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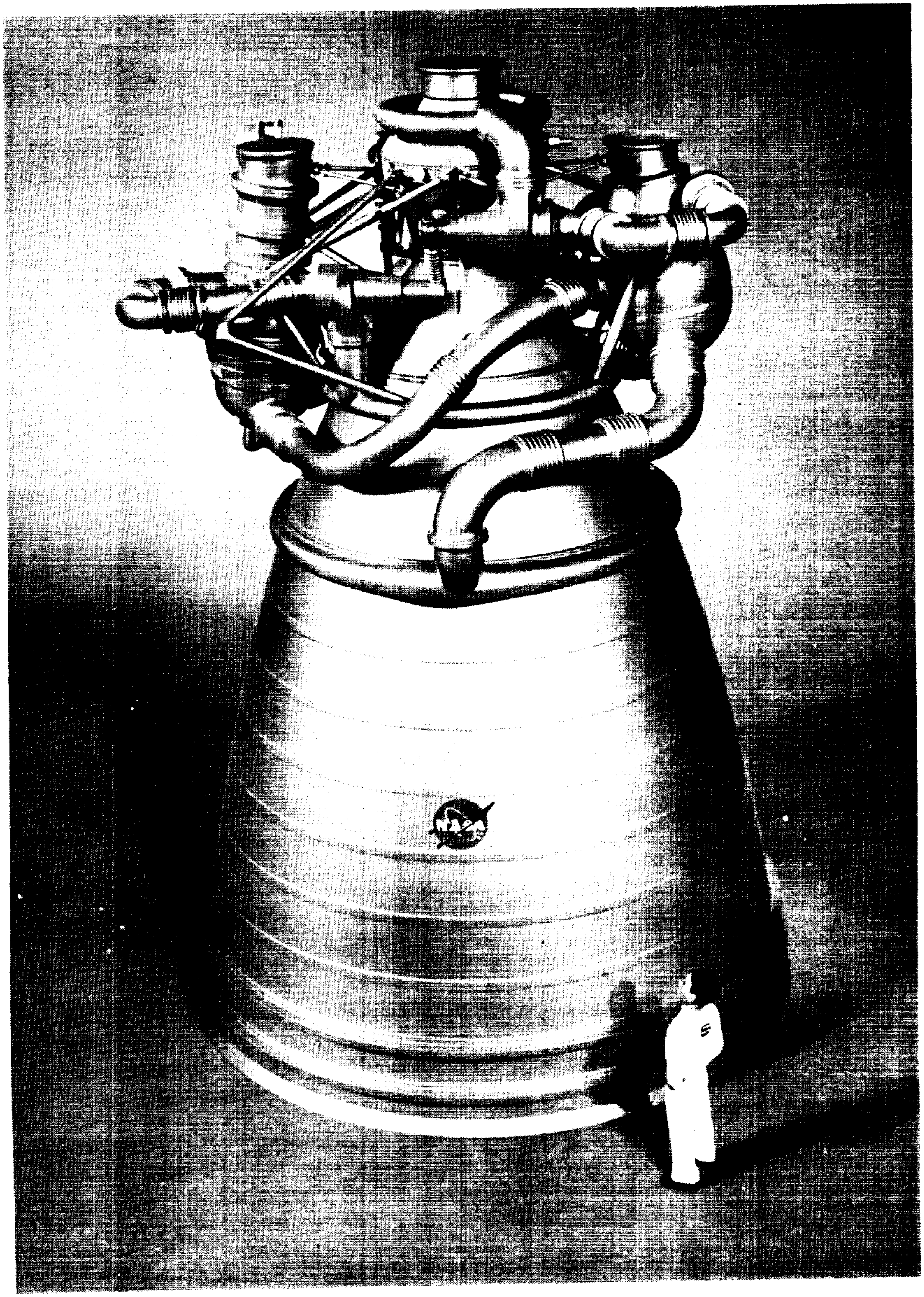

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DEVELOPMENT OF A 1,500,000-LB-THRUST (NOMINAL VACUUM) LIQUID HYDROGEN/LIQUID OXYGEN ENGINE

FINAL REPORT
Report No. 2555-M-1-F

30 April 1962 through 4 August 1966

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I. SUMMARY

The M-1 Rocket Engine Program, under Contract NAS 3-2555 to NASA/LeRC, was intended to produce a large, high-energy rocket engine incorporating a technology that was somewhat advanced beyond that found in the smaller liquid oxygen/liquid hydrogen rocket engines under development.

The M-1 Engine Program commenced on 30 April 1962 and original NASA cognizance was assigned to MSFC: however, this NASA cognizance was transferred to NASA/LeRC during October 1962. A notice of termination was issued in September 1965 and actual cancellation occurred on 24 August 1965. This failure to complete the M-1 Engine Program was largely the result of an indefinite postponement in plans to develop a post-Saturn vehicle.

The M-1 Engine Program was terminated in an orderly manner by NASA/LeRC utilizing a technique which derived a maximum of technological data within the then-available program funding. Although an engine system was not tested, major component tests were successfully accomplished after the necessary large cryogenic test facilities were activated.

The problems associated with high thrust that were significant in the F-1 Program and those problems that are unique to liquid oxygen/liquid hydrogen engines were significant in the RL-10 and J-2 Programs. These problems were combined in the M-1 Engine Program. Typical of these problems were the zero leakage large diameter cryogenic flange seals, combustion stability in the large diameter hydrogen/oxygen combustion chamber, the development of large-size flightweight hydrogen valves, and the development of a high expansion ratio thrust chamber delivering 1.5-million lb thrust.

A. ENGINE SYSTEM

1. Design

In designing the M-1 Engine, it appeared from the component test data that all of the required specifications could be met within the confines of a normal engine development program. The most difficult problem would be in achieving the 22,000 lb wet weight and it appeared that this would require considerable modification of the initial component designs.

The M-1 Engine configuration has opposed oxidizer and fuel turbopumps, which are fed by two 19-in. suction lines that are located 180-degrees apart. The turbopumps are supported by tubular members having primary attachment points on the thrust chamber injector flange and the thrust chamber fuel distribution torus. Although these support locations provide a lighter weight engine, they compromise the thrust chamber design because high unit loads are imposed at these points. However, it was expected that such problems could have been satisfactorily overcome.

Both the cryogenic and hot gas lines utilize three, internally-restrained bellows in each line to obtain line flexibility. However, studies were made which indicated that properly routed solid lines could provide adequate flexibility. This would have resulted in a simpler and lighter engine system and solid lines would probably have been utilized if an engine system had been constructed.

Thrust vector control is achieved by gimbaling the entire engine. Flexibility is provided in the 19-in. engine suction lines. Two alternative thrust vector control methods were considered. These were secondary injection and gimbaling of the thrust chamber assembly.

Secondary injection was too heavy and complicated for the 7 1/2-degree engine gimbaling requirement. It is feasible for an engine of the M-1 type only when a very small requirement, 1-degree to 2-degrees, exists. Although thrust chamber assembly gimbaling, which is used on the Titan Engines, provides the lightest weight and the highest performance, there were problems that precluded its use for the M-1 Engine. There was concern regarding the disposal of the turbine exhaust gas as well as the reliability of large, internally-restrained couplings. If the M-1 Engine were to be utilized as a module feeding a large plug nozzle engine, a gimbaled or hinged thrust chamber would be attractive because the turbine exhaust could be effectively used for base-bleed.

A significant factor in the design of the M-1 Engine was the prediction of dynamic loads. A conservative extrapolation of Titan, F-1, and J-2 engine data was used. This resulted in significant dynamic loads; as high as the equivalent of 6 g's of static load in some cases. Predictions were based upon an estimated random vibration spectral density pattern and estimates of component natural frequencies. In general, expected combustion noise was too high in frequency to couple with the major structural resonances. Dynamic load prediction techniques were utilized in the design of the turbopump and thrust chamber facility structural interfaces. In all cases, the load predictions for the component test environment were conservative.

2. Flow

In the M-1 Engine, separate, direct-driven fuel and oxidizer turbopumps are utilized with the gas generator products passing through the fuel turbopump turbine and the oxidizer turbopump turbine in series. Separate turbopumps were selected because of the major differences in the optimum shaft speed of the two machines as well as the major development problems that were anticipated in producing the high horsepower cryogenic gearbox that would have been required if a common turbine and separately rated pumps were used. The series turbine arrangement was selected to improve cycle efficiency as well as to minimize the weight of the turbine. Propellant utilization control is achieved by by-passing variable amounts of gas around the fuel turbopump turbine. The turbine exhaust gas passes through a gas-cooled nozzle extension and is exhausted at the end of the 40:1 nozzle as a low mixture ratio stream surrounding the primary combustion chamber gas flow. This system was initially selected as a good method for discharging the turbine exhaust gas and providing a nozzle extension for altitude testing that did not couple with the propellants flowing into the injector. However, the wide variations in turbine exhaust temperature and pressure over the complete engine operating envelope as well as the large, lightweight, high-temperature structure involved posed some serious design problems.

Should M-1 Engine development be resumed, serious consideration would be given to the F-1 shingle approach or a regeneratively-cooled system utilizing hydrogen flowing in parallel with the 14:1 regeneratively-cooled combustion chamber.

3. Control System and Engine Transients

There were four primary objectives in designing the M-1 engine control system; to prevent fuel pump stall, to prevent the hydrogen in the injector from becoming excessively cool during the transient, to maintain the mixture ratio between pre-set limits, and to utilize only those fluids that were required for operation of the engine.

The elimination of pump stall and the cold hydrogen problem were both resolved by using a fuel turbopump by-pass valve that is closed by means of high fuel discharge pressure. Mixture ratio limits were maintained by using mechanically-linked thrust chamber valves, mechanically-linked gas generator valves, and by properly positioning the propellant utilization valve during the start transient. These valves are operated by engine fluids; the thrust chamber valves are actuated by the main propellants, the gas generator valves are actuated by hydrogen supplied from the pump discharge, and the helium start valve is actuated by helium from the start tank.

Computer-predicted start transients were developed at both NASA/LeRC and at Aerojet-General. J-2 and Titan data were used to calibrate the computer models. Start is initiated by opening the helium start valve, which admits high pressure helium to the pump turbines. When the sum of the oxidizer and fuel discharge pressures reach 438 psi, the mechanically-linked thrust chamber valves are 18% open. These thrust chamber valves are basically mechanically-linked check valves with a double spring system to provide a step-position operation. The gas generator valves start to open when a pilot valve senses that the fuel discharge pressure exceeds the gas generator pressure by 50 psi to 80 psi. The turbopump system then bootstraps to a higher operating level. The thrust chamber valves resume their opening stroke when the sum of the propellant discharge pressures reaches 760 psi.

The prototype ignition system configuration had not been selected at the time this effort was terminated. However, a fluorine gas hypergolic ignition system for both the gas generator and the thrust chamber appeared to be the best choice. Thrust chamber testing was satisfactorily accomplished using a fluorine hypergolic system.

Shutdown is accomplished by shuttling the gas generator pilot valve causing the gas generator valve to close rapidly. The thrust chamber valves close as the result of the falling pump discharge pressure and the fuel pump by-pass valve opens as the fuel pump pressure decreases. It is expected that the transient achieved as a result of this operation would maintain safe mixture ratio limits and prevent damaging low frequency instability during shutdown.

B. INJECTOR

Maximum use was made of RL-10 and J-2 data in deriving the M-1 coaxial injector design. Such factors as velocity ratio and the selection of the element recess were based upon the J-2 data. Elements were evaluated by means of tests with single elements wherein their freedom from erosion was ascertained. Also, multi-element subscale testing was accomplished at NASA/LeRC, wherein there was some evaluation of element interaction and the adequacy of injector face cooling and baffle cooling was established. Injectors with 3248 elements and 3 1/2-in. baffles were successfully tested at Aerojet-General. Injector performance in full chamber pressure tests was approximately 2 sec above the PFRT requirement of 429 sec (extrapolated to 40:1 altitude operation). No instabilities were noted with start transient fuel temperatures as low as -398°F or with steady-state temperatures in excess of -360°F. The margin between -360°F and the design point temperature of -318°F indicates that the injector possesses adequate stability margin. No injector face erosion occurred during tests under stable operating conditions. There was minor erosion of the copper baffles but this did not progress from test to test.

An ablative chamber was used as the injector test device. The chamber was expected to last for at least four runs with a steady-stage duration of 4 sec. The first chamber survived nine full-thrust runs with a cumulative steady-state of 35.0 sec. This chamber also indicated a streaking characteristic of injector S/N 020. The chamber finally failed by ejection of the ablative section downstream of the throat. At that time, maximum throat char and erosion depth was 0.93-in.

A significant problem encountered during the tests was leakage at the large diameter flanges. Two of the large diameter joints were equipped with double Conoseals and a Flexatallic gasket was used at a third joint. This leakage was probably caused by a combination of pressure loading and a thermal cycle that compresses the seal causing it to yield which is followed by relief from the compression loading. A large diameter RACO hydrogen seal also failed as a result of the Teflon coating shattering. This appears to be a basic problem with the non-metallics when they are subjected to considerable flexing at liquid hydrogen temperatures. This leakage did not cause an excessively hazardous condition nor did it prevent the required data from being obtained.

Therefore, only such "quick fixes" as purging between the two Conoseals and re-torquing the bolts were attempted to stop the leakage. It appears that the best long-term solution for leakage at the cryogenic joints can be the incorporation of a seal weld.

C. COMBUSTION CHAMBER

Three regeneratively-cooled combustion chambers were in various stages of fabrication at the time that the program was terminated. The 14:1 area ratio regeneratively-cooled portion of the cooled combustion chamber is a one-and-one-half pass design equipped with 347 material stainless steel tubes. Hydrogen enters the tube bundle at 1650 psi and -400°F where it is heated to -318°F. The fuel enters at an area ratio of 8:1; it passes down to an area ratio of 14:1 and then returns the full length of the chamber.

A turbine-exhaust-cooled skirt provides for expansion from an area ratio of 14:1 to an area ratio of 40:1. The initial units were made up of a hand-welded and brazed tube bundle. It was welded to an area ratio of 6.7:1. It was planned to braze subsequent units in an electric vacuum furnace. Two tube bundles were hand-welded and brazed without encountering any significant problems.

The major problem uncovered during the design and fabrication of the combustion chamber was the difficulty in providing a close-fitting jacket to surround the tube bundle and which would carry both the axial and the radial loads.

Initially, a close-fitting Inconel 718 jacket was used to provide the axial and radial load carrying capability. This approach was selected over the use of a brazed jacket because it was desirable to use the jacket for more than one tube bundle. Also, problems in developing a satisfactorily brazed unit were anticipated and no suitable brazing furnace was available. It became apparent during component fabrication that tolerances would make jacket fit-up extremely difficult. Immediately before the cooled chamber effort was terminated, it was decided to substitute a wire-wrap design for carrying the radial load and to use an external loose-fitting jacket for the axial path. This appeared to be a more straightforward approach from the fabrication aspect. In this approach, the tube bundle will yield during the tube thermal childown; however, other chambers made from ductile material have satisfactorily accepted similar stresses.

It is expected that if the program had continued, fabrication could have been completed and successful operation obtained with the available hardware and design concept.

D. GAS GENERATOR ASSEMBLY

Three basic gas generator injector types were investigated: multi-orifice, large-thrust-per-element, and coaxial.

The large-thrust-per-element injector was unsatisfactory because of severe injector and chamber erosion. The multi-orifice gas generator injector was tested and results indicated that a satisfactory unit could be developed. Both the multi-orifice and the coaxial injectors presented a high-frequency instability problem. An acoustic liner was evaluated as an instability suppression device. However, the system remained unstable with the frequency of oscillation increasing from 4400 cps to 7200 cps. Minor liner erosion was also experienced. Although the liner could probably have been developed into a satisfactory instability suppression device, the use of baffles successfully eliminated the problem and no further effort was expended in developing the liner.

The gas generator injector selected was a coaxial type with 65 elements and incorporated a five-bladed baffle to suppress high-frequency instability. This injector pattern was used in nine oxidizer and fuel turbopump tests. It was considered satisfactory for initial engine operation.

E. OXIDIZER TURBOPUMP

The M-1 oxidizer turbopump consists of a single-stage radial flow pump that is directly-driven by a two-stage axial impulse turbine through a propellant-cooled power transmission equipped with rolling element radial and thrust bearings. The unshrouded pump impeller is fully-machined from an aluminum forging. It has an axial flow inducer, 35-degree swept-back vanes at the discharge, and radial backvanes for thrust balancing. The initial test article was made up of a single-stage turbine and a cast stainless steel pump housing with a rolled-over volute and nine integral diffuser vanes. The unshrouded turbine manifold is a multi-piece fabrication of formed Inconel sheet with bolted-in nozzles. External structural support is provided to the turbine by means of struts connected to the pump backplate.

The turbopump development program was planned to yield operational information concerning the turbopump (i.e., pump hydraulic performance, thrust balance information, turbine aerodynamic performance, turbopump cooling rate, childown time, control sensitivity, malfunction behavior, and purging or drying requirements. It was also intended to develop the turbopump components to a confidence level that would permit the testing of an engine using these components at nominal thrust conditions for short durations. This would have permitted the determination of the interaction between the pumping system components themselves as well as with the valving, the gas generator, and the thrust chamber. No attempt was made to verify the affects of extended duration operation upon the mechanical integrity of the turbopump.

Two series of tests were conducted. Liquid nitrogen was used as the pumped fluid in both series. The first series consisted of ten tests. The primary objectives of this series, which was conducted during January and February 1965, were to demonstrate the mechanical integrity at two-thirds of design speed and to determine the over-all hydrodynamic as well as aerodynamic performance of the pump and the turbine. There were 14 tests in the second series, which was conducted during July, August, and September 1965. The primary objectives of the second series were to demonstrate the mechanical integrity at full-speed and to verify the turbine aerodynamic performance with hot-gas drive.

Significant results were obtained from both series. Mechanical integrity was demonstrated at both nominal and off-design conditions. The facility operated satisfactorily and bearing performance was excellent despite the very high, short duration loads. The performance and accuracy of the instrumentation and data reduction systems were satisfactory. Pump head rise was approximately 8% below the predicted value but is adequate for engine operation.

The over-all thrust balancing system used during the first test series was adequate for the speeds obtained but appeared to be marginal for higher speeds, especially at low suction pressures. This thrust balancing system was modified by back-vane trimming and proved to be adequate in the second test series.

The bearing coolant system performance was adequate and as predicted. Both the dynamic and the static seals performed satisfactorily.

Although the turbine is an impulse machine, it developed a significant (20,000 lb) axial thrust under the off-design conditions encountered with the gaseous nitrogen drive.

Pump efficiency (based upon subscale unit tests) was 3 1/2 percentage points below the design prediction (61.5% versus 65.0%) and turbine efficiency, also based upon subscale tests, met or exceeded design predictions.

F. FUEL TURBOPUMP

The M-1 liquid hydrogen turbopump is an eight-stage, axial flow pump preceded by a mixed flow, first-stage inducer and an axial flow, second-stage inducer (transition stage). The pump design requirement is 600 lb/sec flow rate with a 1800 psi pressure rise at 13,225 rpm. The initial test unit was driven by a single-stage impulse turbine capable of supplying 60,000 horsepower. The prototype system utilized a two-stage turbine. The entire rotating assembly is supported radially by propellant-cooled roller bearings. Rotor axial thrust is carried by a matched, triple set of propellant-cooled, precision-ground ball bearings. The turbopump is mounted on a thrust chamber simulator for testing by means of two sets of stabilizers as well as upper main struts and lower main struts that are identical to the proposed engine supports.

A series of twelve liquid hydrogen turbopump tests was conducted from 13 May 1965 through 22 December 1965. Individual test durations ranged from 9.3 sec to 57.8 sec and the accumulated duration was 330 sec. The entire test program was conducted using only one turbopump and essentially, only one build-up. The major objectives of the test series were to demonstrate the mechanical integrity of the turbopump and to determine the over-all hydrodynamic and aerodynamic performance of the pump and turbine at rotational speeds of up to 90% of design speed.

Liquid hydrogen was used as the pumped fluid in all of the tests. The single-stage impulse turbine was powered by gaseous nitrogen during the early tests. The combustion products of the M-1 hydrogen/oxygen gas generator were used in two of the later tests.

The performance of the facility, instrumentation, and data reduction was excellent as demonstrated by the successful completion of all test objectives in ten rather than the scheduled 18 tests. One test was terminated because of a minor control circuit wiring problem and another one was terminated because of an inadvertent gas generator pressure switch shutdown during a gaseous nitrogen drive test.

Demonstrated pump pressure rise was 4.5% below the predicted performance for this configuration. Even though the pressure rise was slightly low, the pump performance is satisfactory because the head rise exceeds engine requirements by approximately one percent. The decrease in demonstrated pressure performance over that predicted is attributed to the recirculating flow rates in the internal thrust balance and cooling circuits being higher than predicted. Stall margins in excess of the design objectives were demonstrated during the test series. The best estimate of over-all pump efficiency appeared to be two percentage points below the design prediction.

The single-stage impulse turbine met or exceeded design efficiency predictions as evidenced by the test series as well as scale turbine tests performed at NASA/LeRC.

G. VALVES

1. Thrust Chamber Valves

A propellant-actuated, right-angle gate valve was selected as the thrust chamber valve because of engine integration considerations. Although components were manufactured for this valve, the unit was not assembled. The mechanically-linked fuel and oxidizer valves are nearly identical. They are operated by increasing the pump discharge pressure of both propellants which unseats the valves and compresses the inner valve spring. This causes the valves to stroke to an 18% open position. The valves then dwell in this position until the sum of the pressures equals 760 psi, at which time the valves continue to open. They are in the full-open position when the sum of the pressures is 2200 psi.

The initial M-1 engine control system consisted of pneumatically-actuated thrust chamber valves that were actuated by helium from the start bottle. Both the fuel and oxidizer versions of this valve were constructed and used for the initial thrust chamber testing.

The pneumatically-actuated valves were used for the first injector test at Test Stand C-9. Irregular motion of the valve was detected during cold flow testing, which was the first test of this component under full flow and pressure. Venturi effects during the initial valve opening stroke resulted in the flow forces reversing their direction. Satisfactory operation during cold flow was achieved by increasing the pneumatic system pressure and utilizing a double-acting actuator.

During the initial thrust chamber firing, the liquid oxygen feed system failed. Major damage was incurred by the liquid oxygen thrust chamber valve as well as the stand. The failure could have been initiated at the valve. The hot firing environment differed from the previous cold flow tests primarily in the dynamic loading upon the valve. There was also the possibility of fuel contamination in the oxidizer system downstream of the valve. The direct failure of the valve can possibly be attributed to one or more of the following conditions:

Excessive flange loading during the start transient.

A downstream fire in the liquid oxygen manifold.

Contamination in "dead-ended" pockets within the valve.

Violent impact between the gate and the aluminum valve body.

Possible reversal of the valve motion during opening.

After this test, it was decided to use facility valves for the early phases of injector testing. The conclusions drawn from the results of this test were used as the basis for modifying the prototype valve design. These conclusions are as follows:

It is undesirable to pneumatically operate valves whose position versus time characteristic is critical. This is particularly true when the forces acting upon the valve are unknown or variable. Raising the pneumatic system pressures helps to minimize this problem.

Metal-to-metal contact in a liquid oxygen environment during normal or abnormal operation is to be avoided.

Pockets that cannot be thoroughly cleaned should be eliminated from liquid oxygen parts.

Flange loads must be known for both engine and component operation.

2. Gas Generator Valves

The M-1 gas generator valves were designed to be actuated by super-critical liquid hydrogen during engine operation. Pneumatic and subsequently, hydraulic operation of the valves were used for component testing. Both the fuel and oxidizer valves are contained in a single housing. They are actuated by a single actuator. Propellant lead or lag can be varied by adjusting the relative position of the pintles at the time of assembly. Mixture ratio control during the transient is accomplished by properly shaping the valve pintles. The engine valve operates when the pressure differences between the fuel pump discharge pressure and the gas generator pressure exceeds 50 psi to 80 psi. At this pressure level, a pilot valve shuttles which initiates the flow of super-critical hydrogen to the valve actuator. The valve opening rate is then controlled by the force balance on the actuation system as well as by a flow restriction in the actuator supply line.

The gas generator valves were used for gas generator component testing. Both helium and hydraulic fluid were used as the actuating media and electrically-operated pilot valves were utilized. When it was found that actuation valve position versus time was not reproducible with pneumatic actuation, a hydraulic system was substituted. Heaters were required to prevent the hydraulic fluid from freezing.

Component testing of the gas generator valves demonstrated that they were basically sound and suitable for engine operation.

3. Other Valves

A helium start valve was also subjected to extensive component testing. Motion in this valve is initiated by an electrically-operated pilot valve which admits high-pressure helium to the cavity of the opening piston. The valve then opens rapidly. A check valve is utilized downstream of the helium start valve to prevent backflow of the hot gas to the start valve and the start system. Component testing of this valve indicated that it was satisfactory for engine operation.

An electrically-actuated propellant utilization valve was also under development. Because moderate times were allowable for the fuel stroke, a reasonably sized electrical actuator could be selected. However, the adequacy of this valve for engine operation was not demonstrated because component testing with this valve was extremely limited.

II. RESULTS AND CONCLUSIONS

The testing of major M-1 Engine components firmly established the technology for conventional hydrogen/oxygen rocket engines in the 1,500,000 lb vacuum thrust class. Of particular significance was the demonstration of a high performance injector that was both stable and erosion-free.

An engine based upon the proven M-1 design concepts could be developed at a reasonable cost and on a firm time schedule if the requirement arises.

The following results and conclusions encompass the broad aspect of the M-1 Engine Development Program as opposed to the specifics pertinent to the detailed aspects of the effort.

A. RESULTS

1. The feasibility of all major M-1 Engine components, except for the cooled chamber and the gas-cooled skirt, was demonstrated.

2. Performance data were obtained and the mechanical integrity was established for the injector, the fuel turbopump, the oxidizer turbopump, and the gas generator assembly. Also, it was established that these components are satisfactory for use in a demonstration engine.

3. The thrust chamber injector demonstrated stable operation without incurring any damage at performance levels in excess of the PFRT requirement.

4. Major facilities for testing the fuel and oxidizer turbopumps as well as the thrust chamber assembly were constructed and successfully activated.

B. CONCLUSIONS

1. The analyses performed as well as the experimental results indicate the feasibility of the M-1 Engine concept.

2. Based upon the results from the M-1 Engine Program, orderly development of liquid oxygen/liquid hyrogen engines of the M-1 Class (1000 to 1500 psi chamber pressure and one to two million pounds of thrust) can be undertaken with a high confidence for success.

3. Facilities are available for large engine component development.

4. Facilities for large engine development are partially completed.

III. INTRODUCTION

This is the final report for the M-1 Engine Development Program and is submitted in partial fulfillment of Contract NAS 3-2555. It covers the complete program from inception on 30 April 1962 through completion on 4 August 1966. While it encompasses engine and component design, fabrication, and testing as well as supporting efforts (i.e., ground support equipment, facilities, special test equipment, and materials), it is intended to be a comprehensive chronological summary highlighting the history of appropriate areas rather than a detailed presentation of the specific technological development of any designs. This technological detail is to be found in the numerous individual reports issued throughout the course of the M-1 Engine Program. These reports are referenced in each appropriate instance and no attempt is made to repeat the information contained in them. Thus, this final report provides a broad summary of the program only. It is not intended to be a "profound" technical document, rather it is a master reference source for the M-1 technical documentation.

In keeping with the indicated purpose of this report, the presentation of material herein remains generally within the format used in the formal technical status reports submitted quarterly throughout the program. Further, the Table of Contents is presented in greater depth than ordinarily to permit its use as an index-type of presentation. The bibliography, which is to be found as the last section of this report, is presented in a manner that shows applicable reference material under the same general headings as those of the major sections in the body of the report. Where referenced material applies to more than one area of discussion, it is repeated in each applicable section for reader convenience. Footnotes are provided throughout the text and they follow standard textbook usage; their numbering is consecutive and bears no relationship with the numbers indicated in the bibliography.

IV. ENGINE ASSEMBLY

SUMMARY

The M-1 Engine configuration is shown on Figure No. 1. It is comprised of opposed oxidizer and fuel turbopumps which are fed by two 19-in. suction lines located 180-degrees apart. The turbopumps are supported by tubular members whose primary attachment points are on the thrust chamber injector flange and the thrust chamber fuel distribution torus. Utilization of these support locations provides for a lighter weight engine but they also compromise the thrust chamber design because high unit loads are imposed upon it. However, it appears that problems of this nature could have been satisfactorily resolved. Both cryogenic and hot gas lines utilize three, internally-restrained bellows in each line to obtain line flexibility.

The 10-in. bellows required for the pump discharge lines was manufactured. It was qualified by the supplier, the Fairchild Camera and Instrument Company, in accordance with an Aerojet-General specified test program. The most important elements of this highly-demanding test program were the 500 proof pressure cycles, the 3000 gimbal cycles, and 30 g's of vibration for one hour. Before this test program was successfully completed, there were four major design iterations to correct bellows failures that occurred during the testing. The 10-in. bellows qualification program resulted in a unit that represents a significant technological advancement.

The 11 1/2-in. bellows for the fuel turbine exhaust line was supplier-designed. It suffered repeated failures during its development testing. As a result, emphasis was shifted to the investigation of a semi-rigid line. Studies were made indicating that properly routed solid lines could provide adequate flexibility and would result in a simpler as well as a lighter weight engine system. A 9-in. non-flexible bellows line would have approximately the same pressure drop as the 11 1/2-in. bellows unit and at the same time, it would not overload the mating turbine flanges. If an engine had been constructed, solid lines probably would have been utilized.

The selection of the thrust vector control system was based upon vehicle payload studies as well as the complexity of engine development. Upon completion of these investigations, engine packaging trade-offs were studied wherein engine weight, gimbal angle, and natural frequency as a function of the detailed engine component criteria were interrelated. As a result of these analyses, it was decided to establish the actuator arm length at 90-in., the gimbal bearing radius at 15-in., the gimbal angle at 7 1/2-degrees, and to delete the engine natural frequency of 8 cps in lieu of a spring constant of 450,000 lb/in. Thrust vector control is achieved by gimbaling the entire engine and necessary flexibility is provided in the 19-in. engine suction lines. Two alternative thrust vector control methods were considered; secondary injection and thrust chamber assembly gimbaling.

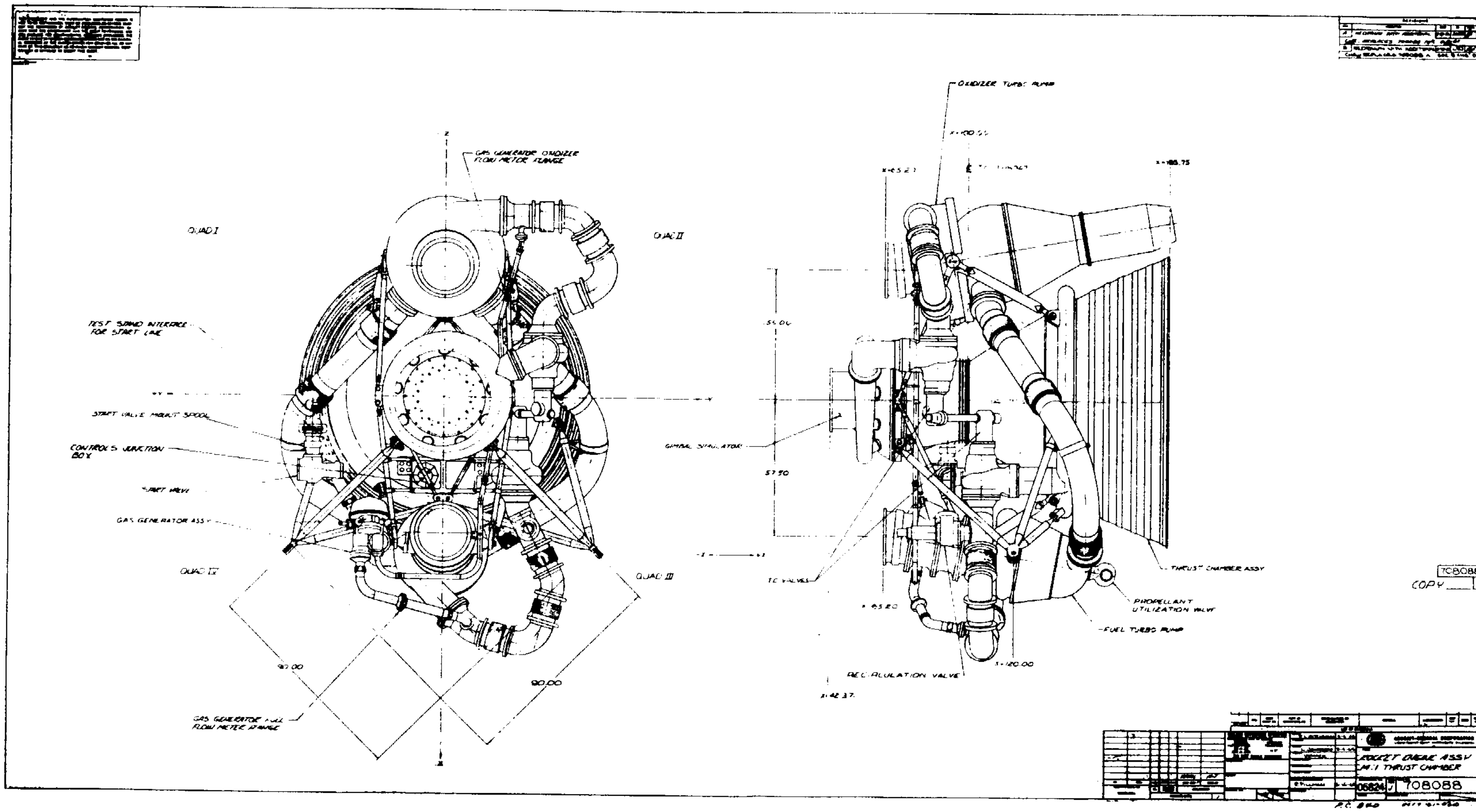

Figure 1. M-1 Engine Configuration

The secondary injection proved to be heavier and more complicated for the 7 1/2-degree engine gimbal requirement. Secondary injection can be considered for an engine of the M-1 type only in the event that the gimbal requirement does not exceed one or two degrees. Thrust chamber assembly gimbaling, which is the technique used for the Titan engines, provides the lightest weight, highest performance system. However, problems in disposing of the turbine exhaust gas plus a concern over the reliability of large, internally-restrained couplings prevented its adoption for the M-1 Engine. A gimbaled or hinged thrust chamber would be attractive if the M-1 were to be utilized as a module feeding a large plug nozzle engine because the turbine exhaust could be effectively utilized for base-bleed.

A major factor in the design of the M-1 Engine was the prediction of dynamic loads. A conservative extrapolation of Titan, F-1, and J-2 engine data was utilized. This resulted in significant dynamic loads; in some cases, equivalent to 6 g's of static load. The predictions were based upon an estimated random vibration spectral density pattern and estimates of component natural frequencies. In general, the expected combustion noise was too high in frequency to couple with the major structural resonances. Dynamic load prediction techniques were utilized in the design of the turbopump and thrust chamber facility structural interfaces. In all cases, load predictions for the component test environment were conservative.

The M-1 Engine flow diagram and control schematic is presented as Figure No. 2. Separate, direct-driven fuel and oxidizer turbopumps are used. The gas generator products pass through the fuel turbopump turbine and the oxidizer turbopump turbine in series. Separate turbopumps were selected because of the major differences in the optimum shaft speed of the two machines and major development problems were anticipated in producing the high horsepower cryogenic gearbox that would be required if a common turbine and separately rated pumps were used. The series turbine arrangement was selected to improve the cycle efficiency as well as to minimize the weight of the turbine. Propellant utilization control is achieved by by-passing various amounts of gas around the fuel turbopump turbine. The turbine exhaust gas passes through a gas-cooled nozzle extension and is exhausted at the end of the 40:1 nozzle as a low mixture ratio stream surrounding the primary combustion chamber gas flow. This system was initially selected as a good method for discharging the turbine exhaust gas as well as to provide a nozzle extension for altitude testing that did not couple with the propellants flowing into the injector. However, the wide variation in turbine exhaust temperature and pressure over the complete engine operating envelope and the large, lightweight, high-temperature structure involved posed serious design problems.

If development of the M-1 engine were to be resumed, serious consideration would be given to the use of the F-1 shingle approach or a regeneratively-cooled system utilizing hydrogen flow in parallel with the 14:1 regeneratively-cooled combustion chamber.

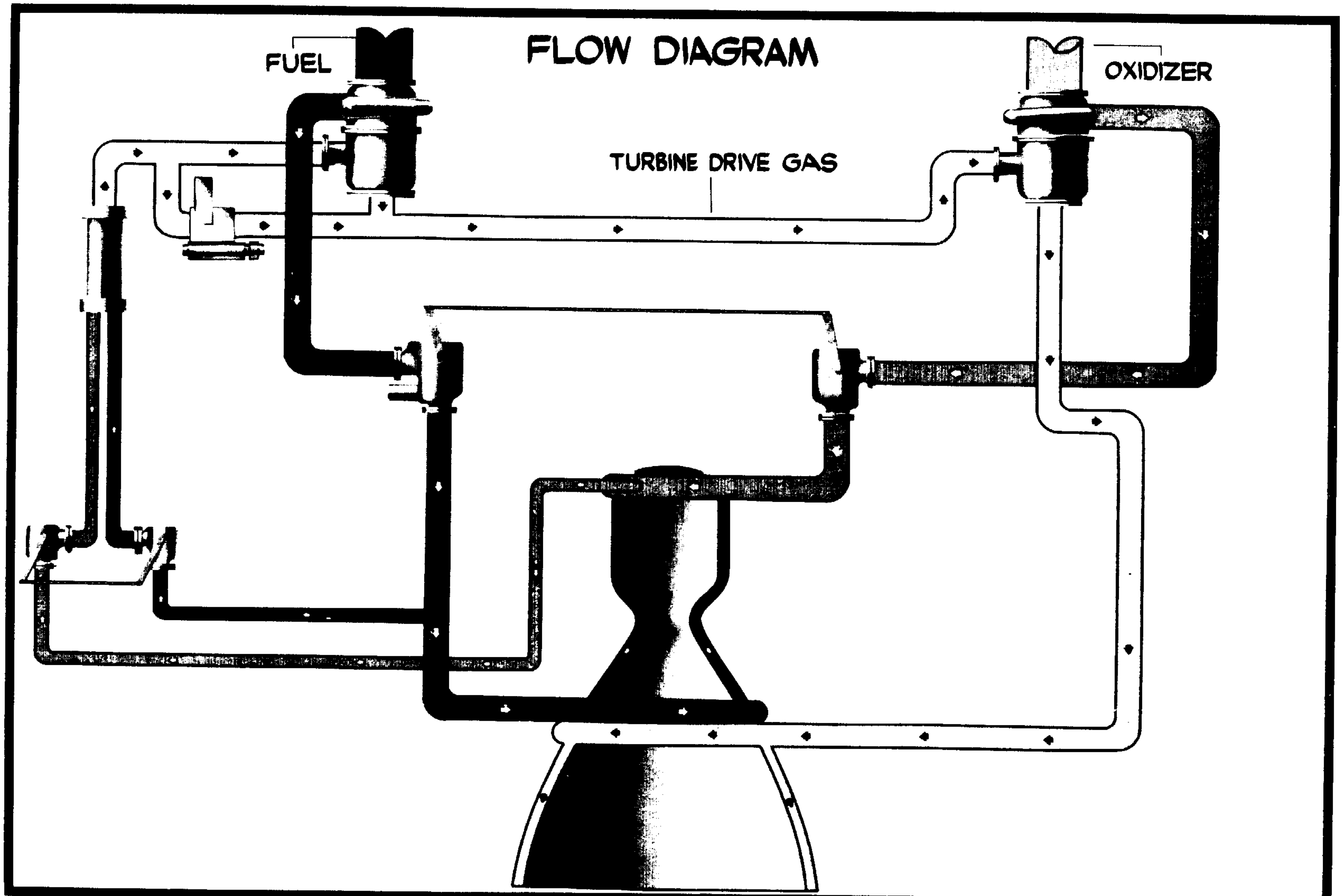

Figure 2. Flow Diagram and Control Schematic

Based upon the component test data, all of the specifications could be within the confines of a normal engine development program. The most difficult problem would be to achieve the 22,000 lb wet weight. Considerable modifications of the initial component designs would be required. In addition, preliminary analyses indicated that the gimbaling accelerations could be as high as 34 rad/sec.^{2} resulting in high engine structural loads and pressure transients on the pump inlet.

At the time of program termination, the detailed design and release of drawings of lines in support of the turbopump assembly test program as well as the initial engine assembly had been 90% completed. Development of these items was 50% completed. The design and development of turbopump support structures for the turbopump assembly test program had been completed. These turbopump support structures consist of Inconel 718 ball end fittings and 9310 steel struts. They adequately satisfied their functional requirements during the turbopump assembly testing. The design of similar structures for the engine was 40% completed.

The technology applied to define, design, and evolve an optimum M-1 engine configuration is detailed in a separate report.(1) Thrust-vector control design criteria are also described as well as the analyses used to relate engine packaging to the vehicle requirements of natural frequency, gimbal angle, and weight.

CHRONOLOGY

1962/2Q

The initial engine design is shown as Figure No. 3. This configuration had thrust-chamber-mounted turbopumps aligned at a 15-degree angle to the thrust chamber axis. The turbopumps were diametrically opposed and the thrust chamber valves were located on the same side of the chamber.

An alternative arrangement was prepared for NASA consideration, in which the turbopumps were mounted on a fixed engine frame to eliminate large pump suction line flex sections and to reduce gimbal requirements. However, it also required articulating pump discharge lines, added an engine frame, and necessitated the direction of the turbine gas toward the vehicle skin.

By the end of the second quarter of 1962, the original engine design had been changed so that the turbopump canted at 10-degrees, each pump discharge line had three axially-restrained bellows assemblies, the two hot-gas inlet flanges on the nozzle extension were oriented 18-degrees opposed to each other, and the thrust chamber valves were relocated to simplify the actuator linkage.

System analyses to define the start system steady-state conditions, means for balancing thrust, and provisions for obtaining propellant utilization were completed.

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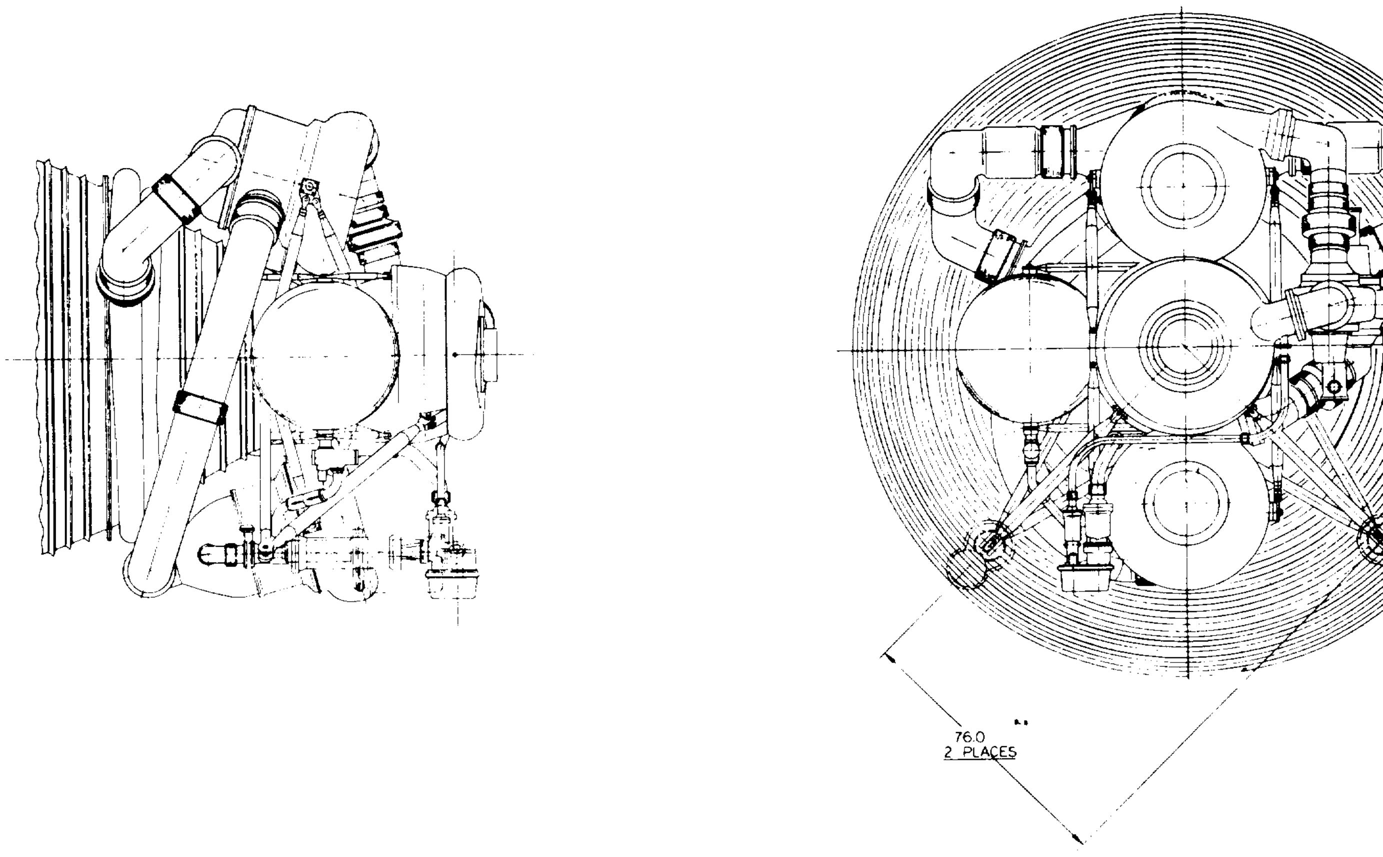
(1)Unmack, K. E., Configuration Design of the 1,500,000 Lb Thrust M-1 Rocket Engine, Aerojet-General Report No. 8800-71, 29 April 1966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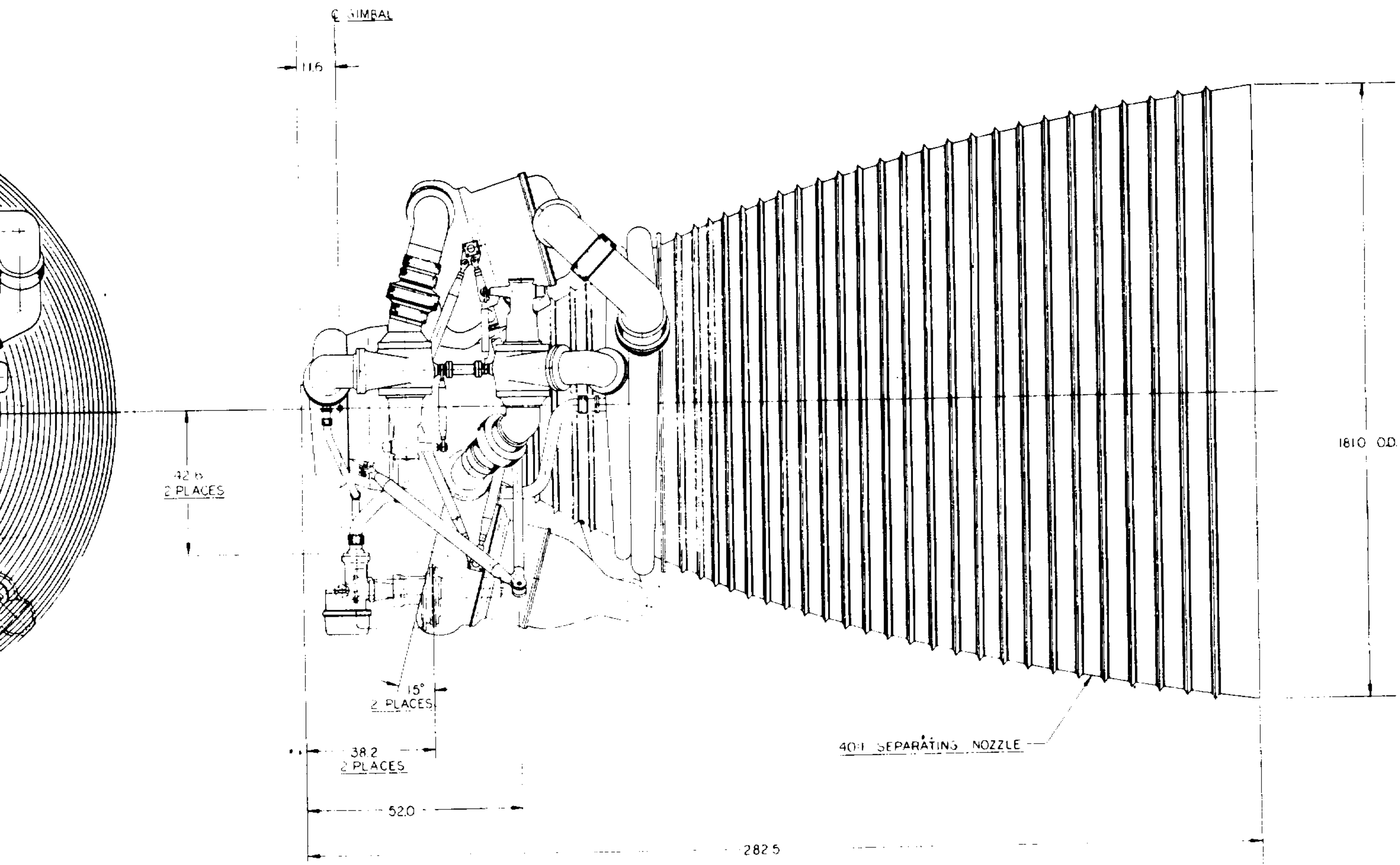


Figure 3. Initial Engine Design

1962/2Q

The initial Design Information Report was submitted to NASA on 28 June 1962.

The engine control systems consisted of the start system, the thrust chamber valves and actuation system, the gas generator valve and actuation system, and the propellant utilization valve. The primary control system is shown schematically in Figure No. 2.

1962/3Q

The M-1 Engine design was again modified during the third quarter of 1962. The length of the actuator arm was increased from 50-in. to 76-in., the turbine flange of the oxidizer turbopump assembly was increased from 10-in. to 14-in. in diameter, the outside space envelope of the thrust chamber assembly at the joint between the injector and the assembly changed in longitudinal location, and a structural ring was included on the thrust chamber at the injector interface to accept turbopump and gimbal actuator mounting loads. These changes also brought a number of configuration problems and an extensive layout investigation was made. This resulted in the creation of five engine configurations. The one shown as Figure No. 4 was selected as the basis for future efforts.

