strong aluminum-alloy casting was selected for the volutes, impellers, and inlets which have operating pressures as high as 2000 psia. For the turbine wheels and manifold, a nickel alloy having high strength at elevated temperatures is used to provide adequate strength for high operating pressure temperatures. Development testing (hundreds of tests were conducted) of the F-1 turbopump assembly, weighing approximately 2500 pounds and delivering 60,000 horsepower, has successfully demonstrated the soundness of the design principles and materials used.

To drive the turbine, the F-1 engine uses a gas generator burning approximately 2.3% of the total engine propellants (Fig. 11). The principal objectives of reliability and safety were furthered in the design of the gas generator body by making the assembly as simple as possible. The gas generator, approximately 10 inches in diameter and burning propellants at the rate approximately required for a 40,000-pound-thrust engine, was first tested in March 1961; it is successfully undergoing testing in the continuing development program.

The control components were selected as the practical minimum number required. In the selection of the fuel and oxidizer valves, the advantages and disadvantages of single, large components were weighed against multiple smaller valving. Two fuel (Fig. 12) and two oxidizer valves (Fig. 13) were selected to minimize sealing areas on each of the valves as well as to provide valving in locations that allow a better distribution of propellant feed to the thrust chamber. All types of valves were considered prior to selection of poppet-type main propellant valves. The basis for this selection was the need of balancing large overturning moments that exist with large butterfly or plug-type valves.

The actuators for the main propellant and gas generator valves utilize engine fuel as the hydraulic fluid. Hydraulic actuation was selected to avoid addition of a pneumatic system in the engine, and fuel was selected as the hydraulic fluid to make use of a fluid which is available in the engine at the required pressures.

The other major control components include dual-linked valves, gas generator controls, and four-way solenoid valve. The valves were selected for the gas generator to provide fast and repeatable actuation while minimizing pressure drop and valve seal forces.

The four-way solenoid valve controls the sequencing of the engine system by controlling the fuel flow to the actuation mechanism of the main propellant and gas generator valve. This valve and the spark ignition system for the gas generator are the only engine components requiring electrical energy.

The primary advancement made in the material areas was the use of high-strength aluminum-alloy forgings rather than castings for the valve bodies. This design provides the propellant valves with a much higher proof pressure capability.

The valves have since demonstrated almost perfect sealing capability and predictable and repeatable actuation times in both component and engine test.

The engine system design objective was to place the components in a minimum size package and still retain maximum accessibility. The engine package configuration was patterned after the Atlas sustainer, the S-4 engine. The 60,000-pound-thrust engine was Rocketdyne's first large, gimbaled, liquid-propellant rocket engine in which the turbopump was mounted directly on the thrust chamber. This engine system package provides a more compact design independent of the vehicle and its specific engine mounting structure. The fixed relationship between the turbopump and thrust chamber permits the use of short high-pressure, nonflexing propellant feedlines.

The selection of gimbaling as the method of thrust vector control was made after an extensive analysis of various methods applicable to liquid-propellant engines. The design consists of a gimbal bearing which provides the engine attach point to the vehicle, and from which the entire engine is hung.

Gimbaling of the engine is accomplished by hydraulic actuators with fixed attach points on the thrust chamber and in the vehicle. In addition to providing a lightweight, proved method of thrust vector control, this system also provides for the flexible lines confinement to the low-pressure engine suction lines.

The turbine exhaust gas disposal into the thrust chamber provided a light compact design and eliminated the need for an attachment of the turbine exhaust duct to the vehicle, in addition to the aforementioned advantages gained by using the gas to cool the skirt extension.

A single major achievement in the engine design was the use of the pressure-ladder sequence combined with fuel-actuated control valves. This simplified, hydraulic, self-sequencing system, represents a significant improvement in the development of engine systems. Pneumatic controls are eliminated and electrical control components are minimized, thereby eliminating more failure modes that would detract from the achievement of the targeted engine reliability.

Engine starting is achieved in the following manner: On start signal, the gas generator spark igniter is energized and this is followed by the electrical actuation of the four-way solenoid valve. Movement of this valve puts in motion the pressure-ladder or start sequencing system, whereby each resulting action is linked to the occurrence of a necessary prior event. The first of these actions is the opening of the oxidizer valves from a ground-supplied high-pressure fuel source. On oxidizer valve open, the ground fuel flows to and opens the gas generator valve. Gas generator combustion then commences with the propellants fed from taps in the high-pressure feedlines upstream of the valve. The gas generator temperature and pressure then build up, accelerating the turbopump. At a preset pressure level, an ignition source in the form of a hypergolic fluid is introduced into the thrust chamber, followed by the opening of fuel valves and the accomplishment of main-propellant ignition in the thrust chamber.

For engine shutdown, the four-way solenoid valve is actuated again and this, in effect, reverses the process achieved in the engine start sequence. This engine starting sequence has been used throughout the early stages of engine development, which now includes more than 400 tests. It has demonstrated a high degree of reliability and repeatability.

The Saturn C-5 vehicle uses a cluster of five F-1 engines. Accordingly, the engine design and operating sequence must consider the use in clusters. First, the engine must not only have a very sensitive and reliable ignition detection system to prevent delayed ignition of one engine from an adjacent engine in the cluster. Secondly, the thrust buildup curve must be reproducible within narrow limits, not only from firing to firing of each engine but also from engine to engine in order to prevent high overturning moments on the launch pad. Obviously, there is a like requirement on shutdown.

A third requirement is insulation of uncooled components of the engine from radiant heating and flame recirculation in the booster boattail region. In some areas, insulating blankets are used. In other areas, moulded ceramic covers are used. In both cases, they are removable. The electric control and instrumentation wiring are enclosed in sealed conduits capable of withstanding flame temperatures for a few minutes. These are a part of the over-all reliability concept for the engine.

The F-1 engine has been undergoing development testing since June 1961 (Fig. 14). Success was encountered in testing the first engine in mid-1961 and improvement continued in the subsequent eight engines tested. The first test at full thrust for the programmed duration of 150 seconds was made on 26 May 1962. A high reliability goal before delivery of flight engines will be met this year.

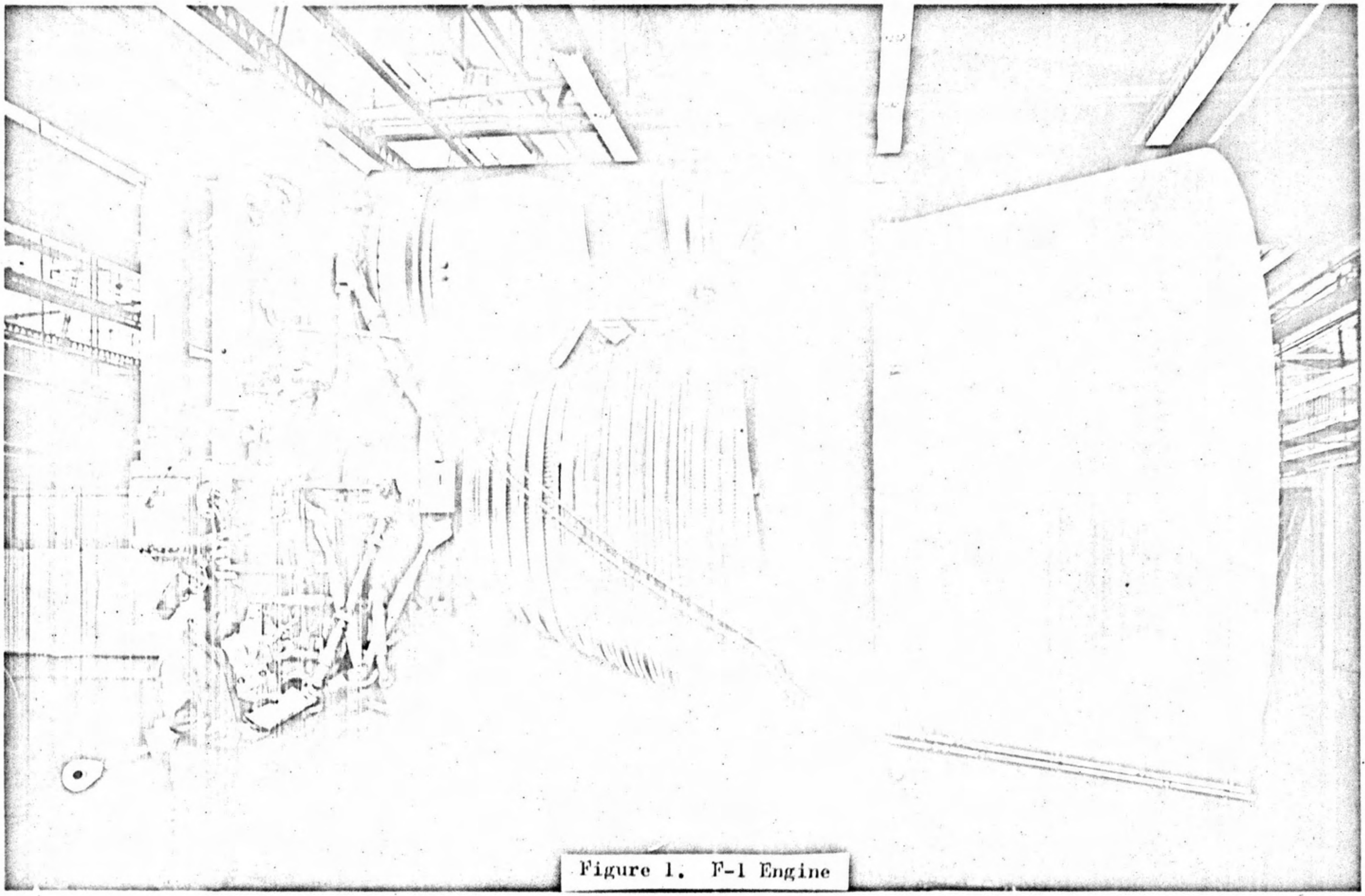

Figure 1. F-1 Engine

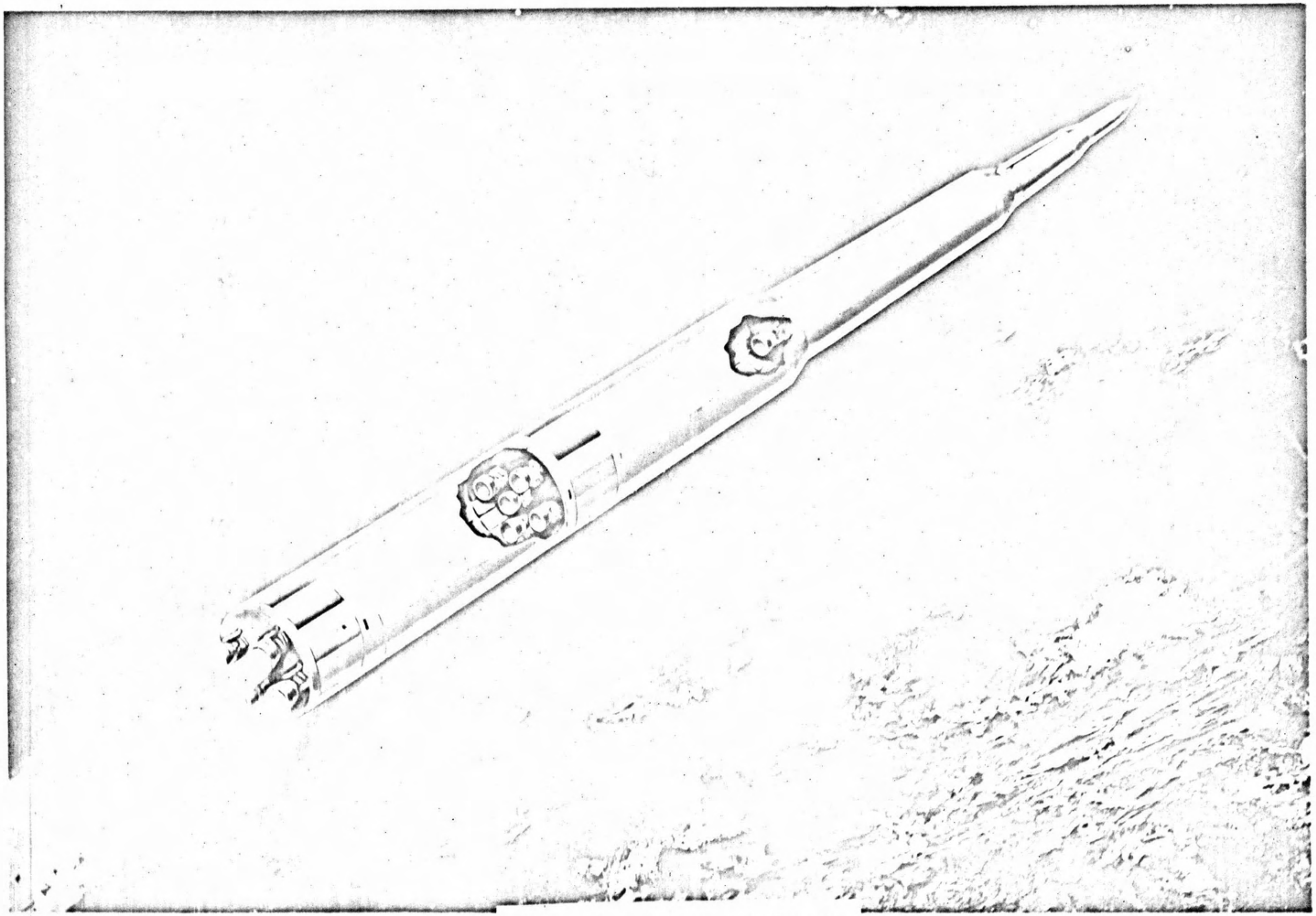

Figure 2. Saturn C-5 Vehicle