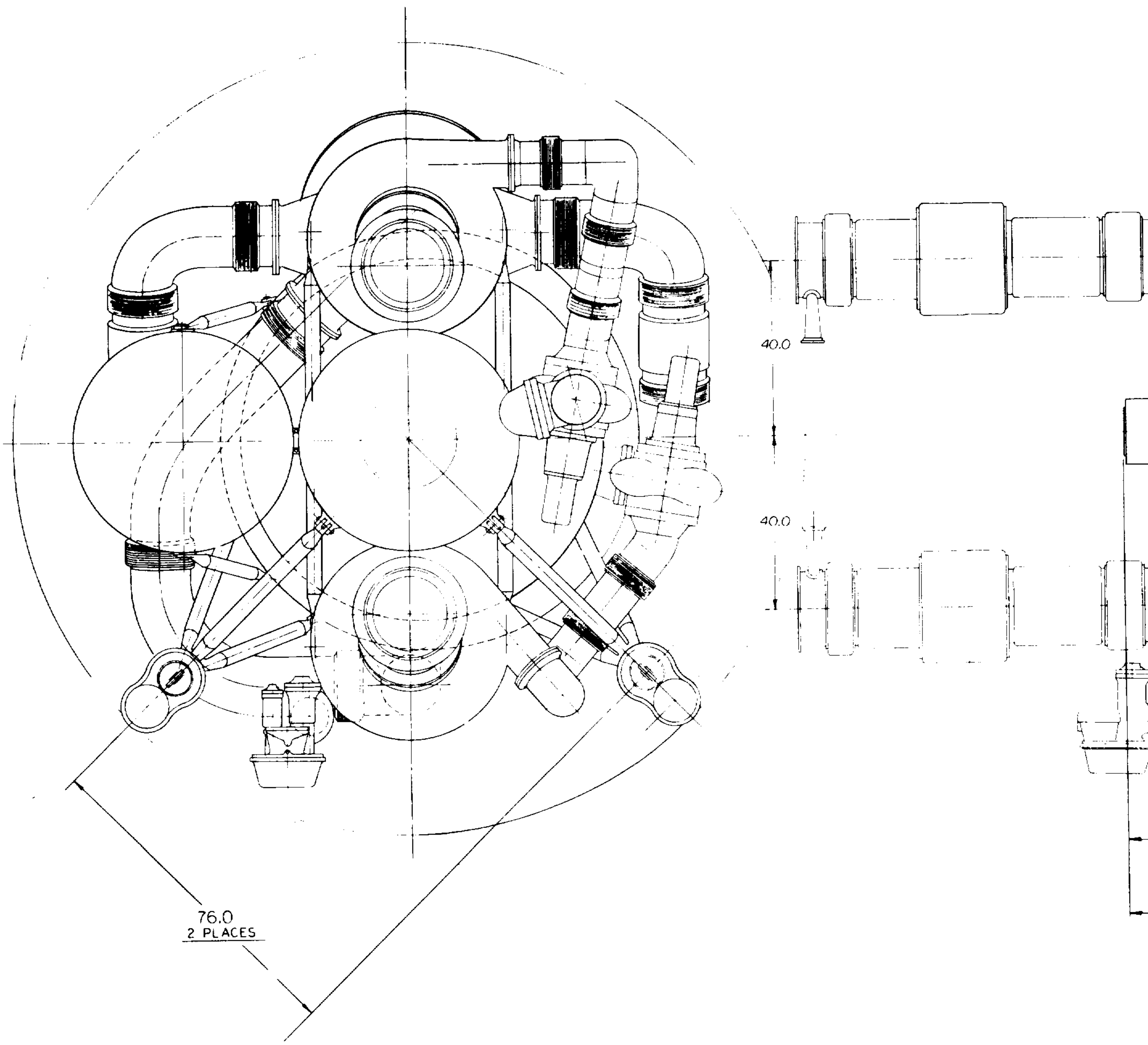
Detailed design of the propellant lines and hot-gas lines was proceeding on schedule. This effort has been detailed in a separate report(2) wherein the investigation and partial evaluation of semi-rigid ducting for hot gas application is presented for comparison with ducts utilizing flexible couplings for duct flexibility.

A one-quarter scale engineering model of the engine, excluding the nozzle extension, was completed during the third quarter of 1962.

1962/4Q

A new gimbaled engine configuration was released in October 1962. Changes from the preceding one included movement of the gimbal center to a point 34.6-in. aft of the thrust face, a reduction in turbopump inlet line length, a change in the angle of the inlet flanges to the gas torus on the nozzle extension from 45-degrees to 0-degrees, removable covers were added for the elbows downstream of the thrust chamber valves, and a structural frame was situated 12.5-in. ahead of the 8:1 area ratio station for the attachment of the turbopump, the actuator, and the start tank support.

(2) Hartley, E. J., M-1 Engine Ducts (Hot Gas and Cryogenic Propellant), Aerojet-General Report No. 8800-31, 6 December 1965.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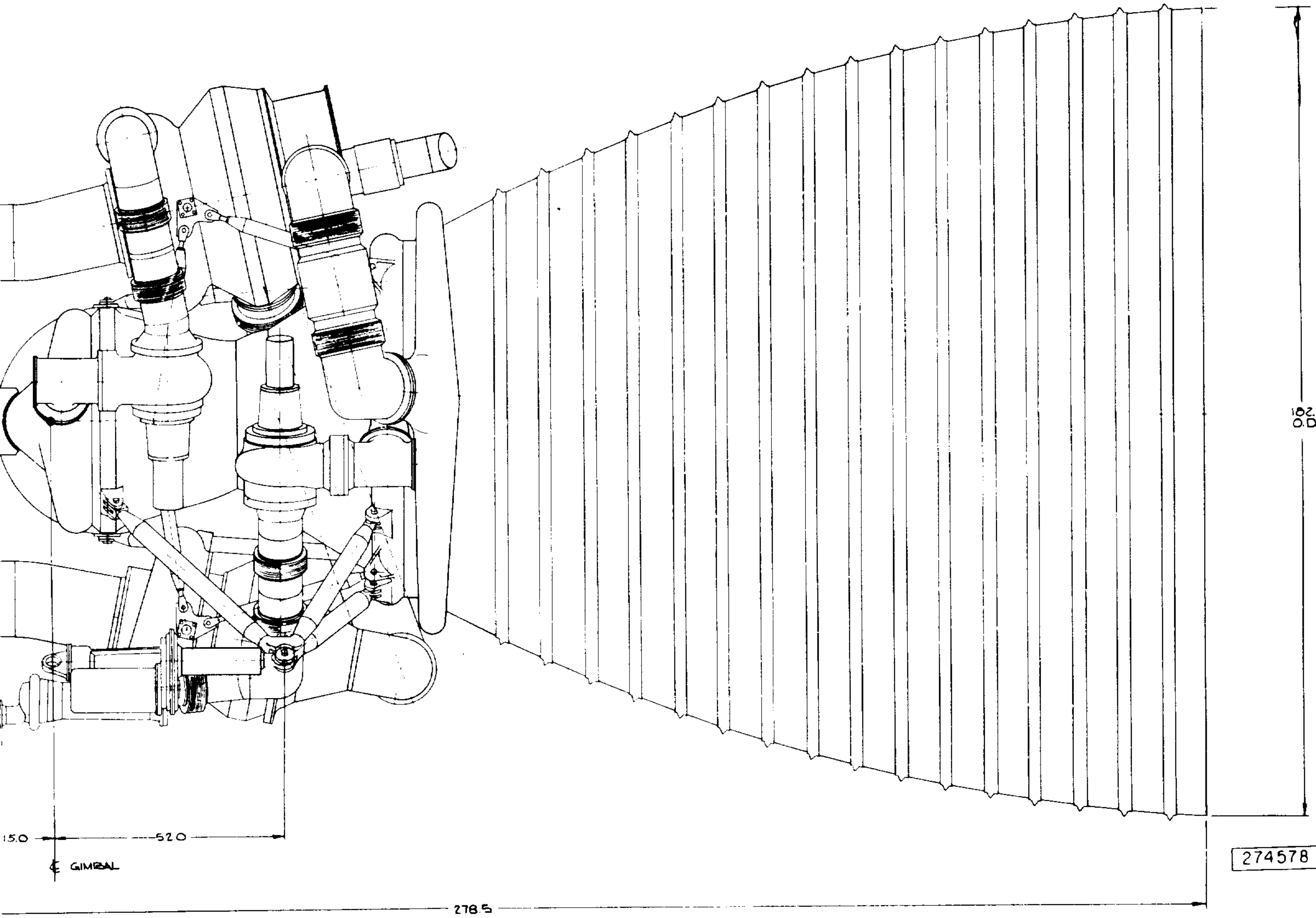

Figure 4. M-1 Engine Assembly, Configuration B-8

However, this configuration was further modified during December 1962 as a result of a new thrust chamber size as well as the inclusion of a new axial flow turbopump. These changes included a nozzle diameter increase from 186-in. to 204-in., relocation of the gas torus from an 8:1 area ratio to a 14:1 area ratio, a 7-in. decrease in the gas generator body, an increase in the suction lines from 17-in. to 19.5-in., and a decrease in the arc between the hot gas inlet flanges to the nozzle extension from 180-degrees to 90-degrees.

The location of the gimbal center, the length of the actuator arm, and the length of the suction lines were firmly established.

A report(3) identifying the problems associated with turbopump inlet flange movement during gimbaling was submitted to MSFC on 9 October 1962.

Thrust-vector control by secondary injection was being studied. The results of this effort have been separately documented.(4) The sources of secondary gas being considered included four auxiliary gas generators, turbine exhaust, main gas generator tap-off, single auxiliary gas generator, turbine exhaust plus the booster unit, thrust chamber tap-off, heated hydrogen, and cold hydrogen.

1963/1Q

During the first quater of 1963, it was found that the engine start system would have to be modified to accommodate the critical stall characteristics of the axial-flow fuel pump. Three control systems were investigated; separately-actuated thrust chamber valves, an oxidizer turbine by-pass valve, and a fuel pump discharge by-pass valve.

The M-1 Engine start and shutdown transient studies are detailed in a previously issued report.(5) The analyses used in these studies, all of which had the objective of determining the required controls and defining their flow control characteristics as well as their sequencing, are discussed. These performance analyses resulted in the final M-1 Engine system.

(3) Aerojet-General Leter 9510-0091, S. C. Datsko to O. E. Williams, dated 9 October 1962, subject: M-1 Turbopump Inlet Flange Movement During Gimbaling

(4) (U) M-1 Engine Secondary Injection Thrust Vector Control (TVC) Study, Aerojet-General Report No. 8225-2, Volumes I and II, 14 August 1963 (Confidential)

(5) Conn, K. R., (U) M-1 Engine Start-Shutdown Transient Studies, Aerojet-General Report No. 8800-34, 19 November 1965 (Confidential)

1963/1Q

Concurrent with the control system studies, a separate investigation was being made to develop a method for predicting the heat input to the hydrogen as it flowed into the relatively warm thrust chamber assembly. The heat transfer investigation is the subject of a separate report(6) wherein the work accomplished in the Heat Transfer Evaluation Program is discussed. This program consisted of fabricating and testing a section of the M-1 hydrogen-cooled thrust chamber for the purpose of obtaining heat transfer and flow rate data during the flow build-up transient.

The engine configuration was again slightly modified during the first quarter of 1963 in that the turbopump inlet was moved 20-in. aft with respect to the thrust chamber assembly.

1963/2Q

A preliminary natural frequency study of the engine gimbal structure loop was completed in April 1963. The results were presented to NASA during May. A restudy of suction line length was made at NASA request and the results were submitted on 18 June 1963.(7)

1963/3Q

The engine natural frequency study was submitted on 12 July 1963.(8)

As a result of the analyses made for natural frequency, gimbal angle, and weight, the engine configuration was modified during the third quarter of 1963. The actuator arm was reduced from 96-in. to 90-in., the gimbal bearing radius was reduced from 26-in. to 15-in., and the 7 cps natural frequency requirement was replaced by a 450,000 lb/in. spring rate for the actuator arm and ring.

1963/4Q

A study to determine methods for reducing the oxidizer turbine exhaust temperature variation was completed during the fourth quarter of 1963. Work was also completed in the study of hydrogen as a start gas.(9)

(6) Farrel, E. C., Heat Transfer Evaluation Program Using M-1 Combustion Chamber Tube Bundle Section (Two-Phase Flow Studies), Aerojet-General Report No. 8800-22, 1 November 1963

(7) Aerojet-General Letter 9617-0292, K. E. Unmack to W. F. Dankhoff, dated 18 June 1963, subject: LeRC Action Item No. 79

(8) Aerojet-General Letter 9617:0132L, K. E. Unmack to W. F. Dankhoff, dated 12 July 1963, subject: LeRC Action Item No. 80

(9) Aerojet-General Letter 8225D:0479, S. C. Datsko to W. F. Dankhoff, dated 8 November 1963, subject: (U) LeRC Action Item No. 207 (Confidential)

1963/4Q

The flange loading analysis was completed for the pump inlet and all other interfaces. Its only immediate effect was an indication that a structural support would probably be required for the turbine exhaust lines.

The basic engine configuration was altered slightly at the end of 1963. The gas generator valve assembly was rotated 180-degrees, the fuel recirculation valve was relocated closer to the thrust chamber valve, the sway brace on the fuel turbopump assembly was lowered, and the start tank was relocated to improve the start line routing.

1964/2Q

Evaluation of the engine helium start bottle was completed and a report published in April 1964.(10) Three methods for loading helium were considered: loading at ambient temperature and 3000 psia, maintaining constant 3000 psia as the helium cooled by the addition of helium; loading at ambient temperature and 3000 psia, allowing the pressure drop as helium cooldown occurred; and loading at -210°F and 3000 psia, allowing the pressure to vary as the helium temperature changed. The constant pressure loading method was recommended. The ambient temperature, variable pressure loading method was recommended as a back-up.

A fuel pump circulating system report was completed and published in June 1964.(11) This system study considered the effect of valve size, line size, and valve closing time upon the fuel pump start transient buildup. The recommended circulation system consisted of a 6-in. line and a valve. This system had a flow coefficient (Kw) of 56.24 and was mechanically-linked to the thrust chamber valve.

A study of thrust chamber skirt cooling, including the effect of fuel turbine seal leakage, was completed during the second quarter of 1964. The primary effect of the seal leakage would be to shift the operating range envelope approximately 25 deg/lb/sec toward lower temperatures.

1964/3Q

The first article fuel turbine simulator was completed during the third quarter of 1964 and delivered to Test Stands E-1 and E-3 for stand activation purposes. Fabrication of the initial set of oxidizer turbopump support struts was also completed. These struts were used to attach the pump housing to the dummy thrust chamber for Test Stands E-1 and E-3 activation. The first article interim suction line was also attached to the pump inlet at that time.

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(10) Evaluation of M-1 Helium Start Bottle, Aerojet-General Systems and Controls Report No. 166, 23 April 1964

(11) M-1 Engine Fuel Pump Circulation System Study, Aerojet-General Systems and Controls Report No. 164, 12 June 1964

The three-tube bundle heat transfer tests and the study of the thrust chamber skirt shutdown cooling were completed in August 1964. Also, the computer simulation of the H-8 gas generator configuration was completed.

An M-1 Engine structural dynamic model was in the process of formulation at the time that the engine design and development aspect was terminated on 24 August 1965. The partially completed model is detailed in a separate report(12) wherein the results of the analysis completed are documented for possible subsequent use. The liquid oxygen and liquid hydrogen turbopump assemblies and thrust chamber valves were analyzed. Natural frequencies and mode shapes were tabulated and illustrated. The report also contains stiffness and mass data as well as turbopump strut load distributions. The method of analysis is discussed briefly and supporting material is provided by listing applicable M-1 drawing numbers as well as the sources of flexibility and weight data.

(12) Newton, R. A., M-1 Engine Structural Dynamic Model, Aerojet-General Report No. 8800-6, 6 July 1965

V. COMPONENTS

A. THRUST CHAMBER ASSEMBLY

SUMMARY

This discussion is limited to the highlights of the thrust chamber assembly development effort from the inception of the M-1 Program through the last phaseout test on 4 August 1966. This testing is continuing as part of an extended program and a detailed report discussing the complete development of the M-1 thrust chamber assembly will be written(13) upon the completion of the extended program.

The M-1 thrust chamber assembly development program was undertaken to provide a unit for the 1,500,000 lb thrust liquid oxygen/liquid hydrogen M-1 Engine. There are three components that make up the thrust chamber assembly; the injector, the cooled combustion chamber, and the nozzle extension. Of these three components, only the injector was fabricated and tested.

A thrust chamber assembly with an uncooled chamber was designed to obtain data regarding starting, operation, performance, and stability parameters of the injector before committing the much more expensive regeneratively-cooled chamber. Thus, ablative-lined metal chambers were used to test the most promising injector design.

An ablative-lined chamber with an area ratio of 2:1 was selected. It provided the necessary operating duration without compromising performance evaluation.

The injector selected for evaluation contained 3248 coaxial elements, each of which has a thrust of approximately 450 lb. These elements were designed to provide a 19:1 injection velocity ratio (hydrogen velocity/oxygen velocity). The injector also incorporates transpiration face cooling, convective and film-cooled baffles, and a bolted-on oxidizer manifold. This design was selected as one of the most likely to provide stability and acceptable performance based upon results obtained from a NASA Stability Program (TND 3373) as well as J-2 engine development.

During the performance testing, the liquid hydrogen was pre-heated before entering the thrust chamber by adding gaseous hydrogen to simulate the temperatures expected at the outlet of the cooled chamber. Ten tests were conducted with the selected injector design using the ablative chamber.

The injector performance was slightly above the nominal PFRT requirement of 429 sec of specific impulse (extrapolated to 40:1 altitude operation). The characteristic exhaust velocity was approximately 97%. It

(13) Scheduled to be a high series NASA Contractor Report.

was demonstrated that the selected injector design was adequate for engine testing and its performance was at least equal to PFRT requirements. Stable operation was achieved during these performance tests.

Although a film-cooled metal chamber was fabricated to provide a short duration testing capability for stability evaluation of the injector, it was not used. Stability evaluation was not accomplished as part of the M-1 Program, rather it was scheduled for the extended injector evaluation effort, which would follow the M-1 Program.

The cooled chamber design for the M-1 Engine was not completed. The major problem encountered was to provide adequate stiffness at the chamber-to-injector flange as well as at the fuel torus to accommodate the loads at the extremities of the gimbal actuators and the pump supports. These loads were calculated based upon the combustion noise vibrations which were estimated to be approximately 5 g's. However, the injector performance tests conducted with the ablative chamber indicated that this vibration level was highly conservative. A chamber design which was capable of absorbing the higher vibration levels without providing for the pump support and actuator attachment was nearing completion at the time of the program termination. This design would permit the injector to be tested with a cooled chamber. There was a potential problem remaining with this design as a result of the thermal stresses produced at the joint between the tubes and the chamber-to-injector flange. The last calculations performed showed that a local permanent set would occur during transients; however, it was predicted that chamber life would exceed 100 cycles.

The nozzle extension design (turbine gas-cooled) was not completed. No design concept had been established which would be capable of absorbing the relative expansion between the nozzle entrance and the chamber exit without incurring excessive stresses. The nozzle tube (Hastalloy X material) wall temperatures were calculated to be approximately 1900°F at the worst of the nominal engine operating points.

CHRONOLOGY

1. Injector

1962/2Q

Initial design effort was directed toward refining the annular ring injector. Concepts for an alternative configuration suitable for coaxial or high-thrust impinging elements were also established. Annular groove injector effort was concentrated upon establishing the details of structural and manifold requirements to obtain adequate propellant distribution.

1962/2Q

It was decided that baffles would be incorporated into the injector to minimize the possibilities of radial or tangential modes of instability and the basic baffle configuration was established.(14)

Injector face rings were designed and eight injector patterns drawn during the second quarter of 1962. Also, the gimbal support plate design was completed as well as the design of the oxidizer torus and internal ribs.

1962/3Q

Several cooling methods were analyzed for the non-removable baffle system. During the third quarter of 1962, the film-cooled system for non-removable baffles was selected. Methods for providing mechanically-attached baffles were also studied and concepts were prepared for baffles that could be removed from the injector face. It was also decided during this time that 10 of the 20 multi-orifice concentric channel injectors would be made with copper face rings.

1962/4Q

An alternative oxidizer torus was selected during the fourth quarter of 1962. This torus was essentially a long conic section formed into a nine-sided polygon (nonagon) with feeder pipes leading to the propellant manifolds. The selection was based upon weight saving and a reduction in fabrication time over the existing design. This torus would be interchangeable between the coaxial and multi-orifice injector designs.

A decision was made, based upon studies for optimum cooling, to increase chamber coolant velocity and reduce the chamber fuel film cooling. This necessitated modification of the orifice pattern, the twenty-third (outermost) channel, and the injector-to-chamber joint so as to reduce metal thickness at the chamber-to-injector joint. Baffle design was also affected by this modification.

Design of the revised injector-chamber interface, which resulted from the decision to increase chamber coolant velocity and reduce chamber fuel film cooling, was completed.

By the end of 1962, three concentric ring injectors (S/N 0001, 0003, and 0004) were being machined concurrently.

1963/1Q

Detailed drawings of an alternative to the first nonagon torus (reported previously) were released early in 1963. This alternative, a smooth, explosively-formed torus, was lighter, had better flow characteristics, and was completely interchangeable between the multi-orifice and coaxial injectors.

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(14) Reardon, F. H., M-1 Thrust Chamber Transverse Modes Combustion Stability Analysis, Aerojet-General Report No. TCR-9621-012, 15 August 1963

1963/1Q

Increases in the pressure schedule necessitated a complete re-analysis of stresses in the coaxial injector and the design was modified to provide the additional strength required. Also, an element design (see Figure No. 5) and orifice grid were selected. The swirler oxidizer element was selected upon the basis of performance and combustion stability experience with liquid oxygen/liquid hydrogen.

A study was completed during the first quarter of 1963 that showed fuel film cooling to be adequate thermal protection for the M-1 injector face baffles.

1963/2Q

Final assembly of all component parts for the first article concentric ring injector (except for baffle bases) was completed during the second quarter of 1963. This included pattern drilling and welding of the oxidizer torus to the injector. Also, the oxidizer torus for the second article concentric ring injector was received. A smooth torus was fabricated and propellant cavity machining as well as manifold cover welding for the first copper face injector were completed along with the propellant cavity machining for the second copper face injector.

1963/4Q

Fabrication of all multi-orifice injectors except for S/N 004 and S/N 019 was stopped during the fourth quarter of 1963 because of program reduction.

Multi-orifice injector S/N 004 was completed and delivered to thrust chamber assembly build-up, where it was mated to the first article lower chamber assembly. Installation of the gimbal pad and machining of the oxidizer inlets were also completed. This assembly (see Figure No. 6) was sent to Test Stand C-9 for final build-up with dummy valve bodies installed. After stand build-up, the functioning valves were installed.

1964/1Q

Thrust chamber assembly valve sequence tests were initiated during Feburary 1964.

1964/2Q

On 20 May 1964, a high pressure thrust chamber oxidizer valve functional test was conducted. The test objective was a high pre-pressure liquid oxygen flow checkout test of the stand actuation system without out-flow. The thrust chamber oxidizer valve functioned satisfactorily during the opening cycle, but the closing time increased by 0.7 sec (1.4 sec actual as opposed to the 0.7 sec specified). A thrust chamber oxidizer valve post-test leak test indicated a leak through the valve. The valve was removed and shipped to the valve shop, repaired, and returned to the test stand.

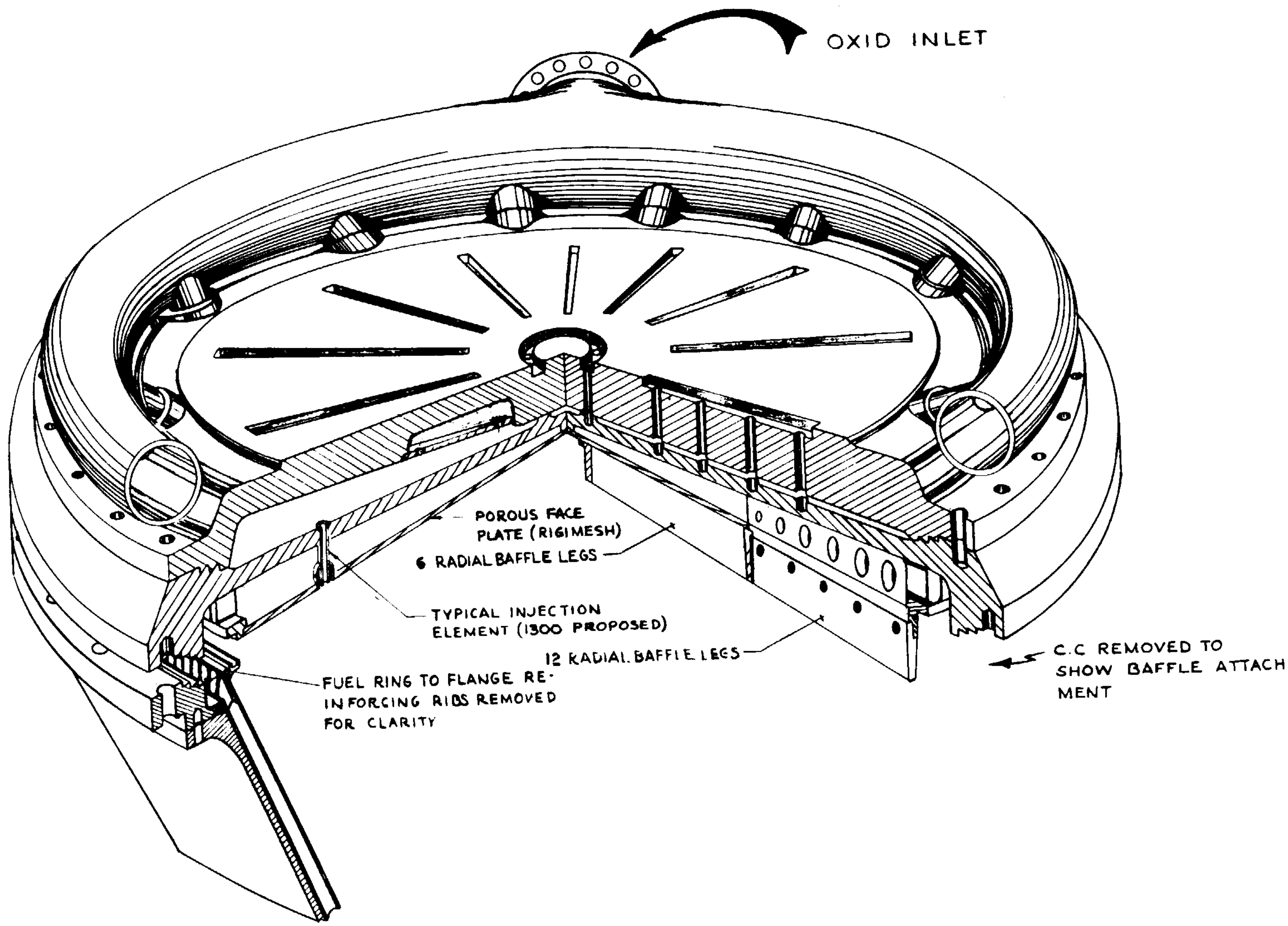

Figure 5. M-1 Coaxial Injector, 12-Rib Bolt-On Dome

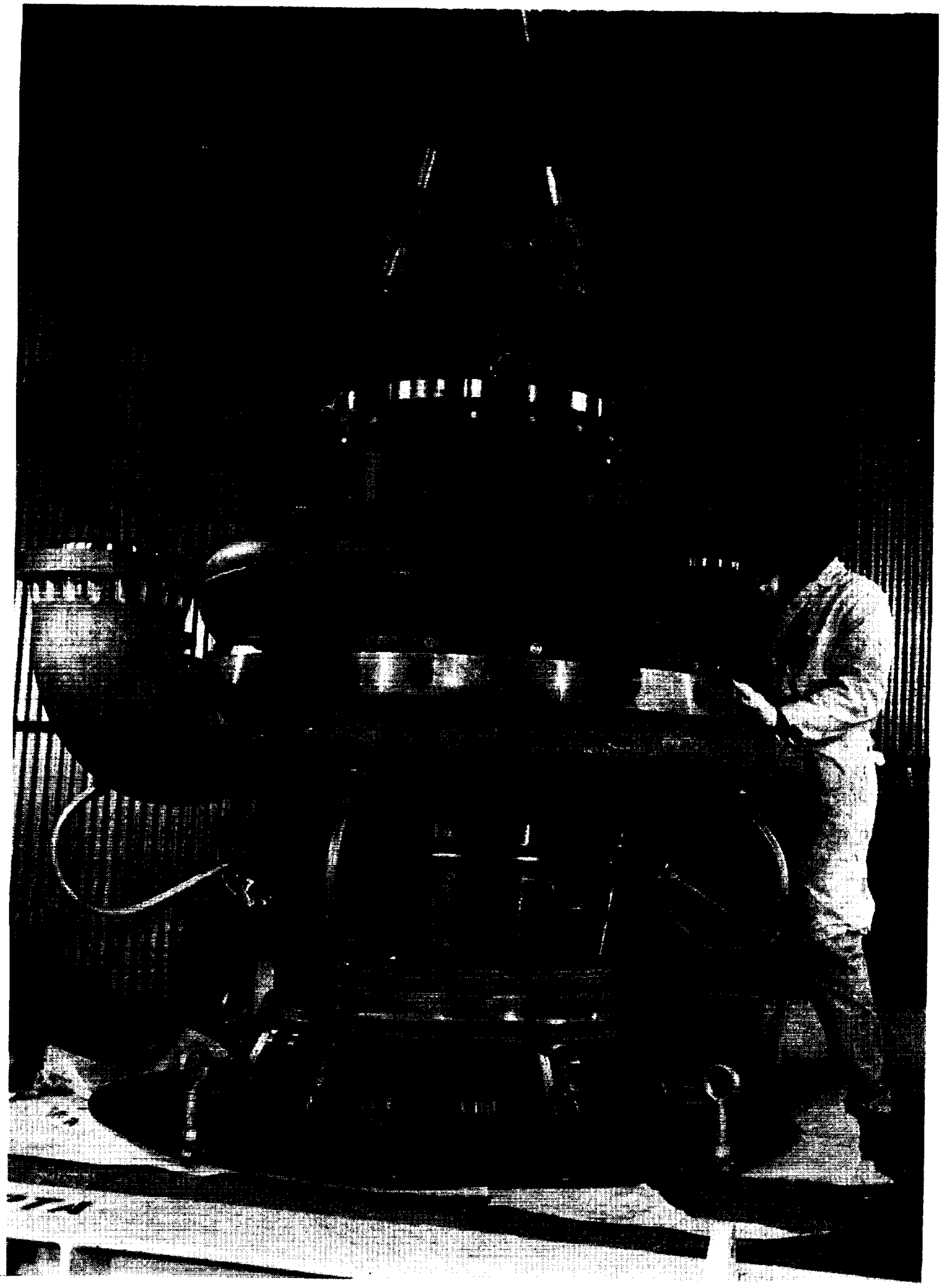

Figure 6. M-1 Uncooled Thrust Chamber Assembly

On 26 May 1964, a high pressure liquid oxygen cold flow test was conducted. Results were satisfactory. A post-test leak test indicated minor leakage through the valve. Because additional high pressure cold flow tests were required to demonstrate repeatability as well as reliability, another test was conducted on 27 May 1964. Results were satisfactory and no additional thrust chamber oxidizer valve leakage was noted after this second test.

The initial hot firing (Test No. 1.1-02-EHM-001) was attempted on 6 June 1964. This test resulted in a premature shutdown (TCPS) as a direct result of not obtaining chamber pressure (P_c) within 0.99 sec. Chamber pressure was not obtained because the thrust chamber oxidizer valve remained closed. The thrust chamber fuel valve opened and allowed partial fuel flow prior to the TCPS signaling shutdown. Post-fire inspection of the hardware indicated no apparent thrust chamber assembly hardware damage. Subsequent leak check and teardown of the oxidizer valve actuation housing revealed actuator piston lipseal failure. The thrust chamber fuel valve was also thoroughly leak checked and found to be satisfactory.

The second hot firing (Test No. 1.1-02-EHM-002) was attempted on 18 June 1966. Records revealed slow initial fuel valve movement, which resulted in a delay of subsequent sequential events. This delay caused the M-1 timer, which was set for FS_1 + 0.826 sec to time out prior to oxidizer valve movement.

Following pre-fire valve functional tests and the verification of start and shutdown sequences, the third hot firing (Test No. 1.1-02-EHM-003) was conducted on 20 June 1964. This test was scheduled for 2.80 sec; however, a premature shutdown occurred at 1.68 sec because of the loss of the oxidizer feed system. Test Stand C-9 incurred severe damage. Post-fire inspection revealed that the thrust chamber oxidizer valve and the oxidizer line from the tank safety valve down to the thrust chamber assembly, including the liquid oxygen flow meter, were severely burned. Numerous pieces of test stand plumbing, including the hydraulic system and electrical system, sustained damage. Damage to the thrust chamber assembly consisted of severe burning of the oxidizer valve, erosion inside the elbow between the valve and the injector, erosion of the injector flange and inlet, injector face erosion in four baffle compartments, bent heat deflector shield, bent fuel film cooling lines on the 2:1 combustion chamber, and external erosion of the fuel valve housing. The thrust chamber assembly was removed from the stand and disassembled. Inspection and evaluation of individual subcomponents was undertaken. Also, the records (oscillograph, high frequency data, accelerometer data, millisadic data, movie coverage, etc.) were investigated by a management-appointed team consisting of Thrust Chamber, Systems and Controls, as well as Combustion Stability Analysis and Stress personnel.(15)

(15) M-1 Thrust Chamber Assembly Accident (Test No. 1.1-02-EHM-003), 20 June 1964, The M-1 Accident Special Investigation Team, Lt. Col. Joe E. Heatherly, Chairman, 28 August 1964

1964/2Q

The uncooled thrust chamber assembly effort was then directed toward completion of the investigation and preparation for testing at Test Stand H-8.

Multi-orifice injector assembly S/N 019 was also completed and delivered to the thrust chamber assembly build-up area during the second quarter of 1964.

1964/4Q

Stop fabrication orders were issued during the fourth quarter of 1964 for all injectors pending resolution of the element and baffle configuration. Uni-element tests indicated that the elements which were inserted but not brazed in injector S/N 017 would erode in the cup. Several corrective design modifications were contemplated, the most promising of which were to be tested as part of the uni-element program. Injector S/N 017 was to remain on a hold status until data was available from these tests. Pattern drilling of injector S/N 015 was completed but this unit was also held pending a decision on an improved element-to-plate joint.

Stop fabrication orders were also issued for all oxidizer domes and components pending completion of stress analysis and testing. A pressure test was conducted on dome S/N 3. After tests at various partial pressures which were required to crack the stress coat and test the strain gages, a full pressure-proof test of 2100 psig was conducted. Some deformation occurred in the area of the inlet. This local yielding was considered to be insignificant because of its small magnitude and the configuration of the component. A thermal shock test was conducted by filling the liquid oxygen torus with liquid nitrogen while insulating the dome and maintaining it at near-ambient temperature. A maximum thermal differential of 270°F was obtained during the first test. Linear potentiometers were installed to monitor the torus shrinkage during chill-down. Maximum deflection of the torus was found to be 0.040-in. to 0.050-in.

1965/1Q

Additional tests were made. Testing of the liquid oxygen torus of an M-1 injector dome assembly continued. The assembly was filled with liquid nitrogen at three different times (stand-pipes to dome were blocked by wooden plugs) resulting in a thermal shrinkage condition which closely approximated operational conditions of the thrust chamber assembly. Deflection and temperature measurements were taken to correlate with the thrust chamber assembly stress analysis. These test results provided considerable information regarding deformation-interaction of the torus-standpipe. Sensitive crystal accelerometers, mounted on the torus during the chill cycles, sensed the pressure and origin of stress waves generated during the chill-test.

1965/1Q

Early in 1965, it was decided to fabricate two injectors, S/N 012 and S/N 020, each having 3248 elements as well as Rigimesh-cooled baffles. New drawings of the injector plate and baffle bases were released and fabrication initiated. The baffle bases were completed and installed into injector S/N 020. Also, the element-to-injector joint configuration was made firm. It consisted of a copper brazed joint with a secondary mechanical (threaded) joint.

A stainless steel valve simulator was also designed. In addition, a throat plug assembly for testing the primary conical seals of the thrust chamber assembly and facility seals downstream of the facility control valves at a pressure of 50 psig was designed. Also, acoustical testing of the M-1 uncooled thrust chamber assembly having a flat-face coaxial injector with 1.5-in., 4.0-in., and 6.0-in. long baffles was completed.

1965/2Q

Pattern drilling of injectors S/N 020 and 012 was completed during the second quarter of 1965.

The mock-up thrust chamber assembly and the facility gimbal simulator adapter were completed and delivered to Test Stand H-8 during the second quarter of 1965. It was installed into the test stand and used to check out and adjust the mating of the stand propellant lines, thrust pad spacer, line dampers, and instrumentation bracketry.

Two test-stand-to-thrust chamber assembly lines with bellows were also completed and delivered to the test stand. Redesign of the gimbal simulator and gimbal simulator spacer ring was also accomplished. This was necessary because of a redesign of the injector dome to solve leakage problems at the radial bolt seals.

Preliminary stress analysis of the thrust chamber assembly indicated all bolts specified on the drawings were adequate.

1965/3Q

In July 1965, Rigimesh baffle fabrication was stopped as a result of evaluations being made at NASA/LeRC. Their subscale test data showed that Rigimesh flow characteristics were such that the projected M-1 design point could not be achieved regardless of the material porosity. Film cooling would be necessary with Rigimesh baffles and in addition, three baffle legs failed during hydraulic testing. The design of copper baffles was initiated. Copper baffles were subsequently fabricated and installed in the injectors used for thrust chamber assembly testing.

Injector S/N 020 was returned from the vendor brazing operation in mid-July 1965 and it was installed in the vertical lathe for swaging of the injection element fuel sleeves. Cracking in the flare was experienced with all of the 19 sleeves upon which swaging was attempted. Metallurgical analysis of sample sleeves and wet chemical analysis of samples removed from the injector fuel distribution ring disclosed that the 347 CRES material in both types of samples was heavily carburized. Subsequent sampling of other major thrust chamber components (i.e., domes, injectors, torus and flange assemblies, and lower chamber assemblies) disclosed no further significant carburization with the exception of injector S/N 12.

Heat treating procedures were investigated. Carburization of the two injectors had resulted from oven cycles at Pyromet wherein a graphite tooling support plate was used in conjunction with inadequate gaseous hydrogen flow distribution. A heat treating cycle was developed which would provide material properties compatible with structural requirements.

In September 1965, both injectors were processed through the prescribed heat treating cycles with subsequent metallurgical sample analysis indicating satisfactory material properties. Fabrication of both injectors was then continued.

Injector S/N 018 (1184 element) was completed during the third quarter of 1965 to the point of assembly of the solid, uncooled baffles. These baffles were modified from a previously specified 6-in. length to 4-in. and fitted to the injector. A gaseous nitrogen flow test was then successfully performed with this injector to ensure that the Rigimesh face plates were free of obstructions and flow balancing restrictors were tack-welded at the inlet to the oxidizer elements.

Required bolt and coolant hole sizes as well as their locations were established for the copper bolt-on baffles.

The instrumentation drawings for the uncooled thrust chamber assembly and uncooled ablative thrust chamber assembly were completed and released during the third quarter of 1965.

Fabrication of thrust chamber components was also completed. This included the throat plug, two stainless steel valve simulators, a flow restrictor, gimbal simulator, and gimbal simulator spacer ring.

Three preliminary fuel tank rise-rate tests were conducted on 14 and 15 September 1965 at 80% ullage with no outflow. Gaseous hydrogen was used as the pressurant and liquid nitrogen as the fluid. The fuel tank was pressurized at 350 psig, then ramped to 1600 psig at 12 sec, 8 sec, and 4 sec intervals. The first test was shut down prematurely when the strip chart recorder, which was improperly ranged, showed the ramp continuing past the programed setting. In all other respects, the three tests were satisfactory.

1965/3Q

Three preliminary oxidizer tank rise-rate tests were conducted on 22 September 1965 at 25% ullage with no outflow. Gaseous nitrogen was used as the pressurant and liquid nitrogen as the fluid. The oxidizer tank was pressurized at 350 psig, then ramped to 1600 psig at 5.5 sec, 2.5 sec, and 1.5 sec intervals. All three tests were satisfactory.

1965/4Q

The assembly of S/N 018 injector and S/N 001 dome, the element and face plate installation of S/N 012 and S/N 020 injectors, and a set of copper baffles were completed in the fourth quarter of 1965.

A simultaneous flow test of both oxidizer circuits of the mock-up thrust chamber assembly utilizing liquid nitrogen was attempted on 5 October 1965; however, the thrust chamber valves failed to open. Three satisfactory thrust chamber valve functional and sequence checks were subsequently completed. A satisfactory 6.05 sec flow test was conducted on 8 October 1965.

During the period 12 October 1965 through 19 October 1965, six facility "twang" link calibration tests and seven thrust measurement system resonant frequency shock tests were conducted without the thrust chamber assembly installed. These tests were for the purpose of determining system constants.

Installation of the thrust chamber (S/N 004) with injector S/N 018 on Test Stand H-8 was complted. The thrust chamber assembly was instrumented for oxidizer cold flow testing. Two liquid oxygen cold flow tests of the oxidizer circuit were conducted. Each had a duration of 6 sec. The first test was satisfactory and no hardware damage was visible. Examination after the second test revealed that the 6-in. burst diaphragm at the oxidizer thrust chamber valve was ruptured and the dump line downstream of the burst diaphragm was damaged. Redesign and repair was completed. Fuel system purge calibrations were initiated using gaseous nitrogen as the purge media. Regulator problems were encountered and disassembly of the pressure regulator revealed that eight seat retainer bolts and a 3-in. section of the seat were missing. Only one of the eight bolts was found in the thrust chamber assembly fuel torus during line disassembly. The remaining parts are assumed to be in the thrust chamber.

Two igniter checkout tests were satisfactorily conducted in a liquid nitrogen media. The igniters were soaked for 20 minutes prior to ignition. Three static thrust calibration tests were completed at ambient pressure and temperature conditions. Also, three static thrust calibration tests were conducted at 1500 psi and cryogenic temperature conditions.

Six thrust chamber fuel valve functional tests were conducted on 29 November 1965 for purposes of evaluating operation under ambient and cryogenic conditions. Of the six tests performed, one was at ambient conditions, four with liquid hydrogen in the line under no pressure, and one with liquid hydrogen in the line at 320 psig. The fuel valve opening time increased by 0.11 sec under pressurized conditions.

One fuel vessel rise-rate test was satisfactorily conducted on 3 December 1965 utilizing liquid hydrogen as the fluid and gaseous hydrogen as the pressurant. The ullage was 10% with no outflow of hydrogen. Tank pressure was ramped from a pre-set of 344 psig at FS_1 to 1650 psig within 1.1 sec. Test duration was 6.2 sec.

An oxidizer system flow test, utilizing liquid oxygen as the media was conducted on 6 December 1965 for a duration of 6.02 sec. Tank pressure was ramped from a pre-set of 790 psig at FS_1 + 1.6 sec to a run pressure of 1770 psig in 1.01 sec. The oxidizer by-pass valve opening time was 1.71 sec as compared with a required 0.40 sec. The closing time was 1.23 sec as compared with a required 0.50 sec. No hardware damage was noted.

A combined liquid nitrogen flow test of both propellant circuits was attempted on 9 December 1965 for a duration of 0.8 sec. Shut-down was initiated by the 0.8 sec malfunction timer, at which time, the oxidizer by-pass valve must be fully open. Because of slow actuation of both the fuel thrust chamber valve and the oxidizer by-pass valve, the 0.8 sec timer terminated the test. No hardware damage was noted. Because of the thrust chamber valve control problems encountered, further flow testing was delayed pending rework of the pilot valves and the hydraulic actuation system.

A gaseous hydrogen mixer flow test was conducted on 11 December 1965 for a duration of 7.20 sec. Shutdown was initiated when the fuel tank pressure rose above the 175 psig pressure limit in the fuel tank. This pressure limit is prescribed by the vent system through which the gaseous hydrogen was being exhausted. The test was satisfactory for the duration of 7.20 sec. At FS_2 + 0.1 sec the pressure drop across the gaseous hydrogen filter rose abruptly for 0.4 sec. Post-fire inspection revealed a collapsed filter element which had extended through the filter body flange into the downstream line. The entire fuel propellant system was disassembled for inspection and cleaning.

Following the failure of the gaseous hydrogen mixer filter element, the liquid hydrogen system was disassembled, flushed with water. reverse "blowdowns" were conducted, and it was vacuum cleaned. The gaseous hydrogen filter element was replaced with an element of new design and the fuel systems were re-assembled.

1966/1Q

Flow testing of S/N 020 and S/N 012 injectors were successfully completed at the Hydraulics Laboratory during the first quarter of 1966. Also, the first set of copper baffles were incorporated into the thrust chamber assembly build-up which was sent to the test area.

Two gaseous hydrogen mixer flow tests were conducted on 8 January 1966 for durations of 6.06 sec and 5.07 sec.

The Critical Experiment Review for the first test of an M-1 uncooled thrust chamber assembly with a 3300 element injector was held on 27 January 1966. The start transient utilizing fluorine ignition was made final, along with the fluorine ignition system.

Three partial liquid nitrogen combined flow tests were conducted on 2 February 1966 for durations of 4.5 sec each to confirm the valve timing. A combined liquid nitrogen flow test of the 1100 element injector was conducted on 10 February 1966 for a duration of 7.5 sec.

The oxidizer flow restrictor was removed and an oxidizer manifold fill test was satisfactorily conducted on 11 February 1966 for a duration of 7.5 sec.

Thrust chamber assembly S/N 004 with coaxial injector S/N 018 was removed from Test Stand H-8 and returned to the Research and Development Shops.

A satisfactory gaseous helium purge calibration was conducted on 10 March 1966 for a duration of 3 sec. The regulator was replaced with a nozzle for flow control. All systems appeared normal.

Thrust chamber assembly S/N 002, with 3300 element injector S/N 020 and the ablative lined combustion chamber, was received in the test area and installed on Test Stand H-8 on 11 March 1966.

A satisfactory refrasil liner material gaseous fluorine compatibility test was conducted. A sample of the liner material used in the combustion chamber was exposed to 3 lb of gaseous fluorine for 3.5 min. No abnormalities or damages to the sample were noted.

The gaseous fluorine ignition manifold was received in the Test Area on 22 March 1966. Assembly of the manifold to the thrust chamber assembly was completed.

1966/2Q

After completing the assembly of the fluorine ignition system manifold to the thrust chamber assembly, a low pressure fluorine flow test was conducted on 12 April 1966 for a duration of 2.65 sec. A maximum pressure of 170 psia was obtained and no damage was noted.

On 14 April 1966, a 750 psia high pressure fluorine flow test was conducted for a duration of 4.41 sec. A burn-out of the check valve and 1/2-in. manifold tubing (upstream of the fluorine injector at the P_c5E location) occurred at FS_1 + 4.25 sec. The inside of the fluorine injector was lined with metal slag from the erosion upstream and the fluorine injector orifice was also eroded by the flowing metal slag. There was minor charring of the ablative liner material which surrounds the fluorine injector tip.

The fluorine ignition manifold was reworked without being removed from the test stand. The primary change was the removal of the check valves. The system was flow tested on 25 April 1966 with gaseous fluorine for a duration of 4.52 sec. The fluorine pressure dropped from 780 psia to 575 psia in 3.7 sec. No abnormalities or hardware damages were noted.

Satisfactory sequence and functional checks were conducted on 28 April 1966. High pressure oxidizer system leak checks were started on 29 April 1966. Upon reaching 530 psig, a flange failure occurred at the dehydration line attachment point of the oxidizer system. Metal burning occurred at the dehydration line flange and at the 2-in. oxidizer system low point bleed valve flange. Subsequent investigation indicated that the failure mode was flange separation with a possibility of severe rubbing. The separation occurred as a result of high water-hammer pressure associated with vapor bubble collapse in the branch line system. There was no evidence of contamination. Appropriate corrective action involved the assurance that there was adequate system bleed prior to pressurization and slow ramping of the liquid oxygen tank pressure during the initial pressurization phase.

The oxidizer system was repaired and satisfactorily cryogenic leak-checked to 1565 psig and the fuel system was satisfactorily cryogenic leak-checked to 1320 psig. Hot gas leak checks as well as satisfactory valve functional and sequence checks were conducted.

The primary objective of the first thrust chamber assembly test at Test Stand H-8 was to achieve liquid ignition utilizing gaseous fluorine hypergolic ignition. The secondary objectives were to verify hardware and test stand integrity, confirm valve operation and sequencing, and to check on the gaseous hydrogen mixer operation.

Test No. 1.2-05-EHM-001 was conducted with S/N 020 injector on 20 May 1966 for a duration of 2.340 sec. The primary objective was satisfactorily met with chamber pressure passing 200 psia and signaling shutdown was planned. All secondary objectives were satisfied except for the gaseous hydrogen mixer operation. Gaseous hydrogen flow to the mixer caused a reversal in the liquid hydrogen flow. Only gaseous hydrogen was delivered to the combustion chamber and combustion occurred at a higher mixture ratio than planned (approximately 5 as compared to 2).

No hardware damage or erosion occurred. A minor increase in heat-marking of the porous injector face surrounding all of the welded element fuel tips and the welds themselves was noted. This was an increase from the original heat-markings caused by welding. Figure No. 7 is a pre-test over-all view of the thrust chamber assembly and injector while Figure No. 8 is a post-test view.

Test No. 1.2-05-EHM-002 was conducted on 27 May 1966. The primary objective was to achieve 450 to 500 psia chamber pressure. Secondary objectives were to verify hardware and tst stand integrity, confirm valve operation and sequencing, as well as to confirm the mixer operation. The mixture operation had been changed from a feed-back to an open-loop system. In addition, safeguards were added to the system to initiate shutdown should an all-gas operation occur. Test duration was 0.635 sec. Shutdown was automatically signalled when the thrust chamber fuel valve opening actuation pressure fell below 2000 psig, after once exceeding 2000 psig.

The thrust chamber assembly start sequence and operation were modified prior to Test No. 1.2-05-EHM-003.

Test No. 1.2-05-EHM-003 was conducted on 1 June 1966 for a duration of 0.843 sec. Shutdown was singalled when the temperature at the thrust chamber fuel valve passed 260°R. The gaseous hydrogen flow stepped to approximately 21 lb/sec within 0.06 sec at initial flow instead of the programed 10 lb/sec in 0.05 sec, with the thrust chamber fuel valve 9% open. Therefore, 0.07 sec later, when the ramp to 69 lb/sec was occurring, the gaseous flow was 25 lb/sec instead of the anticipated 13 lb/sec, with the thrust chamber fuel valve 12% open. Thus, the flow of gaseous hydrogen entering the mixer was sufficient to decrease the liquid flow to exceed the 260°R fuel through the thrust chamber fuel valve. Also, a predominant 2 cps frequency in the fuel system was noted with the fuel temperature downstream of the thrust chamber fuel valve oscillating between 125°R and 280°R.

Two changes were made in the mixer system to resolve the mixer problems. The fuel safety valve was throttled to increase the pressure drop from the tank to the mixer in the liquid system. The second change was to machine a 7-in. diameter orifice in the downstream end of the mixer to decrease the radial flow of gaseous hydrogen thereby directing gas downstream from the mixer.

1966/3Q

Test No. 1.2-05-EHM-004 was conducted on 5 July 1966 for a duration of 2.528 sec. Shutdown was signalled when the temperature at the thrust chamber fuel valve exceeded 260°R. No hardware damage or increase in baffle heat-marking was noticed. The chamber pressure reached 280 psia at a mixture ratio of 3.7.

Test No. 1.2-05-EHM-005 was conducted on 12 July 1966 with increased fuel tank pre-set pressure for a duration of 3.248 sec. The test

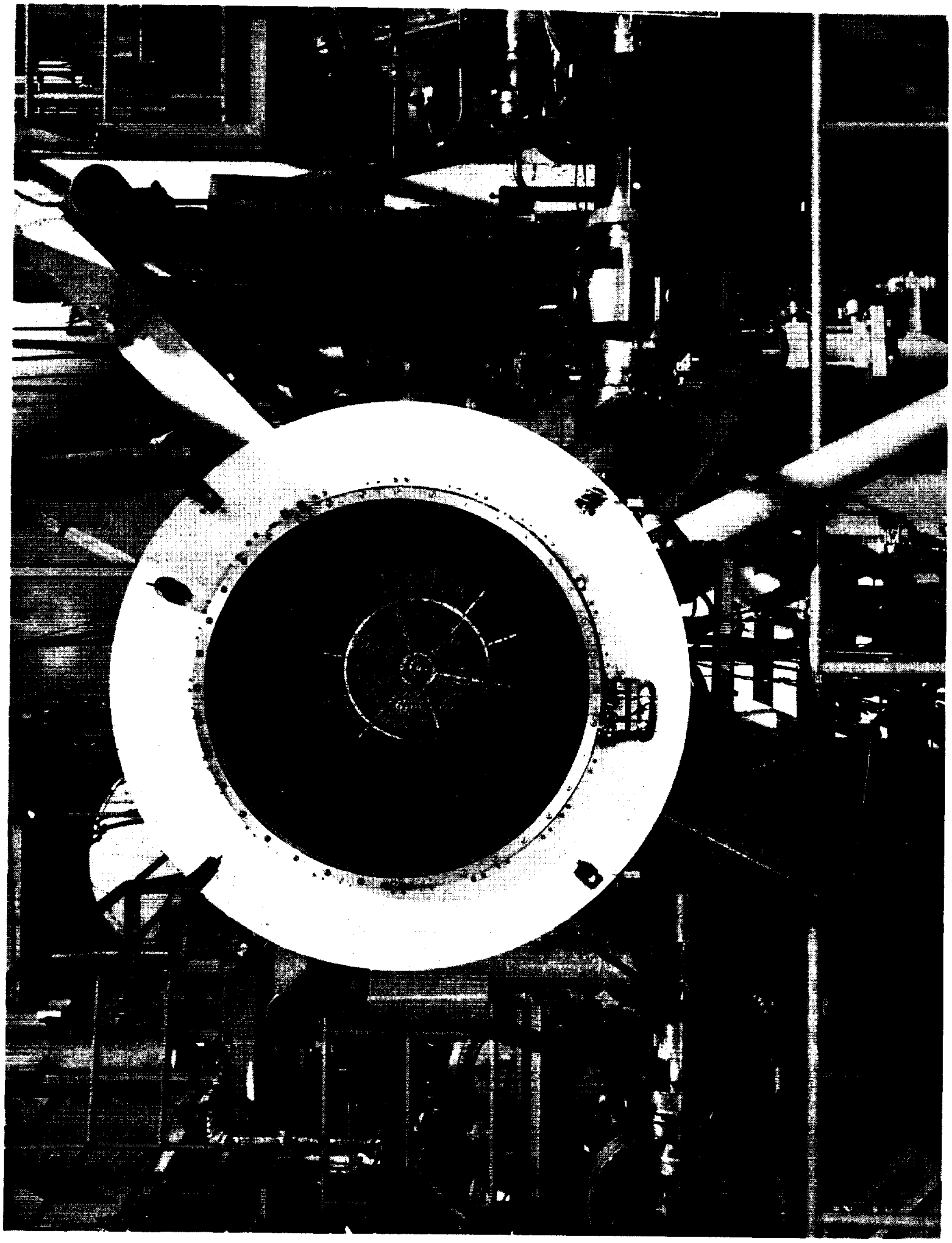

Figure 7. Over-All View of Thrust Chamber Assembly and Injector S/N 004 (Pre-Test)

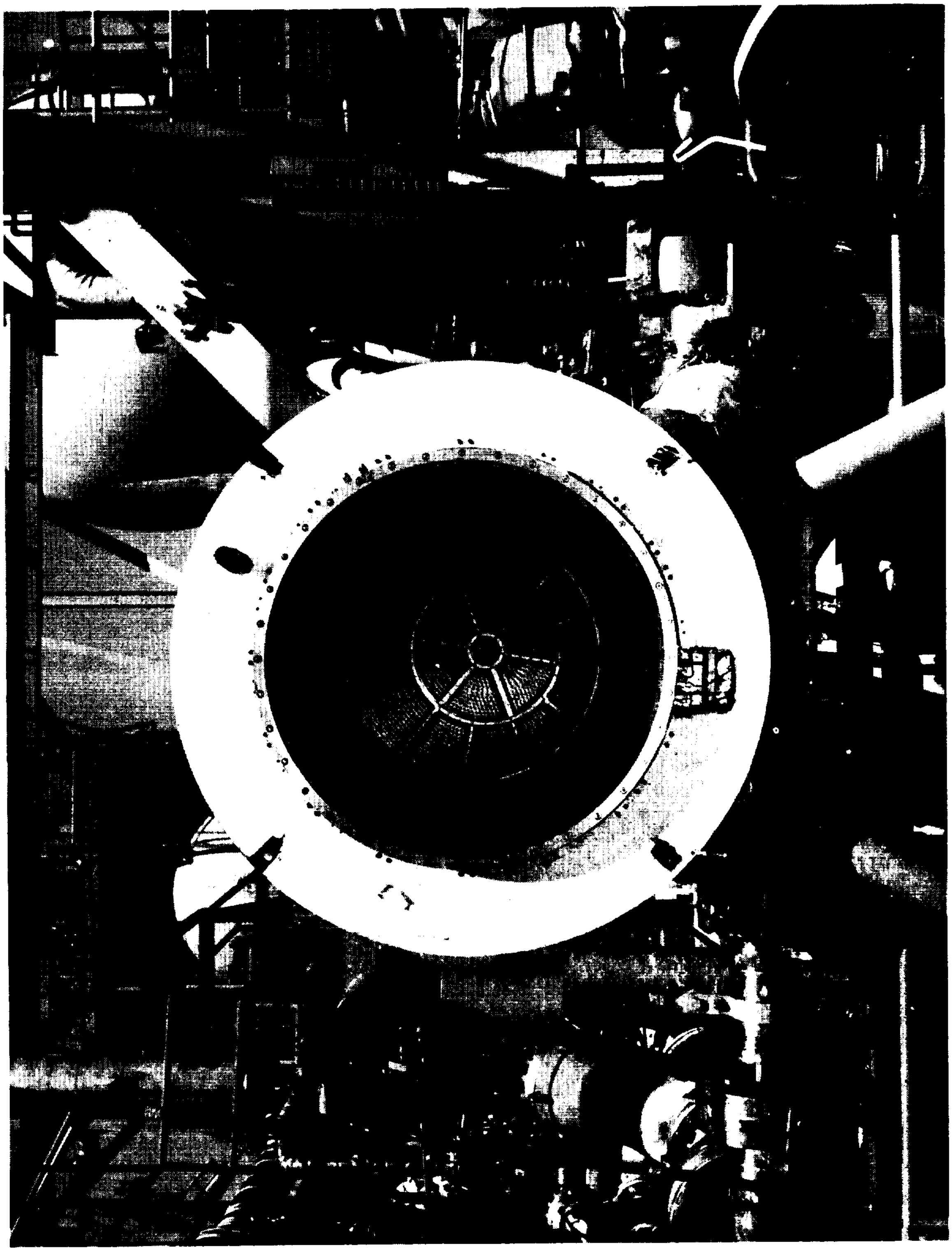

Figure 8. Over-All View of Thrust Chamber Assembly and Injector S/N 004 (Post-Test)

objectives of obtaining approximately half chamber pressure, satisfactory line loads, adequate ignition, and adequate pressure increase were satisfied. Shutdown occurred at 50% of thrust chamber oxidizer valve opening as planned. No hardware damage or increase in baffle heat-marking was noted. A chamber pressure of 450 psia at a mixture ratio of 2.1 was achieved at shutdown. Fuel injector temperature was 154°R.

The first full chamber pressure test (Test No. 1.2-05-EHM-006) was successfully conducted on 20 July 1966 for a duration of 6.558 sec. Minor tip erosion occurred on outer baffles 8 and 11. The tip of the outer ring was eroded in several places. The inner hub and inner baffles were heat-marked but there were no signs of erosion. There was no apparent increase in the injector face heat-marking. Several fuel tip swages were heat-marked in all of the baffle cavities.

Test No. 1.2-05-EHM-007 was conducted on 28 July 1966 for a duration of 4.285 sec. This test was inadvertently terminated when the liquid hydrogen flow decreased from 440 lb/sec to 190 lb/sec. The decrease in flow rate was normal as gaseous hydrogen flow to the mixer ramped and initial liquid ignition occurred. Hardware inspection revealed no visible increase in erosion or heat-marking. Chamber pressure was 1087 psia at a mixture ratio of 5.3.

Test No. 1.2-05-EHM-008 was conducted four hours later and ran for a duration of 7.627 sec. Steady-state duration was 3.10 sec. Chamber pressure was 1100 psia at a mixture ratio of 5.34 and the fuel injector temperature was 134°R. The throat diameter increased from a pre-test value of 29.987-in. to a post-test value of 29.996-in. The throat diameter between baffles ranged from 0.040-in. to 0.130-in. greater than those diameters in line with the baffles.

The final two uncooled thrust chamber assembly tests in the M-1 Engine Phaseout Program were accomplished on 4 August 1966. Test No. 1.2-05-EHM-009 was conducted for a duration of 7.487 sec. Eight hours later, the final test, No. 1.2-05-EHM-010 was conducted for a duration of 7.006 sec. Figure No. 9 shows a full-thrust test in progress.

Two outer baffle legs were modified to increase the S/N 020 injector fuel coolant flow during August 1966. They were installed on S/N 020 injector to repalce two damaged legs (No. 8 and No. 11).

Ten tests were conducted by 4 August 1966, at which time the M-1 Program phaseout effort was completed. Additional testing was initiated on 23 August 1966 as part of the large hydrogen-oxygen injector test program. By the end of August, the uncooled thrust chamber assembly had been subjected to partial thrust tests and eight full-thrust tests. The excellent condition of the injector after this testing is shown on Figure No. 10.

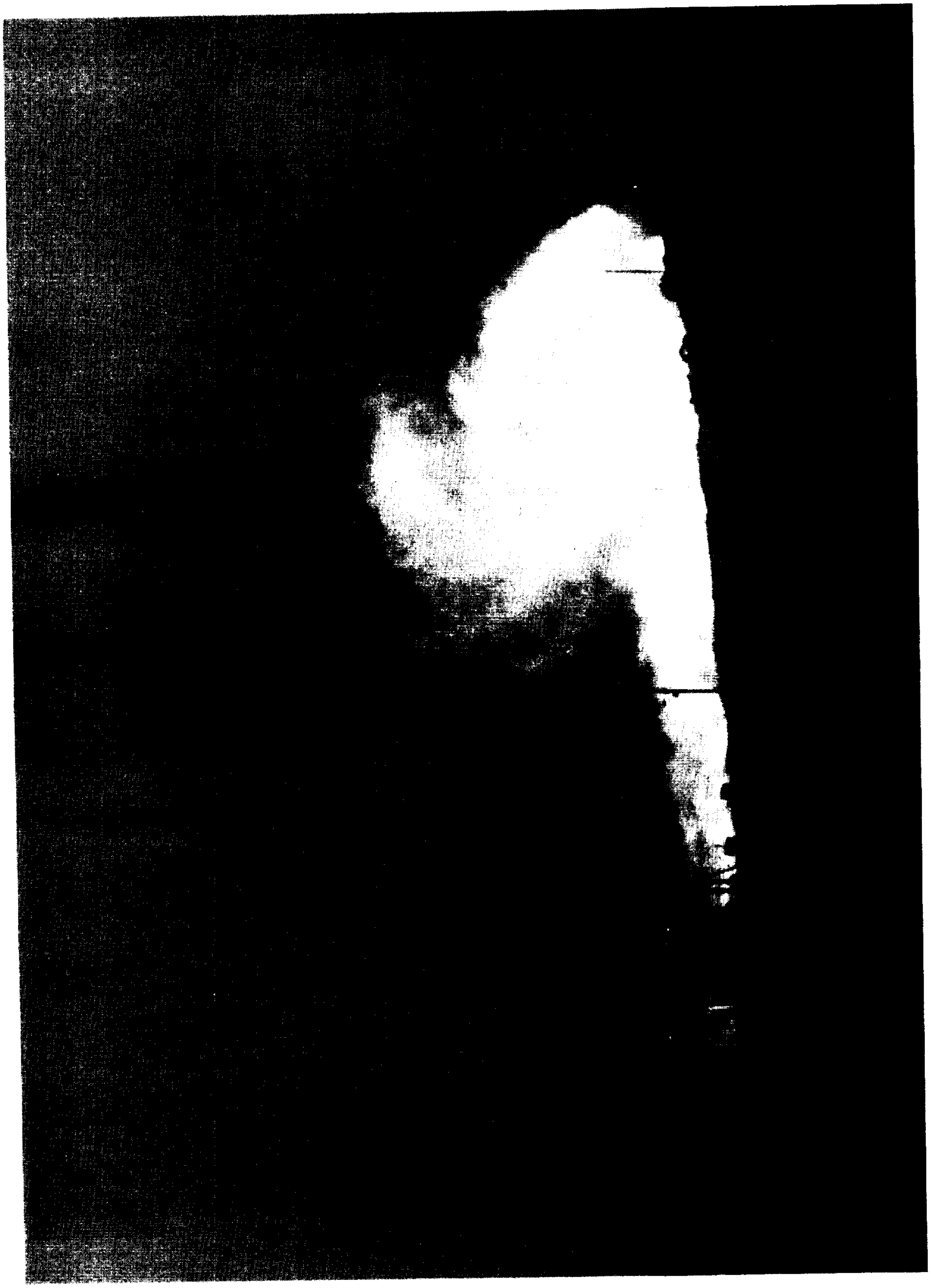

Figure 9. Full-Thrust TCA Test in Progress

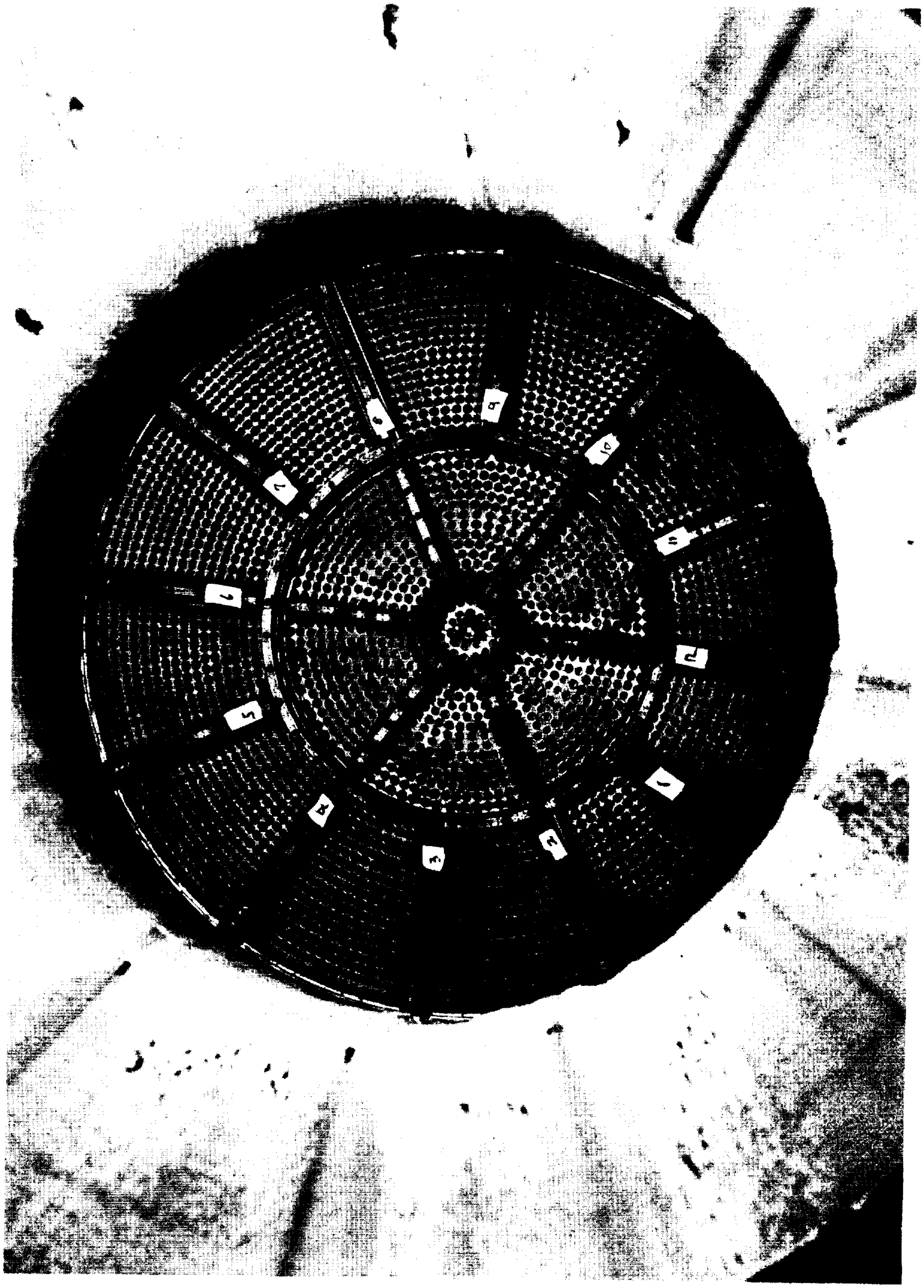

Figure 10. Over-All View of Thrust Chamber Injector S/N 020
After Eight Full-Thrust Firings

1966/3Q

2. Regeneratively-Cooled Combustion Chamber

The design technology and the status of the M-1 regeneratively-cooled combustion chamber from the initiation of the task in January 1962 through its termination during January 1965 are delineated in a separate report.(16)

1962/2Q

Experimental heat transfer studies were initiated at the outset of the program. Evaluations of hydrogen cooling characteristics were made under simulated M-1 Engine operating conditions. Also, procurement orders were placed for test specimens, which became available during the ensuing quarter.

The heat transfer evaluation program was the subject of a separate report.(17)

1962/3Q

Testing of the asymmetrically-heated tubes was initiated during the third quarter of 1962. It was completed on 30 November 1962.

1962/4Q

Several theoretical studies had also been completed by the end of 1962. These included:

The pump discharge requirement as a function of fuel velocity through the combustion chamber coolant jacket was established.

The percentage of film cooling required as a function of fuel velocity through the coolant jacket was established.

The relationship between pump discharge pressure and percentage of film cooling for a constant safety margin was established.

A survey of thrust chamber weight reduction through the use of high-strength materials was made.

The fuel film cooling effect upon the 1,200,000 lb thrust engine thrust chamber and engine performance was determined. This analysis included the determination of pump power and discharge pressure requirements and consequent engine weight. The results indicated that improved performance

(16) Pulliam, W. P., Regeneratively-Cooled Combustion Chamber, M-1 Engine, Aerojet-General Report No. 8800-48, 15 February 1966

(17) Thompson, W. R. and Geery, E. L., An Experimental Investigation on Heat Transfer to Cryogenic Hydrogen Flowing Turbulently in Asymmetrical Heated Straight and Curved Tubes, Aerojet-General Report No. TM 183-63-13, March 1963

1962/4Q

could be obtained with increased coolant velocity and less film cooling. A similar analysis was accomplished for the 1,500,000 lb thrust engine which showed that the combustion chamber in this application would have a pressure drop exceeding 300 psi. This would result in optimum over-all engine performance when turbine power requirements and losses from film cooling were considered.

The effects of nozzle shapes upon thrust chamber performance was studied. A sea-level separating nozzle with an area ratio of 40:1 was selected for the M-1 Engine.

The combustion chamber was redesigned toward the end of 1962 to provide greater engine thrust uprating potential. This new combustion chamber would operate at approximately 800 psi chamber pressure at 1,200,000-lb thrust or 1,000 psi at 1,500,000-lb thrust. Increased coolant velocity was being provided to improve performance potential. Regenerative-cooling will also be provided to the 14:1 area ratio to eliminate the need for film-cooling of the nozzle between the area ratios of 8:1 and 14:1.(18)

The combustion chamber support jacket was also redesigned for the 1,500,000-lb thrust application. This new design included a removable lightweight jacket for greater flexibility as well as lighter weight.

1963/2Q

The M-1 combustion chamber weld-braze program was outlined during the second quarter of 1963. It was planned to furnace braze the combustion chamber; however, the initial units would be hand fabricated because no furnace was available.

A combustion chamber mandrel (14:1 area ratio) was received in June 1963 (see Figure No. 11).

1963/3Q

Tubes for the first-article chamber arrived on 3 September 1963. They were cleaned and "laid-up" on the mandrel and tack welding operations began (see Figure No. 12).

Tubes of 347 stainless steel were obtained during the third quarter of 1963 and cut into equal lengths for a small tube-bundle for the combustion chamber brazing investigation. Also, material for the first combustion chamber jacket for the 14:1 chamber was received and fabrication started.

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(18) Green, M. I. and McGough, C. B., M-1 Thrust Chamber Heat Transfer Analysis and Design, Aerojet-General Report No. TCR 9621-015, 15 May 1963

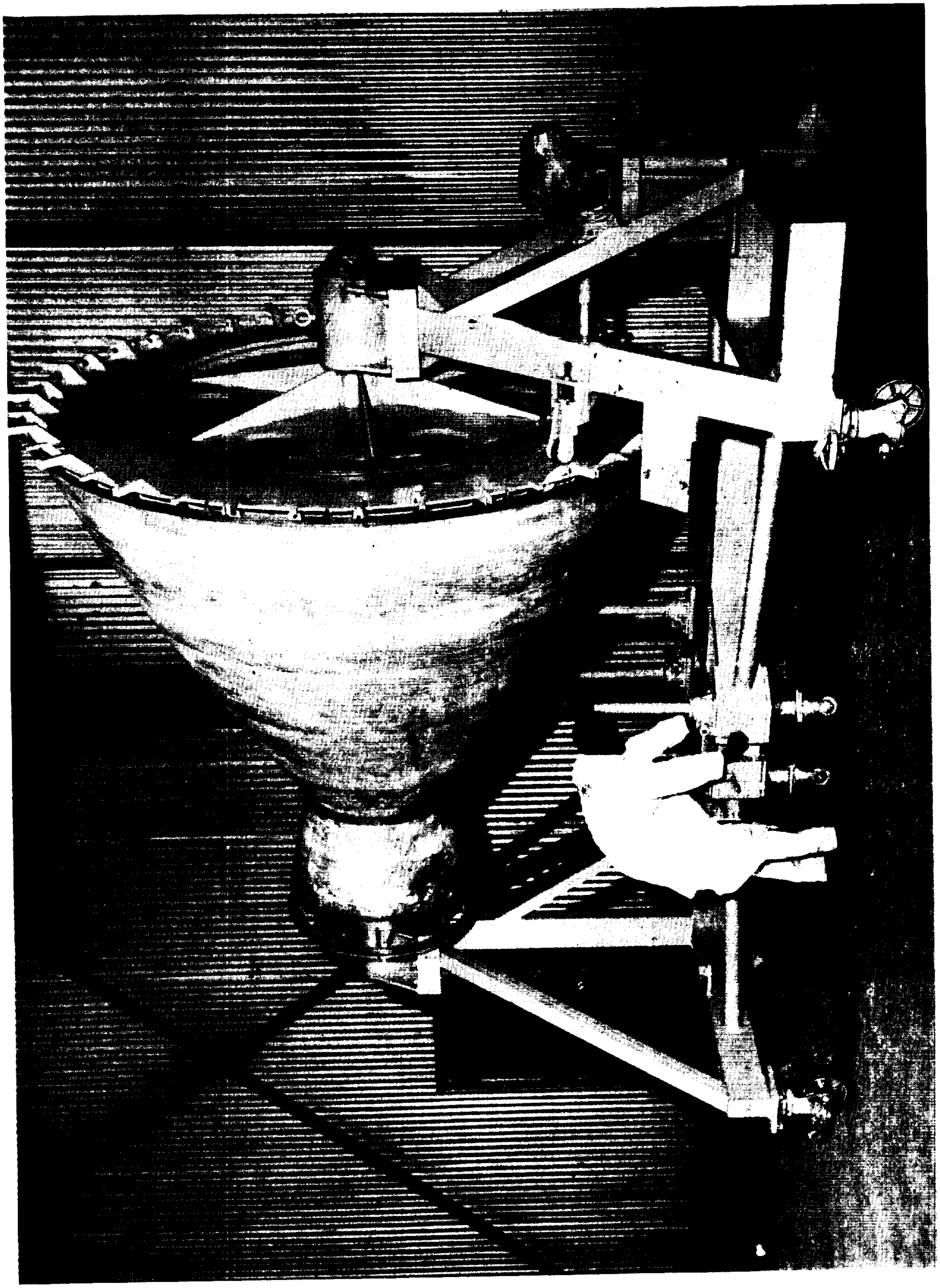

Figure 11. M-1 Combustion Chamber Mandrel

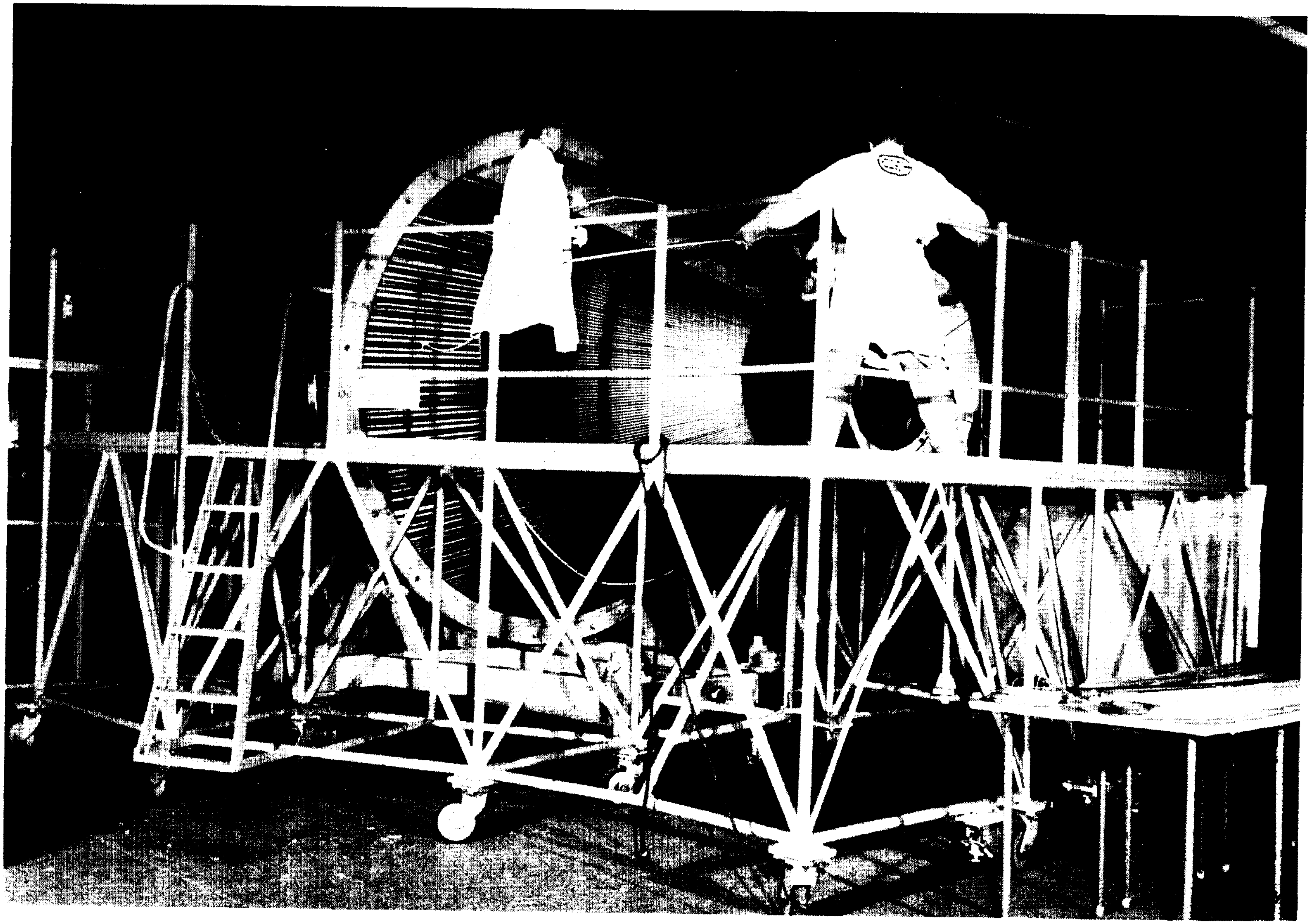

Figure 12. First Article Combustion Chamber Tubes Being "Laid-Up" on Mandrel

1963/4Q

Arc welding of the tube-to-tube joints and hand brazing on the first-article combustion chamber (S/N 013) was completed by the end of 1963. Tack welding of the tube bundle for the second-article combustion chamber (S/N 014) was also completed. The gasket holder, igniter windows, and forward adapter ring were installed.

1964

Installation of the torus, torus inlet transition, clevis gussets, aft flange instrumentation bosses, inlet splitters, inlet splice strip components, and final contour dimensions were accomplished on S/N 013 cooled combustion chamber during the third quarter of 1964; however, combustion chamber S/N 014 was designated as the first article because of torus problems encountered with S/N 013.

1965

All design and fabrication efforts were stopped as of 8 February 1965 because of program redirection. At that time, S/N 013 and 014 were awaiting final stress analysis of their respective components. On S/N 015, the welding of the tube-to-tube joints had progressed approximately 10-in. below the throat (see Figure No. 13).

3. Uncooled Combustion Chamber

1962

The design of an uncooled "workhorse" combustion chamber was initiated during the third quarter of 1962.

1963/2Q

All drawings and fabrication orders, including the final assembly drawing, for the uncooled combustion chamber were released and all long-lead forgings were received during the second quarter of 1963. One fuel flange had been rough-machined and another was approximately 33% completed. Other large components then being machined were the throat forgings and the chamber flanges. The forward and aft cones were received and were awaiting the completion of the throat forging, at which time they would be welded together.

1963/4Q

The first-article uncooled combustion chamber was completed and was successfully hydraulically tested during the fourth quarter of 1963. The second-article combustion chamber and the fuel flange and torus assembly were welded at the Aerojet-General Downey facility and passed X-ray inspection. The hardware was returned to the Sacramento facility for machining and final assembly.

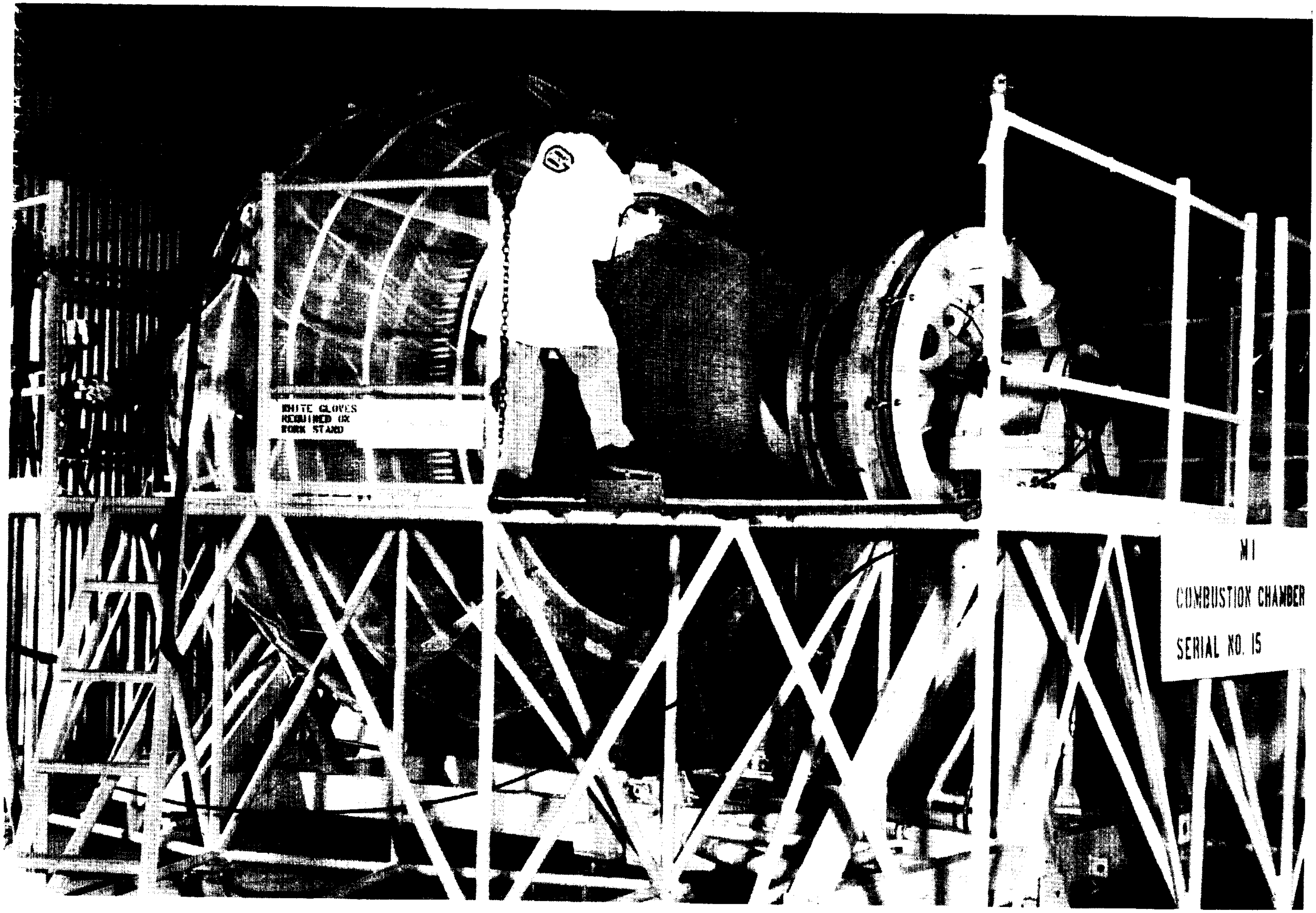

Figure 13. Welding of Tube-to-Tube Joints, S/N 015 Combustion Chamber

1964/2Q

The first-article combustion chamber was used in the uncooled thrust chamber assembly (S/N 001) which was tested on 20 June 1964 at Test Stand C-9. This combustion chamber (S/N 004) suffered repairable damage and directions were issued during the subsequent quarter to make the appropriate repairs.

1964/3Q

The second-article combustion chamber (S/N 001) was completed during the third quarter of 1964 and stored.

All major machining (inside contour and mating flange) was completed for the third-article lower chamber, P/N 293837. The fuel flange and torus assembly, P/N 284890, for this combustion chamber was received from the Aerojet-General Downey facility. All welding was completed on the torus assembly. Because of M-1 program redirection, fabrication of the fourth-article combustion chamber was cancelled and components were stored for possible future use.

1964/4Q

The repair of S/N 004 combustion chamber was completed during the fourth quarter of 1964.

1965/1Q

Because of the program redirection at the outset of 1965, S/N 001 and 004 were to be the only units to be reworked. S/N 001 was being used as a component in the acoustical testing program. Upon completion, the unit would be reworked to add side load clevi, inlet stiffeners, and gasket holder with a reduced film cooling pattern.

The S/N 004 torus and flange assembly had the gasket holder removed, P_c taps were repaired, and the inlet stiffeners were added. The lower chamber, S/N 004, had the secondary film cooling injection ports modified to accept the film coolant. The film coolant rings were changed from circumferential to axially drilled orifices because of heat transfer re-analysis of film coolant conditions. The number of orifices increased from 286 (0.154-in. dia) to 640 (0.109-in. dia) to obtain a more effective coverage. Coolant velocity was 900 ft/sec.

1965/2Q

The S/N 004 torus and S/N 001 lower chamber assembly were completed and delivered to Test Stand H-8 for thrust chamber assembly mock-up during the second quarter of 1965. Rework of S/N 004 lower chamber assembly to add instrumentation bosses, a secondary film coolant ring, and side load clevi with related support plates was also completed.

It was decided to fabricate one uncooled combustion chamber incorporating an ablative liner after a heat transfer analysis, which included

1965/2Q

coatings and ablative materials, was completed. The results showed that only the ablative material would be satisfactory for a 5 sec firing with no film cooling. Serial Number 001 torus and S/N 002 lower chamber assembly were selected as the units to be reworked in support of this effort. The torus was returned from acoustical testing and was reworked to add inlet stiffeners, drain bosses, and a gasket holder.

Lower chamber assembly, S/N 002, was reworked to plug secondary film cooling holes, instrumentation ports, and gun ports. The side load clevi, throat reinforcing rings, and new instrumentation ports were welded in place.

1965/3Q

The inside diameter on the lower chamber assembly S/N 002 was contour-machined and the converging portion of the ablative liner bonded in place during the third quarter of 1965. The converging liner was fabricated at Aerojet-General from billets machined into ring segments. These segments were bonded to each other and then to the chamber.

The diverging liner which was fabricated "in-house" developed circumferential cracks during final machining. At this point, it was decided to terminate work with this unit and to order an additional tape-wrapped diverging liner from a supplier.

The liners fabricated by Haveg-Reinhold, which include one converging cone, two diverging cones, and one torus liner were completed.

All fabrication of S/N 013 torus assembly, including the ablative liner, was completed.

1965/4Q

Serial Number 004 lower chamber was assembled to S/N 004 fuel torus and flange assembly, hydrotested, and built up into S/N 004 thrust chamber assembly (with S/N 018 injector) during the fourth quarter of 1965.

The ablative combustion chamber consisting of S/N 002 lower chamber and S/N 001 fuel torus and flange assembly was also completed.

1966

The design for adapting the fluorine ignition system to the torus and flange assembly was completed early in 1966. Also, the incorporation of the fluorine system to the ablative, which had already been installed into Test Stand H-8, was completed.

Rework of S/N 001 chamber was completed through the level of welding during the second quarter of 1966. The chamber was then stored.

1966

The S/N 004 combustion chamber was hydrotested and cleaned during July 1966. Also, a design modification to incorporate four additional P_c5 bosses into the fuel flange and torus assembly was completed.

4. Nozzle Extension

The design of the M-1 hot-gas cooled nozzle extension is comprehensively detailed in a separate report.(19)

1963/2Q

The design of a sample tube bundle was completed during the second quarter of 1963. This tube bundle consisted of 10 tubes (0.875-in. dia, 0.010-in. wall thickness and 4.8-in. long) brazed together and seven reinforcing bands (0.190-in. thick and 1.50-in. radius). Half of the round torus was attached to the forward end of the tube bundle to provide hot gas entry.

An interim nozzle extension for sea-level testing to evaluate the flange joint, hot gas torus, and heat transfer to the turbine exhaust was also designed. Heat transfer studies of the nozzle extension tubes were completed and a fabrication order was issued for tubes to assemble the five interim nozzles.

1963/3Q

Two brazing test programs were initiated during the third quarter of 1963. The brazing test program (Test Series A) utilized a full-length tube and axial band sample to determine the compatibility of coolant tube and band materials. The nozzle extension program (Test Series B) was used to evolve an optimum brazing procedure and age hardening of Inconel 718 banding.

The interim nozzle design effort has continued on schedule. A design for a scale model of the nozzle-to-chamber interface flanges was initiated.

A brazing symposium was conducted at Aerojet-General to discuss brazing of the M-1 nozzle extension segments. This symposium was open to companies interested in submitting proposals for brazing the nozzle segments.

(19) Lamson, R., M-1 Nozzle Extension, Aerojet-General Report No. 8800-55, 15 March 1966

1963/3Q

By the end of September 1963, the first five components of the nozzle extension program (Test Series B) were completed and the thermal analysis program for the 40:1 nozzle extension coolant tube was continuing on schedule. This program consisted of a parametric heat transfer study of the effect of operating conditions and engine shutdown transients upon the nozzle extension. The thermal analysis of nozzle extension and combustion chamber interfaces also continued as scheduled. This program determined the optimum material and design combination for the flanges, considering the predicted thermal conductance efficiency of the flange joint.

In addition, a vibration study program was initiated for the M-1 thrust chamber assembly for the purpose of computing bell modes and translating them into equations of forced and self-excited dynamic responses in the nozzle. This provided a guide for better definition of potential dynamic problems in the M-1 nozzle.

1964/2Q

A new as well as a revised operating envelope were reviewed during the second quarter of 1964 to determine the effect of the latest changes upon the maximum tube wall temperatures. These envelopes included a reduction of the gas generator combustion efficiency to 97% of the theoretical value and an allowance of 3 lb/sec of liquid hydrogen leakage from the fuel turbine bearings. The new envelope showed the effects of a gas generation mixture ratio control valve. Maximum turbine exhaust temperature was predicted to be approximately 790°F with the additional control valve, and 905°F without it. This compared with a previous maximum value of 1050°F. These reductions would directly affect the temperature of the skirt inlet manifold, flange, and unheated periphery of the tubes.

A layout of the nozzle extension with 480 taper tubes was started. This layout included the addition of shims between the tubes for structural support. The contour of the taper tubes was also determined.

The maximum tube wall temperatures without shims were predicted by taking several possible operating conditions from the high chamber mixture ratio, low coolant weight flow region of the envelopes. The resulting maximum temperature was found to be 2000°F for the envelope without the control valve; this was a reduction of 185°F as compared with the previous envelope. The addition of the gas generator control valve was found to reduce the temperature an additional 30°F.

The relationship between the maximum shim temperature, the thickness, and brazed contact length of the shims, which would be placed between the nozzle extension tubes, was calculated. These calculations covered thicknesses at .060-in. to .120-in. and contact lengths of 0.1-in. to 0.5-in. (see Figure No. 14). Upon the basis of this analysis, a minimum

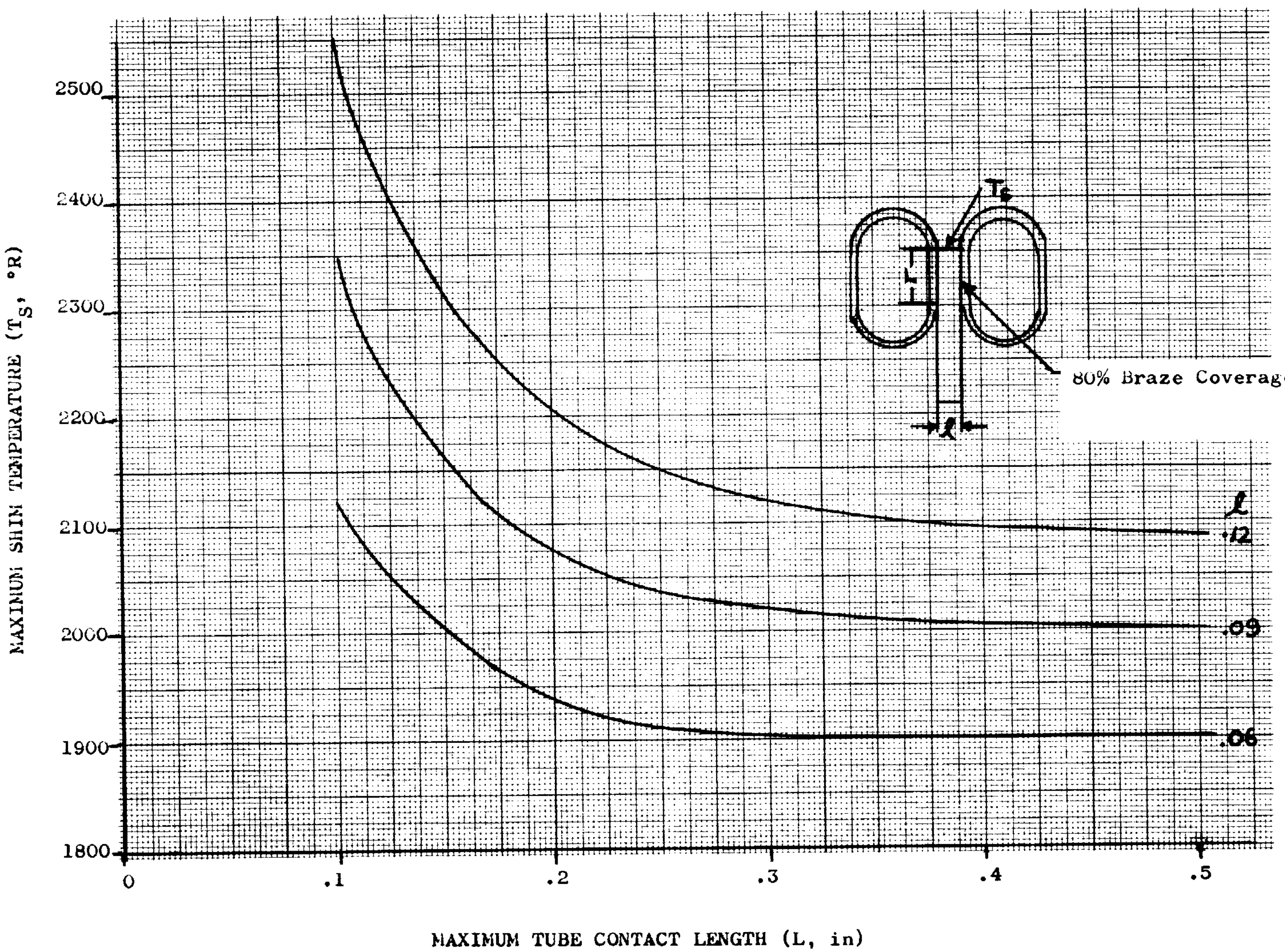

Figure 14. Maximum Shim Temperature vs Maximum Tube Contact Length

1964/2Q

contact length of .2 to .3-in. was recommended. The temperature calculated for a .060-in. thick shim was 1440°F. For thicker shims, the temperatures would increase at the rate of 4°F per .001-in.

The effect of adding 0.083-in. thick by 25-in. long shims and a 3-in. long transition section of the 480 tapered tube nozzle extension to the M-1 gas-cooled nozzle extension design was analyzed during the third quarter of 1964. The over-all change in pressure drop and maximum tube temperature caused by these design changes was not significant. The maximum shim surface temperature would be less than 1550°F. A layout of this tube geometry was completed. However, immediate goals of the M-1 Program were such that further effort in connection with the gas-cooled nozzle extension design was postponed. It was never resumed.

Also, a test fixture was made for testing the flow of gases at the inlet of the tubes. The object of the fixture was to determine the best contour in this area for the nozzle extension. Also, to determine if vanes were required to direct the flow at the inlet.

5. Gimbal

1962/2Q

Two gimbaling device concepts were pursued at the outset of the program. One concept incorporated roller bearings which would allow motion along two axes normal to each other. The other was a ball-and-socket type with an anti-spin lock.

The design and structural analyses of the roller bearing gimbal were completed during the second quarter of 1962. Layouts of both the roller bearing and spherical bearing gimbals were completed during the ensuing quarter. A mock-up of a ball-and-socket gimbal version was also completed.

1962/4Q

All design effort for the roller bearing gimbal was suspended at the end of 1962 pending the completion of a thrust-vector control study then under way.

1963

A decision was made during the third quarter of 1963 to use a 15-in. gimbal bearing radius. This decision was made upon the basis of results from vibration and natural frequency analyses of the engine configuration. Also, material for four gimbal simulators was received.

1963

Two units of the first-article gimbal simulator were completed during the fourth quarter of 1963. One was installed on the first uncooled thrust chamber assembly and the second unit was for the second thrust chamber assembly. A fabrication order for two of the second-article gimbal simulators was released in December 1963. These gimbal simulators were to be used on the 25:1 area ratio thrust chamber assemblies.

1962/2Q

6. Wedge Thrust Chamber

The interim wedge chamber model was test fired during the second quarter of 1962. This test ran for 2.166 sec and achieved 69.3% of the full design operating level (based upon a design pressure of 1000 psia). The injector pattern was like-on-like fuel impingement and showerhead oxidizer orifices.

Post-fire inspection disclosed that 75% of the coolant passage tubing had separated from the aluminum jacket and ruptured at the terminal castings. Also, turbine and injector flange material immediately downstream of the center baffle was severely eroded.

As a result of the data obtained during the first test, design modifications of the interim model combustion chamber tube assembly and the cast aluminum jacket were initiated to preclude the separation of the tube bundle from the aluminum jacket.

The chamber tube bundle for the interim wedge chamber was assembled, wire loops were placed, and the fuel torus was mounted during the third quarter of 1962. The tube bundle was then sent to a vendor for brazing. Upon completion of the operation during the ensuing quarter, the assembly was distorted. It was successfully restored to proper configuration and the final braze cycle was accomplished.

1963

This effort was deleted from the M-1 Program early in 1963.

7. Stage Ignition Device

The development of the stage ignition system for the M-1 thrust chamber is delineated in a separate report.(20)

(20) Trafzer, T., Stage Ignition System, M-1 Engine, Aerojet-General Report No. 8800-11, 15 September 1965

1962

The design of the initial stage ignition device was completed during the third quarter of 1962 and four units were fabricated.

The first-stage igniter test was conducted at the Aerojet-General Azusa test facility on 31 October 1962 for the purpose of ascertaining test stand and procedural problems. Ignition was successful although some erosion of the igniter occurred.

1963/1Q

Final assembly drawings and pattern drawings for the premix igniter were released along with the drawings for the premix igniter test fixture early in 1963. A temperature profile analysis for the thrust chamber igniter injection tube was also completed for a mixture ratio of 0.8:1.

By the end of the first quarter of 1963, 15 stage igniter tests had been completed.

1963/2Q

Three premix igniter test fixtures were completed and delivered to the Azusa facility during the second quarter of 1963.

A gas generator igniter and a thrust chamber assembly swirl-mix igniter were tested during April 1963. Four gas generator igniter tests (two cold flow and two firings) were conducted in May and the first coaxial igniter was tested in June. The second coaxial igniter was successfully fired three times at back pressures of 700 psia to 900 psia. Also, an impinging stream gas generator igniter was tested using the back pressure plate. Post-test inspection revealed severe erosion of the combustion cavity. Testing of this type of gas generator igniter was discontinued.

1963/3Q

The first two prototype coaxial igniters were completed during the third quarter of 1963 and installed on two gas generators. The remaining 10 igniter assemblies were in various stages of fabrication. One experimental premix igniter was also completed.

By the end of September 1963, 47 tests had been completed and testing at the Azusa Proving Grounds was suspended. Testing operations were transferred to Sacramento where more advanced test equipment and improved test operations were available.

1963/4Q

A change in the basic design philosophy was instituted during the fourth quarter of 1963. The design mixture ratio was raised from 0.8 to 3.0-30.0 as a result of the September 1963 M-1 Quarterly Design Review Meeting. Redesign of the igniters was initiated. Redesign of the coaxial igniter was discontinued at this time because it offered little chance of success at high mixture ratios as a result of its high potential for burnout. The use of any cooling technique would have produced unwarranted design complexities.

1964/1Q

Detailed fabrication drawings of the thrust chamber igniter were completed during the first quarter of 1964. The final igniter assembly is shown on Figure No. 15. This assembly consisted of three parts; a body, sleeve, and the spark plug. The body included the barrel, which formed the outside of the oxidizer annulus. The sleeve and barrel formed the fuel annulus. Premixed fuel and oxidizer were injected at an injector ring upstream of the spark plug tip. Fuel film coolant orifices were located at the end of the sleeve. Single conical seals were used in the propellant inlet ports. The body had a double conical seal surface for mating to the thrust chamber igniter ports. The long barrel of the igniter was inserted into the port on the chamber.

The M-1 stage igniter development test plan was completed in January 1964. There were 363 tests planned for Test Stand G-7 at the Sacramento Facility. Cold flow testing was scheduled to begin in June 1964. The program plan included 200 tests for reliability demonstrations starting in mid-1965.

Design of gas generator stage igniters was initiated early in March 1964. Basic design was similar to the thrust chamber design; however, the different operating environment of the gas generator made exact duplication impractical.

The film coolant igniter concept was discontinued because ignition detection was difficult, there was little mixture ratio control, and the cost of developing an alternative concept was not warranted.

1964/2Q

All detailed fabrication drawings for the experimental thrust chamber igniters were released during the second quarter of 1964 and major design effort was considered to be completed. Preliminary design of the gas generator igniter was also completed during this quarter.