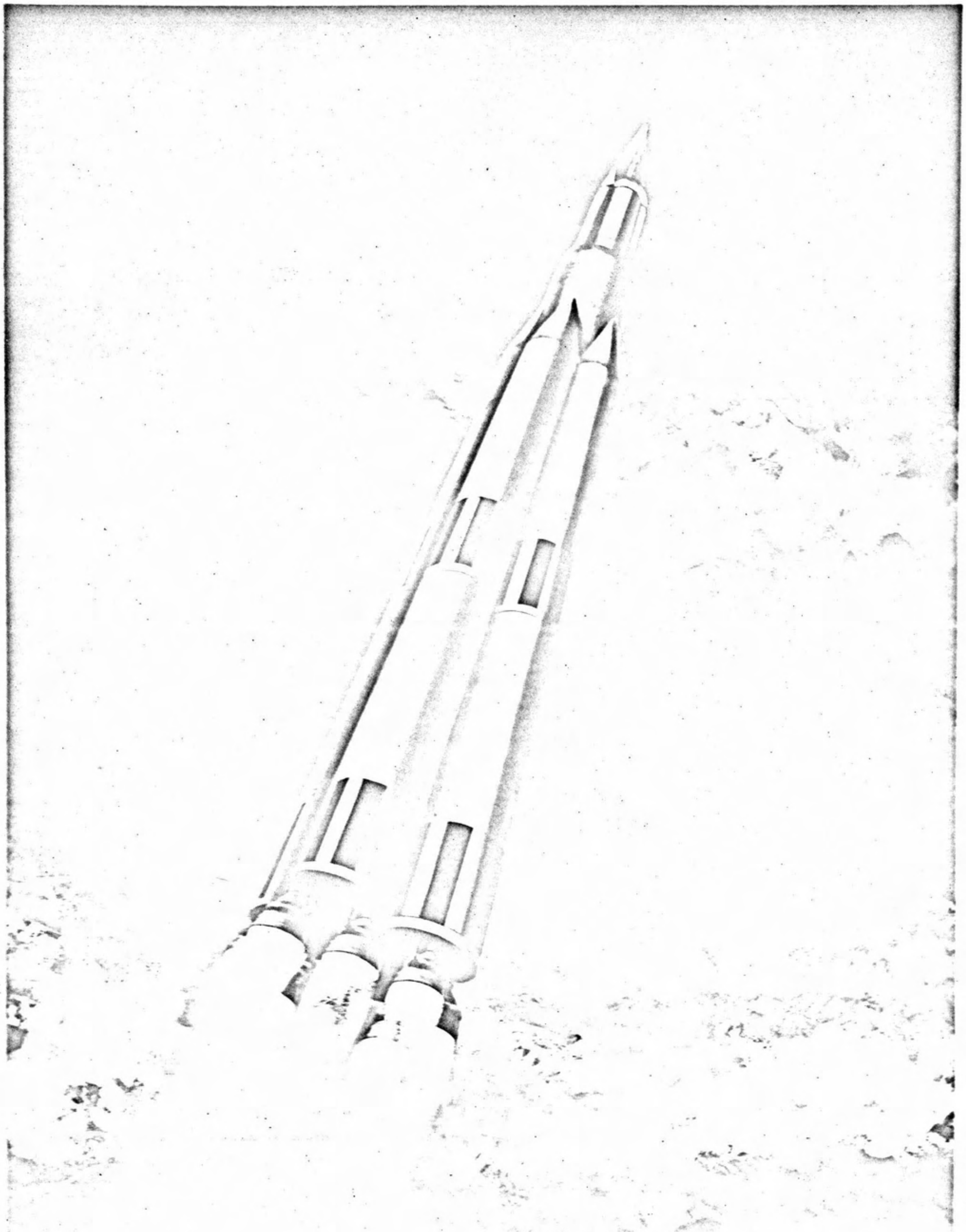

Figure 3. Nova Vehicle

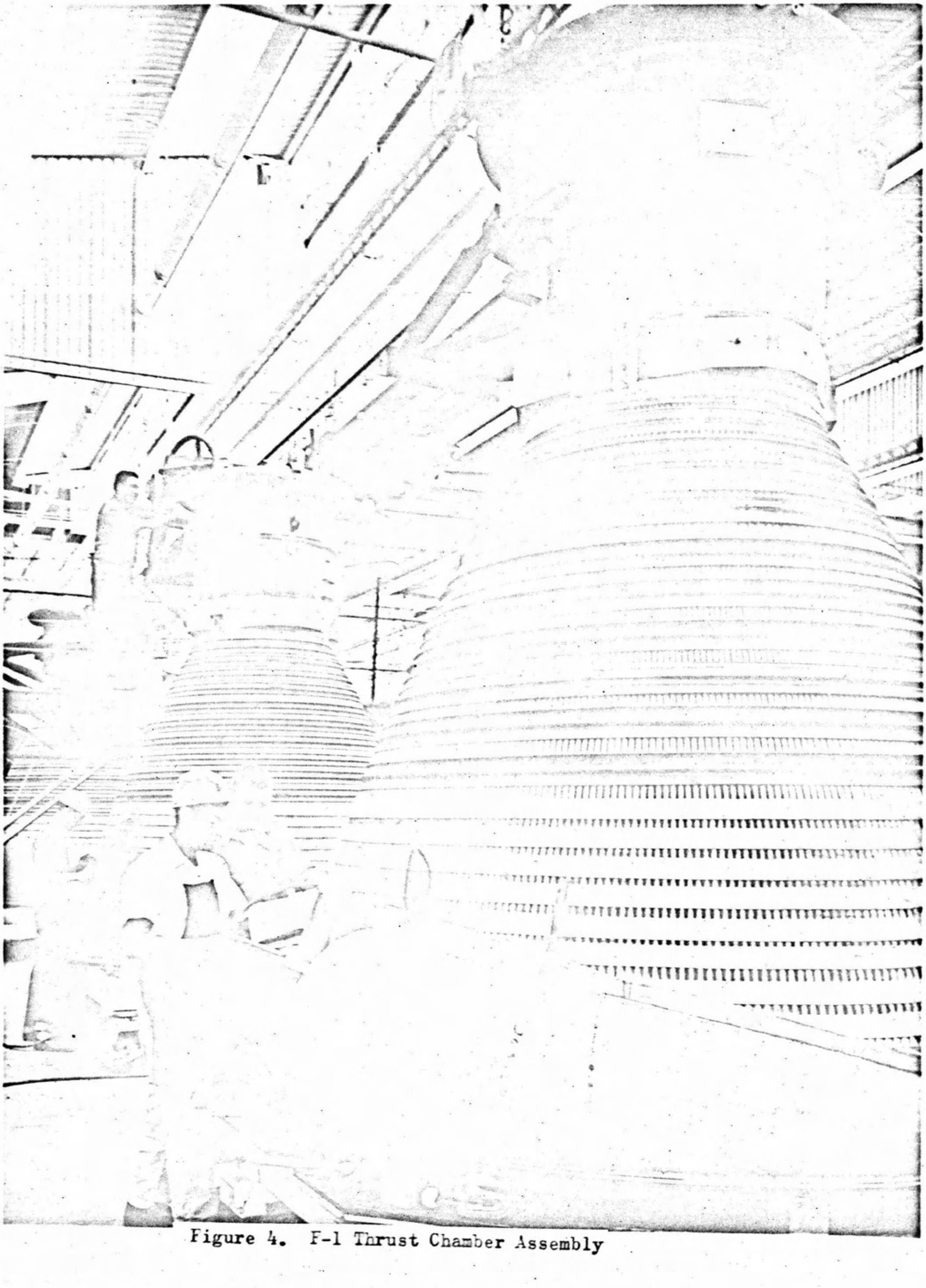

Figure 4. F-1 Thrust Chamber Assembly

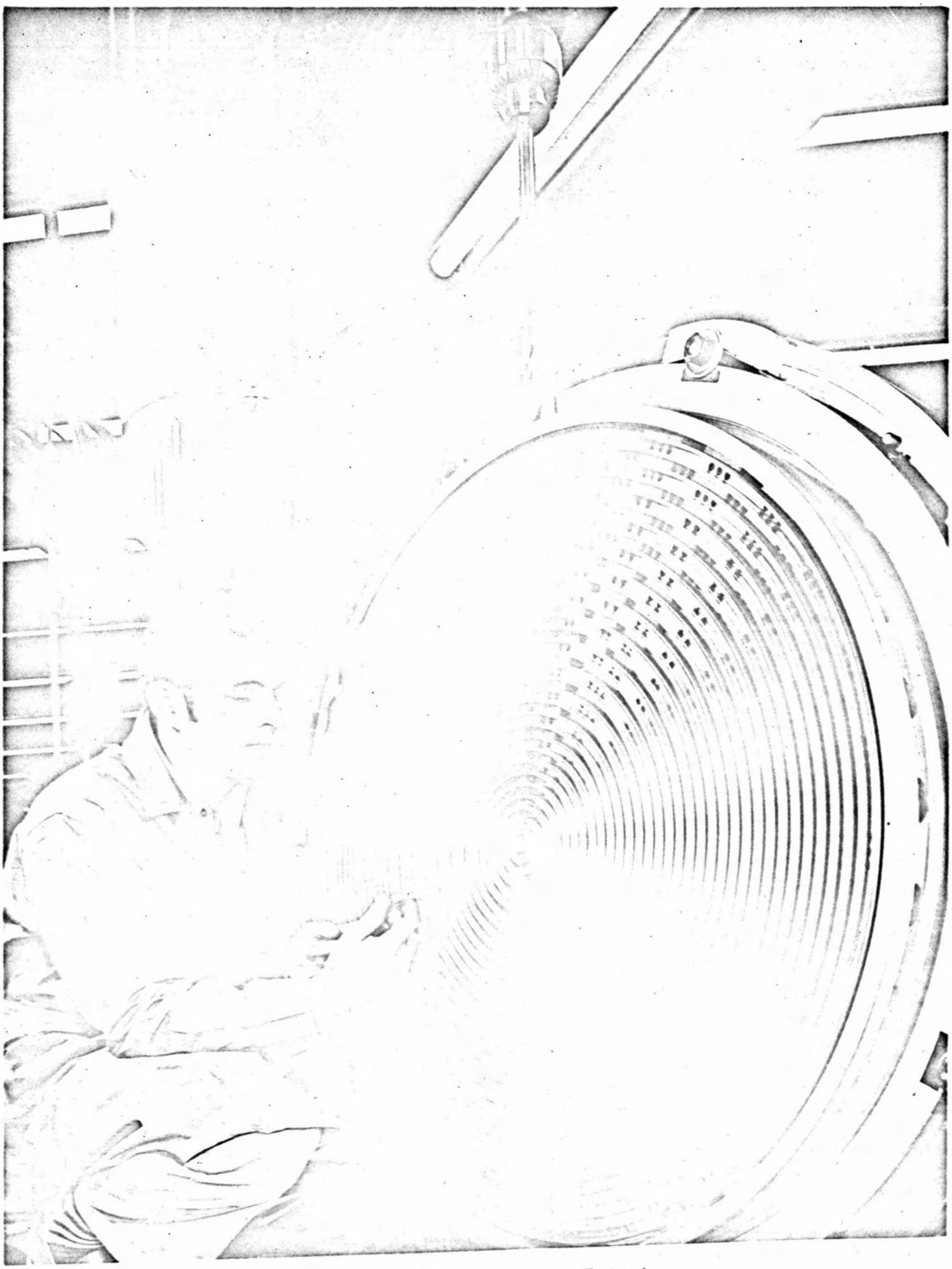

Figure 5. F-1 Injector

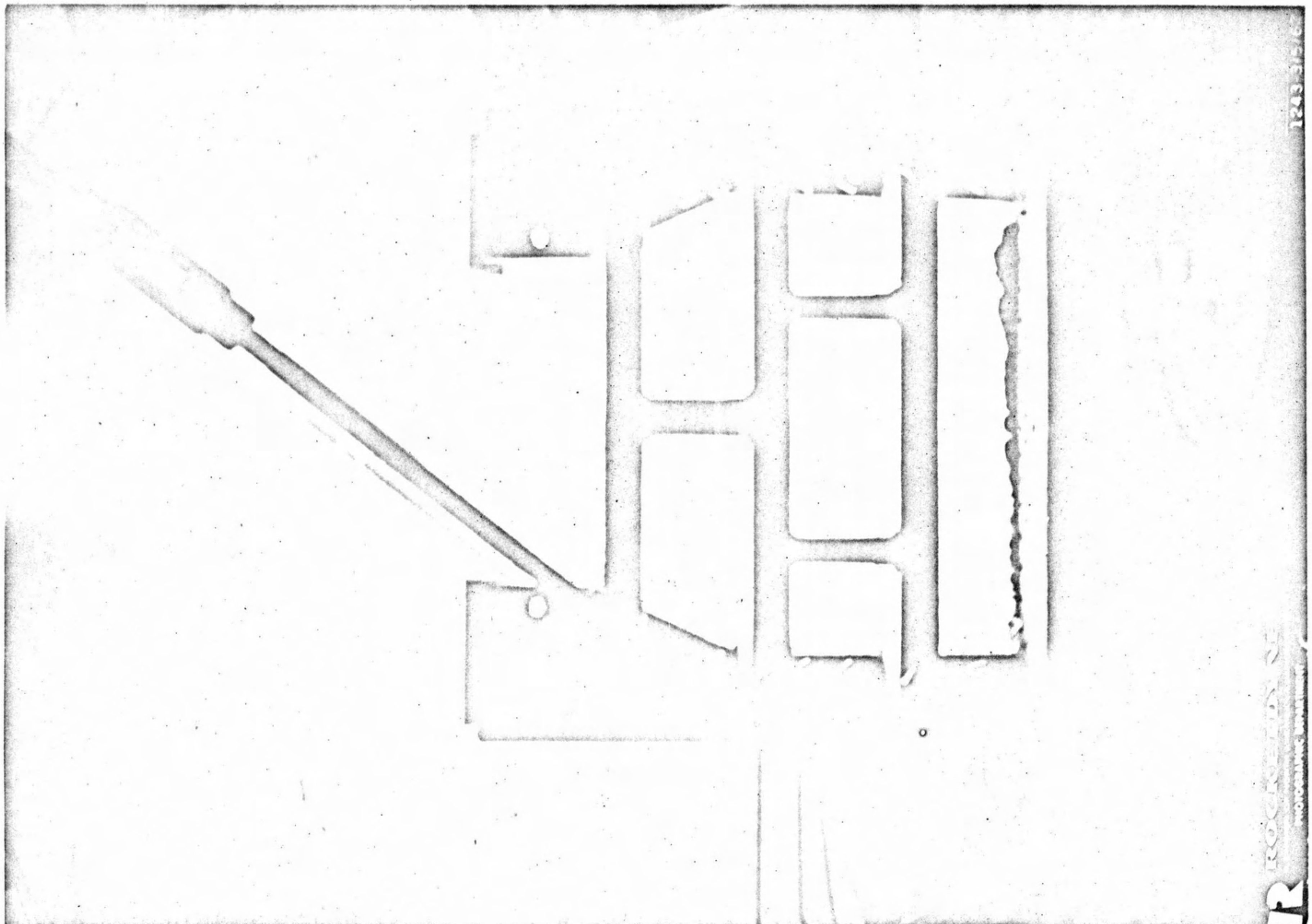

Figure 6. Two-Dimensional Thrust Chamber Firing