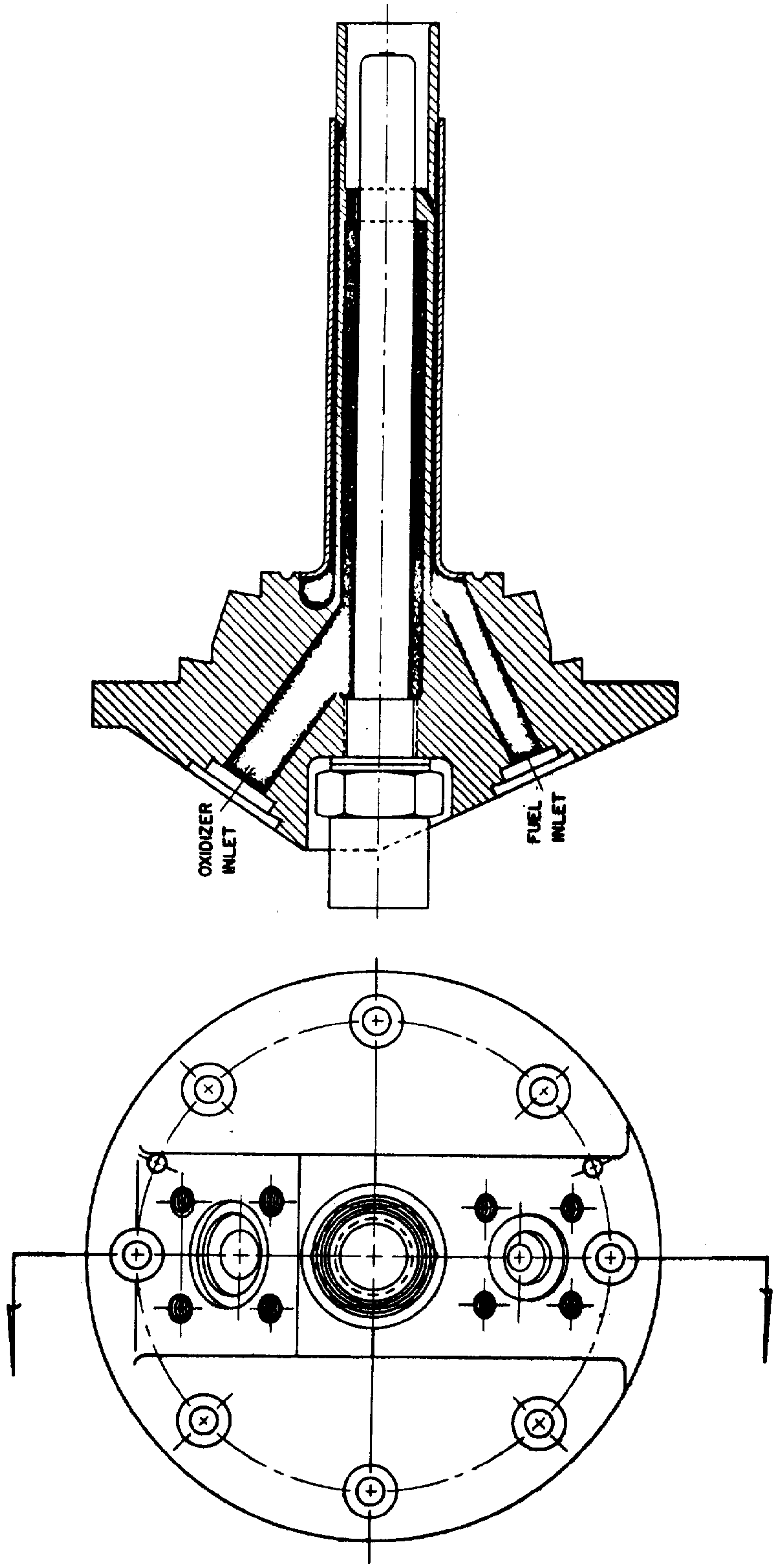

Figure 15. TCA Igniter System

1964/2Q

Fabrication of the first experimental thrust chamber igniter (see Figure No. 16) was completed. It was proof, leak, and flow-tested. Then, it was delivered to Test Stand J-1A and igniter testing commenced on 2 June 1964 with a cold flow test. The first hot firing was accomplished on 14 July 1964.

1964/3Q

In August 1964, all development effort for stage igniters was deferred pending a design decision. This decision concerned hypergolic as opposed to spark ignition techniques. Postponement was also based upon immediate program goals which could be achieved using pyrotechnic igniters. Spark or hypergolic systems were not required during the initial phases of engine development.

All detailed design drawings of the gas generator igniter were released during the third quarter of 1964 and the design effort was considered to be completed. Also, Test Stand J-1A was converted for uni-element testing.

The stage igniter development effort was not resumed.

1964/2Q

8. Uni-Element Development

The design, fabrication, and testing of M-1 uni-element thrust chamber injector assemblies from the initial design concepts on 16 June 1964 through completion of the test program are delineated in a separate report.(21)

Because coaxial injectors required a considerable amount of time to fabricate as well as repair or modify, it was desirable to initiate a coaxial element development test program. This test program was for the purpose of evaluating element characteristics during a hot firing environment prior to the element being committed to a full-scale injector. Appropriate physical alterations were made to each element to eliminate element erosion.

1964/3Q

Fabrication of uncooled hardware was initiated on 3 August 1964. On 21 August 1964, the first assembly was completed, hydraulically tested (see Figure 17), and delivered to Test Stand J-1A, which was converted for uni-element testing. The second assembly was completed, hydraulically tested, and delivered to Test Stand J-1A on 8 September 1964.

(21) Kovach, R., M-1 Uni-Element Program, Aerojet-General Report No. 8800-57, 15 February 1966

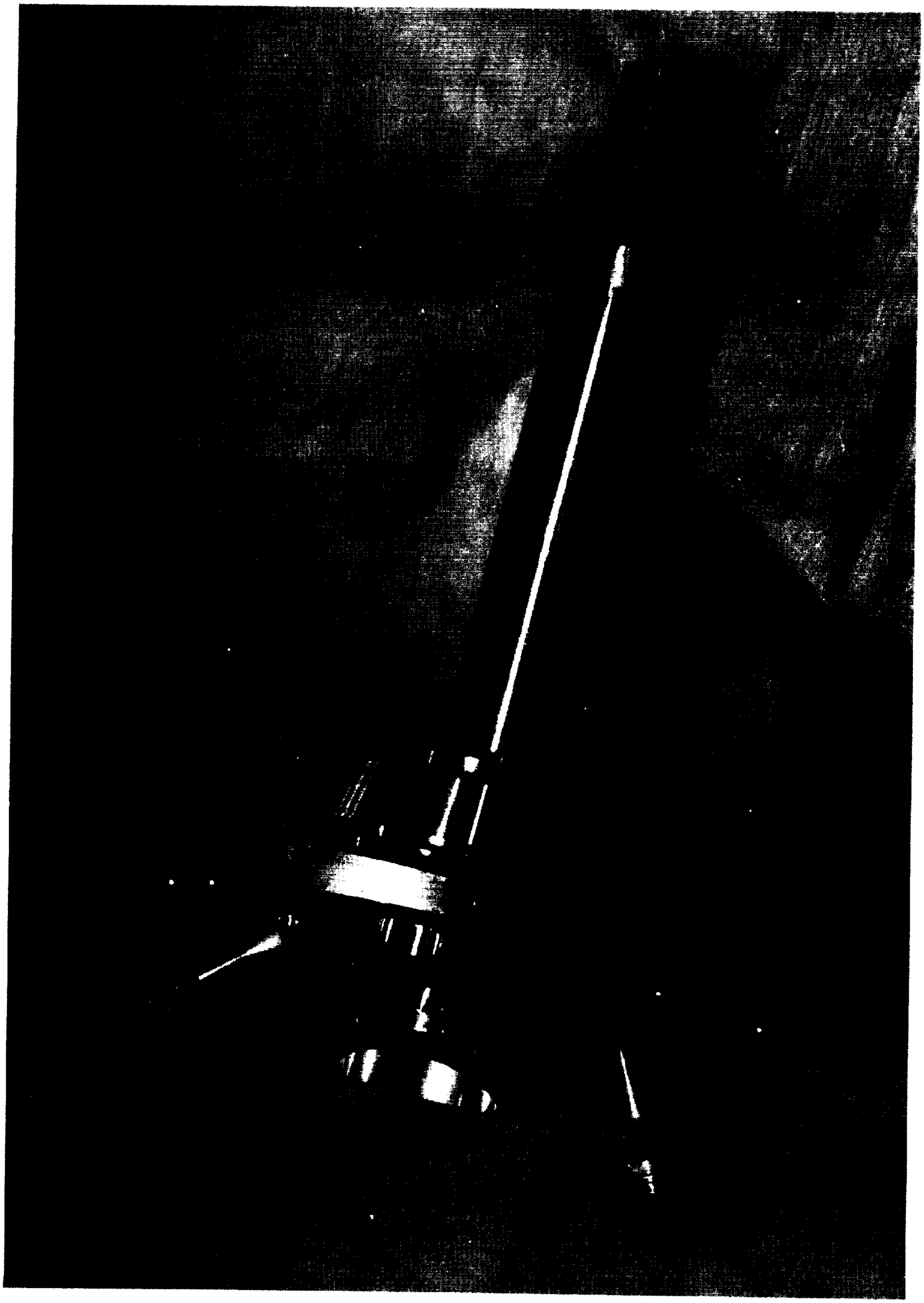

Figure 16. First Experimental Thrust Chamber Igniter

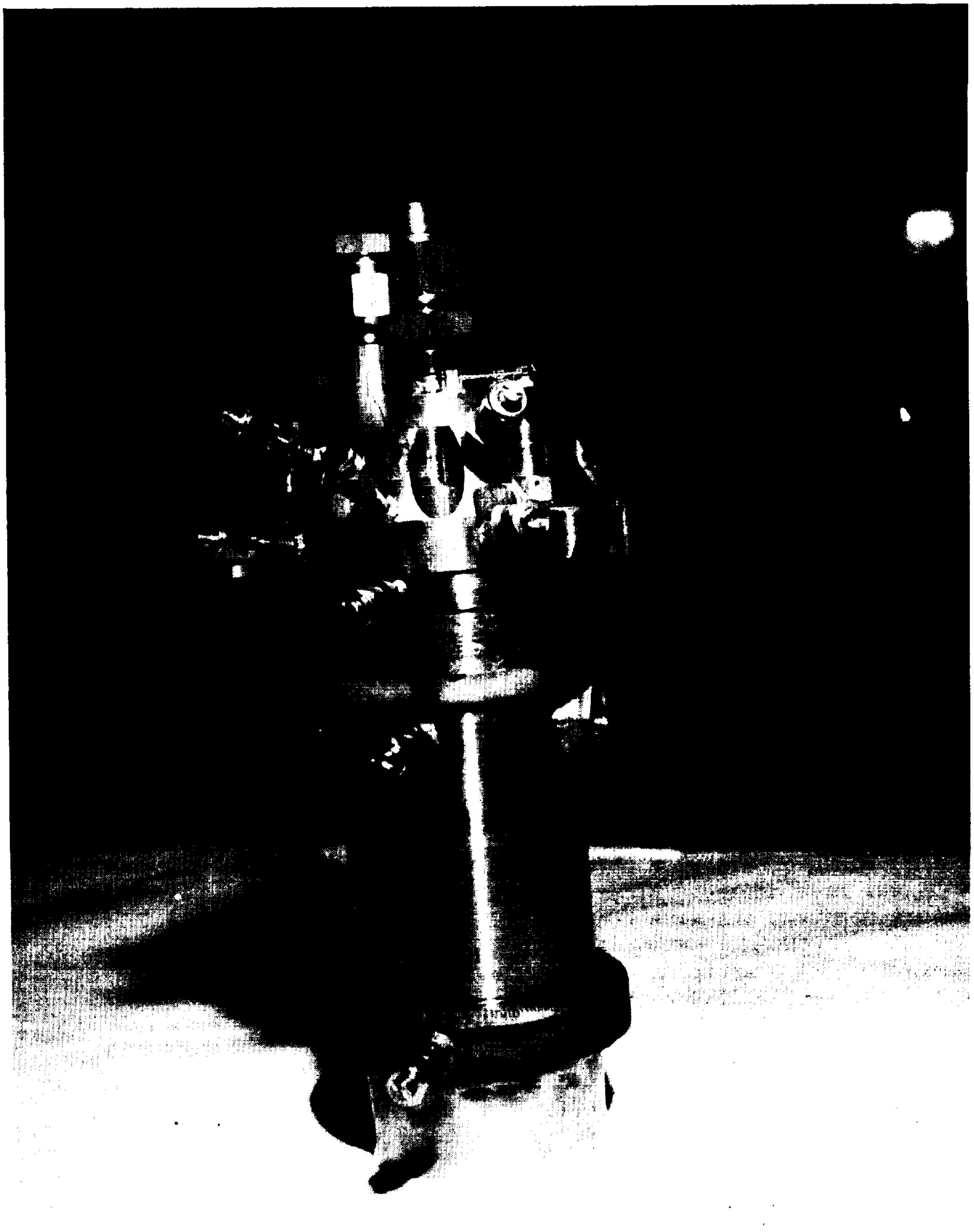

Figure 17. First Uni-Element Hardware Assembly

1964/3Q

The initial hot firing with a S/N 013 type thrust chamber injector element was conducted on 7 October 1964. During the following month, seven elements were subjected to a total of 14 tests. Test Stand J-1A was then modified to test gas generator injector element S/N 022 by removing the gaseous hydrogen mixer and replacing it with a straight line.

1964/4Q

Element S/N 022 was test fired on 16 November 1964. By the end of 1964, four gas generator injector elements has been tested for a total of 13 times.

1965/2Q

The program was completed on 18 May 1965 and all test hardware was cleaned, packaged, and stored. Sixteen thrust chamber injector elements had been tested for a total of 36 tests; six gas generator injector elements were tested for a total of 19 tests.

1964

9. M-1 Steam Hydrogen Experimental Program

This program has been summarized in a separate report.(22) This effort began during the second quarter of 1964 with plastic model tests of the M-1 extension entrance section. The plastic nozzle extension entrance model was installed into a gaseous nitrogen smoke system during the ensuing quarter.

A steam-hydrogen flow loop was completed at the end of July 1964 and the system was cold flowed with gaseous nitrogen. Inconsistent bulk temperature measurements were obtained during initial testing and the instrumentation was re-examined. After corrective action was taken, the bulk temperature measurements were consistent.

A plastic section simulating the torus feed manifold was fabricated and tested during the fourth quarter of 1964.

1965

This effort was cancelled during the first quarter of 1965 as a result of M-1 program redirection.

(22) Visual Studies of M-1 Nozzle Extension Coolant Tube Entrance Designs, Aerojet-General Report No. DTDR 9650-015, 20 January 1965

B. GAS GENERATOR

SUMMARY

The M-1 gas generator development program was initiated to provide a source of high pressure, homogenous combustion gases to drive the fuel and oxidizer turbopump turbines during the operation of the M-1 Engine. Both the fuel and oxidizer turbines were to be driven in series with a single gas generator. The fuel turbine was designed to deliver 90,000 hp and the oxidizer turbine 27,000 hp. The turbines drive their respective fuel and oxidizer pumps which, in turn, supply high pressure liquid hydrogen and liquid oxygen to the engine thrust chamber assembly as well as to the gas generator.

To achieve the turbine horsepower requirements, the gas generator was nominally designed for 110.4 lbm/sec total propellant flow rate at a mixture ratio of 0.8 to supply 1000°F combustion gases. Gas generator chamber pressures were recorded from 0.6 to 1.0 at steady-state conditions. Approximately 98% of theoretical combustion efficiency was achieved with the final coaxial injector design based upon characteristic exhaust velocity calculations. Typical combustion gas exit temperatures measured at the gas generator outlet ranged from 900°F to 1300°F at nominal mixture ratio.

A coaxial injection element injector design with a cylindrical fuel film-cooled combustion chamber proved to be successful and was selected as the prototype gas generator from three basic injector concepts. Other concepts evaluated were the multi-orifice type injector and a pentad, large-thrust-per-element injector design.

Severe injector face and combustion chamber wall erosion occurred during the initial test of the large-thrust-per-element injector concept. Although design modifications could have solved the gas generator erosion problem, no further development was attempted because of the long combustor mixing length that would have been required to achieve homogenous gas temperature in front of the turbine inlet.

Minor injector face erosion occurred with all pattern variations of the multi-orifice injector design. Of the multi-orifice injector patterns tested, the uniformly spaced, like-on-like impinging doublet with radially aligned fuel-oxidizer-fuel impingement fans encountered the least face erosion. It was indicated from work with the J-2 and RL-10 thrust chambers as well as various NASA/LeRC injectors that favorable combustion performance and stability was being obtained with coaxial injection element designs for the liquid oxygen/liquid hydrogen propellant combination. Therefore, it was assumed that a coaxial gas generator could be developed in less time and at lower cost and further development effort with the multi-orifice designs was terminated.

During gas generator development tests of unbaffled injector designs, tangential modes of high frequency combustion instability occurred in four tests. Two of these tests were with multi-orifice injectors and the remaining two tests were with S/N 015 and S/N 020 coaxial injector gas generator assemblies. High frequency combustion instability spontaneously occurred in all four tests during the start transient when the injection velocity ratio (fuel injection velocity/oxidizer injection velocity) was less than four. Because of a shift in the test conditions during all four unstable, unbaffled injector tests, the injection velocity ratio exceeded the normal steady-state values. When the injection velocity ratio exceeded approximately 9 in the three tests with the first tangential mode and when it exceeded 5.7 in the test with the second tangential mode, the high frequency combustion instability was spontaneously suppressed during all four tests. Thereafter, combustion continued with only the normal combustion noise until the end of the tests. Based upon observations during these four unbaffled injector tests, it was suspected that a correlation existed between injection velocity ratios and the occurrence or disappearance of high frequency combustion instability. The stabilizing effect of injection velocity ratio appears to be primarily caused by liquid phase mixing and liquid oxygen droplet vaporization phenomena. Liquid oxygen combustion dynamics are suspected of being the primary cause of high frequency combustion instability.

Several coaxial injection gas generator element designs, one of which was selected for the prototype gas generator, were evaluated by using a single element injector test apparatus. Several element designs with nominal injection velocity ratios from 15 to 20 were rejected because of their severe chugging characteristics. A nominal velocity ratio of 10 was selected for the prototype gas generator assemblies. This lower value was achieved by decreasing the oxidizer injection area to obtain a higher oxidizer injection velocity, thus resulting in a lower fuel/oxidizer velocity ratio.

Throughout the initial gas generator development test series, excellent low frequency combustion stability characteristics were demonstrated by the prototype coaxial gas generator assembly. The measured injector pressure drops of 215 psia and 240 psia for the fuel and oxidizer, respectively, were obtained during nominal gas generator operation. When the gas generator exhaust duct downstream of the sonic gas generator stabilizing nozzle was replaced with the turbopump turbine inlet test manifold, attempts were made to maintain all other test facility and hardware systems intact and to follow earlier, successfully demonstrated test procedures. However, when the turbopump development test series with gas generator drive was initiated, a persistent low frequency combustion oscillation phenomenon was experienced, but the steady-state amplitude of the oscillations ($\pm$30 psi at 1145 P_c, 120 cps) were not detrimental to turbopump operation. Seven oxidizer turbopump and two fuel turbopump development tests were conducted with gas generator drive. However, the nature and origin of the low frequency combustion oscillations were not fully understood at the termination of the effort.

A separate report(23) delineates the technology applied in designing and testing the 120,000 horsepower liquid oxygen/liquid hydrogen gas generator for the M-1 Engine (see Figure No. 18).

CHRONOLOGY

1962/4Q

Design of the large-thrust-per-element injector system was completed during the fourth quarter of 1962 and all long-lead components were ordered. Concentric ring type injector S/N 0001 was completed as well as three chamber assemblies.

1963/1Q

Design of the coaxial injection system was completed during the first quarter of 1963, the top assembly drawing was released, and long-lead component purchase orders placed. The first of the eight multi-orifice injectors being fabricated was completed. One short gas generator combustion chamber was also completed.

1963

Serial Number 004 multi-orifice gas generator assembly was completed in April 1963 and installed into Test Stand C-6. It was cold flow tested on 9 May 1963. The first hot firing was conducted on 17 May 1963 and another was accomplished on 20 May 1963.

Gas Generator S/N 003 was tested on 11 June 1963. It had baffles installed but the splash plate removed.

Nine tests were conducted during the third quarter of 1963 and the other with a large-thrust-per-element pentad injector.

The first cold flow test at Test Stand C-9 was conducted with gas generator S/N 007 on 8 October 1963. The first hot firing was completed on 18 October 1963, the second on 22 October, and the third on 23 October. Two more tests were made on 11 and 18 November 1963, respectively.

By the end of 1963, S/N 001 multi-orifice gas generator assembly was completed, S/N 003 injector body had been reworked, and S/N 016 had been completed.

1964

The next test conducted was on 25 April 1964 at Test Stand C-9 with gas generator assembly S/N 004.

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(23) Ito, J. I., Development of LO_2/LH_2 Gas Generator for the M-1 Engine, NASA Report No. CR 54812, 1 June 1966

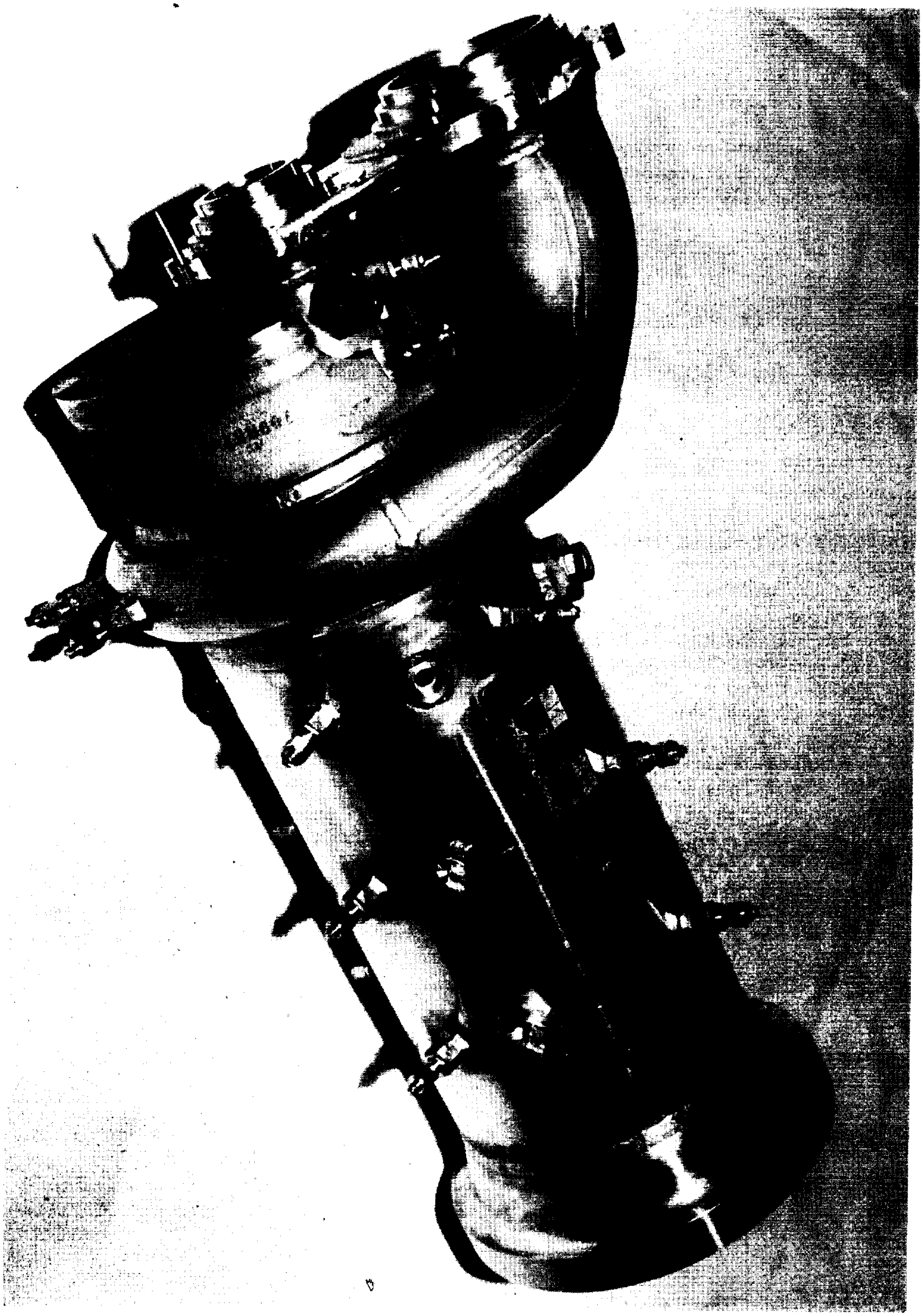

Figure 18. M-1 Gas Generator

1964

The testing, improvement, and fabrication of gas generators continued throughout 1964. Three gas generator assemblies (S/N 017, 017A, and 018) were fired during November and December 1964.

1965

In the first quarter of 1965, three gas generator assemblies (S/N 017A, 020, and 022) were fired at Test Stand H-8. It was demonstrated that S/N 022 was adequate for turbopump testing and it was selected as the prototype. Fabrication was initiated for three additional gas generators that were identical to gas generator S/N 022.

Testing of S/N 022 gas generator assembly with hydraulically-actuated prototype valves was initiated at Test Stand E-3 early in June 1965 with cold flow testing. The first hot test was conducted on 11 June 1965. A second hot firing was successfully conducted on the same day, with a third one accomplished on 25 June 1965.

The gas generator assembly checkout test series at Test Stand E-3 was successfully completed on 6 July 1965. Gas generator assembly S/N 026 was completed on 20 July 1965 and stored as a back-up unit for S/N 002 and 025 gas generator assemblies.

Seven gas generator assembly tests were conducted at Test Stand E-1 during the fourth quarter of 1965. The Test Stand E-1 checkout test series was successfully completed on 26 October 1965 using S/N 026 gas generator assembly. Testing of the fuel turbopump assembly with gas generator drive of the fuel turbine commenced on 6 December 1965 and was concluded on 22 December 1965.

C. HEAT EXCHANGERS

CHRONOLOGY

1962

The analytical method for the design of heat exchanges providing pressurization gases for the propellant tanks was established during the second quarter of 1962.

1963

The investigation of design criteria was started early in 1963 and efforts were extended to include theoretical heat exchanger design analysis.

A preliminary design concept was completed during the second quarter of 1963. This concept was based upon the use of two hot gas ducts feeding the thrust chamber assembly skirt. Two heat exchangers were proposed; one unit in each duct with half the pressurants and half the hot gas flowing in each unit. The two units were designed to the following specifications:

Hot gas flow:	110.4 lb/sec
Coil pressure drop:	10 psi
Hydrogen weight flow:	11.8 lb/sec
Hydrogen outlet temperature:	400°R
Hydrogen outlet pressure:	700 psia
Oxygen weight flow:	29.5 lb/sec
Oxygen outlet temperature:	800°R
Oxygen outlet pressure:	400 psia

Inlet conditions were based upon those specified for the turbopump discharge, considering probable pressure drops from the pump to the heat exchangers.

The principal considerations in the design analysis during the third quarter of 1963 was the location of the heat exchanger. The locations considered included: between the fuel and oxidizer turbines; at the oxidizer turbine outlet; in the hot gas line to the chamber skirt; and in the skirt inlet manifold. A tentative recommendation was made that the heat exchanger be located at the oxidizer turbine outlet.

1963

A heat balance analysis was completed for the M-1 autogenous heat exchangers during the fourth quarter of 1963. Hot gas temperature drop was evaluated as a function of gas generator flow rates, hot gas inlet temperature, and pressurant flow rates.(24)

1964

A parametric analysis was conducted during the first quarter of 1964(25) and a heat transfer analysis was completed during the second quarter of 1964(26).

In the third quarter of 1964, this effort was postponed indefinitely because of the immediate goals of the M-1 Program. The work was not resumed.

(24) M-1 Autogenous Heat Exchanger Heat Balance, Appendix A, Aerojet-General Report No. 4014-02Q-4 (Quarterly Technical Progress Report)

(25) M-1 Heat Exchanger Parametric Analysis, Appendix A, Aerojet-General Report No. 2555-03Q-1 (Quarterly Technical Progress Report) 22 April 1964

(26) M-1 Oxygen and Hydrogen Autogenous Heat Exchangers, Appendix B, Aerojet-General Report No. 2555-03Q-2 (Quarterly Technical Progress Report), 20 July 1964

D. FUEL TURBOPUMP ASSEMBLY

SUMMARY

The M-1 liquid hydrogen turbopump is an eight stage, axial flow pump preceded by a mixed-flow, first-stage inducer and an axial flow second-stage inducer (transition stage). The pump design requirement is 600 lb/sec flow rate with an 1800 psi pressure rise at 13,225 rpm.

The initial test unit was driven by a single-stage impulse turbine capable of 60,000 horsepower. The prototype system utilized a two-stage turbine. The entire rotating assembly is supported radially by propellant-cooled roller bearings. Rotor axial thrust is carried by a matched, triple set of propellant-cooled, precision-ground ball bearings. The turbopump is mounted on a thrust chamber simulator for testing by means of two sets of stabilizers, as well as upper main struts and lower main struts that are identical to the proposed engine supports.

A series of 12 turbopump tests was conducted from 13 May 1965 through 22 December 1965. Test duration ranged from 9.3 sec to 57.8 sec with an accumulated duration of 330 sec. The entire test program was conducted using only one turbopump and essentially, only one buildup.

The major objectives of the test series were to demonstrate the mechanical integrity of the turbopump and to determine the over-all hydrodynamic and aerodynamic performance of the pump and the turbine at rotational speeds of up to 90% of design speed.

All tests were conducted using liquid hydrogen as the pumped fluid. The single-stage impulse turbine was powered by gaseous nitrogen during the early tests and combustion products of the M-1 hydrogen/oxygen gas generator in two of the later tests.

The performance of the facility, instrumentation, and data reduction system was excellent as demonstrated by the successful completion of all test objectives after only ten of the scheduled 18 tests had been completed. One test was terminated because of a minor control circuit wiring problem and another one was terminated because of an inadvertent gas generator pressure switch shutdown during a gaseous nitrogen drive test.

The demonstrated pump pressure rise was 4.5% below the predicted performance for this configuration. Even though the pressure rise was slightly low, pump performance is satisfactory because the head rise exceeds engine requirements by approximately 1%. The decrease in demonstrated pressure performance over that predicted is attributed to recirculating flow rates in the internal thrust balance and coolant circuits being higher than predicted. Stall margin in excess of the design objectives was demonstrated during the test series and the best estimate of over-all pump efficiency appeared to be two percentage points below that of the design prediction.

The single-stage impulse turbine met or exceeded design efficiency predictions as evidenced by both this test series and scale turbine tests performed by NASA.

CHRONOLOGY

1962

A primary objective at the outset of the program was to establish flow-passage sizes and associated pressure drops for the hot-gas system of the series-turbine arrangement. To accomplish this, it was necessary to evolve several system configurations and to evaluate them against the engine operating characteristics. Results of these evaluations established the nominal operating point for the turbine nozzles, blade profiles, and associated flow area. Maximum potential operating conditions were estimated and used to define the mechanical design operating loads. Using these loads, preliminary stress calculations were made to provide assurance that the materials then in common usage for moderate temperature turbines would be adequate for the M-1 application.

Comparative design studies of the turbopumps were initiated during the third quarter of 1962. Preliminary configurations were selected for axial-flow fuel pumps and low shaft speed inducer stage fuel pumps as well as their operating and design specifications.

At the end of 1962, it was decided that in the interest of providing growth potential in excess of 1,500,000 lb of thrust in the design concept of M-1 components, all design and fabrication effort with the fuel turbopump utilizing a centrifugal pump should be discontinued. All subsequent design effort was concentrated upon a multistage axial flow fuel pumping system.

1963

Early in 1963, the turbopump development sequence plan was formulated. The first turbopump assembly units, which were capable of operation to approximately 90% of design speed, were designated as Model I and would be utilized for initial turbopump performance testing. The Model II units, which were capable of operation to design speed, were designated as those to be used in simultaneous system testing of both fuel and oxidizer turbopumps.

The hydraulic design performance requirements for the Model I fuel turbopump assembly were established during the second quarter of 1963.(27) Also, a design study was conducted which resulted in the selection of a fabricated pump rotor. Several welding methods were investigated, including tungsten inert gas (TIG), electron-beam (EB), and Menasco uniweld. TIG welding was selected as the primary method and EB as the back-up method.

(27) Fuel Pump Hydraulic Design, Appendix C, Aerojet-General Report No. 4014-02Q-2 (Quarterly Technical Progress Report), 25 July 1963

1963

The original calculation of the fuel turbopump assembly critical speed, resulting in a value of 16,700 rpm, was found to be in error during the third quarter of 1963. The recalculated critical speed came close to the operating speed of 13,225 rpm and an investigation was undertaken to determine a method of correction. An improved computer program, which accounted for rotor shear deflection in addition to the usual beam deflections, was used in these new computations. While the introduction of this method of analysis further reduced the critical speed, the results were more accurate than those of the original program. It was determined that a major improvement could be obtained from modifications in the pump rotor.

By the end of September 1963, the operational envelopes for both the Model I and Model II fuel turbopumps were determined and the preliminary test request for the Model I fuel turbopump assembly was initiated.

A weight breakdown was completed during the fourth quarter of 1963 for the prototype and target weight configuration. Prototype weight was 4200 lb and target weight was 3340 lb. The instrumentation location drawing for the Model I fuel turbopump assembly defining the type, the range, and the location of all instrumentation required for turbopump assembly testing was also released.

1965

The completed Model I fuel turbopump assembly was shipped to the test area on 13 March 1965. The first tests of the S/N 001 fuel turbopump assembly were successfully completed during the second quarter of 1965. The assembly pumped liquid hydrogen with gaseous nitrogen turbine drive. Twelve tests were conducted during 1965 for a total duration of 330 seconds. The test facility, instrumentation, and turbopump configuration as well as the test procedures, the test results, and the disassembly inspection results are delineated in a separate report. (28)

In addition, a design study was conducted early in 1965 to determine the capability of adapting the M-1 fuel turbopump to satisfy the requirements of the PHOEBUS II Reactor Liquid Hydrogen Feed System. Appropriate design and performance analyses as well as mechanical design effort were performed to the extent needed to permit rapid transition into a development program. A 20% stall margin was predicted for the resulting pump hydraulic geometry at the nominal PHOEBUS operating point. A nominal speed of 11,500 rpm would be required to produce a 1400 psi pressure rise at a hydrogen weight flow of 350 lb/sec. The predicted nominal turbine weight flow was approximately 55 lb/sec. The complete study has been detailed in a separate report. (29)

(28) Blakis, R., Lindley, B. K., Ritter, J. A., and Watters, W. E., Initial Test Evaluation of the M-1 Liquid Hydrogen Turbopump Including Installation, Test Procedures, and Test Results, NASA Report No. CR 54827, 4 September 1966

(29) Lee, C. H. and Knuth, W., Design Study of Modification of M-1 Liquid Hydrogen Turbopump for Use in Nuclear Reactor Facility, NASA Report

1965

1. Fuel Pump

SUMMARY

The hydraulic design as well as the mechanical design of this M-1 pump have been reported separately.

The hydraulic design report presents the design method and the resulting design details as well as performance predictions for a ten-stage, axial flow, hydrogen pump. The pump was designed to supply 600 lb/sec of hydrogen at a pressure rise of 1890 psi. The pump stage complement consisted of a low hub tip ratio inducer stage with untwisted rotor blading followed by a lightly-loaded stage, called the transition stage, and eight main stages. The main stages were designed for free vortex head generation, with 50% reaction at the blade rest and a tip diffusion factor of 0.4. The transition stage was designed to provide uniform radial head distribution for the first main stage. A C-4, or modified C-4 circular arc blading was utilized for all blade rows with the exception of the inducer rotor and discharge housing turning vanes. (30)

The report detailing the mechanical design of the M-1 axial flow liquid hydrogen pump includes descriptions of the pump assembly and major pump components, anticipated stress and vibration levels, materials, fabrication considerations, rotor balancing, and assembly procedures. Emphasis was placed upon the first development unit built. (31)

CHRONOLOGY

1963

Stress analysis of the fuel turbopump rotor assembly was completed during the third quarter of 1963 as was a stress analysis of the discharge housing. A stress analysis of the thrust disc was also completed as well as an analysis of the thrust balance system. The preliminary rotor weld procedure was written.

A study of various details of alternative thrust balancing systems was completed during the fourth quarter of 1963. (32)

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(30) Farquahar, J. and Lindley, B. K., Hydraulic Design of the M-1 Liquid Hydrogen Turbopump, NASA Report No. CR 54822, 15 July 1966

(31) Regan, P. J., Mechanical Design of the M-1 Axial Flow Liquid Hydrogen Fuel Pump, NASA Report No. CR 54823, 15 February 1966

(32) Evaluation of M-1 Thrust Balance Systems, Appendix B, Aerojet-General Report No. 4014-02Q-4 (Quarterly Technical Progress Report), 20 January 1964

1964

During the second quarter of 1964, it was found that there was need for an interim inducer. A revised vane pressure distribution analysis resulted in highly stressed vanes for the original inducer. Also, materials tests revealed lower than anticipated ductility for large forgings of aluminum alloy 7079-T652 at cryogenic temperatures. This combination of high stress values and low material ductility made the original design susceptible to fatigue failure. Layouts for the interim fuel inducer were initiated.

By mid-1964, the fuel pump rotor weld assembly, S/N 01, had been cold cycle tested, the pump discharge housing had been cast, and machining of the pump rotor, the pump rotor blades, and the transition rotor was under way.

A revised thrust balance system analysis was completed during the fourth quarter of 1964. This analysis included all of the latest available data regarding pump pressure distribution including off-design limits, turbine thrust, bearing spring rates, and bearing damping forces.

The fuel pump rotating components were balanced during the second week of December 1964. Some difficulty was experienced in balancing the fuel pump rotor assembly because of a lack of adequate centering control of the turbine tie-bolt. A redesign of the turbine spacer resulted in satisfactory centering of the tie-bolt under varying loading conditions and permitted completion of the balancing in accordance with drawing requirements.

1965

During January 1965, a successful trial assembly of the main stage stator retaining rings and stator blades was completed with a maximum concentricity error of 0.002 TIR. Also, a trial assembly of the discharge housing to mainstage housing was successfully completed. This assembly included both primary and secondary conical seals. The stator unit was then successfully assembled into the main stage and discharge housings without difficulty. The housings were heated to 200°F prior to installing the main stage stator assembly and the components were assembled without any need for axial force. All bolts were installed and torqued to drawing requirements and flange interfaces checked for metal-to-metal contact. The units were disassembled without difficulty.

A fuel turbopump assembly mock-up was also completed during January. This mock-up had an inlet elbow and a discharge housing plus a fabricated spacer to simulate proper inlet and discharge flange orientation. The housings used were those scheduled for fuel turbopump assembly S/N 002. This mock-up fuel turbopump assembly permitted test area personnel to fit lines and check the mounting on the thrust chamber assembly without affecting assembly of fuel turbopump assembly S/N 001.

The assembly of fuel turbopump assembly S/N 001 began in mid-January. The actual assembly of the turbopump assembly required nine weeks. It was completed on 12 March 1965, four weeks ahead of schedule. All critical clearance dimensions were satisfied. The breakaway torque of the rotating components was 105-in.-lb, measured with the lift-off seal in contact with its rotating ring on the turbine shaft. The weight of the assembly, without the balance piston return flow lines, or pump mounting brackets, is 6675 lb. Figure No. 19 shows the completed fuel turbopump assembly being installed in the transport stand immediately before shipment to Test Zone E on 12 March 1965, where it was installed on the dummy thrust chamber assembly (see Figure No. 20). The mated components were then installed in Test Stand E-1.

The test results, procedures, and instrumentation used during childown of the fuel turbopump with liquid hydrogen are delineated in separate reports. (33)(34) During these tests at Test Stand E-1, the turbopump bearings were chilled from ambient temperature to -418°F in 0.9 hour with a liquid hydrogen flow rate of 29 gal/min. These tests with the Model I fuel turbopump were successfully accomplished pumping liquid hydrogen with gaseous nitrogen turbine drive.

Fuel turbopump assembly Model I S/N 001, B/U 001 was returned to the Assembly Room from Test Stand E-1 during July 1965. The unit was partially disassembled to install a new lift-off seal assembly to replace the original which developed a leak in the actuating bellows. The fuel turbopump assembly was then reassembled. After the oxidizer turbopump tests were completed, the fuel turbopump assembly was shipped to the test area.

The turbine thrust analysis for off-design conditions was also completed. Results indicated that negative (toward pump) thrust loads would be developed that exceeded thrust bearing capabilities when the pump was operated at 6300 rpm with gas generator drive. Work was initiated to modify the thrust balancing system to provide maximum ΔP across the thrust disc to reduce the thrust bearing load. This modification consisted of routing the balance piston return flow to a low pressure point in the test stand piping downstream from the pump discharge flange. Design sketches for routing the pump thrust balance system flow lines were completed. Stress analysis of the revised return flow system showed an adequate positive margin of safety.

The pump was tested again during the fourth quarter of 1965. It was disassembled and inspected after this testing was completed.

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(33) Ritter, J. A., Summary of Observed Results When Chilling the M-1 Fuel Turbopump to Liquid Hydrogen Temperature, NASA Report No. CR 54828, 3 June 1966

(34) NASA Report No. CR 54827, op. cit.

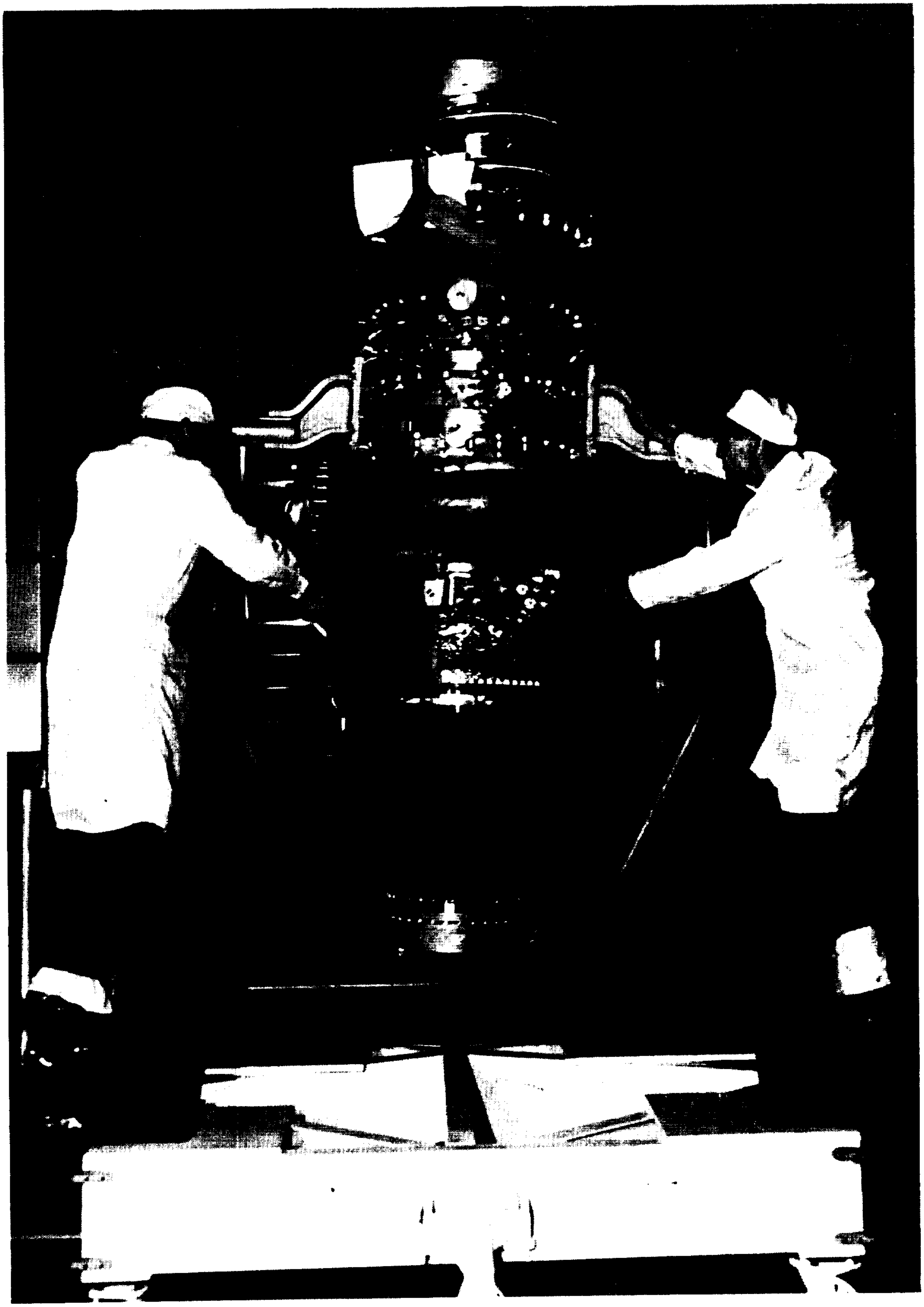

Figure 19. FTPA Being Installed Into Transport Stand

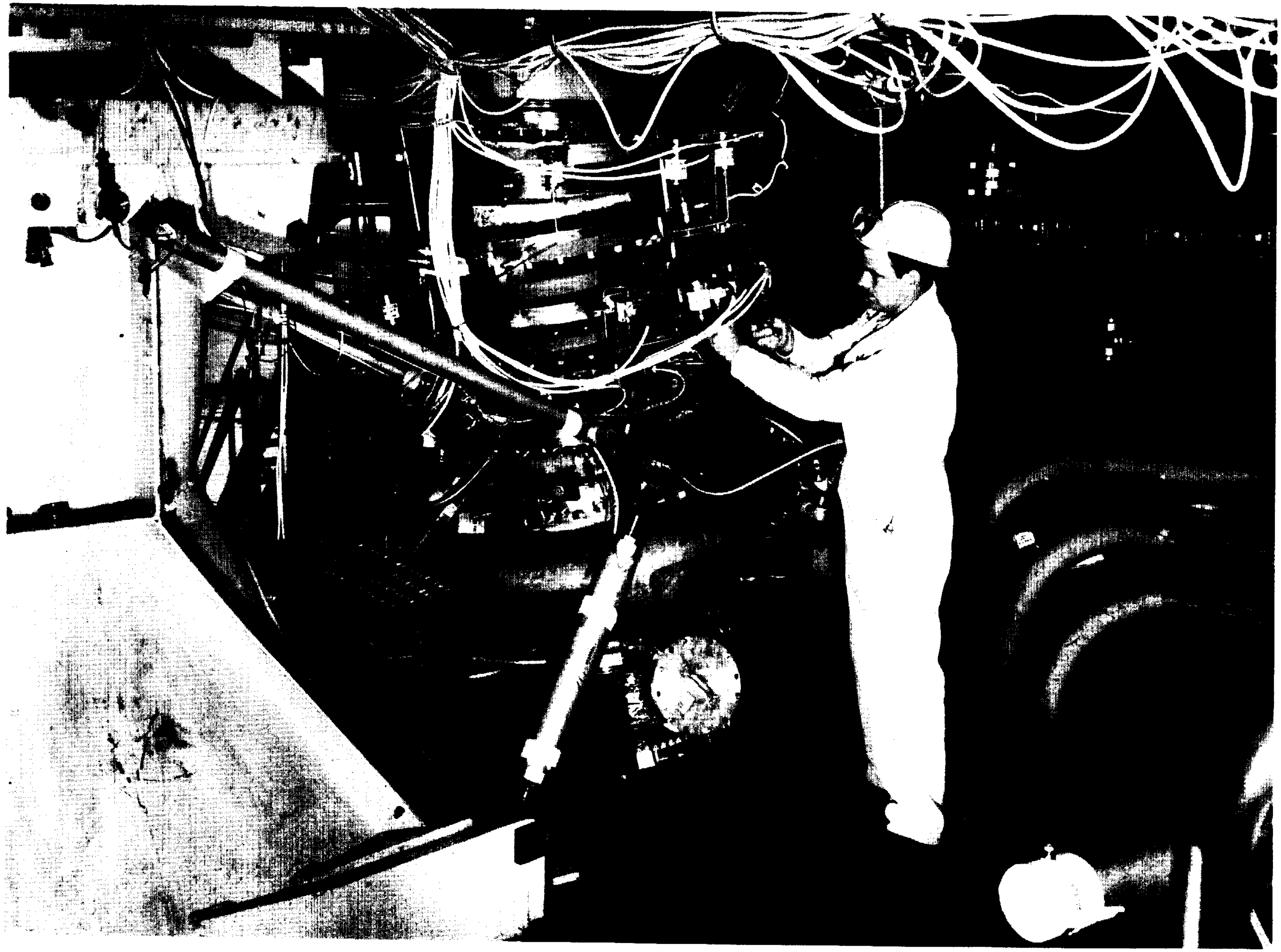

Figure 20. FTPA Being Installed at Test Stand E-1

1965

2. Fuel Scale Pump

The scale pump test program was undertaken at the outset of the program.

An AJ-1000 pump housing casting, approximately 80% of M-1 full-scale, was machined and hydraulically pressure-tested during the third quarter of 1962. Stress patterns were studied using a stress coating. The first scale pump housing instrumented burst testing was completed during the ensuing quarter. The bolts securing the passage closure plate to the housing discharge port failed at 3830 psi; however, there was no housing rupture or bursting at this pressure.

1963

A hydraulic analysis of the subscale fuel pump was completed during the second quarter of 1963 in conjunction with a mechanical analysis of the rotor blades. The hydraulic analysis was undertaken to predict the performance of the pump using water as the test fluid. The method of analysis was similar to that used for the full-size liquid hydrogen pump. The two methods of off-design analysis used were the Howell-Vavra methods and the E. Flagg (General Electric) method. The mechanical stresses in the rotor blades, based upon pressure and centrifugal loads, were found. The maximum stress occurred at the root of the blade with a value of 27,000 psi in the main stage and 25,000 psi in the transition stage for 8000 rpm. The rotor material was aluminum 7075 - T652 with a yield strength of 65,000 psi. Design of the subscale fuel pump was made final and the master layout was completed. This unit was a 3/8 geometric model of the Model I fuel pump with the exception that the axial flow portion was limited to three stages. Detailed drawings for the subscale fuel pump, including the discharge housing casting and the shaft machining (drawings), were then released.

The analysis of an alternative (tandem) inducer for the fuel scale pump was initiated during the third quarter of 1963. It had a twofold objective; to improve the inducer exit fluid velocity distribution and to improve cavitation performance.

The stress analysis for the fuel scale pump inducer was completed by the end of 1963. This analysis was based upon an operating speed of 8,000 rpm, with water as the test fluid. The minimum factor of safety, based upon the yield strength of 7079-T6, was 1.5. The maximum stress occurred at the vane root and was composed of 40,000 psi bending stress caused by hydraulic loads, 10,600 psi radial, and 20,000 psi tangential stresses caused by centrifugal forces. The maximum stress in the hub was less than 1,000 psi.

The fuel scale pump assembly, installation assembly, and discharge housing subassembly drawings were then released. This completed all drawing release requirements for the first five pump configurations.

1964

The first test assembly was completed early in 1964. It consisted of the inducer only. Thrust calibration was accomplished at the transducer laboratory. The data reduction computer program was also completed. This program supplied integrated incidence and deviation angles from traversing data and facilitated performance analysis.

A feasibility study for conducting dynamic strain measurements on inducer blades and pump rotors was completed during the second quarter of 1964. Proposed hardware changes for lead wire routing and a recommended slip ring assembly were evaluated and found suitable for applications in the scale pump. The availability of 10 channels was considered adequate for recording strain gage signals with the existing instrumentation in the test area. This instrumentation would be used to determine interim design operating stress and pressure distributions as well as the development of final design criteria.

Testing of Phase I, Configuration 1 (inducer only, transparent inducer housing) for the fuel scale pump was conducted on 15 April 1964.

1965

The fuel scale pump was built-up again during the first quarter of 1965 and tested with a first-stage inducer, second-stage inducer, and three main stages. The first-stage inducer, second-stage inducer rotor, and the main stage stators were of the original design and did not include the blade changes incorporated into the full size pump.

The M-1 fuel scale pump was built up as P/N 286431-59 during the second quarter of 1965. This build-up included the inducer stage and transition stage only. Spacing of the rotors and stators was such that traversing probes could be placed at the discharge of each element. The pump was installed in Test Bay D-3 on 1 May 1965 and testing was completed on 19 May 1965. This testing included traversing, H-Q, and cavitation tests.

The fuel subscale pump was built up as P/N 286431-279 during the third quarter of 1965 and included the interim inducer, inducer stator, and transition stage. This pump was delivered to Test Bay D-3 on 9 July 1965 and tested on 15 and 16 July 1965. Upon completion of the tests, the pump was sent to the Pump Shop for disassembly and inspection.

3. Fuel Turbine

1962

Aerodynamic and structural design studies of turbine inlet manifolds, rotors, and exhaust manifolds were conducted at the outset of the program.

1962

The turbine-diffuser design for the fuel turbopump initially was a conical configuration. Guide vanes in the diffuser channel were used to increase the efficiency of the diffusion process. This selection was based upon fabrication considerations without accepting performance penalties.

1963

Preliminary calculations for the prototype turbine disc profiles were completed during the first quarter of 1963. The conceptual layouts of the fuel turbines for the Model I configuration were also completed as was the blade stress analysis.

The aerodynamic design of the fuel turbine was completed during the second quarter of 1963. A fuel turbine rotor critical speed analysis, a turbine rotor gyroscopic moment analysis, and a second-stage rotor stress analysis were also completed.