Figure 7. F-1 Thrust Chamber Fabrication

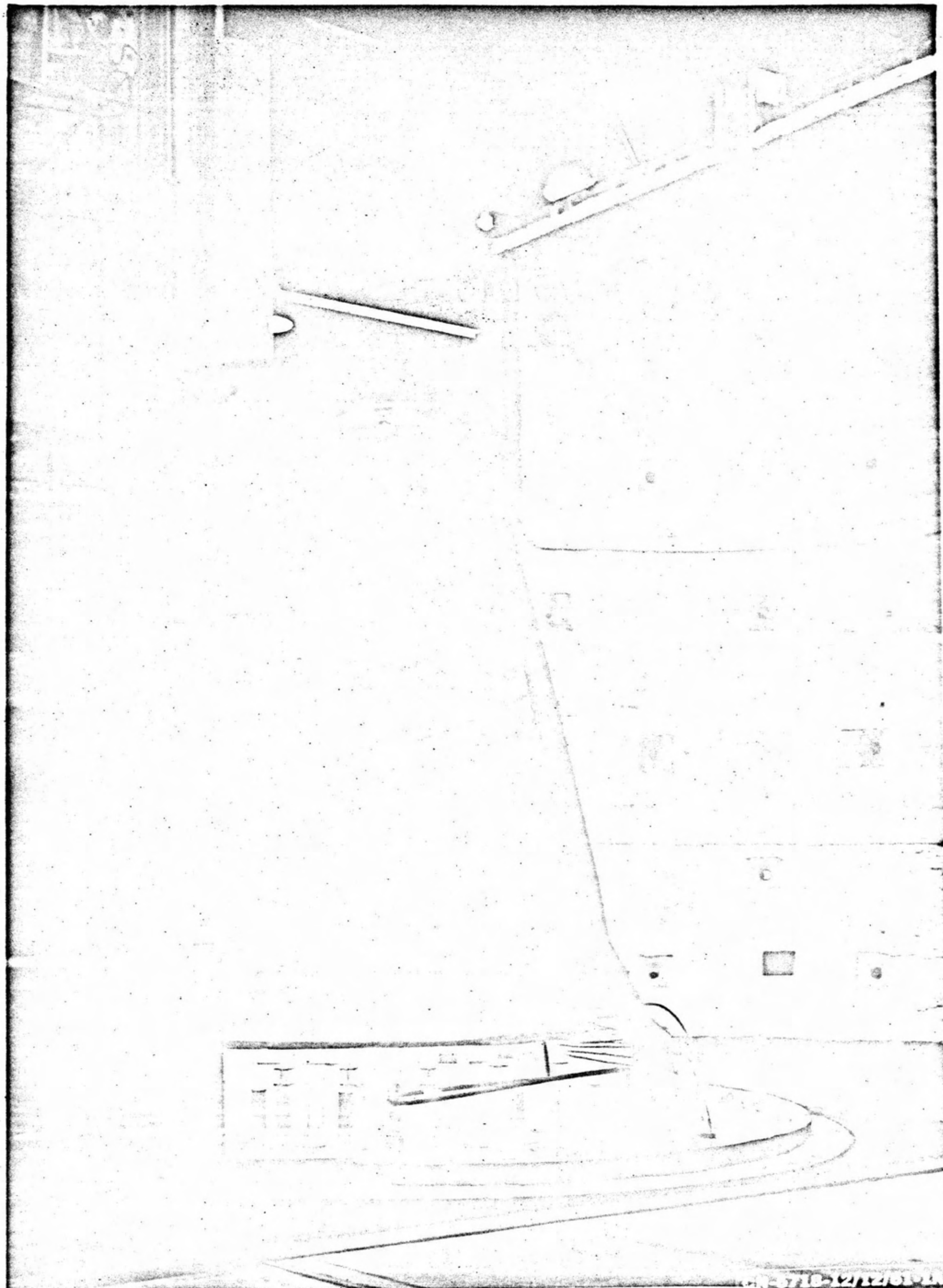

Figure 8. Thrust Chamber Furnace Brazing Operation