A centrifugal and thermal stress analysis of the first-stage rotor for the Model I fuel turbine was completed during the third quarter of 1963. A rotor material study for the Model II fuel turbine was also completed.

By the end of 1963, there was a new Model I and Model II nozzle fabrication concept introduced. The upper and lower shrouds were made integral with the support rings, which were scalloped to facilitate assembly as well as provide improved thermal expansion.

Both the Model I and the Model II fuel turbines have been fully described in separate reports.

The Model I report (35) describes the design, development, and fabrication of an interim, single-stage, full-admission turbine to be used as the prime mover in the initial M-1 liquid hydrogen turbopump tests. It encompasses both the aerodynamic and the mechanical design. It includes the design criteria, the thermodynamic design, the mechanical configuration, and the assembly technique. The prime design objectives in relationship to this turbine were a conservative mechanical structure, a capability for operating with several different gases, and interchangeability of all interfaces with the prototype Model II turbine. The calculated shaft power at the design point of 12,000 rpm was 60,000 hp, with a blade tip speed of 1300 ft/sec.

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(35) Reynolds, T. W., <u>The Aerodynamic and Mechanic Design of a Single-Stage Impulse Turbine for the M-1 Liquid Hydrogen Turbopump</u>, Aerojet-General Report No. 8800-40, 31 January 1966

1963

The aerodynamic design of the Model II turbine is the subject of one of the two reports dealing with the Model II. In this report (36), the aero-thermodynamic design of the two-row Curtis staged turbine, which was intended to directly drive the M-1 liquid hydrogen turbopump, is discussed. The second report describes the mechanical and structural design of this turbine.(37) This turbine was a lightweight direct-drive unit with a design point power output of 88,150 hp at 13,225 rpm, and a thermal efficiency of 62.8% at a velocity ratio of 0.188. The mean blade diameter was 23.0-in. which gave a tip speed of 1490 ft/sec. Operating temperature gradients from -420°F to +1200°F were accommodated by fabricating the housings from thin, welded shells of Inconel 718 alloy and supporting the housings near the axial center point of the blade rows by an external frame.

1964

The final heat transfer analysis of the first-stage rotor blades was completed during the first quarter of 1964. Also, the fuel turbine second-stage blade shroud was redesigned to reduce its overhang and resultant bending stress.

By the end of March 1964, the blade tilt angle calculations for the modified first and second rotor blade, the vibration analysis for the first-stage rotor blade and the second-stage stator vane and rotor blade, and the off-design performance maps were completed. In addition, the first-stage and second-stage blade stresses were recalculated using final profiles, shrouds, and tilt angles. A stress analysis of the flexible thrust bearing support (pump end), an analysis of bearing coolant flow through the externally-mounted filter and venturi meter (38), and an analysis of the turbine end coolant system were completed.

The electron-beam weld program for the turbine rotor-to-shaft weld at Solar Industries was completed during the second quarter of 1964.

A combined stress and thermal analysis of the initial lightweight first-stage and second-stage turbine rotors was completed during the third quarter of 1964. As a result, several design modifications were accomplished. The fuel turbine transient temperature analysis of the lightweight turbine discs was completed. (39) Also, a specification (AGC-46661) for inspecting the fuel turbine rotors curvic couplings was released.

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(36) Reynolds, T. W., <u>Aerodynamic Design, Model II Turbine, M-1 Fuel Turbopump Assembly</u>, NASA Report No. CR 54820, 15 April 1966

(37) Reynolds, T. W., <u>The Mechanical Design of a Two-Stage Impulse Turbine for the Liquid Hydrogen Turbopump of the M-1 Engine</u>, NASA Report No. CR 54821, 30 May 1966

(38) <u>M-1 LH_2 Bearing Coolant Feed Flow Path Analysis</u>, Appendix B, Aerojet-General Report No. 2555-03Q-1 (Quarterly Technical Progress Report), 22 April 1964

(39) <u>Transient Temperature Analysis</u>, Appendix C, Aerojet-General Report 2555-03Q-3 (Quarterly Technical Progress Report), 20 October 1964

1964

The fuel turbine lightweight first-stage and second-stage rotors were thickened at the hub to increase the axial natural frequency to at least 15% above the maximum operating speed of 14,550 rpm. The drawings for these rotors were then released during the fourth quarter of 1964. The Model II fuel turbine first-stage rotor weldment was redesigned to be compatible with the lightweight design.

The simulated Chevron turbine blade platform specimens, which were electron-beam welded by Solar, were tested at 68°F, 800°F, 1,200°F, and -320°F. The results of these tests showed that the weld strength exceeded the parent metal requirements of Inconel 718.

The fuel turbine inlet manifold weld assembly, P/N 286156, was also completed during this quarter. Zyglo inspection of the weldment was satisfactory and the part was successfully proof tested.

The turbine manifold to bearing housing seal assembly (P/N 286116) was proof tested at 562 psig with dial indicator readings recorded during the test. This part was then helium leak tested and no leaks were found.

The Model I single-stage rotor (P/N 286170) was received. It was statically and dynamically balanced. The part was then assembled to the pump rotor and rebalanced as an assembly. During preliminary balancing of the assembly, it was discovered that the tie-bolt (P/N 286134) and the turbine spacer (P/N 286127) were positioned eccentric to the rotor centerline causing excessive unbalance. The spacer and turbine disc were modified to obtain better concentricity and the assembly was then successfully balanced.

1965

The fuel turbine inlet manifold (P/N 286105) was completed by the McGregor Manufacturing Company during the first quarter of 1965 and assembled to the turbopump without difficulty. A thermal analysis of the Model II turbine inlet manifold, the first-stage nozzles, the first-stage blades, and the first-stage disc was also completed. This thermal data was then used in a stress analysis of the Model II turbine inlet manifold.

By the end of March 1965, all Model I turbine components had been assembled into the fuel turbopump without any difficulty.

The Model I turbine rotor disc axial vibration and the blade natural frequency tests were completed during the second quarter of 1965.

The second fuel turbine manifold (P/N 286105) was completed by the McGregor Manufacturing Company during the third quarter of 1965. Also, the first build-up of the Model I turbine was removed from the test stand after the initial gaseous nitrogen tests were completed and inspected. It was in its original condition except for some discoloration on the blades.

1965

The Model II forged turbine blades for both the first-stage and second-stage rotors were successfully electron-beam welded to the rotor discs during this same quarter. These rotors were completed during the ensuing quarter.

4. Fuel Power Transmission Assembly

1962

Power transmission test unit designs were reviewed at the outset of the program. This test unit was to be used for evaluating the complete power transmission assembly under simulated conditions of loads, speeds, temperatures, propellant bearing cooling, and lubrication as well as the transient conditions anticipated during turbopump operation. The evaluation of the surface treatment and plating of shaft materials in cryogenic environments was also under way.

1963/2Q

The design power transmission components for the multi-stage axial flow fuel pumping system was started during the fourth quarter of 1962. The conceptual layouts for the Model I power transmission assembly were completed during the ensuing quarter. The master layout was completed during the second quarter of 1963.

Details of the fuel power transmission assembly frame, which connected the turbine housing assembly to the pump housing assembly, were also established during the second quarter of 1963. In addition, the detailed design of the bearing coolant and seal systems for the environmental conditions, temperatures, pressures, and flow was established.

The bearing coolant circuits for the M-1 fuel turbopump have been described in a separate report.(40). The liquid hydrogen bearing coolant was taken from a relatively high pressure source within the pump, jetted directly on to the bearing rolling elements, and discharged to a lower pressure area within the pump. Satisfactory operation of bearings cooled by liquid hydrogen eliminated the need for a separate bearing coolant pressure source and seals. It also prevented contamination of the pumped fluid. Particular attention was given to the design and calibration of the venturi meter used as the coolant flow measuring device. The design and operation of the lift-off seal was also included as part of this report.

(40) Purdy, C. C., Mechanical Design and Performance Evaluation of M-1 Turbopump Bearing Coolant System, Aerojet-General Report No. 8800-25, 1 November 1965

The Model I fuel coolant filter concepts for pump side and turbine side were completed during the third quarter of 1963. Each filter was designed to remove 98% of 10 micron and 100% of 25 micron particles. These filters were cylindrical wire-woven type with a minimum collapse pressure of 2000 psi. The filter elements on the turbine were replaceable without breaking the coolant line. The Model I fuel frame concept connecting the pump housing and turbine housing was also completed. Three conical segments of 72 degrees each were combined to support the turbine housing from the pump housing. Each frame segment was slotted axially between bolts to accommodate expansion and contraction during temperature change.

A transient temperature analysis for the Model I fuel PTA was performed during this period. Temperature as a function of time from purge, through chilldown, firing of 400 sec, and post-firing was tabulated from computer runs. It was concluded from this analysis that the temperatures of the races during the 400 sec firing and post-firing cycles of 100 sec and 50 sec were in the range -350°F to -420°F. For this range, the curve of coefficient of expansion in relationship to temperature was relatively flat for the bearing and surrounding materials. Bearing fits were then determined using the coefficients for -420°F.

Instrumentation schematic were also made final for the Model I fuel PTA test and PTA showing the location and number of bearing accelerometers and temperature measuring devices. Also made final on the PTA schematic were the pressure pickup points for flow determination in the coolant system. The thrust load was to be measured by strain gages mounted on a thrust ring.

The end thrust control and instrumentation of the M-1 fuel turbopump assembly shaft end are discussed in a separate report. (41) This report contains analyses of the axial forces on the M-1 FTPA rotor. Although results were not verified by testing prior to the termination of the program, valid conclusions were drawn relevant to the existing configuration and recommendations were made for future designs. There is also a discussion of the unique FTPA rotor thrust measuring system which was designed as an integral part of the thrust bearing assembly.

1963/4Q

The Model I PTA instrumentation and wiring concept in the pump and turbine ends were completed during the fourth quarter of 1963. A critical speed analysis considering both shear deflection and bending deflection was also completed.

(41) Laux, N. J., End Thrust Control and Instrumentation, M-1 FTPA Shaft End, Aerojet-General Report No. 8800-75, 30 September 1966

1964

A conceptual layout for a "soft" turbine bearing housing, which was considered as a back-up design in the critical speed evaluation, was completed during the third quarter of 1964.

1965

The calibration of the flexible bearing housing (P/N 289920) was completed during the first quarter of 1965. This housing was used to support the axial load of the thrust bearings in either direction and to deflect under small radial loads so that the ball bearings would be free of any radial loading. It was also used to measure the axial thrust load and bearing friction by means of strain gages wired in wheatstone bridge circuits. The strain gage circuits were also calibrated.(42)

The complete power transmission assembly was assembled into S/N 001 turbopump early in 1965. Because there was approximately 0.004-in. interference fit between the thrust bearing inner races and the rotor shaft at ambient temperature, the assembly procedure included chilling the rotor shaft with liquid nitrogen and heating the bearing package. The chilling resulted in a 0.0098-in. reduction in diameter and the heating provided the additional clearance required for assembly.

New drawings were released for the Model II power transmission assembly during the third quarter of 1965. The proximity detectors were added, one in the pump-end roller bearing housing to measure the axial movement of the thrust bearing housing and the second to measure the axial movement of the lift-off seal.

5. Fuel Bearings

SUMMARY

The results of the liquid hydrogen bearing development program for the M-1 liquid hydrogen turbopump are detailed in a separate report. (43) In this program, roller bearings of 110 mm diameter were loaded to 5000 lb at 13,300 rpm for 2770 sec with a coolant rate of 50 gpm. A triple ball bearing set, 110 mm diameter, was loaded to 36,000 lb at 13,300 rpm for 2700 sec with a coolant rate of 150 gpm. Four roller bearings, 120 mm diameter, were loaded to 15,500 lb at 13,300 rpm for 5880 sec with a coolant rate of 26 gpm. Acceleration tests were successful with rates of 28,000 rpm per second for the 110 mm diameter ball bearings. DN values were in the range of 1.6×10^6.

(42) Instrumentation and Calibration of Flexible Housing (P/N 289720), Appendix A, Aerojet-General Report No. 2555-04Q-1 (Quarterly Technical Progress Report), 20 April 1965

(43) Purdy, C. C., Design and Development of Liquid Hydrogen Cooled 120 mm Roller, 110 mm Roller, and 110 mm Tandem Ball Bearings for the M-1 Fuel Turbopump, NASA Report No. CR 54826, 24 February 1966

1965

In addition, another report (44) presents the shaft whirling critical speeds for the M-1 turbopump assembly. The roller bearing loads caused by this shaft whirling as well as other sources are discussed. The predicted critical speeds were 16,000 rpm for the Model I turbopump and 15,700 rpm for the Model II turbopump with an interim inducer. For the Model II with the final inducer, a critical speed of 18,000 rpm was predicted. Nominal operating speeds for the Model I and Model II turbopumps were 11,700 rpm and 13,225 rpm, respectively.

CHRONOLOGY

1962

Bearing testers were fabricated at the outset of the program and appropriate adapters were designed.

Specification control drawings defining the radial and axial bearings for the fuel turbopump were prepared during the third quarter of 1962. In addition, fabrication orders were issued to procure an initial quantity of prototype bearing sets and an 85 mm ball and roller bearing test unit was placed into operation.

1963

The first full-scale bearing test was conducted during the first quarter of 1963 using a liquid hydrogen coolant.

Testing of various bearings under a variety of conditions continued throughout the program.

An analysis of the Model I PTA roller bearing race stresses caused by fits and temperatures was completed during the fourth quarter of 1963. (45) (46)

1964

A bearing stress analysis was performed during the second quarter of 1964 for a three ball bearing thrust stack. (47) As a result of this analysis, the P/N 288410 bearing was selected for the prototype fuel turbopump assembly.

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(44) Severud, L. K. and Reeser, H. G., Analysis of M-1 Liquid Hydrogen Turbopump Shaft Critical Whirling Speed and Bearing Loads, NASA Report No. CR 54825, 20 December 1965

(45) M-1 Fuel (Liquid Hydrogen) Roller Bearing Stresses, Appendix C, Aerojet-General Report No. 4014-02Q-4 (Quarterly Technical Progress Report), 20 January 1964

(46) M-1 Fuel (Liquid Hydrogen) TPA Roller Bearing Stresses, Appendix D. Aerojet-General Report No. 4014-02Q-4 (Quarterly Technical Progress Report), 20 January 1964

(47) M-1 LH_2 Bearing (P/N 288410/288390) Stress Calculations, Appendix D, Aerojet-General Report No. 2555-03Q-2 (Quarterly Technical Progress Report), 20 July 1964

1964

The turbopump roller bearing internal clearances were calculated during the fourth quarter of 1964 and compared with the clearances used in the bearing tester. The bearing tester shaft and housing materials were of Inconel X whereas the turbopump had an Inconel 718 housing, pump, and shaft and a Rene' 41 turbine-end shaft. As a result, action was initiated to convert the bearing tester shaft and housing to Inconel 718 and to ensure that the bearing fits and clearances in the tester accurately duplicate turbopump conditions at all temperatures.

1965

S/N 001 turbopump was assembled during the first quarter of 1965 and the bearings installed into this assembly were: pump-end roller bearing P/N 288260, S/N 309; turbine-end roller bearing P/N 288340, S/N 369; and ball bearing triple-stack P/N 288410, S/N 853.

The final fuel bearing tests were conducted during the second quarter of 1965. These tests were for the purpose of verifying the adequacy of the pump-end and turbine-end bearings.

6. <u>Fuel Seals</u>

Design studies of the turbopump dynamic seals were initiated at the outset of the program. Initial designs of the static lift-off seals were prepared as well as those for the preliminary labyrinth seals. The design and operation of the lift-off seal is discussed as part of a separate report. (48)

1962

The shaft-seal envelope dimensions were also established early in the program and hydrogen-shaft-seal tests were started to establish the performance and stability of the lift-off seal.

Testing of a double-nose configuration for the fuel lift-off seal was initiated during the fourth quarter of 1962.

1963

A series of tests was initiated during the second quarter of 1963 to select the optimum seal face material combination for the turbine shaft seal.

Layouts for two back-up designs of the Model I PTA lift-off seal assembly were initiated during the fourth quarter of 1963.

(48) Aerojet-General Report No. 8800-25, op. cit.

The testing of liquid hydrogen subscale seals was completed during the fourth quarter of 1964. All subsequent testing was done with full-scale seals.

As a result of previously-conducted development tests, the shaft riding seals were designed during the fourth quarter of 1964 with a 0.003/0.005-in. diametral clearance at cryogenic static conditions. This clearance would be reduced to 0.001/0.003-in. at cryogenic dynamic conditions because of seal ring growth.

Additional seals information is provided in two separate reports published as part of the M-1 Controls effort. One report (49) is a compilation of surveys and investigations of static and dynamic seals as well as joints over a seven-year period. The other report (50) deals with the M-1 Seals Program defining its scope, status, and accomplishments.

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(49) Henson, F. M., An Evaluation of Gaskets, Seals, and Joints for Aerospace Hardware, Aerojet-General Report No. 8800-39, 30 June 1966

(50) Henson, F. M., M-1 Seals Program, Aerojet-General Report No. 8800-62, 22 April 1966

E. OXIDIZER TURBOPUMP ASSEMBLY

SUMMARY

The M-1 Oxidizer Turbopump Assembly consists of a single-stage radial flow pump, directly-driven by a two-stage axial impulse turbine through a propellant-cooled power transmission with rolling element radial and thrust bearings. The unshrouded pump impeller is fully machined from an aluminum forging. It has an axial flow inducer, 35-degree swept-back vanes at the discharge, and radial backvanes for thrust balancing.

The initial test article utilized a single-stage turbine and a cast stainless steel pump housing with a rolled-over volute and nine integral diffuser vanes. The unshrouded turbine rotor was fully machined from an Inconel X forging.

The turbine manifold is a multi-piece fabrication of formed Inconel sheet with nozzles that are bolted in. External structural support is provided to the turbine by struts which are connected to the pump backplate.

The development program was planned to yield turbopump operational information (i.e., hydraulic performance, thrust balance, aerodynamic performance, cooling rate, chilldown time, control sensitivity, malfunction behavior, and purging or drying requirements). It was also intended to develop the components to a confidence level that would permit testing of an engine with these components at nominal thrust conditions for short durations.

Two series of tests were conducted. In both series, liquid nitrogen was used as the pumped fluid. The first series was comprised of ten tests which were conducted during January and February 1965. The primary objectives of this test series were to demonstrate mechanical integrity at two-thirds of design speed and to determine over-all hydrodynamic as well as aerodynamic performance of the pump and the turbine. The second series was made up of 14 tests which were conducted during July, August, and September 1965. The primary objectives of this second series were to demonstrate the mechanical integrity at full speed and to verify the turbine aerodynamic performance with a hot-gas drive.

The following results were obtained from both test series:

Mechanical integrity was demonstrated at nominal and off-design conditions.

Facility operation was satisfactory, including the servo control of speed, flow rate, and pressure.

Bearing performance was excellent.

The performance and accuracy of the instrumentation and data reduction systems were satisfactory.

The pump head rise was adequate for engine operation even though it was approximately 8% below the predicted value.

The over-all thrust balancing system used during the first test series was adequate for the speeds obtained but appeared marginal for higher speeds, especially at low suction pressures. This system was modified (back-vane trimming) for the second test series and proved to be adequate.

The bearing coolant system performance was adequate and as predicted.

Both the dynamic and static seals performed satisfactorily.

The turbine, although an impulse machine, developed a significant (20,000 lb) axial thrust under the off-design conditions encountered with the gaseous nitrogen drive.

Pump efficiency (based upon subscale unit tests) was 3-1/2 percentage points below the design prediction (61.5% vs 65.0%). Turbine efficiency, based upon the subscale tests, either met or exceeded design predictions.

The effort for this turbopump assembly paralleled that for the fuel turbopump assembly at the outset of the program as pertained to the establishment of sizes, pressure drops, arrangements, and the investigation of numerous system configurations.

CHRONOLOGY

1962

Operating and design specifications for the alternative turbopumps selected for study were established during the third quarter of 1962.

1963

During the first quarter of 1963, the initial impeller dimensions were established based upon the nominal design point. Detailed hydraulic analyses would be determined using these initial dimensions. The pump housing diffuser/volute turn and volute section layouts were also completed as was the Model I pump housing casting drawing. The impeller forging drawings were completed and fabrication orders released. The initial velocity distribution study for the pump housing was completed and the pump master layout as well as the bearing cooling system design was started. The housing liner casting and

1963

pump backplate drawings were completed and released for fabrication. Machining drawings were started for the pump housing and pump backplate. Axial thrust calculations as well as the coolant venturi and orifice designs were completed.

The component master layouts for the oxidizer turbopump assembly were completed during the second quarter of 1963.

A test plan covering the initial six months of testing was prepared during the third quarter of 1963. It included the types of oxidizer turbopump tests that would be run, the anticipated schedules, and hardware utilization. The dehumidification requirements for the Model I oxidizer turbopump assembly were also made final. Several of the major forgings and castings had been received and were in work.

A weight breakdown for the prototype and target configurations was completed during the fourth quarter of 1963. The prototype weight was 3900 lb while the target weight was 3000 lb. Also, an analysis was performed to establish the balancing requirements for the oxidizer turbopump assembly rotor. The Model I assembly drawing was released as was its instrumentation location drawing. In addition, a bench model torque meter test program was initiated.

1964

Oxidizer Model I turbopump assembly S/N 001 was assembled during the fourth quarter of 1964. The first test series with this assembly consisted of 10 tests pumping liquid nitrogen with gaseous nitrogen turbine drive. These tests were accomplished during the first quarter of 1965 in Test Stand E-3.

1965

The test results, installation procedures, test procedures, and the instrumentation used during the tests performed with the oxidizer turbopump of the M-1 Engine are delineated in a separate report.(51) The test fluids used were liquid nitrogen for the pump and gaseous nitrogen (at ambient temperature) or the combustion products (hydrogen and water at approximately 900°F) from the gas generator as the turbine drive gas.

(51) Young, M. W., Test Evaluation of the Oxidizer Turbopump of the M-1 Engine Including Test Facility, Procedures, Test Results, and Disassembly Inspection, NASA Report No. CR 54819, 15 September 1966

As a result of the initial testing of the S/N 001 assembly, appropriate modifications were made for B/U 002, which was completed and shipped to the test area on 7 June 1965. The testing of oxidizer turbopump assembly S/N 001, B/U 002 was completed during the third quarter of 1965. This was a series of 14 tests conducted to evaluate the performance of the assembly as well as to determine its ability to operate with a gas generator drive.

The axial thrust characteristics of two oxidizer turbopump assemblies are discussed in a separate report.(52) These characteristics are evaluated over a representative range of speed, flow, and suction pressure. The estimates of thrust changes resulting from impeller backvane modifications are also given. These estimates are based upon thrust verification tests made with a three-eighths size subscale pump using water and an electric motor drive. The full-scale tests were conducted using liquid nitrogen as the pumping fluid and either gaseous nitrogen or gas generator turbine drive. The turbopump consisted of a centrifugal pump directly-driven by a single-stage impulse turbine. The shaft was supported in rolling contact, propellant-cooled bearings. The 27,000 shaft horsepower turbopump had a nominal head generating capability of 3400 ft. The 28.5-in. unshrouded impeller produced a maximum thrust of approximately 70,000 lb towards suction. Net thrust was measured by using calibrated, sleeve-mounted strain gages.

1. Oxidizer Pump

SUMMARY

The mechanical design as well as the hydraulic design of this pump have been reported separately.

In the report describing the mechanical design(53), the detailed design of the major components is discussed and illustrated with respect to the selection of the basic design, the selection of material, as well as fabrication and assembly considerations. Major problems are also discussed as well as conclusions and recommendations given for future designs.

(52) Brunner, J. J., Analysis and Experimental Verification of Axial Thrust on the M-1 Liquid Oxygen Turbopump, NASA Report No. CR 54817, 15 April 1966

(53) Carrington, G. W., Mechanical Design of the Pump for the Liquid Oxygen Turbopump Used on the M-1 Engine, Aerojet-General Report No. 8800-36, 10 December 1965

1965

In the hydraulic design report(54), the design method and the resulting details, along with performance predictions and test results for a single-stage, centrifugal oxidizer pump are presented. The pump design consisted of a backward-swept, centrifugal rotor and a diffusion-type (vaned diffuser) housing. Downstream of the diffuser vanes, the housing was a volute-type design providing for constant velocity. The pump was designed to supply 3000 lb/sec of liquid oxygen at a pressure rise rate of 1528 psi. Its suction specific speed was 20,000; however, the suction eye of the pump was sized to provide for an eventual increase to 40,000 suction specific speed.

CHRONOLOGY

1962

The basic pump dimensions were established at the outset of the program. It was also decided that the oxidizer impeller would be attached by means of a through-bolt joining both the impeller and the inducer.

Studies made of the oxidizer pumping system resulted in the selection of a redesigned centrifugal pump during the fourth quarter of 1962. A swept-back vane, mixed-flow impeller pump was selected because it could provide higher suction specific speed and had lower net positive suction head capabilities. The initial pump parameters were also established and incorporated into the Design Information Report.

1963

A radial load as related to deflection analysis was completed for the Model I pump during the third quarter of 1963. This analysis compared the deflection of the Model I oxidizer pump shaft system to those of the Titan pumps of similar specific speed and configuration. Hydraulic analyses were also completed for the Model I impeller for flow rates of 120% and 80% of design flow. These analyses were accomplished by using a computer program originally supplied by NASA. The program calculated static and total pressures and velocities within the impeller hydraulic passages.

All subassembly and component drawings for the M-1 Model I pump were completed and released, except for the kit drawing which was released during the ensuing quarter.

Initial concepts for the M-1 Model III (flight-type) oxidizer pump were also prepared during this quarter.

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(54) Carrington, G. W. and Farquahr, J., Hydraulic Design and Estimated Performance of a Single-Stage Centrifugal Pump for the Liquid Oxygen Turbopump of the M-1 Engine, Aerojet-General Report No. 8800-63, 1 April 1966

1964

The oxidizer impeller stress analysis at 110% speed was completed during the first quarter of 1964.

An investigation of the vane loading and the material properties of the oxidizer impeller at liquid oxygen temperatures during the second quarter of 1964 indicated that the oxidizer impeller was overstressed. Therefore, an interim design with lightly-loaded inducer vanes was initiated. A feasibility study was also completed for the dynamic strain measurements and installation of micro-pressure transducers on the impeller vanes during actual turbopump assembly testing conditions with liquid oxygen. Analysis of bearing flow requirements was also under way.

A separate report(55) delineates the performance prediction and the orifice sizing of the bearing coolant circuit. The coolant system was designed to supply a total bearing flow of 30 gpm of liquid oxygen at the design speed of 3635 rpm and to maintain a bearing cavity pressure of 400 psia. Design predictions and turbopump test results are also compared.

The hydraulic design of the interim impeller, was completed during the third quarter of 1964.(56)

The first series of proof tests (1160 psi maximum) of pump housing No. 1 (S/N 0026) was completed during the fourth quarter of 1964.(57) Figure No. 21 shows the oxidizer turbopump assembly installed in the transport stand. Figure No. 22 shows the Model I oxidizer turbopump assembly being installed into Test Stand E-3.

1965

The initial oxidizer turbopump tests were accomplished during the first quarter of 1965 and after these tests, effort was directed toward ascertaining the causes of the problems encountered during these tests. Then, appropriate modifications were incorporated into the pump to eliminate the problems.

The second test series was completed on 24 September 1965 and this was followed by disassembly inspection of the pump.

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(55) Beer, R. and Klessig, C. E., Performance Prediction and Orifice Sizing of the Liquid Oxygen Bearing Coolant System of the Oxidizer Turbopump of the M-1 Engine, Aerojet-General Report No. 8800-56, 1 March 1966

(56) Hydraulic Design of M-1 Oxidizer Pump Interim Impeller, Appendix F, Aerojet-General Report No. 2555-03Q-3 (Quarterly Technical Progress Report), 20 October 1964

(57) Proof Test of M-1 Oxidizer Pump Housing, P/N 285204-2, Aerojet-General Report No. 1176-M-1, 16 November 1964

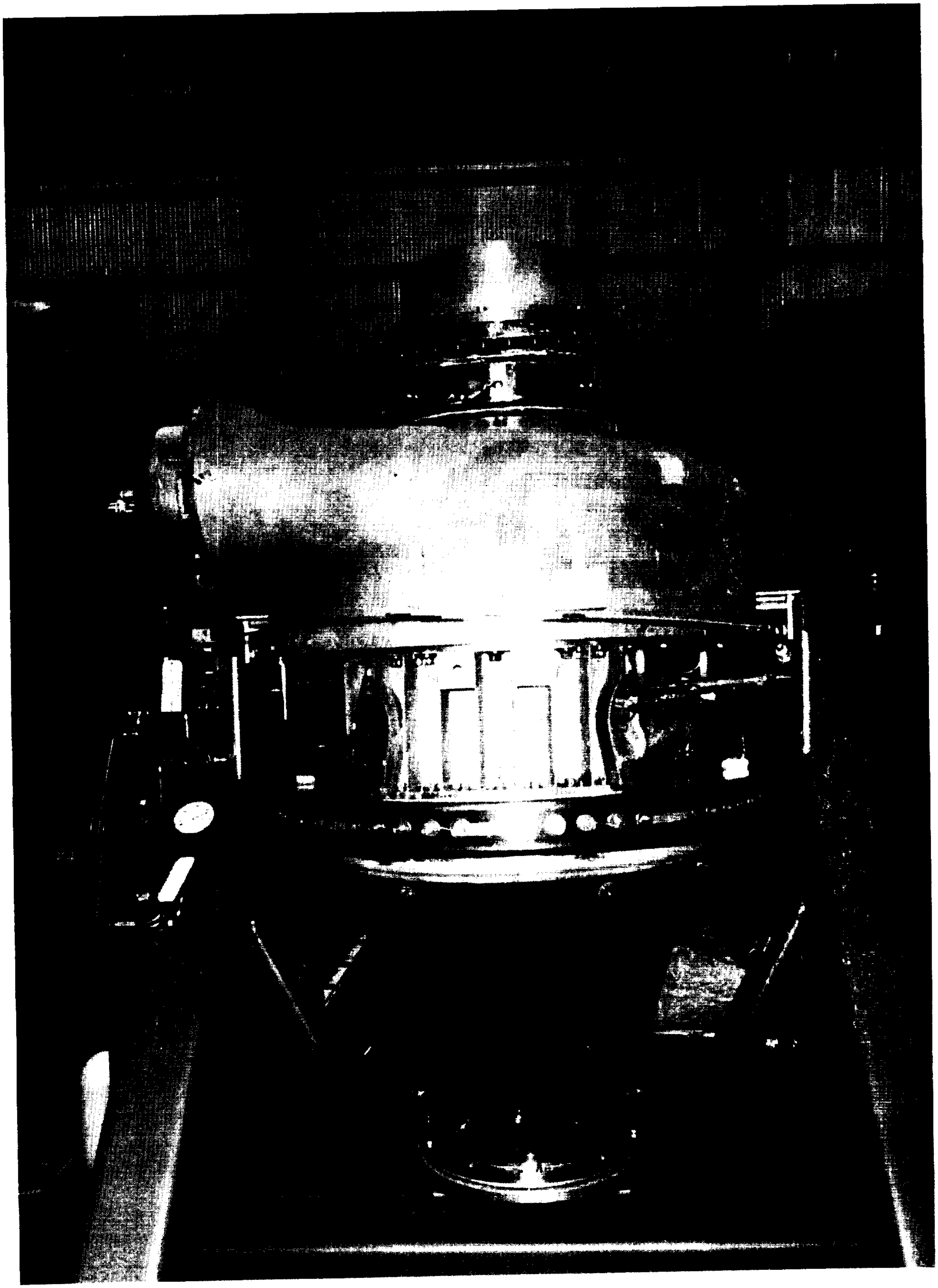

Figure 51. OTPA Installed in Transport Stand

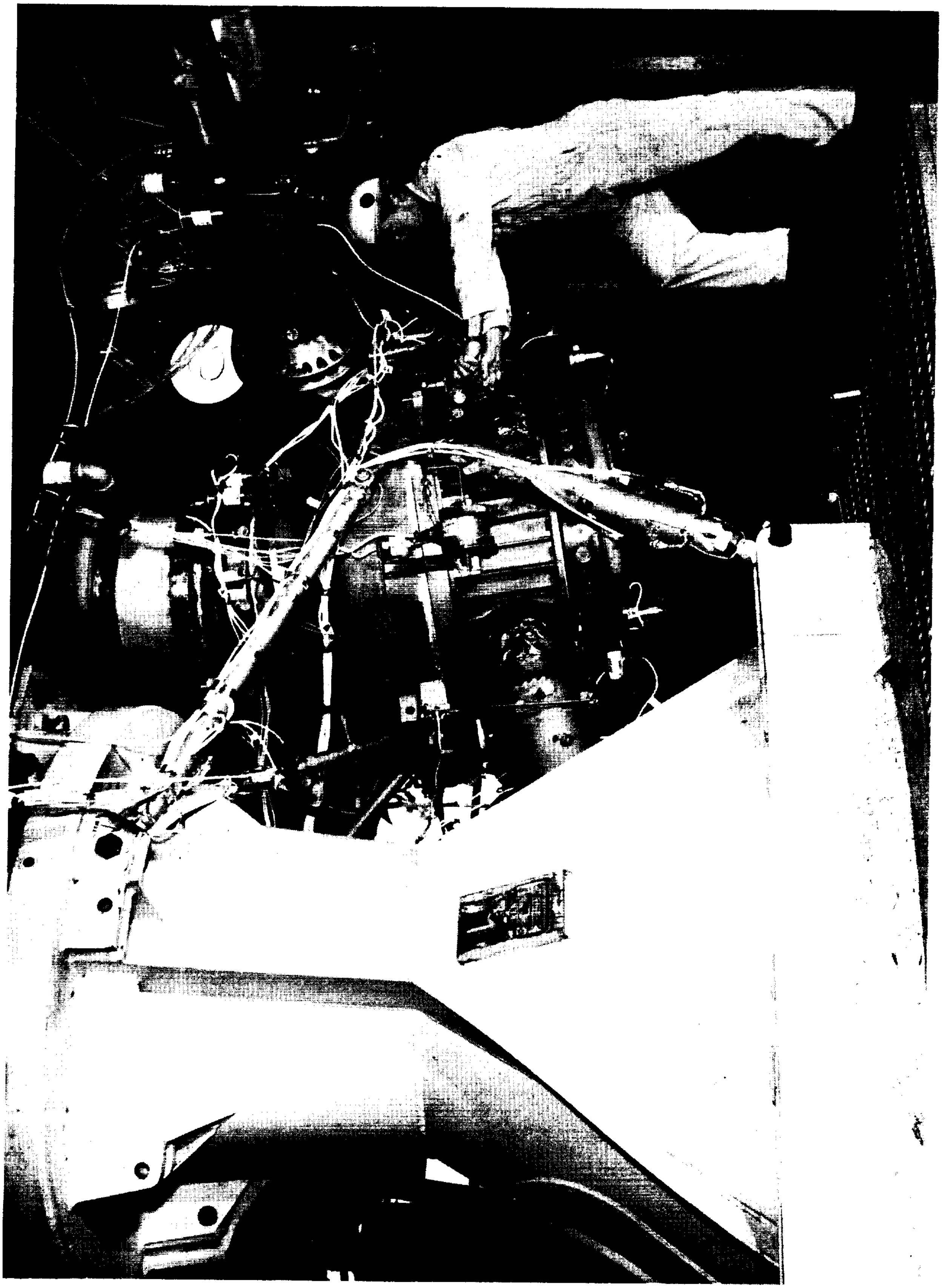

Figure 22. Model I Oxidizer Turbopump Assembly Being
Installed Into Test Stand E-3

2. Oxidizer Scale Pump

1962

The scale pump test program planning was undertaken at the outset of the program.

1963

The master layout of the subscale oxidizer pump was completed during the second quarter of 1963 along with a stress analysis of the scale pump housing. Appropriate documentation for the subscale test program was completed.(58)

Analyses as well as layout of the first five impeller configurations for the scale pump were completed during the third quarter of 1963.

An impeller configuration with a 7-degree (low) inlet angle was investigated during the fourth quarter of 1963. Also, a mock-up was assembled and delivered to the test area to permit completion of the inlet and discharge lines.

1964

A computer program to compute impeller blade surface coordinates was completed during the first quarter of 1964. Also, the first test unit (8-degree inlet angle, vaneless diffusion) was completed. A computer program for data reduction was completed and the test program was revised.

The first scale pump build-up was completed during the second quarter of 1964 and shipped to the test area. Initial scale pump testing was started during the third quarter of 1964.(59) A second build-up was tested during the second quarter of 1965. The M-1 oxidizer scale pump is shown on Figure No. 23.

Impeller rub had occurred in these tests with the scale pump and the causes were investigated.(60)

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(58) M-1 Oxidizer Pump Scale Model Test Program, Appendix E, Aerojet-General Report No. 4014-02Q-2 (Quarterly Technical Progress Report), 25 July 1963

(59) M-1 Oxidizer Scale Pump Performance with Vaneless Diffuser, Appendix A, Aerojet-General Report No. 2555-03Q-4 (Quarterly Technical Progress Report), 20 January 1965

(60) Investigation of Impeller Rub on Oxidizer Scale Pump, Appendix E, Aerojet-General Report No. 2555-04Q-2 (Quarterly Technical Progress Report), 20 July 1965

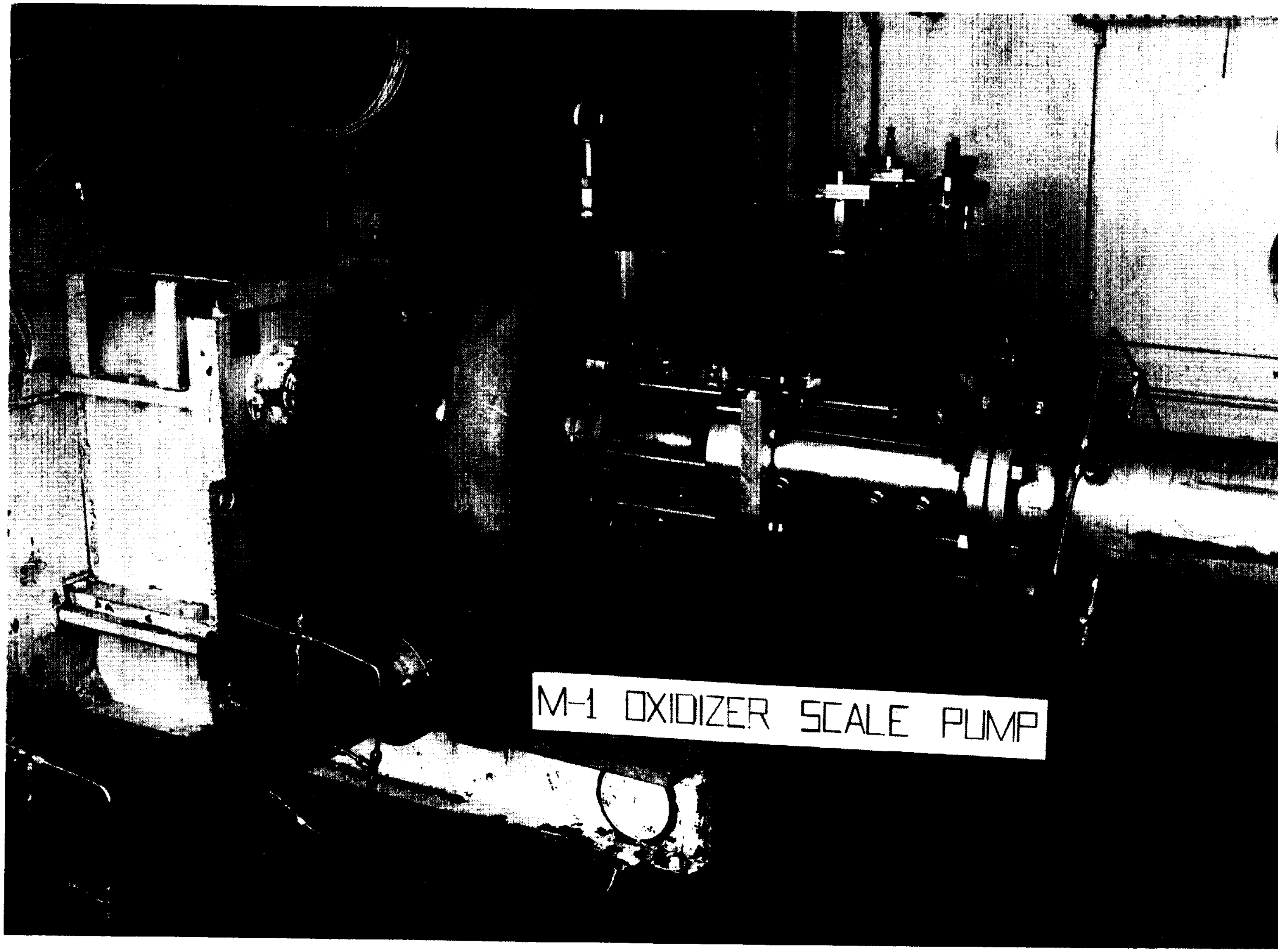

Figure 23. M-1 Oxidizer Scale Pump

1965

Additional scale pump tests were conducted during the first quarter of 1965 with an assembly made up of the original design impeller and the nominal angle vaned diffuser assembly.

3. Oxidizer Turbine

1962

Aerodynamic and structural design studies of turbine inlet manifolds, rotors, and exhaust manifolds were conducted at the outset of the program.

The aerodynamic design of the first concept was completed during the third quarter of 1962. The oxidizer turbine master layout was completed during the next quarter as well as the conceptual drawings of the two-stage turbine arrangement.

1963

Conceptual layouts of the Model II and Model III (flight-type) oxidizer turbine (with an integrated spherical manifold) were completed during the second quarter of 1963.(61)

Separate reports have been issued covering the development of the oxidizer turbine.

The Model I(62) report presents a discussion of the aerodynamic design, the estimated performance, the mechanical design, and a summary of the test results of a one-stage impulse turbine. At the design point, the turbine produced 17,340 hp at a velocity ratio (U/C) of 0.104 and an efficiency of 26%. The turbine was capable of producing 27,000 hp at a design speed of 3635 rpm without any hardware changes.

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(61) Gas Turbine Design Using Integrated Spherical Manifold for M-1 Turbopumps, Appendix F, Aerojet-General Report No. 4014-02Q-2 (Quarterly Technical Progress Report), 25 July 1963

(62) Klessig, C. E., Aerodynamic Design, Estimated Performance, Mechanical Design, and Test Evaluation of a One-Stage Impulse Turbine for the Liquid Oxygen Turbopump of the M-1 Engine, Aerojet-General Report No. 8800-53, 31 January 1966

Two reports deal with the Model II. One details the aero-thermodynamic design of a two-stage Curtis turbine.(63) At its design point, this turbine produced 26,800 hp at a velocity ratio of .133 and an estimated efficiency of 53%. Blunt-edged rotor airfoils were used throughout. Besides superior performance at subsonic Mach numbers, these airfoils (in the form of hollow sheet metal blades) offered advantages in fabrication, thermal fatigue resistance, and weight savings as compared with airfoils having sharp leading and trailing edges.

The second report(64) describes the mechanical design of the 36,000 hp, 400 rpm Curtis turbine. This turbine had unusual design features including an inlet manifold that was integral with the backplate of the adjacent centrifugal pump, and lightweight, hollow blade stator and rotor designs principally made possible by using the electron-beam welding fabrication techniques. Inconel 718 alloy was used almost exclusively. The fabrication technology for the unitized turbine inlet manifold and pump backplate assembly is also discussed.

The electron-beam welding of turbine components is discussed as part of a previously-issued report.(65) The design selection and the fabrication techniques used to produce the turbine components for the oxidizer turbopump are described. The nozzle, reversing vane, and rotor assemblies discussed were fabricated from Inconel 718 alloy. The electron-beam welding process was used to construct the sheet metal blades as well as to attach them to the discs or shrouds.

The basic concept for the Model II and Model III internal manifold insulation was established during the third quarter of 1963. Also, a stress deflection analysis of the inlet manifold as well as a thermal analysis of it were completed.

A computer program for predicting off-design performance for the oxidizer turbine was completed during the fourth quarter of 1963.

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(63) Beer, R., Aerodynamic Design and Estimated Performance of a Two-Stage Curtis Turbine for the Liquid Oxygen Turbopump of the M-1 Engine, NASA Report No. CR 54764, 19 November 1965

(64) Roesch, E., Mechanical Design of a Curtis Turbine for the Oxidizer Turbopump of the M-1 Engine, NASA Report No. CR 54815, 15 June 1966

(65) Beer, R., Fabrication of Lightweight Turbine Components Using Electron-Beam Welding for the Attachment of Sheet Metal Blades to Disc and Shrouds, Aerojet-General Report. No. 8800-49, 17 June 1966

1964

The Model II pump backplate was redesigned during the first quarter of 1964 to facilitate fabrication and reduce cost. The off-design performance maps were also completed.

A stress analysis of the Model II rotor discs was completed during the second quarter of 1964 as well as the rotor and blade analyses.(66)

The Thompson-Ramo-Wooldridge Corporation was awarded a contract to fabricate the turbine rotor assembly and the reversing vane assembly during the third quarter of 1964.

1965

The Model II turbine rotor disc-to-hub attachment was redesigned during the first quarter of 1965. The thermal gradients in the turbine manifold-to-exhaust joint were also established.(67)

The analytical and experimental vibration analysis of the turbine buckets is the subject of a separate report.(68) The shrouded turbine bucket natural frequencies and associated resonant shaft speeds are presented. Both analytical and experimental analysis are discussed and correlated. Using experimental findings, the operating range of 3500 rpm and 4000 rpm were investigated for resonant points. Analysis showed that the first-stage blade package could develop a secondary resonant problem as a result of its second mode of tangential vibration becoming excited by the second harmonic of the upstream 43 nozzle blade stimulus. No resonant problems were expected for the second-stage blade packages.

4. Oxidizer Power Transmission Assembly

1965

Power transmission test unit designs were reviewed at the outset of the program. This test unit was to be used for evaluating the complete power transmission assembly under simulated conditions of loads, speeds, temperatures, propellant bearing cooling, and lubrication as well as the transient conditions anticipated during turbopump operation.

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(66) Aerojet-General Letter 9690:0659, F. F. Bauer to W. F. Dankhoff, dated 18 June 1964, subject: LeRC Action Item #269, Stress and Vibration Report

(67) Thermal Analysis of the Oxidizer Turbine Manifold-to-Exhaust Cone Joint, Appendix B, Aerojet-General Report No. 2555-04Q-1 (Quarterly Technical Progress Report), 20 April 1965

(68) Severud, L. K. and Chinn, T., Analytical and Experimental Vibration Analysis of the Turbine Buckets for the M-1 Liquid Oxygen Turbopump, NASA Report No. CR 54830, 15 June 1966

1962

The master layout of the oxidizer power transmission assembly was completed and released during the fourth quarter of 1962.

1963

All support detailed drawings and the frame kit drawings for the Model I assembly were released during the second quarter of 1963. In addition, a stress analysis of the assembly shaft was completed and the Model I purging system concept was made final. Details of the coolant system were also established.

The effects of shaft radial displacements under axial load, including an analysis of the dynamics of the system, was accomplished during the third quarter of 1963.(69) An analysis of the thermal effect of the purging cycle upon the bearing internal clearance and an analysis of the assembly coolant system were also completed.

A Model II pump-end alternative bearing stack arrangement concept was completed during the fourth quarter of 1963. Also, concepts for modifying the power transmission tester to permit Model I power transmission assembly testing in this unit were also completed as was the Model II conceptual layout. Two test unit power transmission assemblies were assembled during this quarter.

1964

A stress analysis of the Model II shaft was completed during the first quarter of 1964. Testing of an oxidizer power transmission assembly test unit was initiated on 24 January 1964. Two additional test units were completed during this quarter.

Conceptual sketches and a layout were completed during the second quarter of 1964 for the design of a slip-ring assembly and supporting members for instrumenting the oxidizer turbopump rotating components. Also, an assembly incorporating prototype seals and bearings was tested in liquid oxygen.

Design revisions to the oxidizer power transmission assembly were made during the fourth quarter of 1964 because there was a reversal in the direction of the predicted thrust acting upon the oxidizer power transmission assembly shaft.

(69) Radial Loads Transmitted to the Thrust Bearings in the M-1 LO_2 TPA, Appendix B, Aerojet-General Report No. 4014-02Q-3 (Quarterly Technical Progress Report), 19 October 1963

1965

The torque meter system which was to be used in oxidizer turbopump assembly S/N 001, B/U 002, was successfully assembled and calibrated during the second quarter of 1965.

5. Oxidizer Bearings

The program for evaluating the suitability of the bearing package designed for the M-1 liquid oxygen turbopump has been described in a separate report.(70) The test results indicated that the bearing performance was adequate as compared with the predictions made during the design phase. The 110 mm roller and tandem ball bearings were demonstrated at $.5 \times 10^6$ DN values, radial loads of 15,000 lb and thrust loads up to 70,000 lb (twice the rated load). Liquid oxygen and liquid nitrogen were used as coolants. The bearing materials were 440C stainless steel with Armalon cages.

1962-1963

Bearing evaluation began at the outset of the program. Oxidizer subscale bearing tests began during the fourth quarter of 1962. These tests continued through January and February 1963. The first full-scale M-1 bearing test with liquid oxygen coolant was conducted during March 1963.

In the second quarter of 1963, bearing testing was directed toward developing and testing ball and roller bearings that would have a minimum life of 10 hr while operating in a cryogenic environment and subjected to the M-1 design loads.

A new pump-end alternative bearing stack arrangement concept for the Model II power transmission assembly was completed during the fourth quarter of 1963. Also, the bearing stress analysis was completed.(71) (72) In addition, a study was made to determine bearing fits and clearance during childown.

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(70) Young, M. W. and Kirby, L. F., Development of Liquid Oxygen Cooled 110 mm Roller and Tandem Bearings at up to $.5 \times 10^6$ DN Values for the Oxidizer Turbopump of the M-1 Engine, NASA Report No. CR 54816, 11 March 1966

(71) M-1 Oxidizer (Liquid Oxygen) Roller Bearing Stresses (Hertz), Appendix F, Aerojet-General Report No. 4014-02Q-4 (Quarterly Technical Progress Report), 20 January 1964

(72) M-1 Oxidizer TPA Roller Bearing Stresses, Appendix G, Aerojet-General Report No. 4014-02Q-4 (Quarterly Technical Progress Report), 20 January 1964

1965

The first oxidizer turbopump assembly test series was conducted during the first quarter of 1965. Post-test inspection showed all the bearings to be in excellent condition and they were deemed reusable. The second test series was conducted during the ensuing quarter with the same excellent bearing results.

6. Oxidizer Seals

A four-element dynamic face seal was evaluated for application in a turbopump where positive separation of cryogenic bearing coolant and hot gas was required. This development program is described in a separate report.(73) The development objective was accomplished by separating the media via a neutral gaseous nitrogen purge and when this system was applied to the M-1 turbopump, it performed successfully. Initially, it was attempted to develop the seal leakage control without relying upon a buffer gas; however, this effort was discontinued when it became apparent that extrapolation of conventional and small size seal technology would not provide the minimum leakage performance required for this critical application.

1962

Seal requirements as well as the evaluation of available seals were investigated at the outset of the program. Drawings of the prototype oxidizer turbopump assembly shaft riding seals were released during the third quarter of 1962.

1963

The first two seal tests were conducted during the first quarter of 1963 using liquid oxygen as part of the seal face material evaluation.

Cryogenics Laboratory Bay 1 was activated for seal testing during the fourth quarter of 1963.

1964

As a result of the testing, which continued throughout the program, design concepts were prepared during the third quarter of 1964 for reworking the Model I and Model II power transmission seals. Modified seals were then subjected to testing.

The shaft riding seal was redesigned during the fourth quarter of 1964 to extend its sealing surface in accordance with new shaft requirements.

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(73) Roesch, E., Development of Large Size Bellows Face Type Seals for Liquid Oxygen and Oxygen/Hydrogen Hot Gas Service at Moderate to High Pressures, NASA Report No. CR 54818, 10 February 1966

The first oxidizer turbopump assembly test series was conducted during the first quarter of 1965. Post-test inspection showed that all turbine seal carbon rubbing patterns were very clean and uniform. The second test series was conducted during the ensuing quarter with the same excellent seal results.