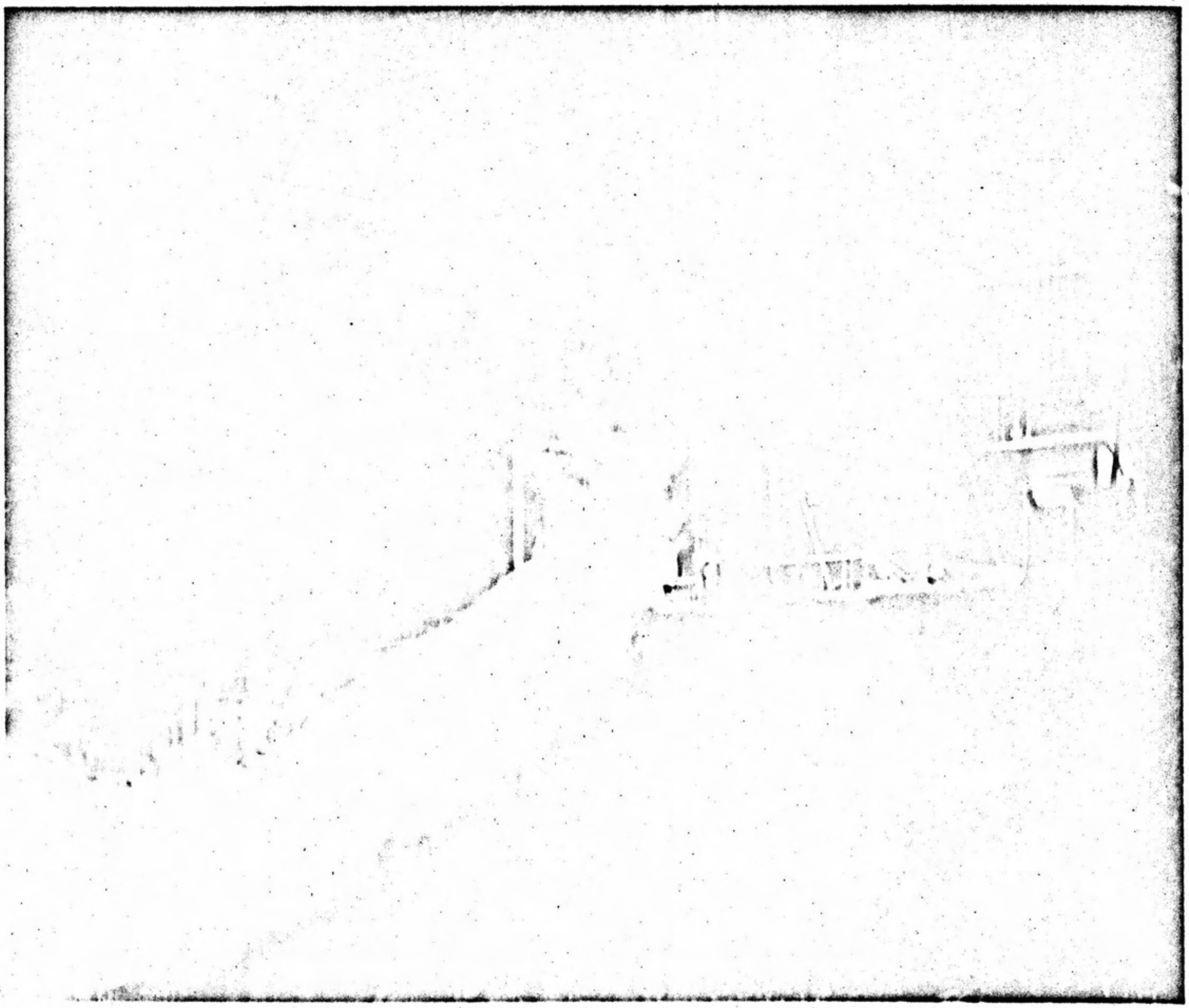

Figure 9. Thrust Chamber Stand 2A at EAFB

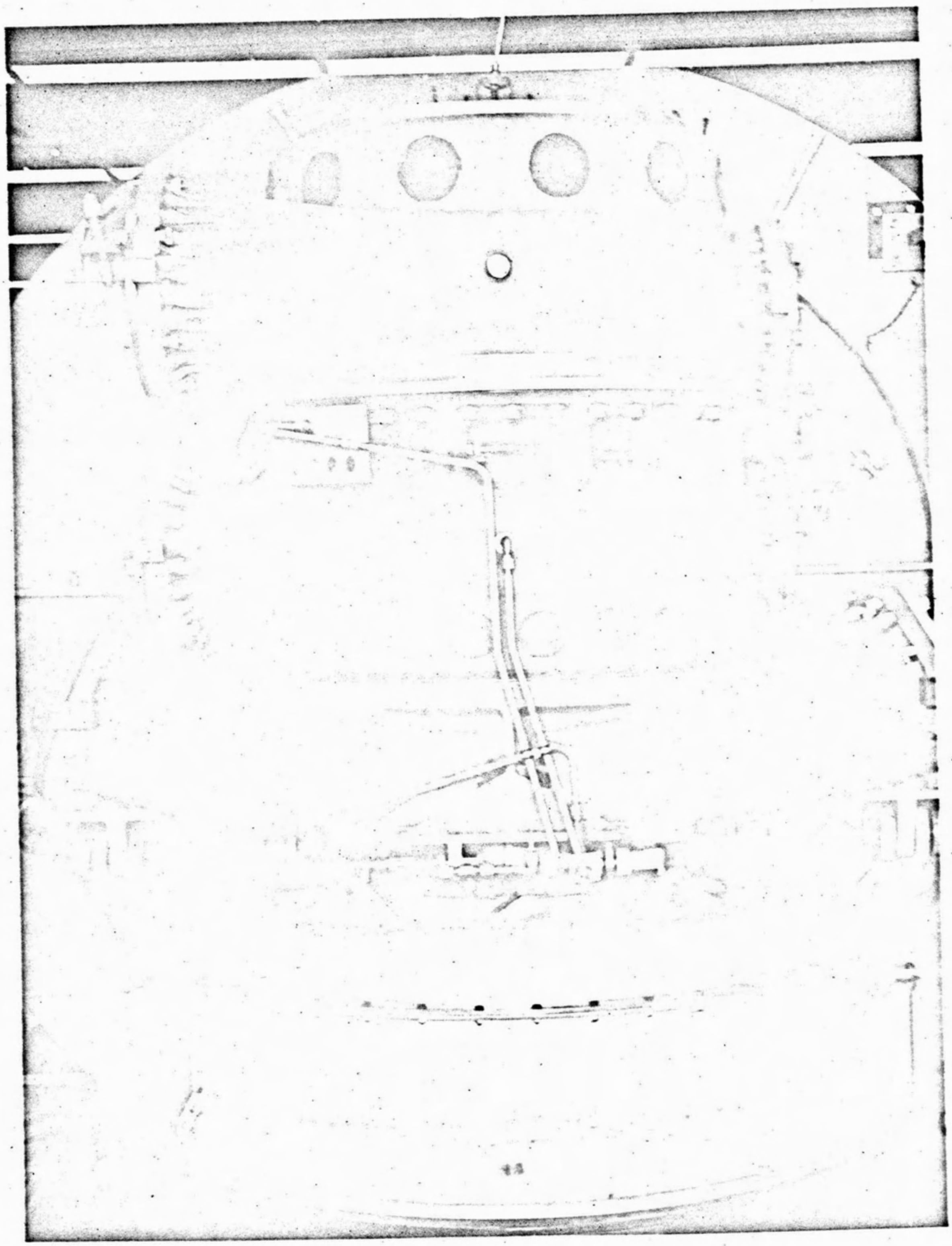