Additional seals information is provided in two separate reports published as part of the M-1 Controls effort. One report (74) is a compilation of surveys and investigations of static and dynamic seals as well as joints over a seven-year period. The other report (75) deals with the M-1 Seals Program defining its scope, status, and accomplishments.

(74) Aerojet-General Report No. 8800-39, op. cit.
(75) Aerojet-General Report No. 8800-62, op. cit.

F. CONTROLS

SUMMARY

The M-1 controls effort encompassed a broad area of design and development activity in connection with the various mechanical and electrical components that comprised the M-1 Engine Controls System. For the purposes of clarity, these controls components can be classified into three major areas; mechanical controls, electrical controls, and system seals.

1. Mechanical Controls

This category included the control valves for the engine fuel and oxidizer, hot gas, and helium start systems. It also included various support components, such as small cryogenic check valves, the start tank fill and vent valve, the start tank relief valve, and the stage ignition valves.

a. Thrust Chamber Valves and Actuation System

Two different thrust chamber valve designs were generated during the M-1 Engine Program; these were the sleeve-type and the poppet-type valves. A change in the thrust chamber valve concept, wherein the poppet-type design was incorporated, resulted from a change in the primary engine system to include "propellant actuation" of the thrust chamber valves. The original sleeve-type thrust chamber valve design was based upon the need for controlling propellant flow only during preliminary thrust chamber assembly component testing. All of the planned engine testing was to be accomplished using poppet-type thrust chamber valves.

The sleeve-type thrust chamber valves were developed to that point where problems areas had been identified and appropriate corrective action initiated. A program plan to complete the development of this valve within an approximate additional six months was released; however, during a joint NASA LeRC/Aerojet-General Design Review conducted during September 1964, it was decided that heavy-duty, test stand valves rather than the flight-weight sleeve-type thrust chamber valves would be used for thrust chamber assembly testing. As a result, all of the development effort that was solely applicable to the sleeve-type thrust chamber valve was terminated.

The design for both the oxidizer and fuel poppet-type thrust chamber valves was completed, released, and scheduled for fabrication and development. However, only a few parts had been fabricated when the thrust chamber valve program was terminated in February 1965. As a result, there were no development tests conducted with a poppet-type valve assembly. However, one sleeve-type thrust chamber valve assembly was modified before the effort was terminated and simulated poppet-seat seals were tested under cryogenic conditions for evaluation. In addition, water flow tests were conducted with this modified sleeve valve to estimate the over-all effects of flow forces upon the valve opening transient. A 3-in. seal and bearing tested was used to determine the preliminary friction forces and leakage rates for the poppet shaft. Results from the development testing of both the modified sleeve valve and the seal tester were useful in obtaining preliminary design data. The flow testing of the modified sleeve valve was particularly helpful in identifying potential flow force problems and providing input for design improvements. Friction forces and seal leakage rates were found to be higher than expected and a plan for corrective action was initiated.

The full scope of development testing, as outlined in the released component test plan, remains to be accomplished to complete the poppet-type thrust chamber valve development program. In addition, there are four areas that would require immediate design and development effort; the poppet deflection plate design, the shaft seal friction analysis, the poppet-shutoff-seal retention, and methods for dampening the poppet during the engine start transient. It would also be necessary to conduct component testing at rated flow and simulated transient conditions with liquid nitrogen or liquid hydrogen, as appropriate. Actual transient opening and closing characteristics during the test could be compared with values predicted by the computer analyses. These data comparisons could then be used to refine the pertinent analysis techniques as well as to improve the validity of engine analysis methods relative to poppet-type thrust chamber valve systems.

(1) Sleeve Design

CHRONOLOGY

The history of the sleeve-type thrust chamber valve from its inception in 1962 through termination during the fall of 1964 is detailed in a separate report.(76) Particular emphasis has been placed upon the development testing from October 1963 through September 1964.

1962

Detailed design of seal chambers to simulate the shut-off and dynamic sealing was completed during the second quarter of 1962. Also, leakage and functional testing of the 8-in. valve was started.

Detailed drawings of the thrust chamber oxidizer and fuel valve assemblies were released during the third quarter of 1962. Evaluation testing of lipseal configurations for the shaft seal had also been undertaken.

(76) Hetrick, J. K., Thrust Chamber Valves (Sleeve), M-1 Engine, Aerojet-General Report No. 8800-19, 12 October 1965

1962

The design investigation undertaken in an attempt to develop a zero leakage dynamic cryogenic seal has been reported separately.(77) Background theory, seal evaluation, as well as design and development method are discussed. The results of this investigation indicated that an advance in the technology of dynamic cryogenic seals had been made; however, further effort was needed to produce an optimum design. This work was recommended as being feasible.

1963

The lightweight sleeve design effort was discontinued during April 1963 when the poppet-type valve was selected as the primary design. Design effort during the ensuing quarter was directed toward the improvement of the dynamic seals in the sleeve-type valve.

Fabrication of all first-article long-lead hardware items was completed during the third quarter of 1963, except for the actuator piston.

1964

Thrust chamber oxidizer valves (P/N 705520, S/N 1 and S/N 2) were assembled and acceptance tested during the first quarter of 1964 for the first thrust chamber assembly firing. These valves were then installed on the thrust chamber assembly. Figure No. 24 shows a thrust chamber valve being assembled.

Sleeve-type thrust chamber valve, P/N 705520-49, S/N 1, used in the thrust chamber oxidizer circuit sustained extensive fire damage during the first thrust chamber assembly firing at Test Stand C-9 on 20 June 1964. The external surfaces of hybrid thrust chamber valve, P/N 705520-39, S/N 4, used in the fuel circuit were also burned. Investigations were conducted to determine the possible causes of the malfunction which occurred during the firing.(78)(79)(80)(81)(82)(83)

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(77) Hetrick, J. K. and Linn, C. G., Valve Lipseals, M-1 Sleeve-Type Thrust Chamber Valve, NASA Report No. CR 54808, 1 March 1966

(78) M-1 Thrust Chamber Assembly Accident (Test No. 1.1-02-EHM-003), 20 June 1964, Final Report Prepared by the M-1 Accident Special Investigation Team, Lt. Col. Joe E. Heatherly, Chairman, 28 August 1964

(79) Incident Report on Test Run No. 1.1-02-EHM-003, Aerojet-General Systems and Controls Report No. 173, 18 August 1964

(80) Summary of the C-9 Failure Investigation, Thrust Chamber Valves, Aerojet-General Systems and Controls Report No. 174, 17 August 1964

(81) Lipseal, P/N 705458-1, S/N 001025, Aerojet-General Failure Analysis Report No. 64-3, 6 August 1964

(82) Lipseal Sleeve, P/N 705458-1, S/N 0001022, Aerojet-General Failure Analysis Report No. 64-4, 6 August 1964

(83) Lipseal Sleeve, P/N 704631, S/N 0000952, Aerojet-General Failure Analysis Report No. 64-5, 6 August 1964

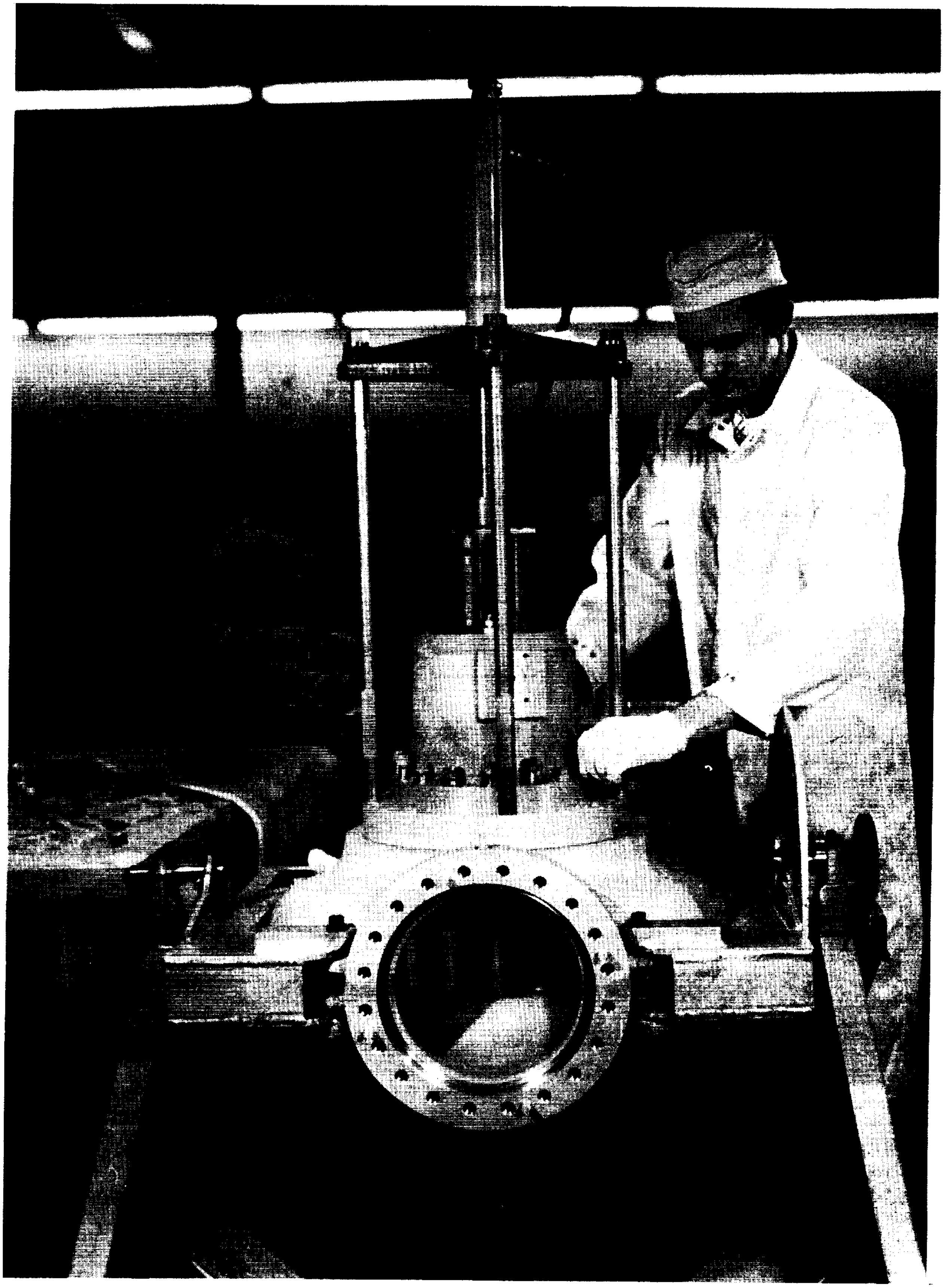

Figure 24. Thrust Chamber Valve Being Assembled

1964

As a result of these investigations, appropriate modifications were incorporated into the hybrid thrust chamber valve to increase its reliability during operation under the environmental conditions experienced at Test Stand C-9.

The sleeve-type thrust chamber valve development was terminated during the fourth quarter of 1964 and all efforts were directed toward the poppet-type thrust chamber valve.

1965

The final development report(84) for the sleeve-type thrust chamber valve as well as the final report (85) covering the associated work done in connection with the seat tester for the sleeve-type thrust chamber valve were published during January 1965.

(2) Poppet Design

A separate report(86) summarizes the development effort completed for the poppet valve configuration. It also details the design, analysis, and fabrication effort accomplished through the pre-test phase.

1962

Design investigations and layouts to determine the feasibility of this type of design were initiated at the outset of the program.

The preliminary design layout of the first configuration poppet valve was completed during the fourth quarter of 1962.

1963

The poppet design was selected as the primary design for the thrust chamber valve in April 1963 as the result of the M-1 Engine start transient analysis.

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(84) M-1 Thrust Chamber Valve (Sleeve) Evaluation, Aerojet-General Report No. CPR-2, 8 January 1965

(85) M-1 Thrust Chamber Valve Seal Tester Evaluation, Aerojet-General Report No. CPR-1, 6 January 1965

(86) Hetrick, J. K., M-1 Thrust Chamber Valve, Poppet-Type, Aerojet-General Report No. 8800-45, 31 January 1966

1963

A final design review of the poppet-type thrust chamber valve was held during the third quarter of 1963. This design incorporated two dynamic seals on the poppet to achieve the desired pressure balancing. Although a shear-type burst diaphragm was not included in the design, provision was made to incorporate such a device should it be needed subsequently to achieve the required level of sealing.

1964

Assembly and detailed drawings of the poppet-type thrust chamber valves using a single-piece body were completed during the first quarter of 1964. Because seal friction and friction repeatability were critical in this valve, a poppet thrust chamber valve seal evaluation program was also initiated.

Detailed drawings of the one-piece poppet-type thrust chamber valve were released during the second quarter of 1964. Also, a change in the valve body was incorporated to increase the wall thickness as a result of a stress re-analysis.

A pneumatic actuator was an integral part of the valve design for the preliminary checkout operations and development testing. This actuator was detachable and could be removed if necessary for special testing. The unit was intended to be used for actuation of the valves throughout the pre-flight checkout of the engine system but not during an engine hot-firing operation.

The test plan for the poppet thrust chamber valve was completed during the third quarter of 1964. It was released during the ensuing quarter.

1965

A report of the simulated poppet seat/seal evaluation(87) and one detailing the results of the flow-induced side-load tests(88) were published during February 1965.

(87) M-1 Thrust Chamber Valve (Poppet) Evaluation, Aerojet-General Report No. CPR-5, 19 February 1965

(88) Side-Load Flow Effects on the M-1 Thrust Chamber Valve, Aerojet-General Report No. CPR-4, 12 February 1965

(3) Thrust Chamber Valve Actuation Systems

1962

The preliminary design layout of the transfer valve, a lapped spool shuttle design, was completed during the second quarter of 1962. Also, detailing of the pilot valve specification control drawing was started. It was completed during the ensuing quarter.

A Titan II thrust chamber valve pilot valve was te tested under cryogenic conditions during the third quarter of 1962 for the purpose of evaluating the lapped-spool transfer valve design concept.

Evaluation of vendor proposals during the fourth quarter of 1962 regarding the helium fuel actuation system pressure regulator and the pneumatic actuation system pressure regulator revealed that one regulator would fulfill the needs of both actuation systems. Also, a new transfer valve design layout was completed. The new transfer valve was a double-poppet-type utilizing a bellows seal for minimum internal friction at cryogenic temperatures.

1964

Early in 1964 the W. O. Leonard thrust chamber valve pilot valve was replaced by a Futurecraft solenoid valve for actuating the sleeve-type thrust chamber valves during thrust chamber testing. The original vendor was unable to deliver the hardware.

b. Gas Generator Valves and Actuation System

SUMMARY

Considerable development effort was accomplished with the mechanically-linked gas generator valve assembly. Fifty-four ambient and 28 liquid nitrogen temperature acceptance tests were conducted. These included proof, leak, and functional checks of both the gas generator fuel valve and the gas generator oxidizer valve. Two-hundred-and-twenty-one cryogenic cycles were performed under rated liquid nitrogen flow conditions. Water flow testing was conducted with both the fuel and oxidizer valves to establish flow characteristics as well as to develop the pintle contours. One gas generator valve assembly was thoroughly vibration tested at both ambient and cryogenic temperatures. Seven cold flow tests and 17 hot-firing tests were conducted with the valves assembled into the gas generator.

The performance of the gas generator valves during the gas generator hot-firings was generally satisfactory. Although the gas generator valves were to be actuated by fuel in the M-1 Engine System, a hydraulic (1800 psig Freon) actuator was adapted to the valve assembly to satisfy the stringent timing delay and linearity criteria while opening at

test stand pressures which were significantly higher than engine transient values. The main poppet-shutoff-seals sealed effectively during all pre-fire and post-fire operations. Although some galling occurred in the rocker arm linkage portion of the valve because of the relatively high loads while testing with pressure-fed test stand systems, these effects were not detrimental to the operation of the valves during their limited testing experience. The greatest problem encountered with the valves during hot firing was the occasional failure (fracture) of the primary pintle lipseal on the oxidizer valve. This problem of lipseal breakage remains unsolved. Additionally, further development of the rocker arm bearings and link pins is needed to provide a longer endurance life under heavy loads.

The gas generator valve fuel (liquid hydrogen) actuation system is made up of three valves; the gas generator valve pilot valve, the gas generator valve pilot valve control valve, and the gas generator pre-flight pressurizing checkout valve. Although initial components were received for each of these valve designs, no development work was accomplished. Various types of problems were encountered with these designs during their initial acceptance testing. This, coupled with delays in receiving hardware from supplier sources, precluded the accomplishment of any development testing with any of the components in the gas generator valve fuel actuation system.

The design and development of the gas generator valve has been the subject of a separate report.(89) A second report(90) discusses the gas generator valve actuation system, which is made up of the gas generator valve pilot valve, the gas generator pilot valve control valve, and the gas generator valve pre-flight pressurizing checkout valve.

CHRONOLOGY

1962

Design layouts of the gas generator valve and actuator were completed during the second quarter of 1962.

Detailing of the gas generator fuel valve and actuator was completed during the third quarter of 1962. It was also accomplished for the gas generator oxidizer valve.

1963

The gas generator valve position indicator specification was released during the first quarter of 1963.

(89) Badenkirch, B. O. and Linn, C. G., Gas Generator Valve, M-1 Engine, Aerojet-General Report No. 8800-46, 31 January 1966

(90) Crosier, R. L. and Linn, C. G., M-1 Gas Generator Valve Actuation System, Aerojet-General Report No. 8800-43, 31 January 1966

1963

The preliminary layout of the Aerojet-General designed gas generator valve pilot valve was completed in April 1963.

A design featuring the sequencing of the gas generator valve pilot valve with the gas generator valve fuel inlet pressure was completed during the fourth quarter of 1963. New body drawings were then released for the gas generator fuel valve and the gas generator oxidizer valve wherein 6061-T6 material was specified in place of 7075-T73. Problems were encountered in the form of critical permanent dimensional changes in grades 6061 and 7075 aluminum alloy valve bodies that occurred during cryogenic thermal cycling. This problem is detailed in a separate report. (91) A special fabrication process to reduce the distortion within tolerable design levels was developed and applied to the test hardware. Test data confirmed that the special process reduced distortions in both aluminum alloys, but the degree of distortion was dependent upon body shape.

Two dry endurance tests (at ambient temperature) and actuation tests were performed with the gas generator valve assembly during the fourth quarter of 1963. Water flow testing was also completed with both the gas generator oxidizer valve and the gas generator fuel valve at the full-open position.

1964

Two dry endurance test series (100 cycle) were performed with the gas generator valve assembly during the first quarter of 1964.

The body material of the gas generator valve pilot valve was changed from a forging to barstock during the second quarter of 1964. This was done to expedite procurement and decrease costs. Also, the bellows assembly was redesignated as the gas generator igniter valve pilot valve. Static and dynamic leakage tests were performed at ambient, liquid nitrogen, and then again at ambient temperatures with a gas generator oxidizer valve using different seals.

The first prototype gas generator valve assembly (fuel-lead configuration) scheduled for test stand gas generator assembly firings was assembled during the third quarter of 1964 to Q-CAL III requirements and acceptance tests at ambient and liquid nitrogen temperatures. Excessive leakage occurred and the second (back-up) prototype gas generator valve assembly (fuel-lead configuration) for test stand gas generator assembly firings was assembled to Q-CAL III requirements and acceptance tested. Again, there was excessive leakage.

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(91) Henson, F. M. and Inouye, F., Dimensional Instability of Aluminum Alloys for Extreme Low Temperature Cycling Applications (GGV Material Instability Problem), NASA Report No. 54829, 31 January 1966

1964

As a result of the excessive leakage experienced with the dynamic Omniseals, these seals were changed to lipseals in all places except on the actuation piston. Test experience with the lipseal valves was limited, but showed an improvement in performance.

An actuation test series was completed during the fourth quarter of 1964 with prototype gas generator valve assembly Kit No. 1. Prototype gas generator valve assemblies were also endurance tested.

1965

As a result of program redirection during the first quarter of 1965, effort in connection with supporting components (i.e., gas generator pilot valve control valve and gas generator pre-flight checkout valve) intended for operation of the gas generator valves on the engine was discontinued.

Acceptance testing, vibration testing, full-rated flow and endurance testing, as well as cold flow tests were conducted with various gas generator valve kits early in 1965. A design change was also completed which incorporated a hydraulic gas generator valve actuation system to operate the valves during hot firings in support of gas generator and turbopump development. This was a conversion from the 1200 psig helium actuation system previously used.

Cold flows as well as hot firings were accomplished with prototype gas generator valves during the second quarter of 1965. Operation of these valves was satisfactory during all of these tests. Figure No. 25 shows the gas generator valves mated to the gas generator.

c. Propellant Utilization Valve and By-Pass Shut-Off Valve

The propellant utilization hot gas valve was not subjected to component acceptance or development testing. All of the parts for the original propellant utilization valve design were received and available for the assembly of the initial development valve except for the valve body which was rejected because of machining discrepancies. No identification of problem areas was possible because no development testing was conducted.

The design of the propellant utilization valve actuator and controller is discussed in a separate report. (92) Another separate report (93) describes the fuel pump circulating valve, whose primary function was to by-pass fuel from the pump discharge to the fuel tank during the engine

(92) Linn, C. G. and Koons, V. W., Propellant Utilization Valve Actuator and Controller, Aerojet-General Report No. 8800-42, 31 March 1966

(93) Roberts, R. J., Fuel Pump Circulating Valve, M-1 Engine, Aerojet-General Report No. 8800-9, 9 August 1965

Figure 25. Gas Generator Valves Mated to Gas Generator

1965

start and shutdown transients. This precluded pump operation in the stall region. An auxiliary function of the by-pass system was to provide a fuel bleed and chilldown circuit through the pump.

1962

Design layouts for both valves were started at the outset of the program. Preliminary specifications for the propellant utilization valve actuator-motor gearbox were determined and the preparation of a component specification was initiated. Also, the type of electrical control system to be used for testing was established.

Detailing of the by-pass shut-off valve was completed during the third quarter of 1962. Also, the design layout of the propellant utilization valve, which incorporated the by-pass valve function, was completed.

1963

The propellant utilization valve electrical actuator specification was completed during the first quarter of 1963. Also, the design of the flow control element was changed from a poppet-type to a sleeve-type to reduce the electrical actuator power requirements.

Detailing of the propellant utilization valve design was completed in April 1963 and the drawings were released. The valve body and sleeve gate, which were long-lead items, were also ordered in April. Vendor proposals for the propellant utilization valve actuator were evaluated in May and the Consolidated Controls Corporation design was selected. Eight propellant utilization valve actuators were ordered in June 1963 for evaluation.

Because of program re-direction at the end of 1963, the fuel pump circulation valve design activity was suspended until requirements for the component design were established based upon fuel turbopump assembly testing.

d. Helium Start System

<u>SUMMARY</u>

The controls system components in the engine helium start system consist of the helium start valve and the start check valve. The design of the helium start valve was refined during a rigorous and successful component development test program. All leakage requirements were adequately satisfied and over 35 functional tests were conducted under simulated engine start conditions using a test pad system. All system flow and valve response requirements were achieved; however, a number of development tests (i.e., vibration, endurance, and icing) remained to be performed at the time that the component development program was terminated. A single start valve pilot valve was used for 20 start system helium "blowdown" tests without any failures

or discrepancies. This pilot valve still requires additional development testing. The pilot valve controls the operation of the start valve.

The internal poppet guide support webs in the start system check valves yielded and remained permanently deflected as a result of start system "blowdown" impact loading. Although redesigned valves were obtained, this portion of the M-1 Engine Program was terminated before the modified valve could be evaluated.

The M-1 Engine helium start system has been detailed in a separate report. (94) This system consisted primarily of a 20-30 ft^3, 3000 psig spherical helium start bottle; a start valve; a start check valve; and associated plumbing to direct the helium gas from the start bottle to the fuel and oxidizer turbopump turbines during the engine start transient.

CHRONOLOGY

1962

Detailing of the start valve was completed during the third quarter of 1962.

1963

The helium start valve was redesigned during the first quarter of 1963 in accordance with reduced factors which was intended to lessen the weight. The testing of various valves and switches for the system was also under way.

Proof, functional and leak tests were being conducted during the fourth quarter of 1963 with various component parts as appropriate.

1964

Two helium start valve assemblies were subjected to functional tests using gaseous nitrogen and then, gaseous helium during the first quarter of 1964. The position indicator potentiometer design was also completed.

1965

The requirement for using helium start system controls components in support of turbopump testing in E-Area was deleted during the first quarter of 1965. As a result, all development effort for the helium start valve, helium check valve, and helium start valve pilot valve was discontinued.

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(94) Linn, C. G. and Oldenburger, K. P., Helium Start System, M-1 Engine, Aerojet-General Report No. 8800-44, 31 January 1966

e. Miscellaneous Mechanical Controls

SUMMARY

This area of controls included small cryogenic check valves, the start tank fill and vent valve, the start tank relief valve, and the stage ignition valves. The development of the check valves was discontinued relatively early in the engine program because they were at a point that was judged to be adequate for the initial engine configuration. A primary problem encountered during the development of the check valve was poppet chatter. The valves were modified to eliminate this problem by incorporating a Teflon washer on the poppet shaft. The washer was threaded through the spring coils to dampen the natural frequency of the spring. The development testing of the check valves was terminated before this modification to eliminate chatter could be evaluated.

The development testing of the start tank fill and vent valve had not progressed to a point where the design could be thoroughly evaluated before the program was terminated. Several development problems had been defined. These were either corrected or were being corrected when the effort was terminated.

Start tank relief valves were not subjected to component development testing. A minor problem that was uncovered during their limited acceptance testing was leakage at the outlet port check valve. This check valve prevents moisture from entering into the relief valve from the overboard vent system. At the termination of the M-1 Engine Program, these valves were being reworked to replace the metal-to-metal seal arrangement with a softer (KYNAR) seat material.

The design of the prototype stage ignition valves was not completed before the program was terminated. Design specification sheets for four valves and two actuators had been issued. In addition, one incomplete design layout was available.

As indicated, the miscellaneous mechanical controls consisted of the small cryogenic check valves (1/4-in. to 1.0-in. nominal size). the start tank fill and vent valve, the start tank relief valve, and the stage ignition valves. These controls are discussed in a separate report. (95)

CHRONOLOGY

1962

Development component specifications defining quick-disconnects, pre-flight checkout valves, pressure switches, and check valves were completed and released during the second quarter of 1962.

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(95) Crosier, R. L. and Bazinet, G. D., Miscellaneous Mechanical Controls, M-1 Engine, Aerojet-General Report No. 8800-21, 1 November 1965

1963

In April 1963, the 210-psia and the 936-psia thrust chamber pressure switch designs were changed to the calibration pressure switch design developed by NASA and Southwestern Industries, Inc. This design allowed pre-flight checkout of the switch without the use of special shuttle valves in the system. The thrust chamber pressure switch specification control drawings were revised to incorporate these design changes.

Check valve evaluation tests were started in June 1963.

1963-1964

Three thrust chamber pressure switches were received and accepted during the fourth quarter of 1963. Also, check valves were being subjected to leak, internal icing, humidity, and accelerated storage tests. Varied appropriate testing of mechanical controls components continued throughout the program.

1965

The test plan for the fuel pump circulating valve was released during the first quarter of 1965.

2. Electrical Controls

SUMMARY

These controls consisted of the propellant utilization valve actuator and controller, the electrical harness, turbine speed sensors, spark ignition, the in-flight sequence unit, and the thrust chamber pressure switch.

The design of the propellant utilization valve actuator and controller is discussed in a separate report.(96) The remaining electrical controls are the subject of another report.(97)

a. Propellant Utilization Valve Actuator and Controller

The propellant utilization valve actuator is designed to position the valve in response to the propellant utilization system signals so as to provide the desired control of the engine mixture ratio. Development testing of this component was conducted to evaluate its performance, stall, temperature rise, low temperature, and life-cycle operating characteristics. The actuator design and development problems were defined and components were returned to the supplier for appropriate design modifications and corrective action. The design of the motor bearings and the motor drive pinion was

(96) Aerojet-General Report No. 8800-42, op. cit.

(97) Hetrick, J. K., Hill, E. H., and Ozment, W. L., M-1 Electrical Controls, Aerojet-General Report No. 8800-41, 21 January 1966

1965

modified to improve the actuator life-cycle capabilities. Clearances in the drive screw bearing assembly were increased to relieve a binding condition detected at a low temperature soak of -250°F. An electrical noise which exceeded the design specification limits was detected in the potentiometers of the first actuators during acceptance testing. These potentiometers were reworked to eliminate this excessively high noise level. Although these design modifications were incorporated into the hardware, they were not evaluated before the program was terminated.

An experimental propellant utilization valve controller for valve positioning during engine firings was designed and fabricated for preliminary testing. This controller was made up of three basic systems; an error sensor, an amplifier, and a polarity sensitive micropositioner relay. This initial experimental design was modified by replacing the micropositioner to decrease the vibration sensitivity and to incorporate a rate feedback to improve the system stability. Hardware was procured to fabricate two of the new design controller units, but they were never assembled because the propellant utilization valve effort was terminated.

The pertinent chronological information for the propellant utilization valve actuator and controller was presented under the Propellant Utilization Valve and By-Pass Shut-Off Valve discussion (Section V, F, 1, c).

b. Electrical Harness

Two harness designs were evaluated to satisfy the temperature environment range requirement of -240°F to +170°F. Harness samples were obtained from the Bendix Corporation and GLA (General Laboratory Association). The Bendix unit proved to be superior to the GLA design. The Bendix Assembly satisfied all of the component design requirements and was selected as being suitable for M-1 Engine testing. The major remaining problem area in connection with harness design would be the requirement for the harness to withstand a temperature of 1000°F for five minutes. Special design techniques and considerable development effort would be necessary to satisfy this severe requirement. Design of the harness junction box incorporated welded aluminum Z-sections with a welded cover plat on one side and a Teflon-sealed bolted cover on the opposite side. Although the drawings were completed, no fabrication was accomplished.

CHRONOLOGY

1964

The specification drawing and the specification control drawing covering low temperature harness requirements were issued during the first quarter of 1964.

1964

A fixture for testing the controls harness test unit was fabricated during the third quarter of 1964.

1965

As a result of program redirection during the first quarter of 1965, the engine controls harness effort was discontinued.

c. Electrical Speed Sensors

The original oxidizer pump sensor design performed satisfactorily during all of the development and pump testing. However, some problems were encountered by the supplier in providing the required output for the fuel pump speed sensor because of its restricted size. Subsequent testing of the pump overspeed detection system indicated that a lower output of the speed sensor was acceptable; therefore, the fuel pump speed sensor as originally designed was considered to be adequate.

1963

The component specification for the turbine speed sensor was released during the first quarter of 1963.

The fuel turbine speed sensor specification control drawings were completed during the fourth quarter of 1963.

Vendor proposals for the turbine speed sensor were evaluated during May 1963 and the Space Instrumentation Corporation was the vendor selected.

1964

The oxidizer turbine speed probe test fixture was completed during the first quarter of 1964 and probe development testing was initiated.

d. Spark Ignition System

A Bendix low-tension igniter system and a GLA high-tension igniter system were evaluated over a wide range of operating conditions. The Bendix low-tension igniter was selected because of its excellent performance over a wide range of pressures whereas the GLA high-tension system was found to be more limited. The spark plug ignition units were to be ultimately utilized in separate high mixture ratio stage ignition chambers with individual igniters. Testing was accomplished to define the ignition problems for a high mixture ratio igniter. The most significant problem area uncovered was a high temperature at the plug tip. As a result, some investigation of high temperature materials was conducted.

1962

The design of the spark ignition system for the thrust chamber, the wedge chamber, and the gas generator was completed during the second quarter of 1962.

Special equipment was constructed during the third quarter of 1962 for measuring the approximate amount of energy in the ignition spark system.

1963

The high-voltage coupling adapter for high-tension spark energy measurements were fabricated during the first quarter of 1963. Also, the Bendix and GLA ignition systems were undergoing tests.

Thrust chamber assembly igniter assembly firing tests were conducted at the Aerojet-General Azusa facility during the second quarter of 1963.

The component specification for the gas generator and the thrust chamber ignition system was released in May 1963.

e. Electrical In-Flight Sequence Unit

A thorough evaluation of available component operating characteristics was completed to determine their suitability for use in the severe environmental conditions. Data was obtained for many components, particularly the semi-conductors, for use in designing a satisfactory logic system for the in-flight sequence unit. Components demonstrating satisfactory performance characteristics were selected and no significant development problems were anticipated in establishing a design for the in-flight sequence unit assembly.

1962

A preliminary in-flight sequence unit design using NOR logic and power amplifier modules was completed during the third quarter of 1962 as was the circuit breadboard. Other specialized semi-conductors were also being tested. These included silicon-controlled rectifiers, avalanche diodes, tunnel diodes, and field effect transistors.

1963

The test plan for environmental testing of the in-flight sequence unit module assemblies was completed in April 1963. Fabrication of the assemblies was also completed and testing began.

Testing of the in-flight sequencer component parts as well as the logic and power gate circuits at cryogenic temperatures was completed during the fourth quarter of 1963.

1964

Design of the NOR logic, OR logic, power gate, patch panel, and back mounting panel for the in-flight sequence unit was completed during the fourth quarter of 1964.

1965

The electrical sequencer development effort was terminated during the first quarter of 1965 as a result of program redirection.

f. Thrust Chamber Pressure Switch

The Calips pressure switch was preferred by NASA/LeRC and it satisfied the design requirements for pre-fire checkout capability. The supplier (Southwestern Industries, Inc.) provided two different internal diaphragm linkage configurations, neither of which was completely satisfactory. One switch of each design failed to actuate within specification limits; and one of the two switches failed the diaphragm leakage tests. However, the remaining eight switches performed satisfactorily during functional testing. One switch successfully underwent an extensive testing program, which included vibration and extreme temperature tests.

3. System Seals

SUMMARY

Three general types of seals were evaluated for the M-1 engine; static seals, small tubing joints, and dynamic seals.

The scope, the status, and the M-1 seal program accomplishments are discussed in a separate report.(98) Another report (99) has also been issued in connection with this effort. It is a compilation of surveys and investigations of static and dynamic seals as well as joints performed over a seven-year period.

The static seals discussed in the second report referenced contains pertinent design information for static joints, required installation and handling procedures for the seals and flanges, and the results of tests conducted with different types of gaskets and joints. It deals primarily with lightweight joint designs larger than 1.00-in. sizes. The small tubing joint discussion deals primarily with tubing connections of 1.00-in. sizes and smaller. Welded, brazed, swaged, as well as the proprietary and military

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(98) Henson, F. M., M-1 Seals Program, Aerojet-General Report Aerojet-General Report No. 8800-62, 22 April 1966

(99) Henson, F. M., An Evaluation of Gaskets, Seals, and Joints for Aerospace Hardware, Aerojet-General Report No. 8800-39, 30 June 1966. (To be published as a NASA Low Series Contractor Report)

standard connections are included. The results of tests conducted with these connections are provided. The dynamic seal discussion is concerned with sliding and rotating shaft and piston seals for running speeds of 1000-in. per minute or less. Test results are also included for the seal designs tested.

a. Static Seals

The static seal effort was primarily concentrated upon the dual seal joint configurations utilizing Conoseal gaskets. Considerable development testing was accomplished with these joint designs using seal test cells. Component testing of the Conoseal test cell joints disclosed five basic problem areas. It was noted that leakage increased as thermal cycling was accomplished with dissimilar joints, the grooves appeared to deteriorate with the assembly cycles, bare aluminum grooves and gaskets corroded with time, the critical dimensions within the groove changed with thermal cycling, and that different radial deflections caused excessive leakage. The first four problem areas were satisfactorily resolved during the seals development program. The fifth problem, the differential radial deflection, remains unsolved. It was decided that the use of a soft secondary seal rather than a secondary Conoseal would minimize the adverse effects upon the deflection capabilities of the primary Conoseal gasket. However, this area of investigation was never resolved before the M-1 seals program was terminated.

CHRONOLOGY

1962

Based upon results from other programs, the Conoseal[R] was selected as the most suitable static seal at the outset of the program. A double-Conoseal joint design was selected which included a primary seal and a secondary seal. The purpose of the secondary seal was to monitor any leakage from the primary seal. In addition, the design layout of seal development hardware to simulate gas generator discharge and fuel turbine seal joints was completed.

Conoseal testing began during the third quarter of 1962. Testing of Teflon O-Rings, Omniseals, and Voi-Shan seals was also under way.

The seal identification and location charts for the M-1 engine were published during the fourth quarter of 1962.

(R) Registered trade name of the Aeroquip Corporation

1963

A 10-inch. double-conical seal test cell was received in April 1963. Two 12-in. double-conical test cells were tested. An alternative, the Infundibuli (funnel-shaped) seal (R) for the conical seal was being tested.

A revised Conoseal standard (AGC-STD-4808) was approved during the fourth quarter of 1963. Also, the fabrication of the 45-in. thrust-chamber-to-injector joint simulator and Conoseal gaskets were completed. Three gas generator valve joint simulator test cells in the 6-in. double-Conoseal configuration were completed.

1964

The fabrication of back-up seals (Infundibular, Aero-Cell, and dynamic seals) was suspended during the first quarter of 1964 because of the satisfactory results obtained with the Conoseal.

As part of the investigation of the C-9 failure, assembly tests were conducted during the third quarter of 1964 to simulate the condition of the distorted conical seals found in the elbow-to-oxidizer torus joint after the firing.(100)

1965

As a result of program redirection during the first quarter of 1965, all seals development and evaluation testing (45-in., 12-in., 10-in., and 6-in. test cells as well as small-bore joints and component hardware) was suspended and efforts were directed toward trouble and failure investigations of components used in the test program. One such effort was the investigation of pre-fire and post-fire leakage problems with the gas generator assembly during simulator tests at Test Stand H-8.(101)

An investigation was conducted during the second quarter of 1965 concerning the cause of a fire (resulting in premature shutdown) during a gas generator test at Test Stand H-8.(V102)(103)

(R) Registered trade name of the Futurecraft Corporation

(100) Failure Analysis of the Oxidizer Torus-to-Elbow Joint Conical Seals from M-1 Thrust Chamber Assembly Firing 1.1-02-EHM-003, Aerojet-General Systems and Controls Report No. 171, 30 July 1964

(101) M-1 Investigation of Gas Generator-Turbine Simulator Assembly Conoseal Joint Leakage Problems, Aerojet-General Report No. CPR-3, 16 February 1965

(102) M-1 Investigation of the Causes of Leakage and Subsequent Fire During Test No. 1.2-04-EHG-007 on Stand H-8, Aerojet-General Report No. CPR-7, 20 April 1965

(103) Investigation of the Gas Generator Boss-to-Dynisco Transducer Hot Gas Joint at H-8, Aerojet-General Report No. CPR-8, 27 April 1965

b. Small Tubing Joints

SUMMARY

The single-seal bolted joints as well as the many types of available threaded, brazed, and swaged connections were evaluated in relationship to the stringent sealing requirements of the M-1 Engine system.

The designs for test hardware were primarily concerned with the smaller tubing joint sizes. Designs were completed for single-seal bolted joints of 1.50-in. and smaller. Also, for threaded connections of 1.00-in. and smaller.

The problems encountered with the small tubing joints were related primarily to the quality control of fabrication and installation procedures.

c. Dynamic Seals

There was only limited testing accomplished with dynamic seals for controls components applications. Dynamic seal testers were used for this purpose.

CHRONOLOGY

1962

A supplier survey to determine the availability of dynamic seals for cryogenic service was initiated at the outset of the program.

Detailed drawings of the dynamic seal test cells were completed during the fourth quarter of 1962.

1963

The preliminary development test plan for dynamic seal testing was completed in June 1963.

A thorough program was planned for the testing of the many types of reciprocating dynamic seal design for the M-1 Engine. A test cell was designed and testing was to encompass the variances of pressure, temperature, surface speed, dynamic and static friction, leakage rates, squeeze or interference sealing stresses, life cycling, and combinations of these variables. However, these tests were not conducted because of the reduction in M-1 program scope. Useful lipseal data was collected in connection with the testing of the sleeve-type thrust chamber valve and this information was reported separately.(104)

(104) NASA Report No. 54808, op. cit.

VI. GROUND SUPPORT EQUIPMENT

SUMMARY

A final and highly detailed report providing a functional as well as a technical description of the ground support equipment (GSE) developed in support of the M-1 Engine Program was prepared as part of the M-1 Engine Phaseout effort.(105) This final GSE report also identified the status of the equipment in relationship to engineering, fabrication, and testing. Therefore, this presentation is limited to a brief chronological history of the development of this equipment along with sufficient technical description to identify appropriate items.

The ground support equipment effort spanned the period from the outset at contract award through the third quarter of 1965. During this period of time, a variety of transportation, handling, and decontamination equipment was developed along with appropriate maintenance and tool kits. The over-all requirements for ground support equipment and the items developed to satisfy these requirements are briefly discussed herein.

Over-all GSE requirements were established at the outset of the program and this planning data was extended to prepare listings of all projected ground support equipment that would be needed for the M-1 Program as originally defined. In addition, preliminary interface requirements between the engine and the GSE were established for checkout, monitoring, control, handling, and transportation of the engine. As M-1 Engine parameters became specifically defined, detailed interface requirements were fixed.

CHRONOLOGY

1962

A. HANDLING AND TRANSPORTATION EQUIPMENT

1. Requirements

The following basic requirements were established at the outset of the program.

a. The engine will be assembled, transported, and stored on a basic engine-handling frame.

b. The engine will be installed on the test stand by using slings and a lifting device.

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(105) Hartwig, R. A., Ground Support Equipment, M-1 Program, Aerojet-General Report No. 8800-32, 22 November 1965

1962

c. The engine without the nozzle extension, will be transported on a low-boy semitrailer. The engine frame will be used to support the engine and appropriate cushioning will be provided for the necessary shock isolation.

d. A hydraulic boom will be used to install large components on the engine as well as to remove them from the engine.

2. Investigations and Studies

Helicopters were investigated early in the program as a means of transporting an assembled M-1 Engine nozzle extension. They were found to be capable of providing such transport.

1963

During the first quarter of 1963, a transportation study covering the movement of the M-1 Engine to its test site at Aerojet-General, Sacramento, as well as to Cape Canaveral, Florida, was accomplished.

The route for transporting the thrust chamber assembly from the fabrication area to Test Stand C-9 using the rocket engine semitrailer (GSE Item 110) was outlined. This routing was also extended to Test Stand H-8 and K-Area. Figure No. 26 shows this on-plant routing to Test Stand C-9.

For off-plant airway transportation, Aero Spacelines, Inc. of Van Nuys, California, provided information about the Boeing 377-PG "Guppy." The cargo space of this aircraft is 19-ft-8-in. in diameter by 42 ft in length. A pallet, 9 ft by 42 ft, is used to secure large shipments to the aircraft. The range of this plane is 1,500 miles with a payload of 40,000 lb.

Also for off-plant transportation, the Bigge Drayage Company of San Leandro, California, prepared a proposal covering truck transport of the M-1 Engine to Alameda California, dockside and/or cross-country to Cape Canaveral.

Off-plant transportation by means of a water route was investigated. A route survey between Aerojet-General, Sacramento, and the Sacramento River was completed (see Figure No. 27). The River Line, Inc. of Sacramento, California, was capable of providing transportation from the loading dock on the Sacramento River to dockside in San Francisco. The Smith Rice Company of San Francisco could provide floating derricks to transfer the M-1 Engine from the barge to shipboard. The States-Marine Isthmian Agency, Inc. of San Francisco would provide the transportation from San Francisco to Cape Canaveral, Florida.

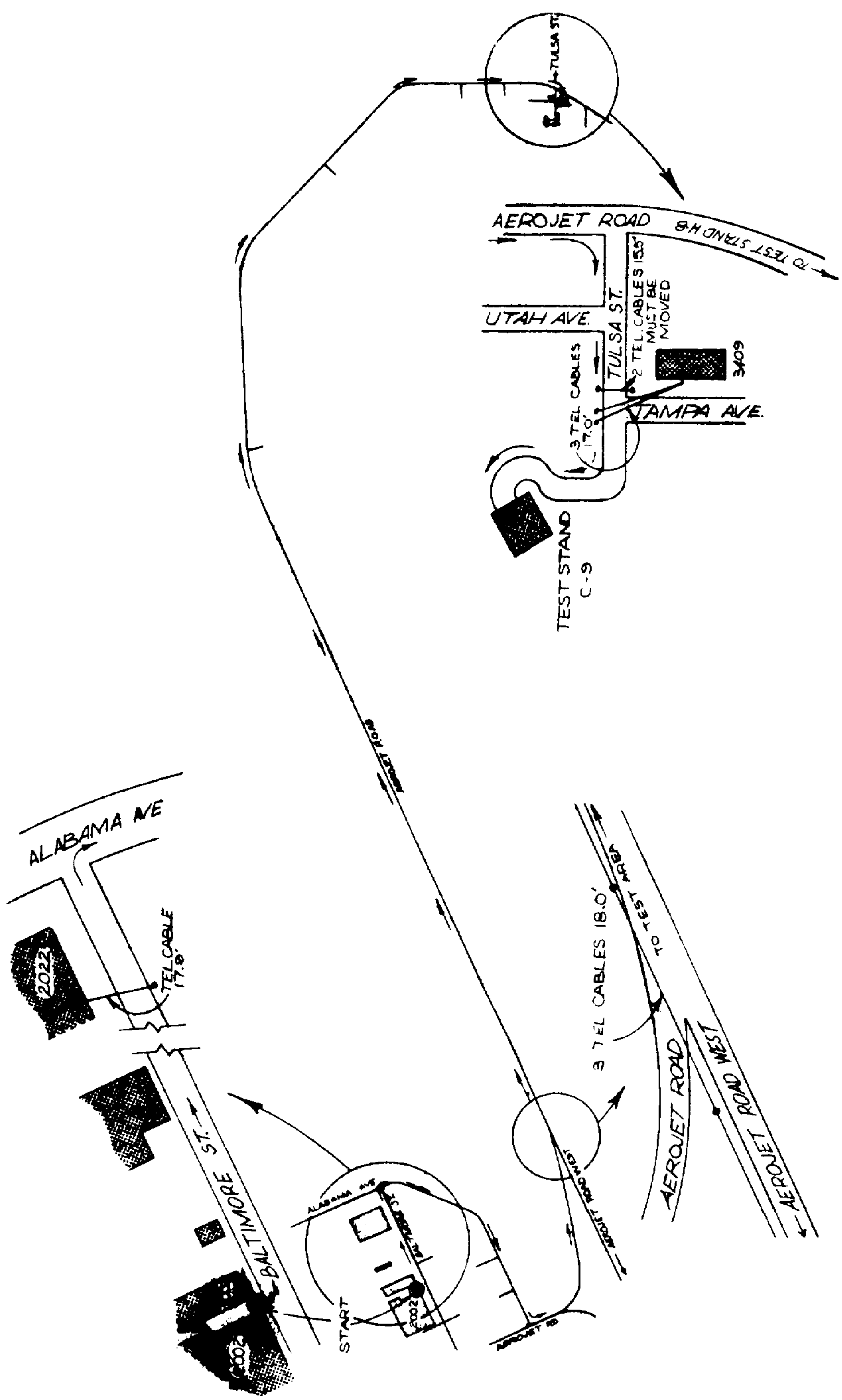


Figure 26. On-Plant Routing for M-1 Engine

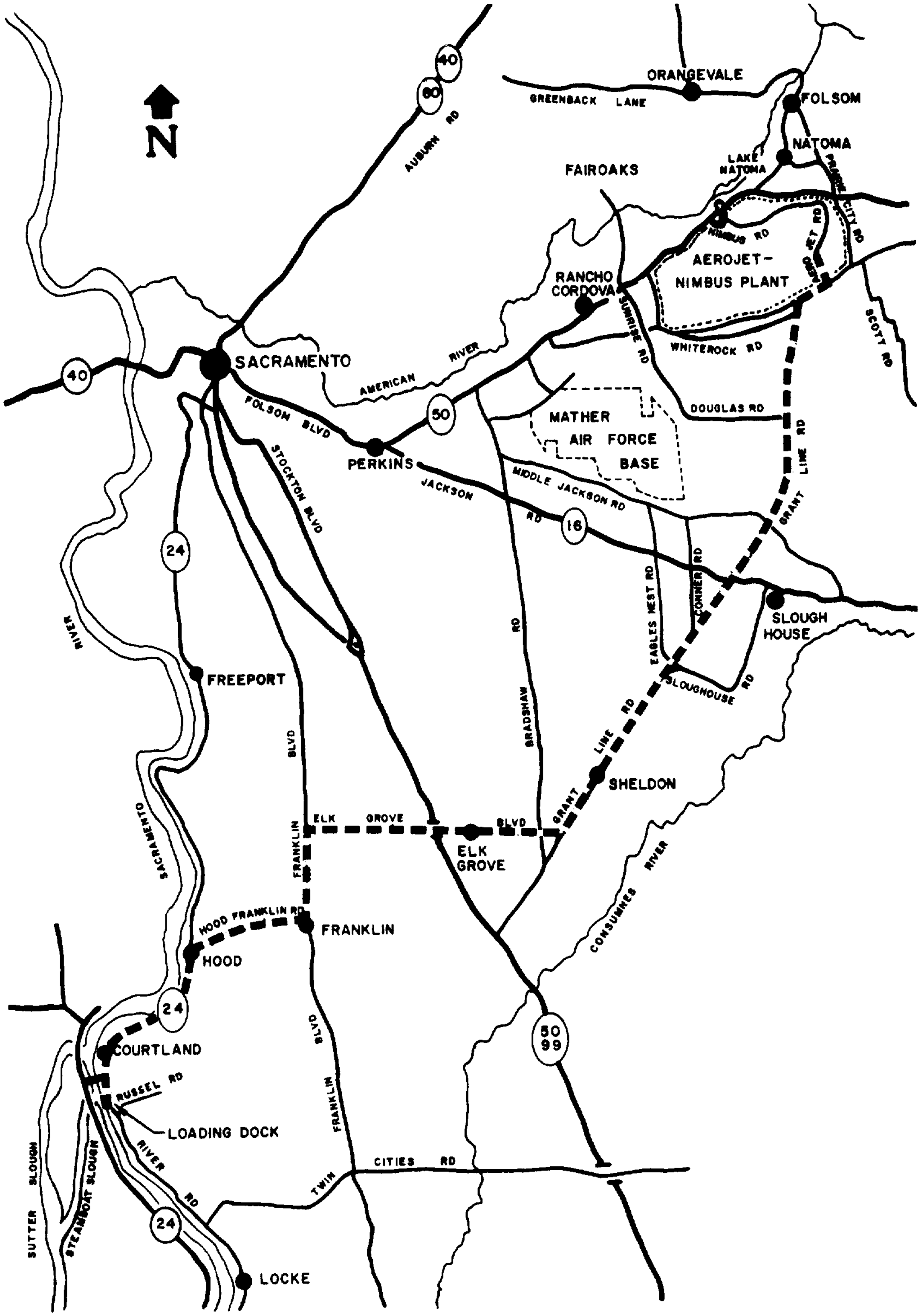


Figure 27. Off-Plant Transport Route to the Sacramento ... for the M-1 Engine

1963

At the outset of the program, it was decided to investigate "Air'Ride" suspension for use in highway transportation of the M-1 Engine. This was accomplished by means of tractor-trailer combination road tests conducted during the third quarter of 1962. A full "Air-Ride" suspension Kenworth-Peterbilt Air-Ride Tractor was used in a 60 minute test to determine the input levels of acceleration transmitted from the trailer and tractor axles to the trailer bed. It was found that the registered acceleration recorded with this van bed was equivalent to the registered acceleration recorded during previous van tests were leaf-spring suspension and combination "Air-Ride" and leaf-spring suspension had been used. Therefore, any need for an "Air-Ride" suspension system was eliminated.

3. M-110 Semitrailer (P/N 270000)

1963/1964

This equipment (see Figure No. 28) was designed specifically for transportating the M-1 Engine or the 14:1 Thrust Chamber Assembly set in the Rocket Engine Stand (M-111) in a vertical position between the various facilities of the Aerojet-General Sacramento Plant. It is also used to rotate the engine or thrust chamber assembly to a horizontal position upon delivery to a horizontal test stand.

During loading or unloading operations, the two retractable landing gears at the gooseneck of the trailer and the two leveling jacks at the rear of the trailer are lowered to provide stability.

Twelve tie-down points are provided on the chassis; two forward, four on each side, and two on the rear. These can be used to lift the vehicle, secure it during transport, or for attaching safety lanyards to the M-1 Engine when it is being transported.

Compartments, one on each side, are provided on the chassis for storage purposes.

4. M-111 Rocket Engine Stand (P/N 270001)

1962/1965

The M/111 Rocket Engine Stand was specifically designed to support the M-1 Engine and to protect the thrust chamber aft flange during engine build-up, transport, test stand installation, and storage. The stand can be placed on a shop dolly during engine build-up for ease of movement and upon completion of build-up, it can be moved onto the semitrailer for transport of the engine. The rubber shock isolation mounts in the legs of the stand protect the engine from vibratory energy inputs during build-up and transport.

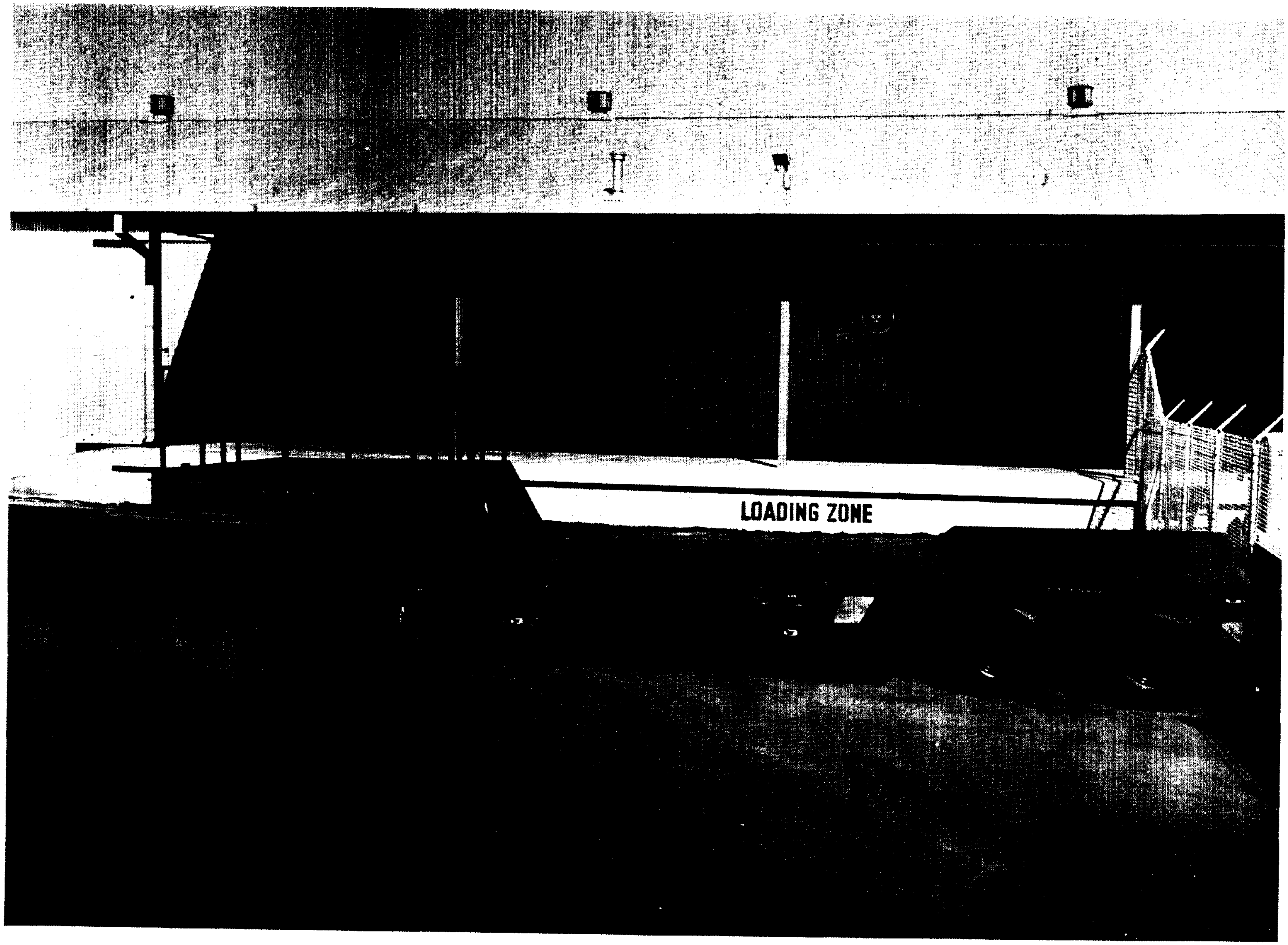


Figure 28. M-110 Semitrailer

1962/1965

Final design reviews for the rocket engine stand were completed by the end of 1962. Drawings were completed and released along with fabrication orders for four (subsequently reduced to two) M-111 Rocket Engine Stands by the end of the second quarter of 1963.

A study was completed in November 1963 in connection with the in-plant transportation of the 25:1 area ratio nozzle extension using the M-110 Semitrailer and the M-111 Rocket Engine Assembly Stand. It was found that these items of ground support equipment would be applicable; however, an adapter (not identified as an item of GSE) would be required between the M-111 stand and the forward flange of the 25:1 nozzle extension. The final design for this modification was completed in August 1964.

All work on the two M-111 stands was suspended in February 1965 as a result of the M-1 phaseout direction. At that time, the only work remaining was the fabrication and installation of 16 clamps and two sling brackets. The clamps are used to secure the thrust chamber aft flange to the stand and the slings are used for rotation of the chamber.

5. M-114 Rocket Engine Cover (P/N 270190)

1962/1964

This cover was designed to protect the M-1 Engine from casual observation, dirt, and dust while being transported or stored on the M-111 stand. The cover is installed by slipping the lower half under the M-111 and the upper half over the top of the engine. The two halves are then zipped together and drawstrings are tightened. A bottom inspection panel is removable without disturbing the rest of the cover. This barrel-shaped cover is made from 13 oz neophrene-impregnated nylon cloth which is aluminized on the outside. It is approximately 14 ft high and 15 ft in diameter.

Drawings for the cover were completed by the end of 1962. These drawings were released along with a fabrication order for two covers during June 1963. Two covers were completed in 1964.

6. M-120 Horizontal Engine Sling (P/N 1119685)

The M-120 sling was designed to support the aft end of the M-1 Engine whenever it is lifted into a horizontal attitude. It can also be used for the 14:1 area ratio thrust chamber to assist in rotating it. The M-120 is a two-legged sling with an 81-in. wide spreader bar. A pear-shaped link of 16-1/2-ton capacity connects the bridle legs and provides an interface for a hoisting device. Its over-all height is 97-1/2-in. The sling is designed for a capacity of 30,000 lb with a safety factor of 3 based upon the material yield strength.

1962/1964

The original design consisted of two slings with one (P/N 270002-9) attaching to clevises on the thrust chamber forward flange and the other (P/N 270002-19) attaching to the M-111 Rocket Engine Stand. One of each sling was manufactured in 1963. However, this original design became obsolete in 1965 as a result of modifications in the thrust chamber interfaces. No manufacture of the new design was ever ordered because of the M-1 Program phaseout.

7. <u>M-121 Vertical Engine Sling (P/N 1119618)</u>

1962/1963

The M-121 slings was designed to lift the 14:1 area ratio thrust chamber assembly into a vertical attitude. It is also used to assist in rotating this thrust chamber. Its ultimate use is to lift a complete M-1 Engine in a vertical attitude; however, to accomplish this, the sling capacity requires an increase and new interface fittings would have to be designed. The M-121 is a two-legged sling with a 36-in. wide spreader bar. A pear-shaped link of 16-1/2-ton capacity connects the bridle legs and provides an interface for a hoisting device. Its over-all height is 88-in. A swivel fitting is attached to the lower end of each lifting leg to provide an interface with the thrust chamber assembly dummy gimbal. The sling is designed for a capacity of 20,000 lb with a safety factor of 3 based upon the material yield strength.

The original M-121 configuration consisted of a four-legged sling that attached to clevises on the thrust chamber forward flange. Three M-121 slings were manufactured in 1963. However, this original design became obsolete in 1965 as a result of modifications in the thrust chamber interfaces. No manufacture of the new design was ever ordered because of the M-1 Program phaseout.

8. <u>M-130 Multi-Purpose Maintenance Stand (P/N 270003)</u>

This multi-purpose maintenance stand is a mobile unit having a rectangular chassis (see Figure No. 29). Four tubular uprights, which support a head that can be rotated manually, are mounted on the chassis. Its over-all dimensions are: length, 5 ft; width, 5 ft; and height, 3-1/2 ft. A two-man "push" handle, which can be removed and stored on the chassis is provided for moving the stand short distances. It will support a maximum load of 1000 lb up to a cantilever distance of 18-in. measured from the face-plate of the rotatable head. Beyond 18-in., the maximum load (lb) is calculated by dividing 18,000 by the distance (in.) of the load center-of-gravity from the faceplate. The output torque capacity of the gear reducer is 6000 in. -lb at 100 rpm or less.

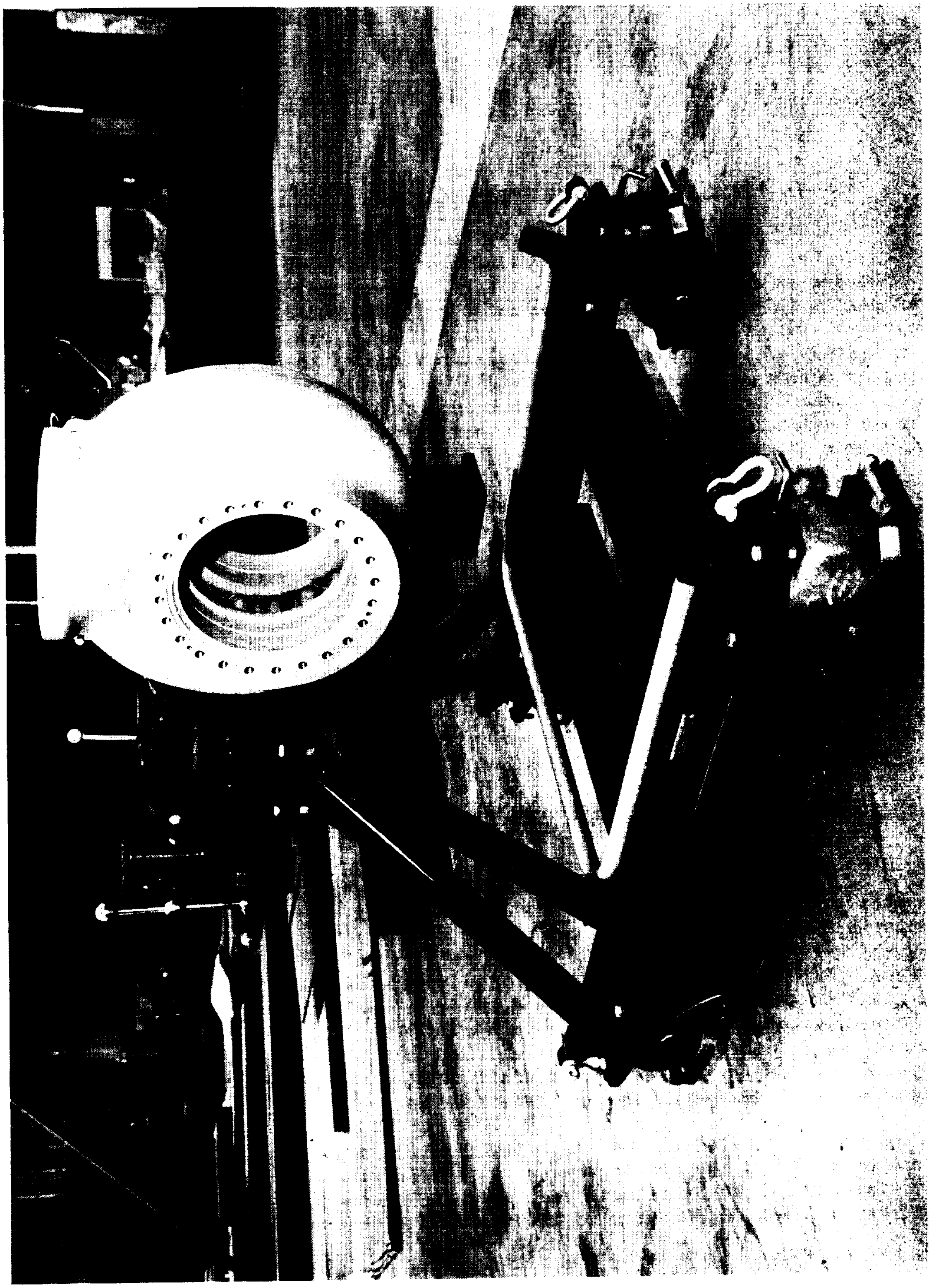


Figure 29. M-130 Multi-Purpose Maintenance Stand

1962/1964

9. M-131 Thrust Chamber Valve Adapter (P/N 270450)

The M-131 adapter provides a means for attaching the M-1 thrust chamber valves to the M-130 Maintenance Stand and/or the M-170 sling. This adapter is made up of 3-in. diameter aluminum tubing with faceplates, which permit this attachment to the inter-related ground support equipment. Its over-all dimensions are: 26-in. across, 31-in. deep, and it weighs approximately 35 lb. Threaded holes are provided at the flange ends of the tubing to permit attachment to the M-1 thrust chamber valve body. The bolts used to fasten the adapter to the valve body are secured to the flanges by means of chains. A second set of holes is provided in the flanges to permit the attachment of a hoisting device.

Because the M-1 fuel and oxidizer valves utilize identical bodies, the adapter is used to service both valves.

The final design review for this adapter was completed by the end of 1962 and the drawings were released during the first quarter of 1963. Three M-131 adapters of the P/N 270450-9 configuration were completed in October 1963. This configuration was used for handling sleeve-type thrust chamber valves. In November 1964, a minor redesign of the adapter converted it to handle both the sleeve-type and the poppet-type of thrust chamber valve. This modified adapter was assigned the identification of P/N 270450-19; however, as a result of the M-1 Program phaseout, no manufacturing of the new design was ever ordered.

10. M-132 Gas Generator Adapter (P/N 270460)

1962/1963

The M-132 adapter provides a means for attaching the M-1 Gas Generator to the M-130 stand and/or the M-170 sling. It is a swing-type clamp with two integral faceplates. The clamp, which is 10-in. deep, 17-in. wide, and weighs approximately 30 lb, solidly grips an 8.67 $\pm$ 0.05-in. diameter. Rubber strips are bonded to the inside of the adapter to allow for variations in the gas generator body diameter as well as to prevent nicks and scratches. A dowel pin, which fits into a hole drilled into the gas generator, acts as a positioner for the adapter and prevents slippage. The vise-type handle of the adapter is designed to fail by bending before over-torquing can inflict crushing damage to the gas generator body.

The final design review for this adapter was completed by the end of 1962 and the drawings were released during the first quarter of 1963. Three M-132 adapters were completed in December 1963 and they were structurally proof load tested to 1100 lb during that same month.

1962/1963

11. M-133 Gas Generator Valve Adapter (P/N 270470)

This adapter was designed to provide a means for attaching the gas generator fuel valves to the M-130 stand and/or M-170 sling. It is a tubular framework with clamps and faceplates. The adapter is 25-in. in diameter, 20-in. long, and weighs approximately 33 lb.

Shortly after the M-133 adapter design was completed early in 1963, it was decided to subassemble the valve on the gas generator. As a result, the need for this adapter was eliminated. Raw stock and commercial hardware for three M-133 adapters had been ordered in January 1963, but all of the material on hand was declared surplus in April 1963 when the unit was no longer needed.

12. M-140 Turbopump Assembly Stand (P/N 270800)

1963/1965

The M-140 stand provides a means for assembling, disassembling, and maintaining the M-1 turbopumps. It is made up of four separate assemblies; a roll-away elevator, a variable height work platform, an access ladder, and a trunnion assembly mounted on two stanchions. None of these assemblies are mechanically connected to each other except for the access ladder which has a safety attachment to the work platform. All of the assemblies are mounted on a common floor base.

Figure No. 30 shows a fully assembled oxidizer turbopump assembly installed into the stand.

13. M-141 Fuel Turbopump Assembly Adapter Kit (P/N 270489)

1963/1964

The M-141 kit is located and attached to the M-140 stand via four bolt holes in the stand trunnion assembly faceplate and the bearing on the left hand stanchion. It is used in conjunction with the stand to assemble, disassemble, and/or maintain the M-1 Fuel Turbopumps. The kit consists of two table adapters and one yoke adapter; all fabricated from aluminum castings. One table adapter is 40-in. in diameter and 30-in. high while the other is 28-in. in diameter and 12-in. high. The yoke adapter is approximately 75-in. long, 32-in. wide, and 22-in. high.

All adapters in the M-141 kit are designed to support 8000 lb with a safety factor of 4 based upon the material yield strength.

All surfaces of the aluminum castings are anodized to keep the pores from collecting dust, thereby assuring compliance with the cleanliness specification requirements of the Pump Assembly Room.

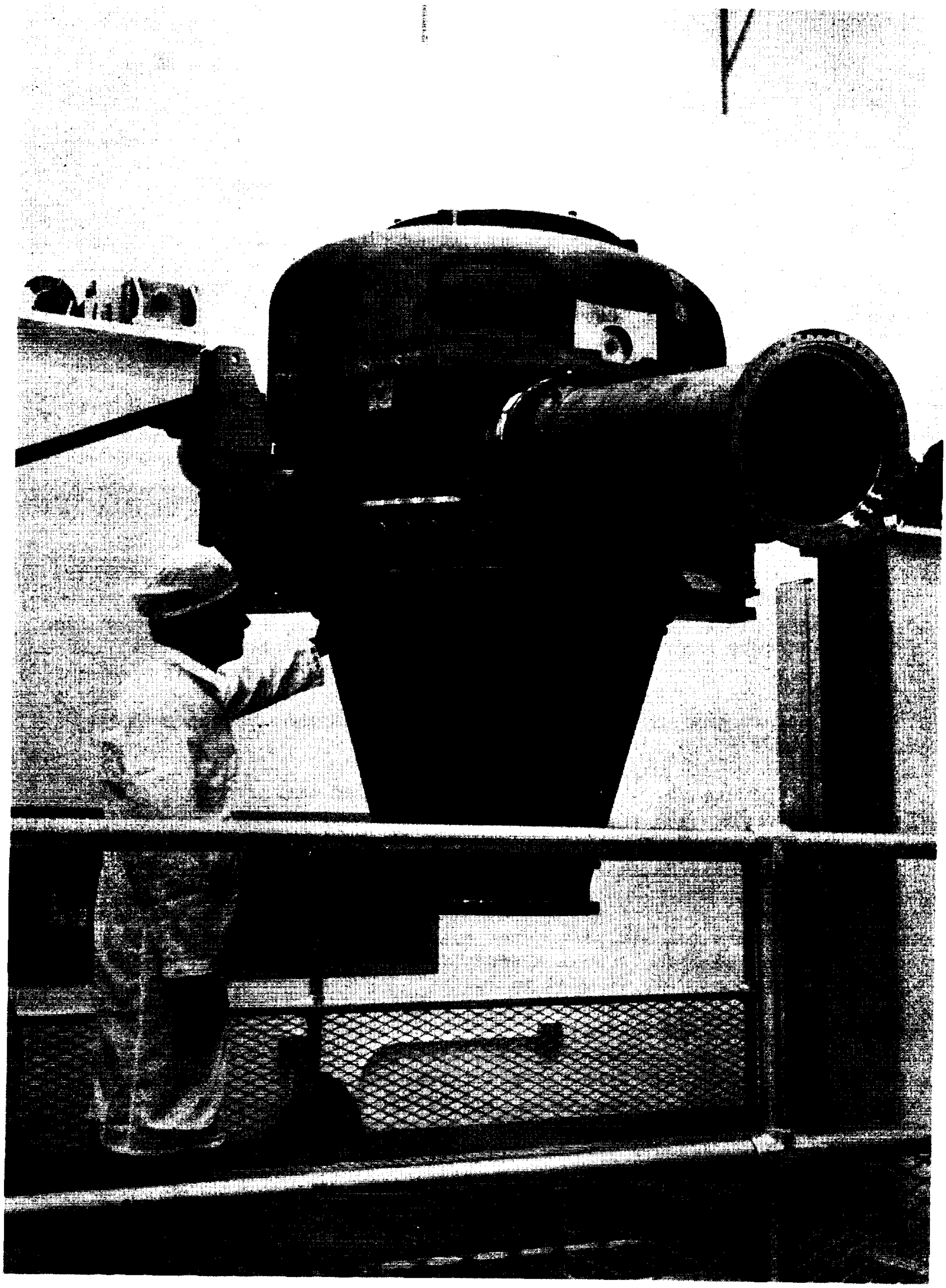


Figure 30. Fully-Assembled M-1 OTPA Installed in the M-140 Stand

1963/1964

The design of the M-141 kit was initiated at the end of 1963. Drawings and fabrication orders were released early in 1964. One kit was manufactured in 1964 and completed all structural and compatibility tests.

14. M-142 Oxidizer Turbopump Assembly Adapter Kits (P/N 270310 and P/N 270499)

1963

Both kits are used in conjunction with the M-140 turbopump assembly stand for assembly, disassembly, and/or maintenance purposes. The P/N 270499 kit is used for the Model I oxidizer turbopumps while the P/N 270310 kit is used for the Model II oxidizer turbopumps.

The M-142 P/N 270499 OTPA adapter kit consists of one yoke adapter and one table adapter; both fabricated from low carbon steel. The yoke adapter is approximately 75-in. long, 32-in. wide, and 22-in. high. The table adapter is 42-in. in diameter and 30-1/2-in. high.

The M-142 P/N 270310 OTPA adapter kit consists of one yoke adapter and two table adapters; all are fabricated from aluminum castings. The yoke adapter is similar to the P/N 270499 one both in size and shape. One table adapter is 42-in. in diameter and 20-in. high while the other is 42-in. in diameter and 10-in. high. Both adapters are similar in appearance to the P/N 270499 table adapter.

All adapters in both kits are designed to support 10,000 lb with a minimum safety factor of 3 based upon the material yield strength.

The aluminum casting surfaces are anodized and the steel parts are electroless-nickel plated to satisfy the cleanliness specification requirements of the Pump Assembly Room.

The final design of the M-142 kit was completed and fabrication orders released during the second quarter of 1963. The kit was redesigned during the fourth quarter of 1963. One P/N 270499 kit was manufactured in 1963 and subjected to structural proof load and compatibility tests. Although three P/N 270310 kits were ordered, no fabrication was ever initiated because the M-1 Program entered its phaseout stage.

15. M-146 Transport and Storage Stand (P/N 270210)

1963/1964

The M-146 stand provides a means for handling, storing, or in-plant transportation of an M-1 fuel or oxidizer turbopump in the vertical position. However, because no provisions were made for shock isolation,

1963/1964

transport by truck is limited to five miles-per-hour over graded gravel roads and ten miles-per-hour over paved roads. The stand consists of one set of oxidizer turbopump adapters and one set of fuel turbopump adapters. Figure No. 31 shows the stand with the fuel turbopump adapters.

16. M-150 Turbopump Assembly Slings (P/N 1120089 and P/N 1120045)

1964/1965

The M-150 P/N 1120089 sling is a four-legged sling with a 55-in. spreader bar. A movable pear-shaped link with a 19-ton rated capacity provides the interface for the hoisting device. Two sets of end fittings act as the interfaces for the fuel and oxidizer turbopumps. All sling end fittings are fabricated from stainless steel or aluminum. The wire ropes are of stainless steel and the spreader bar is made from electroless-nickel plated structural steel. This sling is intended for permanent use in the Turbopump Assembly Clean Room. It can be used for lifting complete pump assemblies or individual pump components. This sling has a capacity of 16,000 lb and a minimum safety factor of 3 based upon the material yield strength.

17. M-170 Multi-Purpose Sling (P/N 270007)

1963/1964

The M-170 sling is designed in four configurations to satisfy the different requirements for which it is needed. For example, during engine assembly, the sling must support engine components so that they are perfectly aligned to permit mating of the conical seal joints whereas during normal shop handling, alignment is of no concern. Thus, for reasons of economy, the sling is provided in the P/N 270007-9, P/N 270007-19, P/N 270007-29, and P/N 270007-39 configurations. These configurations differ only in the extent of the adjustment provided.

B. DECONTAMINATION EQUIPMENT

1. M-440 Decontaminator (P/N 271803)

The M-440 Decontaminator is used to heat, regulate, and supply facility nitrogen to critical components of the M-1 Engine to eliminate both air and moisture. If necessary, this is followed by a helium purge to replace the nitrogen atmosphere in the fuel side of the engine system.

The M-440 is a mobile unit containing an electrically-controlled hydraulic/pneumatic system for supplying gases to purge moisture from the M-1 Engine. This system is housed in a weather-resistant all-metal enclosure mounted on a four-wheel chassis equipped with pneumatic tires and parking

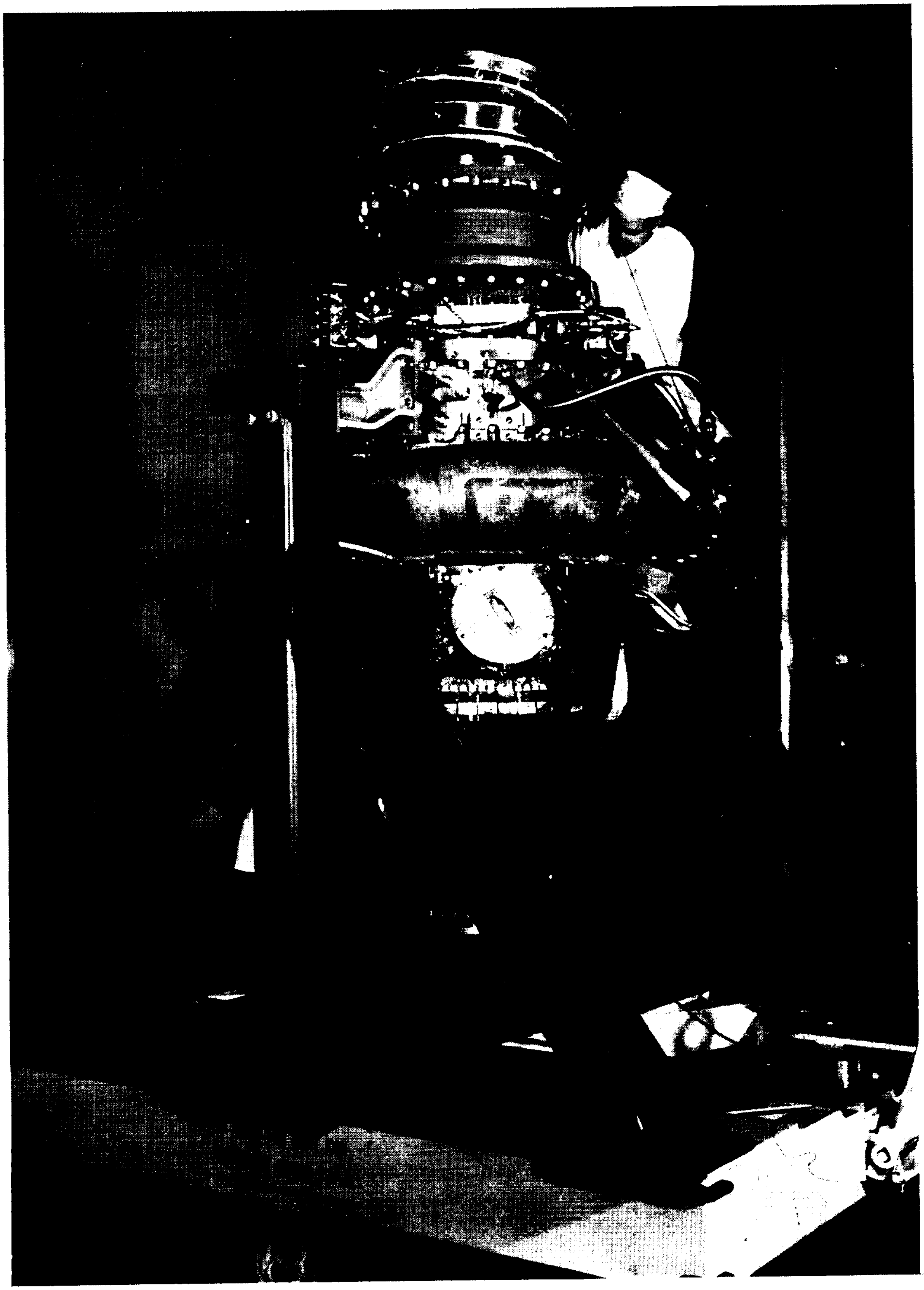


Figure 31. M-146 Transport and Storage Stand (P/N 270210)

brakes. The unit has a drawbar which permits towing at a maximum speed of 7-1/2 mph. Its over-all dimensions are: width, 6 ft; length, 8 ft; and height, 7 ft. Figures No. 32 and No. 33 are exterior views of the unit while Figure No. 34 is an interior view.

2. M-445 Vacuum Unit - Kinney Vacuum Co. (P/N MVP 130 TF)

The M-445 is a commercially-procured unit that is used in conjunction with the M-440 Decontaminator to lower the air pressure in the engine to reduce the nitrogen purge drying time. It is a vacuum pump mounted on a two-wheel trailer. Its over-all size is: width, 5 ft 6-in.; length, 8 ft; and height, 5 ft.

C. MAINTENANCE AND TOOL KITS

The general needs for maintenance and tool kits were defined at the outset of the contract. Specific requirements were dependent upon the final engine configuration and the component configurations as well as the engine mock-up.

1. M-710 Engine Maintenance Kit

It is intended that this kit will be made up of the special tools, fixtures, and adapter required to maintain the M-1 Engine as well as those needed to support installation or removal of the engine from the test stand. Only one tool was completed, the Suction Line Compressor (P/N 270550-9).

2. M-730 Oxidizer Turbopump Assembly Pressure Test Kit (P/N 297300)

1963/1964

This kit provides the closures needed to seal the oxidizer turbopump openings so as to retain the pressures applied within the turbopump during leak tests of the seal assemblies and the pressure sealing areas. Actually, the M-730 is made up of two kits. One, the P/N 297300-9 kit is used for servicing the Model I Oxidizer Turbopump Assembly; the other, P/N 297300-19 kit is used to service the Model II Oxidizer Turbopump Assembly.

The M-730 kit is intended for use in the Pump Assembly Clean Room; therefore, all of the closures contained in the kit were designed to satisfy the appropriate cleanliness requirements.

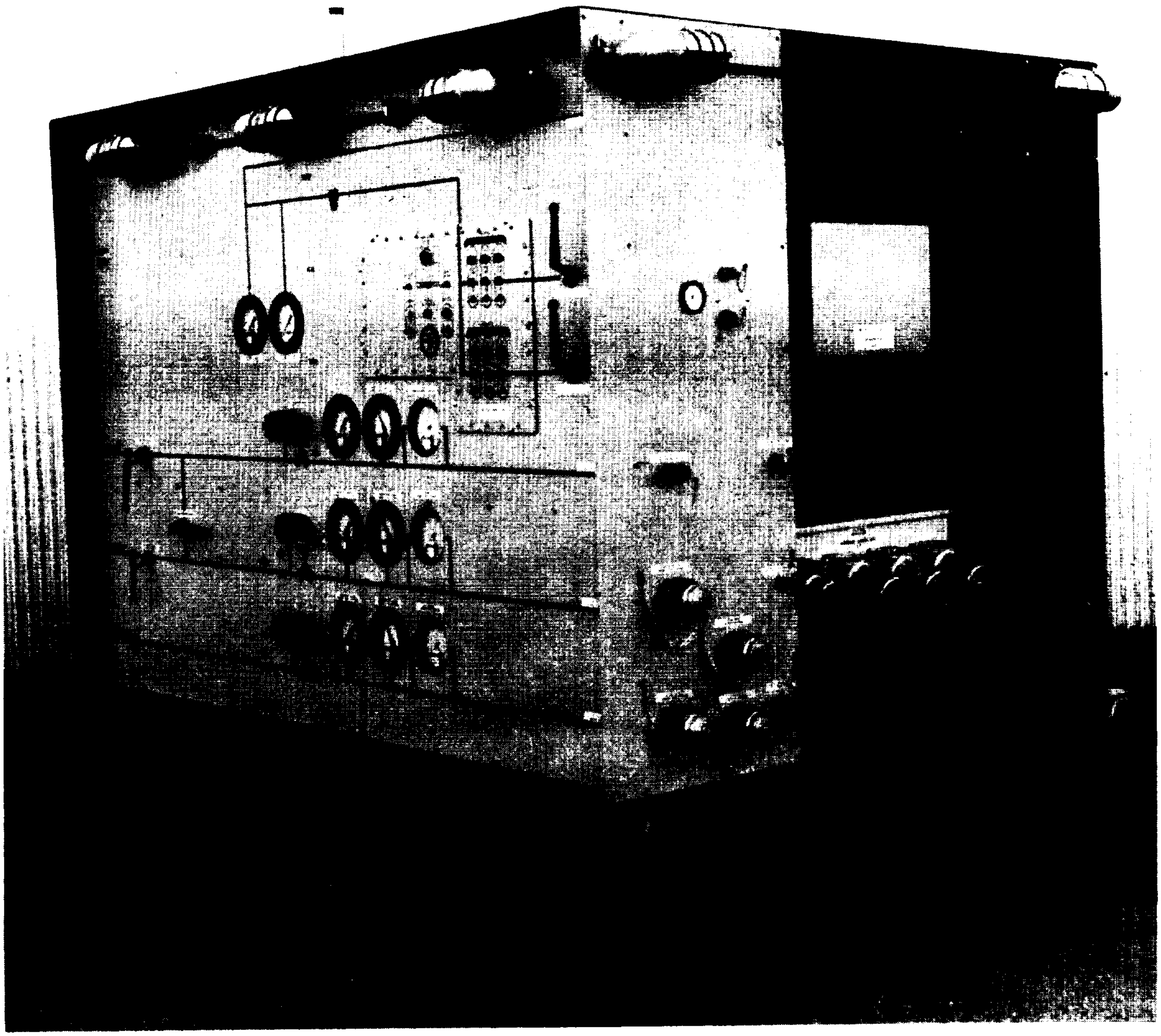
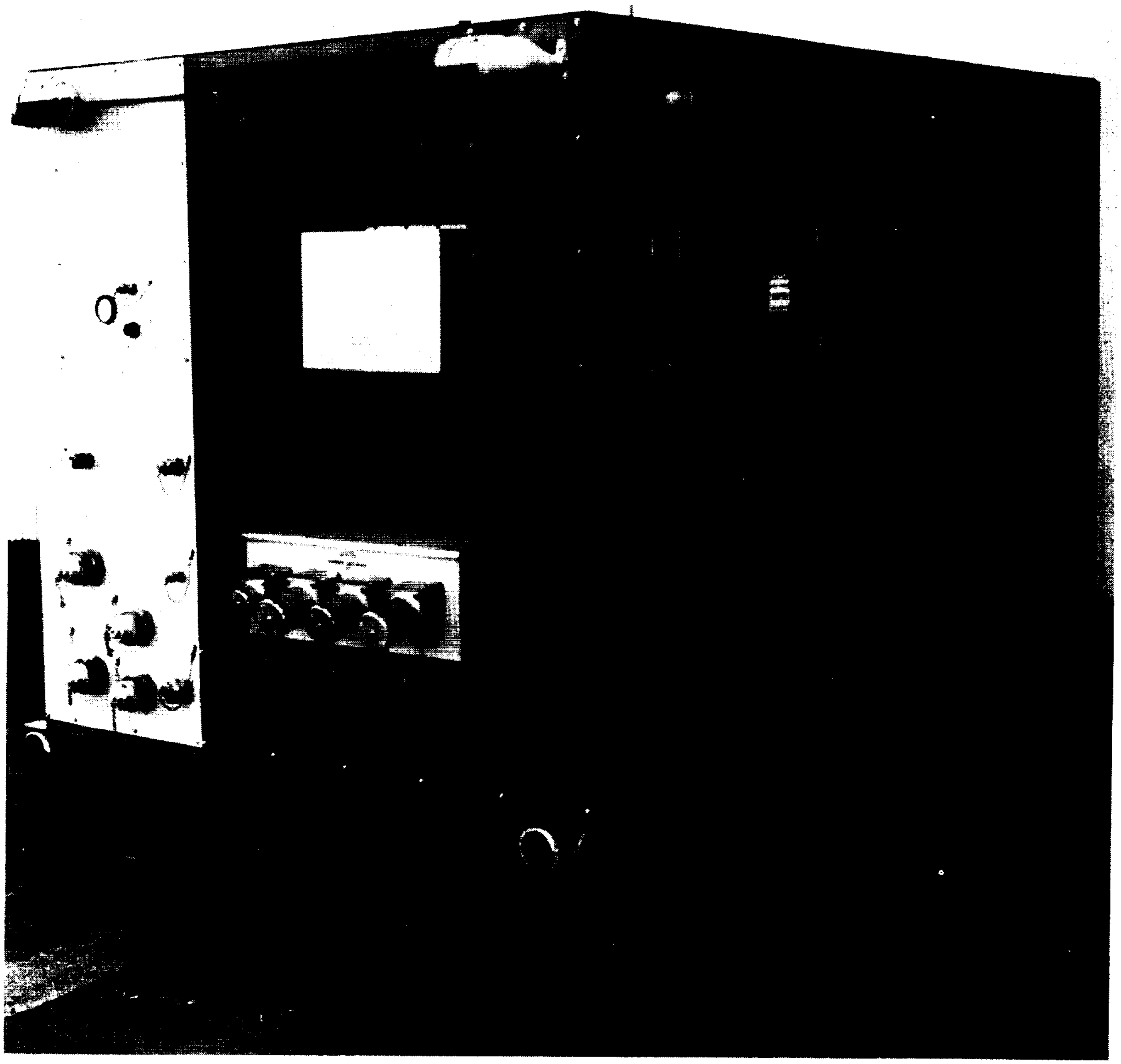


Figure 32. M-440 Decontaminator (P/N 271803) (Front View)



: 33. M-440 Decontaiminator (Rear

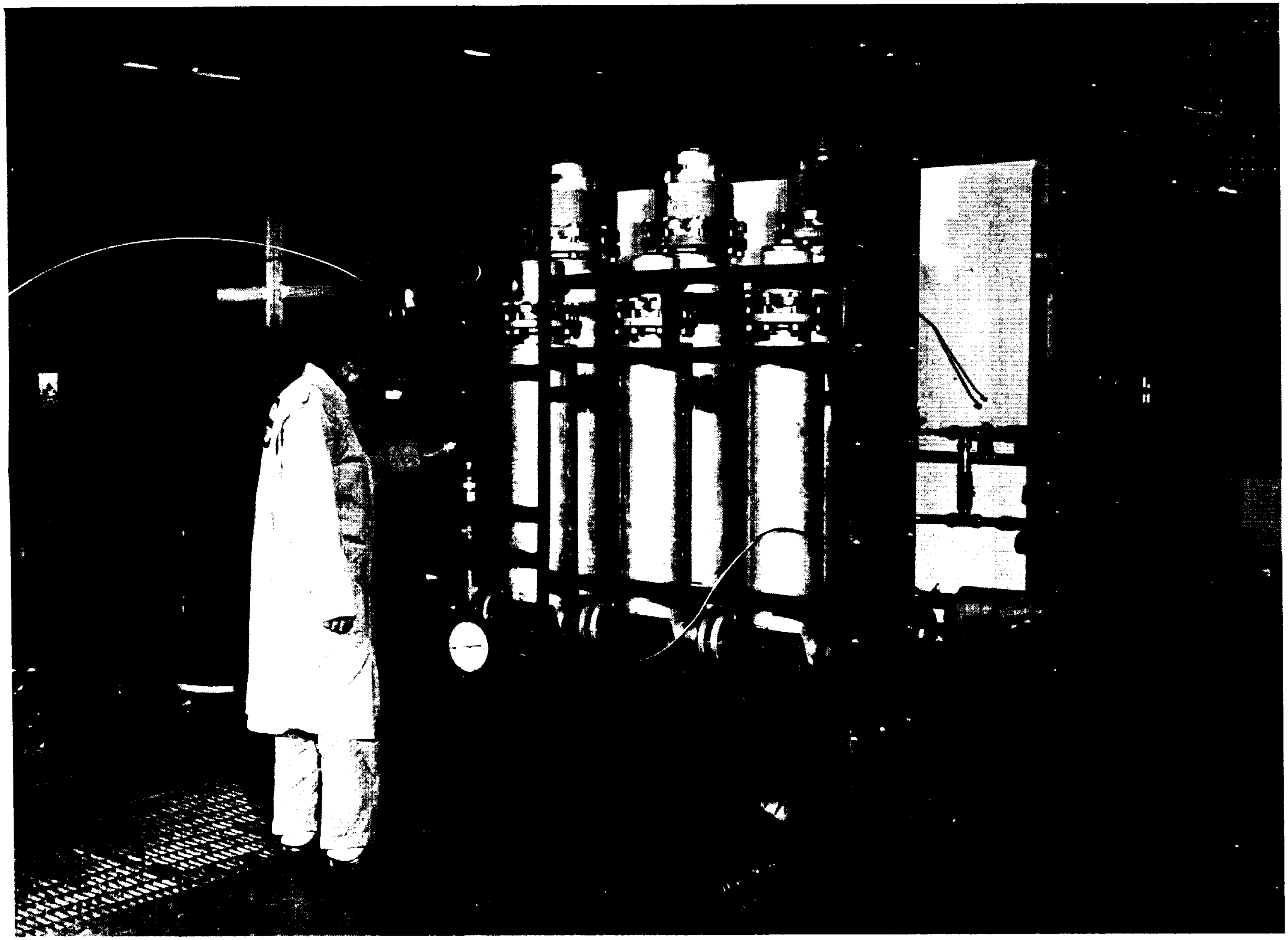


Figure 34. M-440 Decontaiminator (Internal View)

1963/1964

The M-730 kit drawings were completed at the end of 1963 and fabrication orders were released. All designs were completed during the third quarter of 1964 and one P/N 297300-9 kit was manufactured by the end of 1964. All proof pressure and compatibility tests were completed for this kit.

No fabrication activity was initiated for the P/N 297300-19 kit.

3. M-731 Fuel Turbopump Assembly Pressure Test Kit (P/N 297400)

1963

This kit provides the closures needed to seal the fuel turbopump assembly openings so as to retain the pressures applied within the turbopump during leak tests of the seal assemblies and the pressure sealing areas. Actually, the M-731 consists of two kits. One, the P/N 297400-9 kit, consists of ten closures to be used with the Model I Fuel Turbopump Assembly; the other, the P/N 297400-19 kit, contains eleven closures and is to be used with the Model II Fuel Turbopump Assembly.

The M-731 kit is intended for use in the Pump Assembly Clean Room; therefore, all of the closures contained in the kit were designed to satisfy the appropriate cleanliness specifications.

The conceptual design for this kit was completed, reviewed, and accepted by the end of 1963. One P/N 297400-9 kit was manufactured in 1964. All proof pressure and compatibility tests were completed for this kit.

No fabrication activity was initiated for the P/N 297400-19 kit.

4. M-740 Oxidizer Turbopump Assembly Maintenance Equipment Kit (P/N 273350)

1963/1964

This kit contains the special equipment needed to assemble, disassemble, and maintain both the Model I and the Model II Oxidizer Turbopump Assemblies. The equipment is used in replacing impellers, housings, seals, deflectors, turbine rotors, couplings, nozzles, and manifolds. Actually, the M-740 consists of two kits. One, the P/N 273350-9 kit, supplies 19 items of special equipment used in the assembly of the Model I Oxidizer Turbopump Assembly; the other, the P/N 273350-19 kit, supplies 18 items of special equipment used in the assembly of the Model II Oxidizer Turbopump Assembly.

The M-740 kit is intended for use in the Pump Assembly Clean Room; therefore, all of the items contained in the kit were designed to satisfy the appropriate cleanliness specifications.

The M-740 kit drawings were completed at the end of 1963 and fabrication orders were released. All designs were completed during the third quarter of 1964 and one P/N 273350-9 kit was manufactured in 1964. All appropriate testing, including compatibility tests, were completed for this kit.

No fabrication activity was initiated for the P/N 273350-19 kit.

5. M-741 Fuel Turbopump Assembly Maintenance Equipment Kit (P/N 273200)

This kit is intended for maintenance used with the Fuel Turbopump Assembly. Actually, the M-741 consists of two kits. One, the P/N 273200-9 kit, supplies 54 items of special equipment used in the maintenance of the Model I Fuel Turbopump Assembly; the other, the P/N 273200-19 kit, supplies 54 items of special equipment used in the maintenance of the Model II Fuel Turbopump Assembly.

The M-741 kit is intended for use in the Pump Assembly Clean Room; therefore, all of the items contained in the kit were designed to satisfy the requirements of the appropriate cleanliness specifications.

The conceptual design for this kit was completed, reviewed, and accepted by the end of 1963. One P/N 273200-9 kit was manufactured in 1964. All proof pressure and compatibility tests were completed for this kit.

No fabrication activity was initiated for the P/N 273200-19 kit.

6. M-750 Thrust Chamber Sleeve Valve Maintenance Equipment Kit (P/N 296000)

This kit provides the necessary tools for the assembly, disassembly, and pressure testing of the sleeve type thrust chamber valve. It is made up of eleven special tools.

The M-750 valve kit is intended for use in the Pump Assembly Clean Room; therefore, all of the items contained in the kit were designed to satisfy the requirements of the appropriate cleanliness specifications.

Preliminary equipment specifications were completed and released early in 1963. Drawings were completed and assembly fabrication orders were released in the second quarter of 1963. Three M-750 kits were manufactured in 1964 and all appropriate testing was completed.

1963

7. M-751 Gas Generator Valve Maintenance Equipment Kit (P/N 296250)

This kit provides the necessary tools for assembly, disassembly, and pressure testing of the gas generator fuel and oxidizer valves. It consists of 13 special equipment items.

The M-751 valve kit is intended for use in the Pump Assembly Clean Room; therefore, all of the items contained in the kit were designed to satisfy the requirements of the appropriate cleanliness specifications.

Detailed design of this kit was completed early in 1963. Drawings were completed and assembly fabrication orders were released during the second quarter of 1963. Three kits were manufactured during the third quarter of 1963 and all appropriate testing was completed.

8. M-752 Poppet Thrust Chamber Valve Maintenance Equipment Kit (P/N 296160)

1963/1964

This kit provides the necessary special equipment for assembling, disassembling and pressure testing the poppet type thrust chamber valve. It consists of 14 special equipment items, including spring compressors, guide pins, spacer rings, and pressure test closures.

The M-752 valve kit is intended for use in the Pump Assembly Clean Room; therefore, all of the items contained in the kit were designed to satisfy the requirements of the appropriate cleanliness specifications.

Preliminary equipment specifications were completed and released early in 1963. The detailed design was also completed during the first quarter of 1963. Fabrication orders for two kits were released in 1964; however, as a result of the M-1 Program entering its phaseout stage, only seven of the 14 items were completed. The completed items are as follows:

a. P/N 296164, Seal Inserter Power Unit (2)
b. P/N 296172-1, Housing Yoke Guide Pin (4)
c. P/N 296190-1, Position 1 Poppet Seat Spreader
d. P/N 296194-1, Valve Piston Positioning Adapter
e. P/N 296200-9, Poppet Thrust Chamber Valve Yoke
f. P/N 296204-1, Seal Serrator Guide Pin
g. P/N 296206-9, Seal Serrator Spacer Ring

None of the completed items were tested.

1963/1964

9. M-754 Start Valve Maintenance Equipment Kit

When completed, this kit will provide all of the necessary special tools for assembling, disassembling, and pressure testing the M-1 start valve. Only one tool, a valve spring compressor, was designed.

The M-754 valve kit is intended for use in the Pump Assembly Clean Room; therefore, all of the items making up this kit should be designed to satisfy the requirements of the appropriate cleanliness specifications.

Four P/N 296435-9 valve spring compressors were manufactured in 1964, but no tests were conducted with them.

D. CHECKOUT AND OPERATING EQUIPMENT

1962

The interface requirements for this equipment were coordinated with the appropriate design groups at the outset of the M-1 Program. Automatic logic diagrams were prepared and maintained on a current basis. Simplified schematics were initiated to define the ground support equipment requirements for leak checking, servicing, and cleaning the engine.

During the third quarter of 1962, the basic logic diagrams were changed to conform with the Marshall Space Flight Center Specifications and effort continued for the preparation of the schematics and determining the interface requirements. Preliminary layouts were started for the master operating console, the recorder, the pre-fire and post-fire checker, and the controller. Some procurement was initiated in connection with electrical items so they could be evaluated for use in the checkout control equipment. Fabrication orders were released for breadboard systems of the checkout and operating equipment end items. Also, test evaluation of vendor-supplied items was under way.

By the end of 1962, design reviews had been held for the leak detector test set, the hydraulic-pneumatic control unit, and the engine simulator. Some potentiometers as well as relays and switches had been tested. It was at this point that all effort in connection with the checkout and operating equipment was terminated because it was decided that the functions this equipment would provide would be built into the test facilities.

VII. PROGRAM SUPPORT REQUIREMENTS

A. EMERGENCY DETECTION SYSTEM

CHRONOLOGY

1962/1963

It was intended that this system would be used to detect imminent in-flight malfunctions of a catastrophic nature. The effort was initiated at the outset of the contract award. A preparatory design of the logic modules was completed for the preliminary layout of the emergency detection system logic unit. The mechanical layout of the unit itself was completed during the second quarter of 1962. At the same time, procurement specifications for the overspeed detector and pressure switches were initiated along with interface design. However, major design and development test efforts were delayed in the third quarter of 1962 until an engine malfunction analysis, then under way, was completed.

Approximately 85% of the transistors and diodes required to perform circuit breadboard tests were ordered by June of 1962. This was done in conjunction with the fabrication of the engine sequence control unit.

Also, approximately 100 samples of solid-state components were temperature-cycled (ambient to -300°F). Testing indicated that a number of components can undergo and withstand this temperature cycling. These tests were combined with component testing for the sequencer unit. The testing of transistors and diodes was also being accomplished to establish a basis for their selection.

A breadboard test chamber was set up and a suitable control system was assembled to permit circuit testing. Three types of resistors were evaluated for low temperature (-260°F) operation. Sixty-nine tests were conducted with 23 resistors. Preliminary results indicated that two types were suitable for use in low temperature environments. All of this effort was accomplished in conjunction with engine control unit tests.

During the month of October 1962, NASA cognizance of the M-1 Engine Development Program was transferred from the Marshall Space Flight Center to the Lewis Research Center. As a result, the emergency detection system task was eliminated early in 1963 during contract negotiations with NASA/LeRC.

B. FLIGHT INSTRUMENTATION SYSTEM

1962/1963

The design of a flight instrumentation system was initiated at the outset of the program. Anticipated ranges and nominal values for flight instrumentation parameters were selected. A list of flight parameters requiring measurement was compiled during the third quarter of 1962 and then revised along with Aerojet-General-assigned symbols and anticipated nominal readings. Design studies were conducted to obtain these nominal readings.

The original specification for this system was that it be a high level system (0-5 volt); however, this requirement was changed early in 1963 to make the system a low level (0-40 millivolt) one. Specifications and specification control drawings were revised during the first quarter of 1963 to reflect this change.

During the second quarter of 1963, it was decided to defer a major portion of the flight instrumentation program for one year. Data from the completed tests were analyzed and appropriate informal reports prepared. However, this task was not re-initiated because the M-1 Program entered its phaseout stage before there was any requirement for completing the system.

The design of a force ring was started as part of the flight instrumentation program during the second quarter of 1962. This force ring was to be used in measuring axial thrust on the pump bearings. The initial effort involved a literature search to identify bonding agents that function satisfactorily at liquid oxygen and liquid hydrogen temperatures. The data uncovered indicated that the EPY 400 bonding agent was successfully used at temperatures ranging as low as -420°F.