Figure 10. F-1 Turbopump

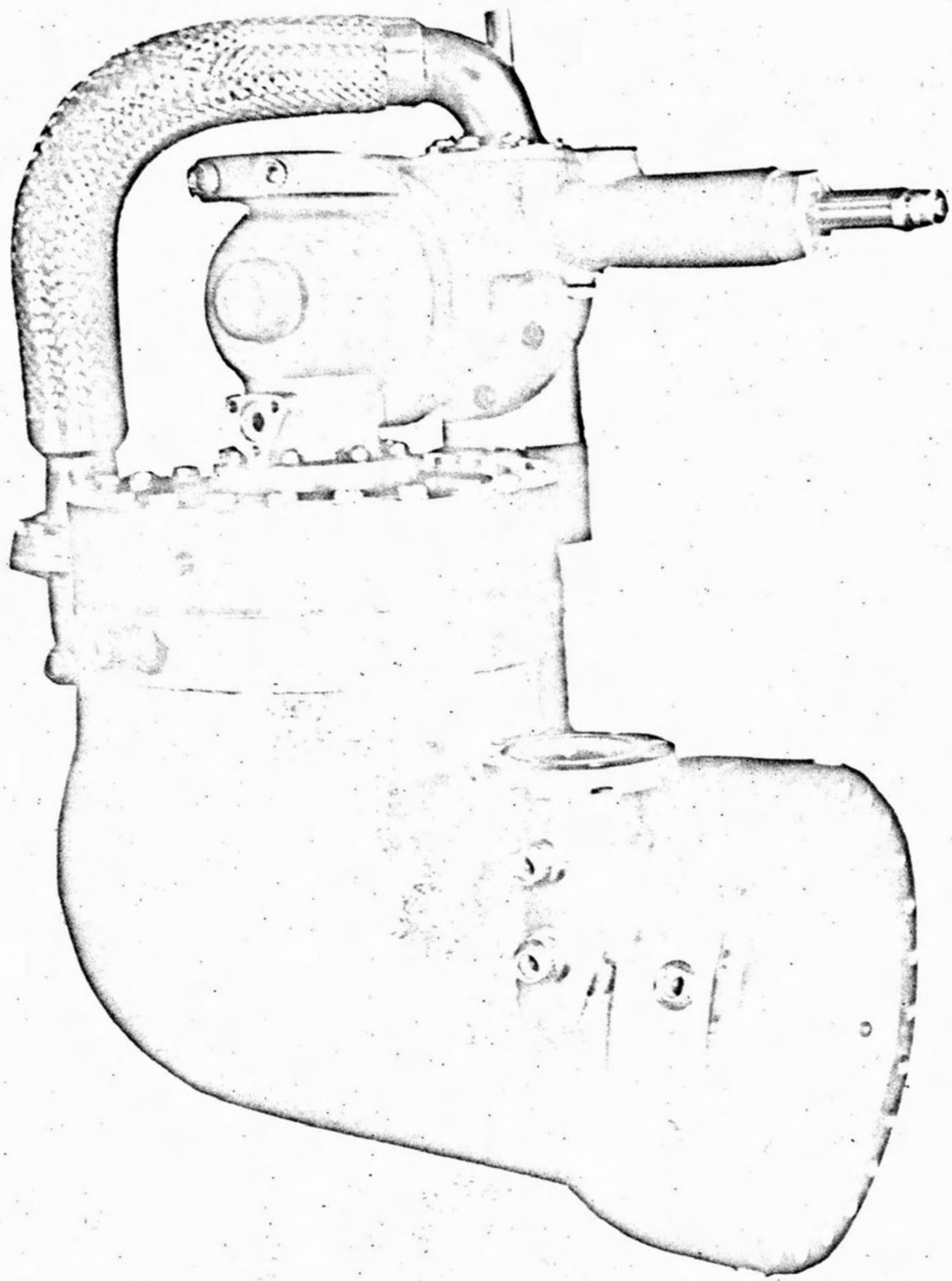

Figure 11. F-1 Gas Generator

Figure 12. F-1 Fuel Valve

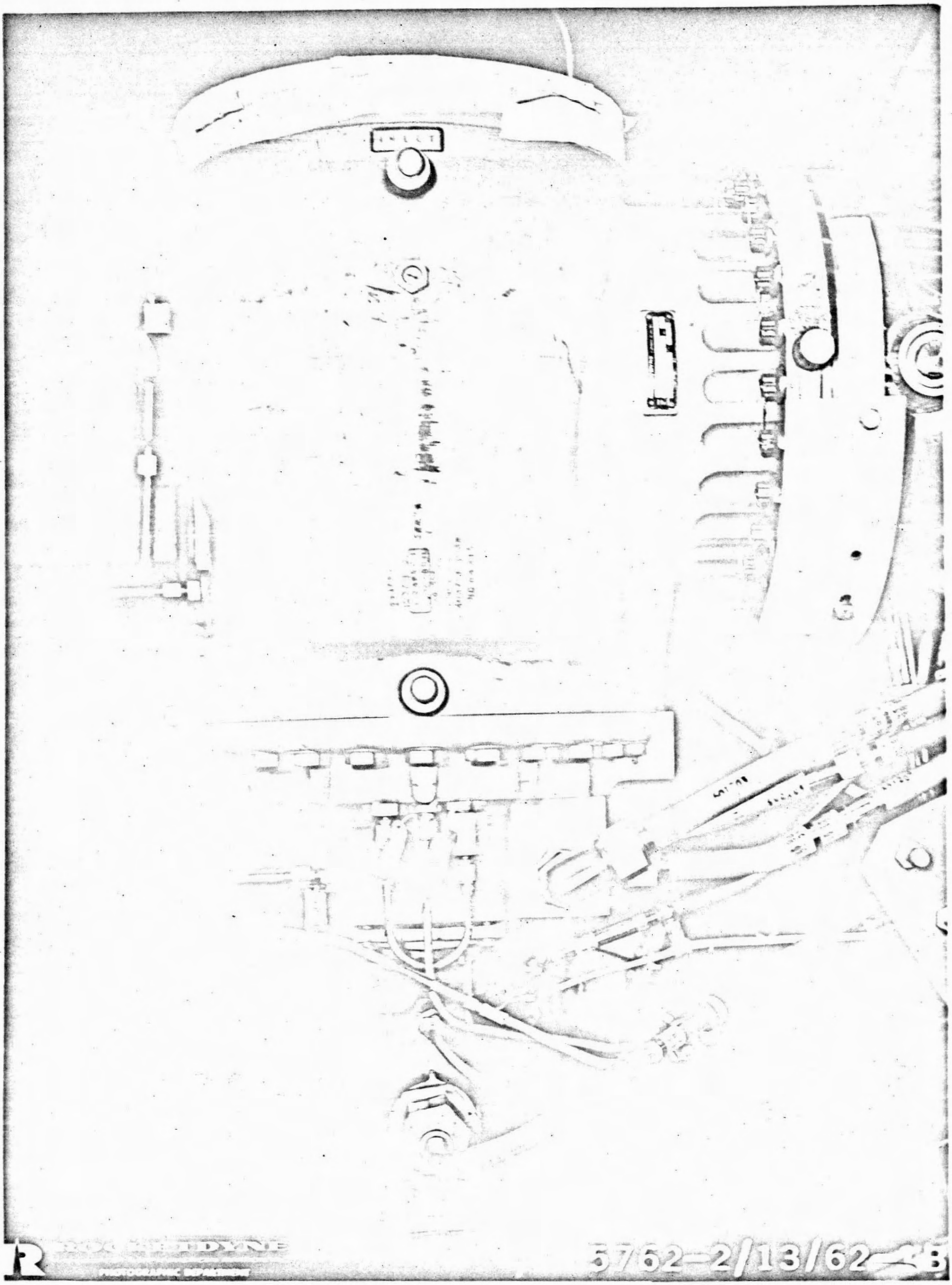

Figure 13. F-1 Liquid Oxygen Valve

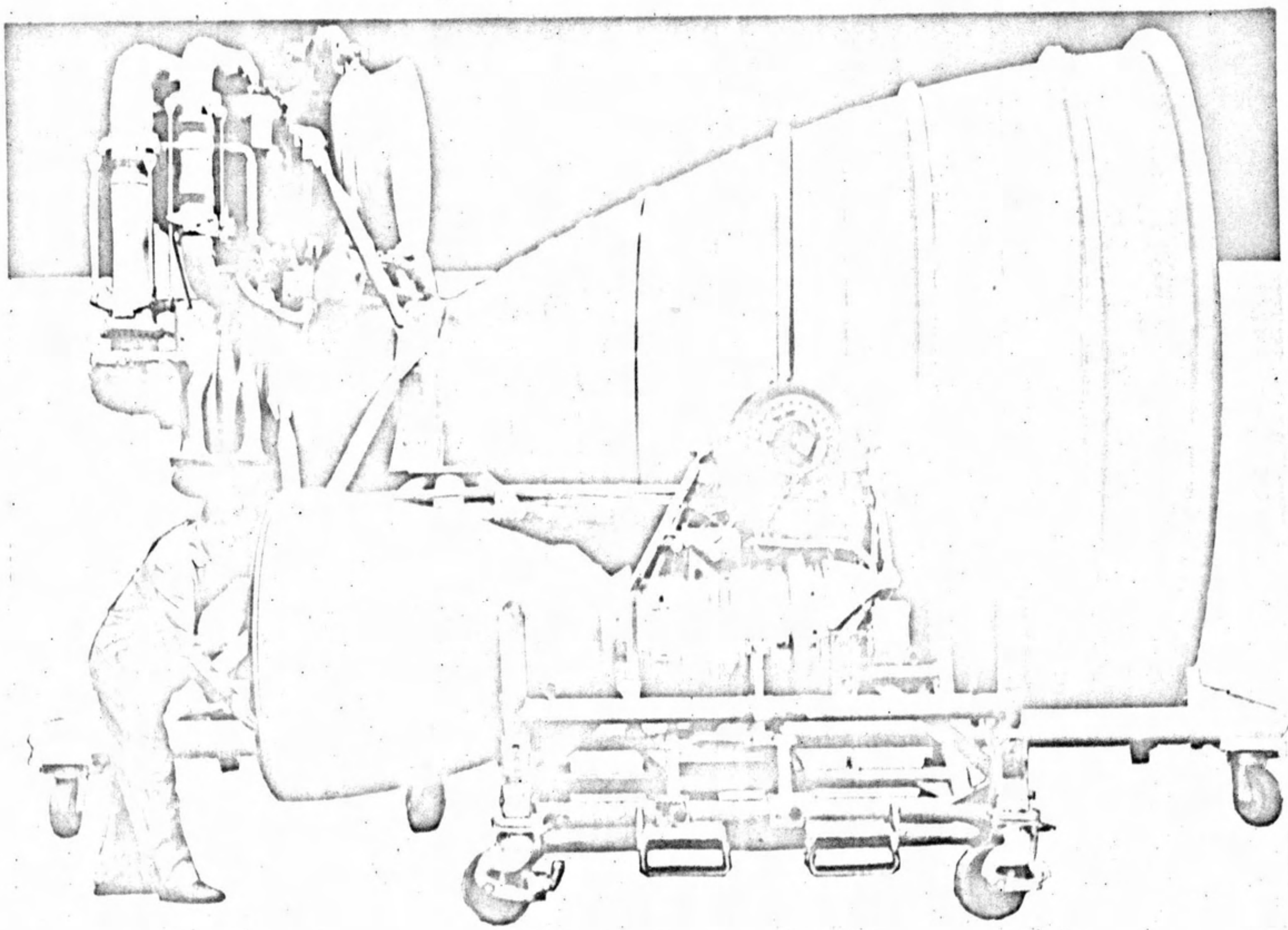

Figure 14 F-1 Mockup With H-1 Engine