Preliminary specifications for a helium-damped, strain-gage accelerometer with a five-volt output were completed during the fourth quarter of 1962. The specification control drawing for this accelerometer was initiated in January 1963. In April 1963, the specification for strain-gage linear accelerometers was revised to reflect the latest program requirements. Also, the preparation of the specifications for piezo-electric cryogenic accelerometers was initiated.

Purchase requisitions were initiated in May 1963 for 12 linear strain-gage accelerometers.

The possible use of the flowmeters that were being supplied to the Rocketdyne Corporation and the Pratt & Whitney Corporation for M-1 instrumentation was investigated. Also, the possible use of J-2 or RL-10 flowmeters for the M-1 gas generator was investigated; however, the unavailability of detailed information precluded any decision.

At the time that the flight instrumentation program was deferred, a study of methods for conducting transducer vibration tests at cryogenic temperatures was under way. Specifications had been completed for axial load transducers and had been submitted to various vendors for quotations.

C. TANK INSTRUMENTATION

At the outset of the M-1 Program, requirements were established with MSFC regarding the definition of standard test equipment to be installed in the run tanks for Test Stand E-1A.

Liquid oxygen and liquid hydrogen testing would be done under simulated vehicle pressurization rather than with vehicle flow rates. The pattern for required temperature profiles was defined and the requirement for initial pressurization was established at a 5% ullage point with helium.

Tank instrumentation design for the special test equipment was initiated. It was intended that this special test equipment system would be integrated into the normal test system instrumentation to minimize installation costs.

The investigation of pressurization gas requirements as well as its associated control system was also initiated.

By the third quarter of 1962, the formal criteria for the special test equipment was in preparation. Also, the criteria for the instrumentation systems were formally transmitted to the Aerojet-General AETRON Division for appropriate action. The final designs for the probes to be used in this application were sent to vendors for price quotations at this same time.

The purchase specification for the total immersion resistance temperature transmitter (RTT) was completed and forwarded to NASA by the end of the third quarter of 1962.

The development of an over-all test plan was undertaken. Studies indicated that a two-phase program would be required; a low pressure investigation to satisfy immediate needs and studies of the existing technology to investigate the effects of propellants when subjected to high pressures.

A literature search was initiated at the end of 1962 to obtain information about small tankage instrumentation.

During the month of October 1962, NASA cognizance of the M-1 Program was transferred from MSFC to NASA/LeRC. As a result, the tank instrumentation program was eliminated early in 1963 during contract negotiations with NASA/LeRC.

1962/1963

D. THRUST ALIGNMENT

Alignment studies were undertaken at the outset of the M-1 Program for Test Stand J-5 using a six-component thrust-eccentricity measurement system. It was found that a computer program study was required to define the system as well as to calculate the points for gimbal arm correction. Also, an analysis was conducted of results from a similar thrust-eccentricity measuring program study made for the Aerojet-General Solid Rocket Plant.

The alignment studies in connection with the J-5 test stand were suspended during the third quarter of 1962 pending the results of Zone J design studies and criteria development.

By the end of 1962, it was determined that misalignment of the altitude skirt would cause a thrust eccentricity; therefore, the determination of skirt alignment would have to take place in an altitude test run facility. This requirement dictated that the M-1 testing be conducted on the J-2 horizontal test stand and it was decided that a three-component thrust measuring system would be used to measure thrust alighment.

The two systems for measuring thrust to determine thrust alignment were reviewed during the first quarter of 1963. The three-component system would be more economical to build, data processing would be simpler, and reliability would be greater than with the six-component system. However, the use of the three-component system would require the assumption of a calculated point of application of the line of thrust. Thus, the accuracy of the system would depend entirely upon the accuracy of the assumed point of application. The use of the six-component system would permit exact determintion of the magnitude and line of thrust as well as all radial thrust components. In either system, propellant line configuration and variation of pressures in the altitude chamber during the run will have a bearing upon the force measurements.

In June 1963, it was decided that the design of the six-component thrust alignment measuring system would be continued by the AETRON Division, where a computer program would be formulated for the data reduction of the thrust alignment measurements.

The thrust alignment effort was completely deleted from the contract in December 1963 by NASA/LeRC.

E. SUCTION LINE ARRANGEMENT

1962/1963

Investigation of the use of modified F-1 liquid oxygen suction lines (vacuum insulated) for both the M-1 liquid oxygen and liquid hydrogen lines was undertaken at the outset of the program. It was determined that the 17-in. diameter F-1 oxygen propellant flexible suction line sections could be used in both the oxygen and hydrogen suction lines during M-1 engine development. Studies were initiated to establish what modifications to the F-1 suction line section would be required to provide proper insulation. The heat flow rate into the hydrogen propellant system during propellant loading and standby were being investigated.

Another study was started to determine the applicability of the Titan I liquid oxygen suction conditions to the M-1 non-bleed system.

An analysis was made during the third quarter of 1962 of a preliminary NOVA suction line arrangement to establish the feasibility of a non-bleed system. The oxygen line used in the analysis was a straight 15 ft line with an F-1 suction bellows. The only modifications needed to obtain the existing net positive suction head (NPSH) pre-fire requirement at the oxygen pump was to add a thickness of insulation to the suction line, bellows (flexible insulation), and a major area of the turbopump assembly and discharge line. The hydrogen line configuration was assumed to be approximately 40 ft long, routed around the outside of the oxygen tank, and containing three bends. The optimum method for obtaining the required pre-fire NPSH was to insulate the hydrogen suction line, bellows, and pump with a moderate thickness of insulation. Also, to install a parallel 4-in. diameter line the length of the suction line to act as a riser to induce circulation. This riser would require less insulation than the main suction line but would need a thin insulation to prevent surface liquefaction of air.

Another analysis was completed. This was of the liquid oxygen and the liquid hydrogen non-bleed systems to determine the effects of vehicle acceleration from 1 g to 10 g, and vehicle compartment air velocity from 0 to 50 ft/sec. Results of the analysis indicated that the operation of the suction line systems was adequate within these ranges.

A final report of the M-1 inlet line investigation (106) was submitted to NASA in November 1962. At the same time, analyses began for the preliminary and gimbal center locations upon the suction line length would be shown.

The preliminary analysis of the length and the weight of gimbaling suction lines as a function of required line movement was completed during the first quarter of 1963. An extrapolation of F-1 suction line parameters was made and it was shown that the M-1 lines will require a greater degree of flexibility to maintain a reasonable size and weight.

(106) M-1 Engine Inlet Line Investigation, Aerojet-General Report No. 9510-5, 1 November 1962

1962/1963

Engineering study contracts to analyze suction line lengths were awarded to two specialty suppliers, the Flexonics Company and Arrowhead Products. A complete set of data, consisting of the Flexonics report, the Arrowhead report, an Aerojet-General analysis, and a comparison of the three studies was provided to NASA/LeRC in May 1963.

Fabrication of twelve 19.5-in. diameter interim suction lines was ordered at the end of the first quarter of 1963.

By the end of 1963, NASA/LeRC determined that volume-compensated suction lines would not be needed for the M-1 program.

F. PREVALVE AND SUCTION FLOWMETER

At the outset of the program, it was resolved with MSFC personnel that the prevalve and suction flowmeter would be furnished by NASA for M-1 Engine testing.

A study was initiated to determine the feasibility of using the F-1 suction flowmeter and prevalve for the M-1 liquid oxygen and liquid hydrogen suction lines. It was determined that a 17-in. prevalve similar to the F-1 oxygen propellant prevalve would be suitable for the M-1 suction lines.

In November of 1962, it was agreed that Aerojet-General should propose on this item because there had been an increase in the oxidizer turbopump inlet diameter to provide for a lower minimum NPSP.

Both of these items were deleted from the program early in 1963 at the request of NASA/LeRC.

G. GIMBAL ACTUATORS

1962

Preliminary calculations of actuator loads were made at the outset of the program. These calculations did not include the effects of flight, aerodynamic, and vehicle turning loads. Also, effort was initiated to prepare the specification for a pneumatic gimbal actuator, but this work was suspended at the end of 1962 pending the results of a study program then in process for thrust-vector control by secondary injection methods. This effort was not resumed.

H. SCALE MODEL PUMP TRAVERSING SYSTEM

1963/1965

A special traverse system for the scale model pump was installed in Test Area D-3 at the end of 1963 to obtain accurate pressure profile measurements. The system was received and checked out before installation into the test area.

The complete installation and operating procedures for the traverse systems were released on 20 February 1964, including all drawings and schematics.

The oxidizer scale model pump traversing system was completed by the end of the first quarter of 1964.

Preliminary design of the traversing position stepping control was completed in September 1964. Although the traversing system can be operated without this step position control, positioning of the six actuators manually during the tests requires additional time.

Traversing probes of increased strength were received in October 1964. These probes had a strong internal structure using Inconel 718 which has a yield strength four times greater than the materials used in the earlier models.

During the first quarter of 1965, all traverse system controls were modified for use with strain-gage transducers. This was necessary because a problem of hydraulic response existed as a result of a volumetric displacement in the sensing lines and transducer cavity of the angle sensing transducers being used. A modified unit was tested and satisfactory sensitivity and response were obtained.

This task was completed during the third quarter of 1965 when three scale model fuel pump tests were conducted in Test Stand D-3 to complete the scale model pump test program.

I. PRESSURE AND TEMPERATURE PROBES FOR MODEL I OXIDIZER AND FUEL TURBINES

The design of special total pressure probes and temperature probes for measurement at the inlet and discharge points of the Model I oxidizer and fuel turbines was completed by the end of 1963. The drawings for these special probes were released and a purchase order initiated.

An evaluation was made during the first quarter of 1964 of the ceramic cements that could be used as a protective coating on the probes. It was found that Astroceram cements were unsuitable for this application because of repeated cracking of the ceramic cements during the curing cycle. It appeared that there was a mismatch of the expansion coefficients.

The first tests of these probes occurred during the fourth quarter of 1964 when two types of thermocouples (closed junction and bare wire junction) were evaluated during Run Nos. 1.2-04-EHG-001 through -007 of the gas generator test series on Test Stand H-8.

1963/1965

Figures No. 35 and No. 36 show the total pressure probe after removal from the gas generator. This probe was immersed in the flow stream for three inches. Its condition after removal indicated excessively high loading. Analysis of the data indicated that this excessive loading probably occurred during the start or shutdown transient. The tip of the probe had started to burn as shown in Figure No. 36. This tendency to burn was removed in subsequent probes by removing the sharp edge which slightly lessened flow-direction sensitivity. Also, it was planned that in future tests, thermocouples and pressure probes would be immersed only two inches to reduce the loading.

The remaining probes, approximately 20%, required for this program were ordered early in 1965. All of the available probes were delivered to E-Area for the scheduled turbine testing. The remaining probes were received during the ensuing quarter. They were also delivered to E-Area and this task was completed.

J. LIQUID LEVEL CONVERTER UNITS

The installation, adjustment, and calibration of liquid level converters was initiated during the fourth quarter of 1963. As the liquid level converters were received, they were checked out, installed into the tanks, and calibrated. Liquid level converters were installed in the following tanks: VE-1, VE-2, VE-10, VE-11, VE-12, VE-32, VE-34, VH-70, VH-71, and VH-75. This effort was completed during the first quarter of 1965.

K. TOTAL PRESSURE PROBE FOR LIQUID OXYGEN FLOW

1964

The preliminary design for a liquid oxygen total pressure probe was initiated during the first quarter of 1964. An investigation of transducers capable of fulfilling the requirements of the pressure system was also under way.

The drawings for the liquid oxygen total pressure probe were released in May of 1964. At the same time, a purchase order was submitted so as to obtain bid quotations from vendors. The order for manufacture of the probes was placed with Advanced Dynamics, Cleveland, Ohio. It was decided that the mounting hardware would be fabricated in-plant.

The mechanical design of the probe was reviewed during the third quarter of 1964 and changes were incorporated to increase the safety factor to 2 when the probe is installed 90-degrees out of flow alignment (the worst possible position for adverse loading conditions). The diameter of the probe stem base was increased and the stem shape was tapered. The safety factor for correct installation was approximately 5.

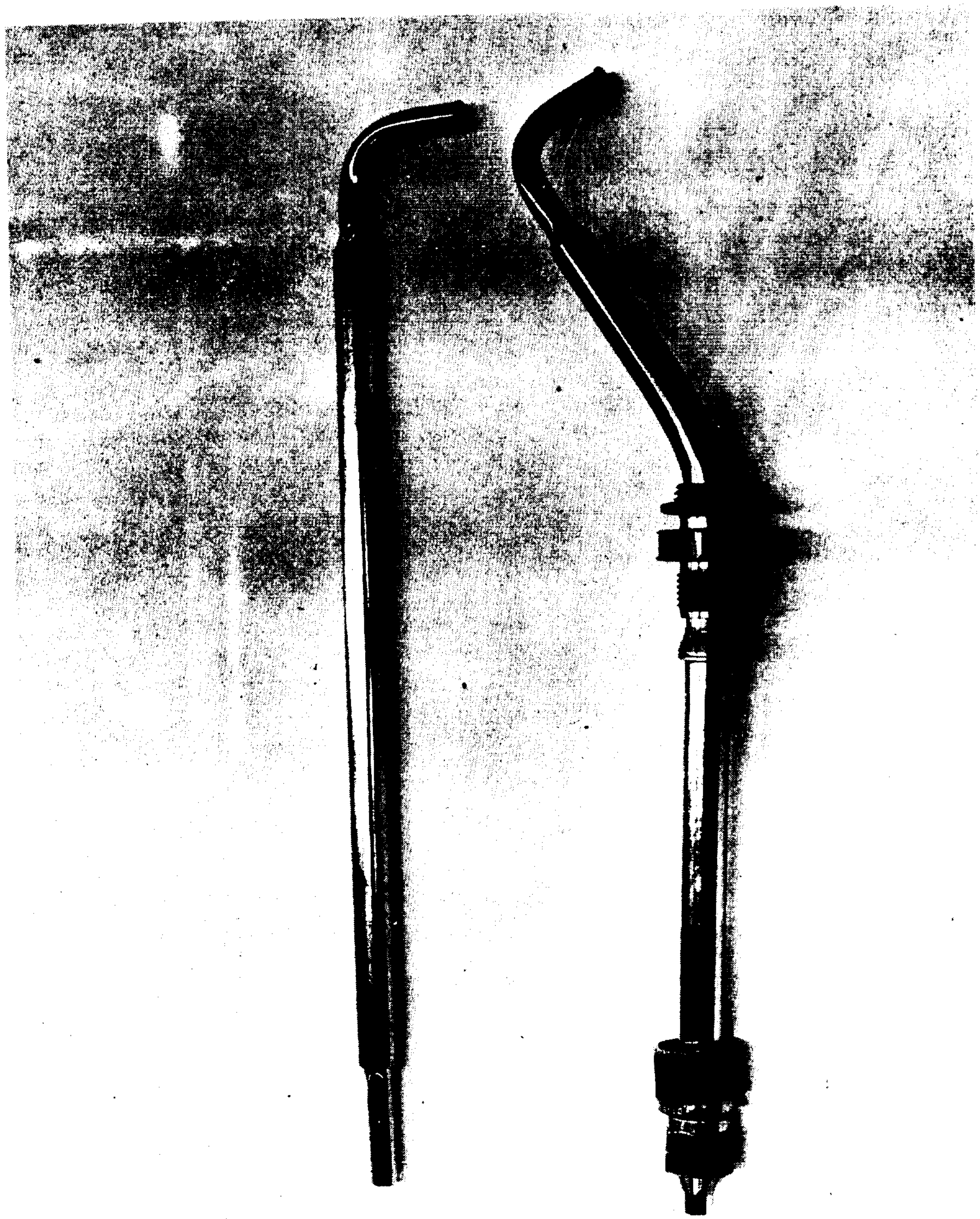


Figure 35. Total Pressure Probe after GGA Test

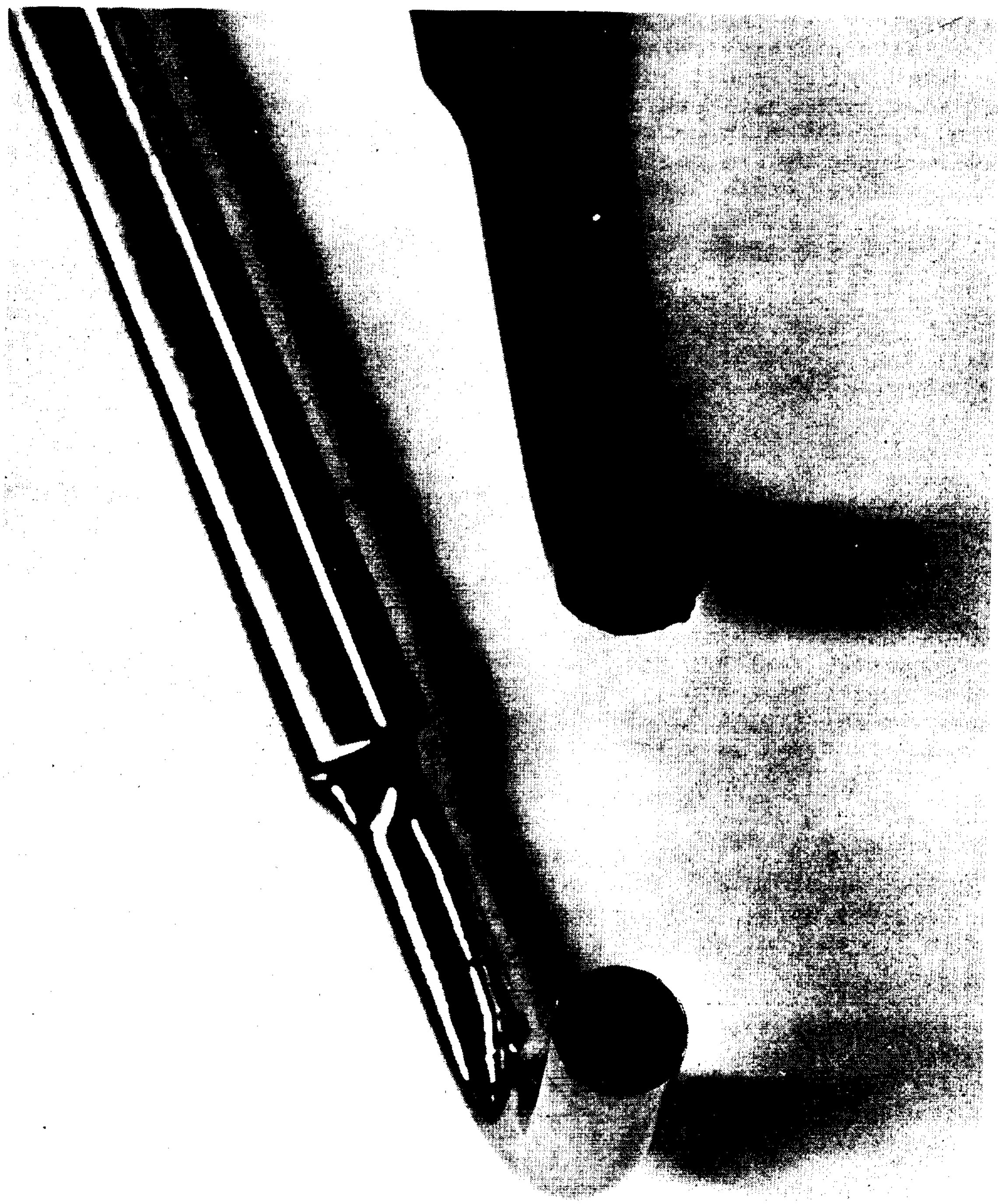


Figure 36. Total Pressure Probe after GGA Test

1964

Both the design as well as the detailed drawings of a new method for mounting the total pressure probe (flare fittings) were completed by the end of September 1964.

The probes were received from the vendor during the fourth quarter of 1964. By the end of the year, the complete package of probes, mounting hardware, and drawings were delivered to E-Area for installation on the oxidizer turbopump assembly. This completed the support effort for this task.

L. EVALUATION OF CRYOGENIC TRANSDUCERS

Six Micro-Systems transducers, Model PT3J-3C, were tested during the first quarter of 1964 at temperatures ranging from +130°F to -425°F. They operated satisfactorily in the low temperature environment at changes of pressure throughout their 0 to 1500 psia range. Various full-scale shifts were recorded with an extreme, in one case, of -43% of full-scale at -415°F. This same unit had a zero shift of +9% at -415°F. Non-linearity for all transducers ranged from -1% at ambient to +3% at the low temperature.

Five additional Micro-System PT3J-3C pressure transducers were received in March 1964 (one 0 to 100 psia unit and four 0 to 1500 psia units). Performance testing was conducted with these units at +130°F, +77°F, -30°F, -100°F, -200°F, and -300°F.

A Gulton Industries high frequency transducer was also tested within the temperature range of +200°F to -100°F with favorable results. Terminal-based non-linearity and hysteresis throughout the factory-compensated temperature range (+190°F to -30°F) was approximately ±0.4% of full-scale and repeatability was approximately 0.3% of full-scale.

Additional testing and calibration of the Micro-System transducers continued throughout the second quarter of 1964.

This task was completed during November 1964 with the preparation of an evaluation report(107).

M. GAS GENERATOR THERMOCOUPLES

1964/1965

This effort commenced during the second quarter of 1964 and was in support of the gas generator testing.

(107) MSIPT3C-C2, Cryogenic Evaluation, Aerojet-General Flash Report No. 9271:2839, 16 November 1964

Two insulated thermocouples, designed for measuring gas generator temperatures, were modified in April 1964 to obtain a faster response time as well as to increase the mechanical strength of the probe. Also, a new design and fabrication of a high strength and fast response probe was completed. The sheath material for this new probe was rolled tantalum. A study of refractory ceramic coatings was also undertaken because the protective coating of tungsten disilicide on tantalum-sheathed thermocouples showed signs of flaking.

Thermocouples were being destroyed during gas generator tests and it was necessary to reduce their immersion depth. This proved to be a satisfactory expedient. After a test with this lesser immersion depth, it was found that all of the chromel-alumel thermocouples were in good condition whereas all of the tantalum-sheathed, tungsten/tungsten-rhenium thermocouples were damaged although not burned.

During the fourth quarter of 1964, evaluation thermocouples were installed and operated during gas generator test runs (Test Run Nos. 1.2-04-EHG-001 through -007). The location of these thermocouples is shown in Figure No. 37.

The TOTS-4C parameter was measured using an Inconel-sheathed (0.25-in. OD, 0.030-in. wall thickness) chromel-alumel (C/A) thermocouple, immersed 1-1/4-in. into the gas stream. The junction was exposed and the sheath tip was cut to a 45-degree angle and positioned to face the gas flow. Despite some shortcomings in mechanical structure, the thermocouple yielded continuous satisfactory data.

The TOTS-4E parameter was measured by using a tantalum-sheathed (0.060-in. OD, 0.010-in. wall thickness) tungsten/tungsten 20% rhenium (W/WRe) thermocouple as shown in Figure No. 38. The thermocouple was encased in a second tantalum sheath (0.350-in. OD, 0.125-in. wall thickness) which had a coating of 0.010-in. alumina. The unit had a grounded junction and was immersed 2-1/4-in. into the gas stream. This thermocouple also yielded continuous data throughout the test series.

The TgTs-2T parameter was measured using a shielded, grounded C/A thermocouple in the first test and an exposed junction C/A thermocouple in the remaining tests. Although some problems were encountered with the latter thermocouple, which was immersed three inches into the gas stream, it yielded better data than the one used in the first test.

All of the tungsten/tungsten-rhenium, tantalum-sheathed thermocouples (manufactured by Electrowest) failed because of apparent embrittlement of the tantalum, which caused cracking of the sheath at the high stress point of the thermocouple stem.

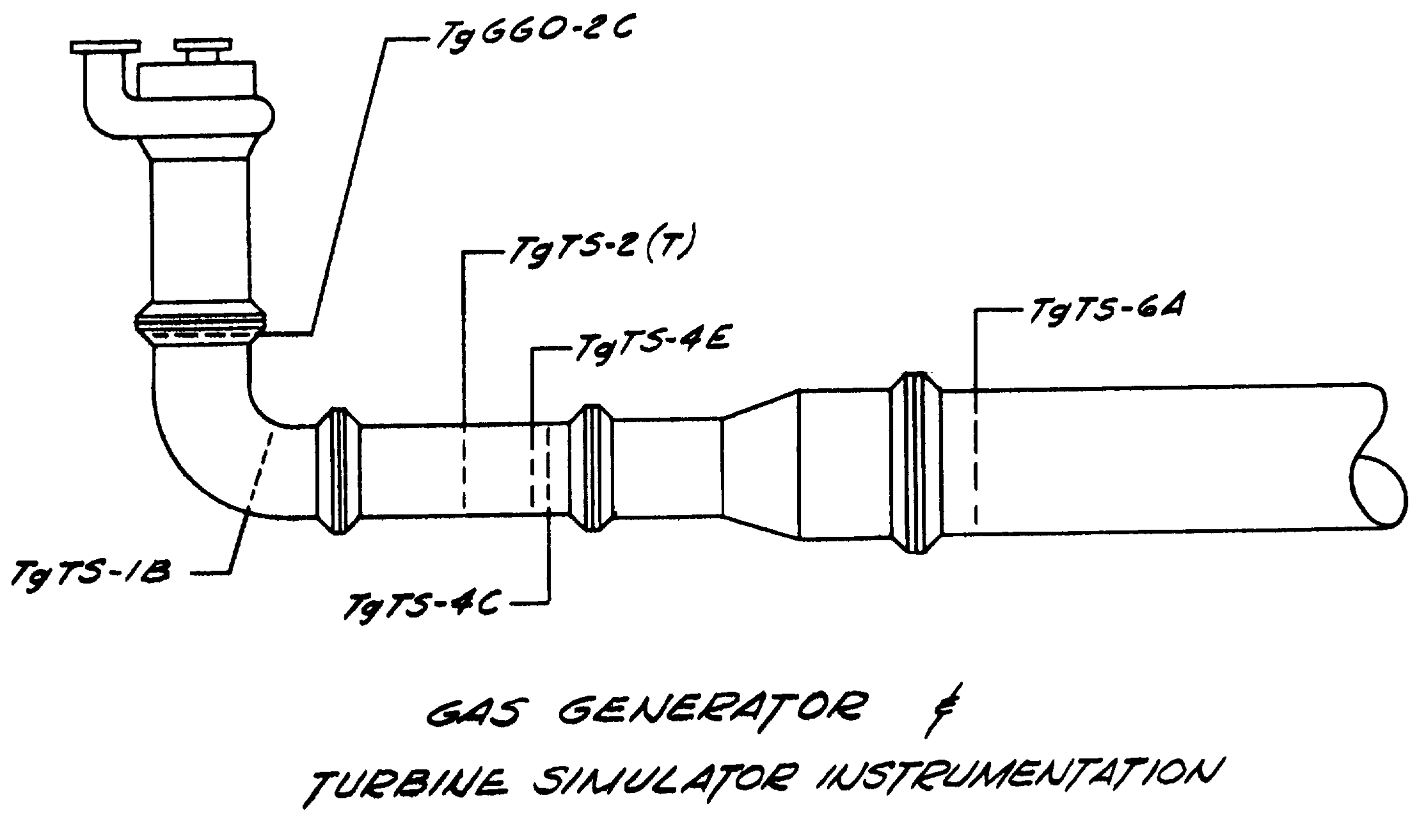


Figure 37. Gas Generator and Turbine Simulator Instrumentation

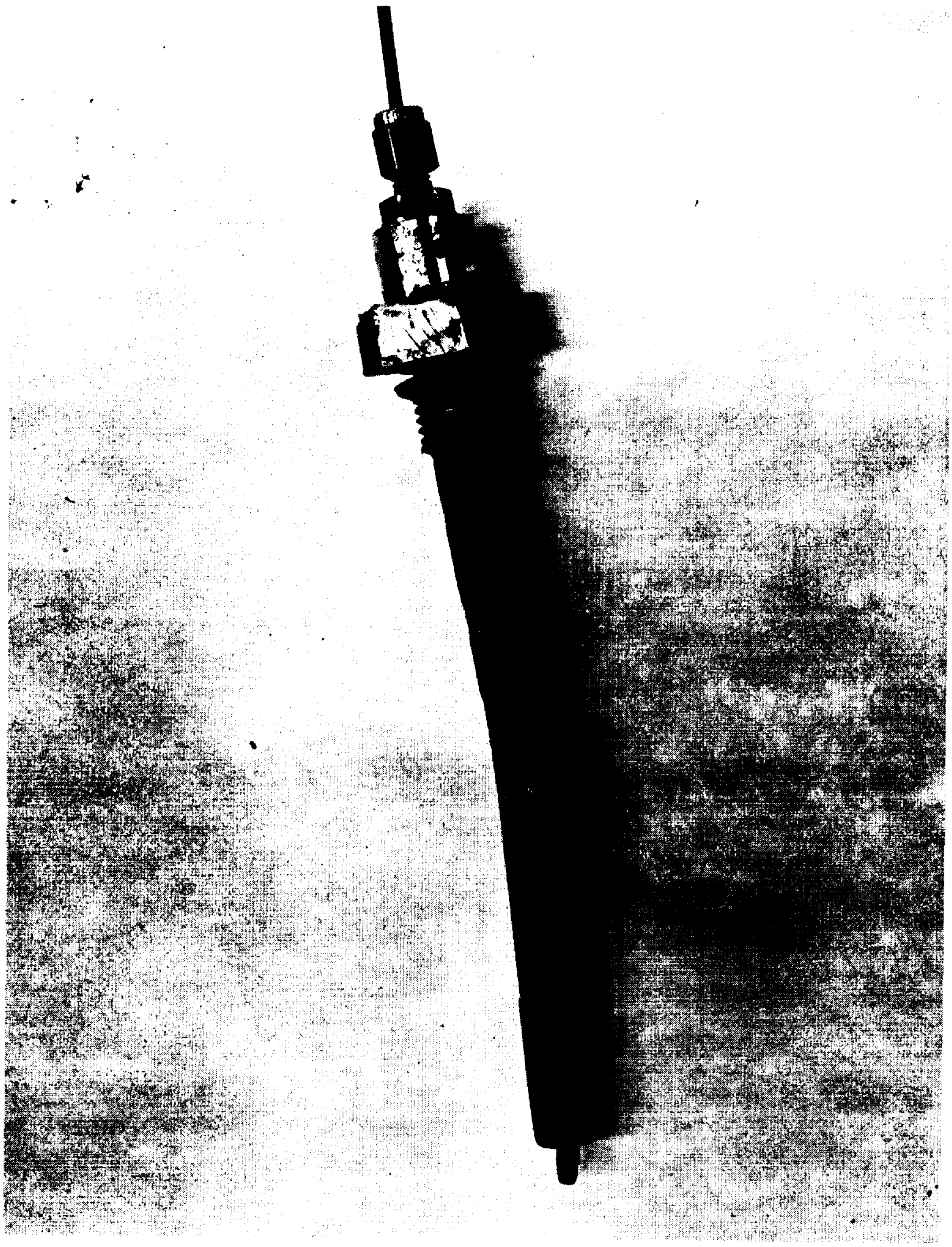


Figure 38. Gas Generator Thermocouple after Test

1964/1965

The TgGGO-2C parameter was measured by using a standard chromel/alumel thermocouple (1/4-in. outside diameter, 0.030-in. wall thickness) Haynes-25-sheathed, with a shielded grounded junction.

The performance of the thermocouples continued to be monitored throughout the remainder of the gas generator test program which was completed at the end of 1965.

N. HIGH-FREQUENCY PRESSURE TRANSDUCERS

1964/1966

Early in the second quarter of 1964, a review was made of available hot gas high frequency pressure transducers for application on the uncooled and the cooled thrust chamber assemblies. The helium bleed technique was definitely workable and the Photocon transducers appeared to be superior to the Dynesco transducers. Photocon made a unit that was physically interchangeable with the Dynesco.

A program for the evaluation of the high frequency transducers was also formulated. In addition, a review of the Princeton calibration procedure for the checkout of a hydrogen-cooled transducer manufactured by Electro Optical Systems was made.

During the third quarter of 1964, the design of the prototype helium bleed adapter (Model HB3X) was completed and a purchase order issued for three units. This model was designed to fit the basic cooled-engine boss and an adapter was designed to allow this model to be used with the uncooled thrust chamber assembly.

At this same time, the calculations necessary for stress analysis of the cooled thrust chamber assembly boss configuration were made and preliminary layout design was completed. A prototype transducer was also designed for the gas generator assembly tests. An adapter to fit this transducer to the Dynesco boss was also designed. Three of these transducer adapter units were procured.

The market search for high frequency pressure transducers was completed. New equipment was continually being introduced by manufacturers and was considered as it became available. At that time, the third quarter of 1964, Kistler, Photocon, and certain MSI and Gulton Industries transducers were considered for meeting the M-1 program requirements.

Preliminary procedures for evaluation of non-cryogenic high frequency pressure transducers were prepared. Also, a procedure for evaluation of the Electro Optical Systems hydrogen-cooled transducer was written.

A preliminary layout design for adapting the shock tube to high and low temperature environments was accomplished and a proposal from the Houston Engineering Research Company (HERCO) for such an installation was requested. HERCO was the firm that built the shock tube. A computer program was received from Princeton University.

The sinusoidal pressure generators in use were investigated as well as several design approaches evaluated. It was found that the system used at the Guggenheim Laboratories at Princeton University was the most suitable of those then available. Accordingly, a detailed design and cost proposal was solicited from Princeton.

The requirement for helium bleed transducers for use in the uncooled M-1 thrust chamber assembly was reviewed during the first quarter of 1965. Also, the design of the Pc-5 adapter was initiated. This design was completed during the ensuing quarter and procurement was initiated.

Designs of the helium bleed transducer assemblies for Pc1 through Pc5 for the ablative-lined thrust chamber assembly were initiated in June 1965.

Four helium bleed transducers and shock tube adapters for Pc-5 of the unlined thrust chamber assembly were received during the third quarter of 1965. The 16 Kistler transducers for the Pc-1 and Pc-5 of the thrust chamber assembly were also received. Orders for the helium bleed adapters for the lined thrust chamber assembly were placed.

By the end of 1965, it was found that there was a degradation of the expected frequency response of the Pc-5 helium bleed transducers for the unlined thrust chamber assembly. This resulted from the electrical connectors becoming contaminated with "Loctite." As a result, new connectors were procured and the "Loctite" was replaced with Teflon tape.

Also, by the end of the year, the Pc-5 transducers for the lined thrust chamber assembly were reworked, flow checked, and static calibrated. The Pc-2 through Pc-4 helium bleed transducers had been flow checked and static calibrated.

The Pc-2 through Pc-4 helium bleed transducers were installed in the lined thrust chamber assembly during the first quarter of 1966. The Pc-5 transducers had also been reassembled; however, they were deleted from the test requirements.

Seven helium bleed transducers had been installed on the thrust chamber assembly for its test in June 1966. Also during the second quarter of 1966, effort was initiated to prepare the Pc-5 transducers for installation in the unlined thrust chamber assembly. A vibrational test was requested to evaluate these transducers for any possible effects resulting from their length.

This evaluation of high frequency transducers is a continuing effort and will extend through the thrust chamber assembly testing at Test Stand H-8, which is beyond the scope of this report.

O. SPRING-LOADED BEARING THERMOCOUPLES

1964

Evaluation units for the investigation of spring-loaded bearing thermocouples were ordered during the third quarter of 1964. Vibration testing of two spring-loaded bearing thermocouple designs was completed by the end of the year. One, a thermocouple made by Electrowest Incorporated, had a permanent set in the spring arrangement which made this design undesirable for the proposed application. Had this set occurred during actual usage, it would have released the contact between the thermocouple junction and the bearing resulting in a loss of test data. The other design, furnished by the Nugget Manufacturing Company, successfully completed the vibration tests without any structural failure. Several minor resonance points were recorded but no degradation of test data was observed. This design was considered satisfactory for further evaluation in the bearing tester.

The detailed aspects of this effort have been delineated in a separate report.(108)

P. TEMPERATURE PROFILE PROBES

1964/1966

During the fourth quarter of 1964, effort was initiated to adapt the Test Stand C-9 probe design for use and installation at Test Stand H-8. Drawings were completed early in 1965 and requests for vendor quotations were issued during the second quarter of 1965.

The temperature rakes for the Test Stand H-8 hydrogen mixer discharge were calibrated during the fourth quarter of 1965 and delivered to the H-8 test area.

(108) Young, E. A., Test Report, Spring-Loaded Thermocouple for M-1 Bearing Tester, Aerojet-General Report No. 0830-94, 1 August 1966

1964/1966

A review was conducted in January 1966 to determine the loading upon the probe if it were inadvertently turned crosswise in the flow. A stress analysis indicated that the probe would be marginal in a cross-flow. When properly installed, the probe has a safety factor of 2.

Nine thermocouples were installed for the first thrust chamber assembly tests at Test Stand H-8 during May 1966. Data obtained from these rakes showed good response and temperature correlation with six of the nine. The three inoperative thermocouples were repaired.

Effort in connection with the temperature profile probes will be continued in support of the thrust chamber assembly testing at Test Stand H-8, which is beyond the scope of this report.

Q. STATIC PRESSURE PROBES FOR THE FTPA INTERSTAGE

1964/1965

Quotations were received, evaluated, and an order was placed during the last week of December 1964 for 34 interstage pressure probes.

A separate report(109) has been issued which summarizes this effort.

R. FTPA TOTAL PRESSURE PROBE

1964/1965

The design of the fuel turbopump assembly total pressure probe was completed during the fourth quarter of 1964.

Because of delay in delivery of total pressure probes for the first fuel turbopump assembly test which was scheduled for mid-April 1965, an in-house effort to build three probes was initiated. The probes were completed and installed in the fuel turbopump assembly. They operated satisfactorily during the testing of the pump.

This effort has been detailed in a separate report.(110)

S. MINIATURE RTT FOR FUEL TURBOPUMP ASSEMBLY BALANCE CHAMBER FLOW

1965

Three RTTs were provided by the NERVA Program for use in the M-1 Program as part of their effort to gather additional performance data for these units. All three RTTs were calibrated in fixed point baths and in the cryostat

(109) Birner, R. A., Interstage Static Pressure Probe for M-1 Fuel Pump, Aerojet-General Report No. 0830-65, 28 December 1965

(110) Birner, R. A., Design of Total Pressure Probe for M-1 FTPA Balanced Chamber Return Flow Line, Aerojet-General Report No. 9830-39, 18 October 1965

during the first quarter of 1965. Repeatability tests were conducted in liquid neon by cycling from ambient to liquid neon temperatures; two units appeared stable. The third unit shifted during calibration, but appeared to stabilize after further testing. Final reduced data from each unit was used to complete a mathematical equation for the resistance versus temperature relationship.

A detailed accounting of the RTT performance and history was included in a report issued for the NERVA Program.(111)

T. INSTRUMENTATION PROBE STRUCTURAL INTEGRITY INVESTIGATION

1965

Early in 1965, expressions were derived defining the safety factors for the Rosemount Corporation Model 134CT RTT probes. Safety factors were defined for these probes in full-flow and partial cross-flow of liquid hydrogen, liquid nitrogen, and liquid oxygen. A monograph was drawn to allow rapid evaluations of RTTs for a particular installation and set of flow conditions. A minimum safety factor of 3 was recommended for all applications. Flow conditions were also defined for a safety factor of 2.

The detailed aspects of this investigation have been reported separately.(112)

(111) Chandon, H., Report on Microsystems Silicon Resistance Temperature Transducer Model TS06A-1 and -2 for Cryogenic Service, Aerojet-General Report No. 9271-20, 15 April 1965

(112) Lehmberg, A. E., Structural Integrity of 134 RTTs in LOX and LH_2 Cross-Flow, Aerojet-General Report No. 9271-10, 25 February 1965

VIII. QUALITY CONTROL

CHRONOLOGY

1962

A preliminary Quality Control Program Plan for the M-1 propulsion system was prepared and submitted to NASA at the outset of the program. This plan was prepared so as to conform to the requirements of NASA Quality Publication NPC 200-2, April 1962. This plan was revised after discussion with NASA personnel and resubmitted on 15 September 1962.

During the third quarter of 1962, Quality Control Standard 9-1-4 was implemented in connection with the vendors and the End-Item Test and Inspection Plan as well as the monthly Quality Status Report formats were established. Also, a system for Quality Control review of drawings before their release was agreed upon. Appropriate action was initiated to certify soldering personnel and vendors were required to submit all drawings and specifications defining an article of supplier design.

A third revision to the M-1 Quality Control Program Plan was submitted to the Marshall Space Flight Center on 15 October 1962. This was followed by meetings with NASA/LeRC Quality Control personnel to work out an agreement on the plan among Lewis, MSFC, and Aerojet-General. Also, during the fourth quarter of 1962, the drawing review procedure and checklists were released as well as implemented.

1963

The M-1 Quality Control Program Plan was approved by NASA/LeRC in January 1963.

Quality Control audits for the M-1 Program were initiated during the first quarter of 1963. A number were completed during that same time period, including: non-conforming material processed in receiving inspection; quality control records; configuration control; and anodizing process. Also at this time, appropriate personnel completed the NASA Soldering Requirements introductory course.

1963

The first Monthly Quality Status Report was released in March 1963 in compliance with NPC 200-2.

Formulation of a factory inspection and test plan for the gas generator assembly was undertaken in April 1963. The NASA Western Office certified the Aerojet-General Soldering School (per Specification No. MSFC-PROG-158A) during that same month.

A program was initiated in June 1963 to train and certify radiographic inspectors.

1963

The Corporate Quality Control Standard H-3-2, "Interplant Quality Program Administration Under NASA Quality Publication NPC 200-2," was issued and implemented during the third quarter of 1963. During this same quarter, 324 engineering drawings were reviewed, 313 drawing changes were audited, and 21 engineering change requests were initiated. Also, X-Ray Standard B-10-286332 for the fuel rotor assembly and the fuel pump weldment was prepared and the second Quarterly Audit Summary Report was issued.

By the end of the quarter, the Fabrication and Inspection Plan from the Straza Industries for the thrust chamber jacket had been reviewed and necessary revisions were being coordinated.

In the fourth quarter of 1963, 312 engineering drawings were reviewed, 403 drawing changes were audited, and 107 supporting documents (specifications, standards, instructions, and procedures) were prepared or approved. Also, the factory inspection test plans were received from Straza Industries for the M-1 coolant tubes and the jacket. A program to determine the quality of combustion chamber tubes was established. A detailed study was initiated during this same quarter to determine quality control gaging requirements for the fuel turbopump assembly and the final design package for the X-Ray facility was approved.

1964

The M-1 Quality Control Program Plan was revised in January 1964 and the Engineering Review Board Procedure (QCS 15-1-11) was issued.

During the first quarter of 1964, 346 engineering drawings were reviewed, 405 drawing changes were audited, 67 engineering change requests were initiated, and 97 supporting documents were prepared or reviewed. An organization was established to control the disposition and corrective action pertaining to fired hardware. Also, Factory Inspection Test Plans were prepared for the fuel pump rotor and the cooled thrust chamber. A purchase order was issued to Automation Industries in Boulder, Colorado to develop an ultrasonic standard for electron-beam welding of the fuel rotor.

The standard for ultrasonic inspection of the electron-beam welded fuel rotor was completed during the second quarter of 1964. It was used by Solar Industries for the first article fabricated.

The use of SKL-HF for Type II Penetrant Inspection was begun during the third quarter of 1964 and procedures were initiated for spray-can use of ZL44B for Type I Penetrant Inspection.

The Type I Penetrant Inspection facility was completed during the fourth quarter of 1964. An activation plan was initiated during this same period to check out the equipment installed in the new X-ray facility.

1964

Critical process controls for balancing were incorporated into the manufacturing and quality control systems at the end of 1964. This required the preparation and verification of Product Inspection Instructions (PII) for Proof of Station and Proof of Balance.

An M-1 bolt problem investigation for the gas generator assembly and the oxidizer turbopump assembly was completed early in December 1964 and the results were detailed in separate reports.(113)(114) Also, the pre-assembly buildup of the fuel turbopump assembly was being monitored.

1965

At the outset of 1965, the Quality Control functions and tasks were re-evaluated so that they could be realigned for most effective usage within the scope of the M-1 Program redirection. As part of this realignment, the Drawing and Specification Review and Change Board participation was combined with Inspection Planning under the direction of the M-1 Program Quality Engineer. Also, Supplier Quality Engineering modified its quality tasks and functions to permit an orderly withdrawal of the source acceptance functions from the suppliers' plants.

Quality Control policy was clarified, combined, and published as Quality Control Instructions during the first quarter of 1965. The same was done for the procedures used to implement these policies. A single Quality Control System standard was imposed upon subcontractors. The Aerojet-General Standard Clause No. 29 (QCS B-9-1-4) was replaced by Clause No. 32 (QCS H-1-6).

Appropriate support effort for procurement fabrication and test operation was continued throughout the quarter as was the test inspection effort. The use of Spray Lubricant S-122 (Threads) was authorized and special nondestructive test methods for evaluating the integrity of the welds and parent metal of the fuel turbine rotor were developed. The application of these methods constitutes a significant technological advancement in the application of ultrasonic inspection techniques.

A report(115) was prepared in March 1965 which identified all the facilities that Quality Control planned to use in support of the fabrication and inspection of M-1 hardware.

NASA/LeRC approved the use of Level "O" type drawings during the second quarter of 1965 as well as a system for accepting components from the Controls

(113) M-1 Bolt Investigation, Gas Generator Assembly, Aerojet-General Report No. 1720-3628, 10 December 1964

(114) M-1 Bolt Investigation, Oxidizer Turbopump Assembly, Aerojet-General Report No. 1720-3626, 10 December 1964

(115) Impact Analysis of M-1 Funded Facilities for Quality Control (1700), Aerojet-General Memorandum Report No. 1701:5136, 29 March 1965

Laboratory based upon certification by the cognizant M-1 Task Manager. Appropriate Quality Control effort continued in support of procurement, fabrication, and testing.

During the third quarter of 1965, action was initiated to define the individual Quality Control Tasks that would fulfill the requirements of NPC 200-2 as well as those imposed by NASA for the Phaseout Development Program and the Engineering Board activity. This definition of tasks was in lieu of refining the Quality Control Program Plan of 2 July 1965. Appropriate support was again provided for procurement, fabrication, and testing. At the same time, work was initiated to determine the impact of M-1 contract termination upon the availability of M-1 Quality Control facilities for other NASA programs within the Sacramento Plant. Also, the X-Omat film processing unit was returned to NASA/LeRC in accordance with their instructions.

The definition of the individual Quality Control Tasks was completed during the fourth quarter of 1965. Activity was also under way to implement the inspection and disposition of terminated components. Appropriate support for procurement, fabrication, and testing continued.

1966

The Quality Control requirements for the Extended Thrust Chamber Assembly Program were prepared during the first quarter of 1966. Support for procurement, fabrication, and testing also continued along with the effort in connection with the analysis and justification for retention of Quality Control facilities.

Major Quality Control effort during the second quarter of 1966 was largely concentrated upon support of the thrust chamber assembly tests.

IX. RELIABILITY

1962

The Reliability Program Plan was reviewed with MSFC personnel at the outset of the program. At the same time, procedures were formulated for collecting, storing, and disseminating M-1 Program data.

A preliminary statistical design for conducting the engine testing program was completed during the second quarter of 1962 and preliminary studies were under way for the statistical design of experiments for component testing. Procedures for economically assessing the reliability of the engine were also being studied.

A tentative listing of the detailed information to be machine stored was prepared during the third quarter of 1962 while the studies and investigations previously initiated were continued.

In the fourth quarter of 1962, preliminary activity began in connection with reliability apportionment. An engine system block diagram had been prepared as part of the system design analysis completed for the selection of components for automated checkout. This diagram was modified to make it suitable for use in reliability apportionment. Modes of failure and failure effects analysis material(116) were reviewed. Also, a relative reliabilities evaluation was completed for six proposed systems of secondary gas injection into the combustion chamber for thrust vector control. The relative reliabilities were studied using an "equivalent component" method.

The Reliability Program Plan was revised in accordance with the NASA/LeRC document LeRC-REL-1, "Reliability Program Provisions for Research and Development Contracts," dated 20 August 1962. The statistical test planning at the end of 1962 was directed toward the preparation of preliminary statistically-designed experiments for the gas generator assembly, the thrust chamber valves, and the propellant utilization valves.

A meeting was held with personnel from the Rocketdyne Division of the North American Aviation Corporation and from the Pratt and Whitney Company early in the fourth quarter of 1962. Reliability demonstration problems were discussed and it was learned that Rocketdyne was using a success-rate method for reliability assessment in the F-1 Program. This method was similar to one

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(116) Bell, J. H., Generalized Failure Effects Analysis and Selection of Components Requiring Provision for Remote Automatic Checkout for the M-1 Rocket Engine, Aerojet-General Report No. 9430-2, 17 January 1963

considered by Aerojet-General for the M-1 Program but utilized a different mathematical model. An investigation of the merits of the Rocketdyne equation was undertaken during December 1962. Additional study was also devoted to the various aspects of calculating M-1 Engine reliability as a product of the best estimate of hardware as well as performance reliability.

1963

The statistical portion of a test plan for evaluating four igniter designs for the M-1 Engine was completed in January 1963. This test plan provided for the selection of the best igniter for future development. Also, the nomenclature for engine testing success-failure criteria was established.

An operating procedure defining the method to be used in accomplishing the reliability auditing function was released in February 1963.

In March 1963, detailed preliminary planning was conducted to establish the reliability data system for the M-1 Program. This would be a central source of information concerning hardware test experience and includes manual logs, mode of failure files, problem folders, and electronic data processing.

Work was initiated in February 1963 for the apportionment of engine reliability and additional analysis was performed in the reliability evaluation study of various systems for thrust vector control. Also, the paired comparisons of various engine parts which were made to determine relative failure potentials for the reliability apportionment were completed.

By the end of the quarter, a sampling plan which provided for more efficient vendor selection and testing of burst diaphragms was designed and implemented. Also, failure data from previous programs was assembled in support of the verification of selected leak test pressures, thrust chamber design, and hardware provisioning.

The failure checklists used for design review were revised in April 1963.

Apportionment of the reliability goals for the PFRT M-1 Engine was completed during the second quarter of 1963. It was accomplished using the Wilder Method recommended by MSFC.(117)

(117) A Generalized Approach for Systems Design Analysis and Selection of Components for Automation, MSFC Technical Report No. MTP-P&VE-E-62-2, 4 January 1962

1963

During the third quarter of 1963, Reliability personnel assumed the responsibility for assembling problem summaries. Reliability information for the then current M-1 engine configuration was prepared for use in NOVA vehicle studies. This included a predicted mortality curve (frequency of anticipated failures in relationship to engine operating duration), a study of inherent reliability, the effect of increasing run duration upon reliability, and the effect of starting at sea-level upon reliability.

Failure reporting and failure analysis reporting were accorded a comprehensive review by both NASA and Aerojet-General personnel during the fourth quarter of 1963. It was agreed that initial failure reports and failure analysis reports would be furnished to LeRC in a format, and in accordance with procedures, to be approved by LeRC. This agreement was reflected in a revised reliability program plan. Also, modes of failure analysis work was initiated.

Revised failure check lists for the turbopump assemblies, the combustion chamber, the thrust chamber assembly injector, the gas generator, the thrust chamber valves, the gas generator valves, propellant lines greater than 1.5-in. in diameter, hot gas lines greater than 1.5-in. in diameter, and tubes and fittings of less than 1.5-in. in diameter were distributed to the various cognizant engineering groups.

By the end of 1963, a reliability analysis had been conducted to compare the over-all M-1 engine system reliability with an 8.0 oxidizer-to-fuel ratio igniter and the reliability for an M-1 engine with an 0.8 oxidizer-to-fuel ratio igniter.

1964

The revised Reliability Program Plan was formally submitted to NASA/LeRC during the first quarter of 1964 as part of Report LRP 256, Volume 2, Revision D, the M-1 Engine Development Program Plan. During this same period, an informal report about the effects of mixture ratio upon M-1 Engine igniter reliability was prepared. Also, the reliability criteria used in the design of the turbopump bearings was summarized.

During the third quarter of 1964, a comprehensive technical assessment of the M-1 Engine was undertaken for the purpose of uncovering possible technical problems not previously identified. Both NASA/LeRC and Aerojet-General personnel participated. Seven Failure Analysis Reports and one Initial Failure Report were submitted in this same time period.

1964

Technical Assessment Reviews were conducted during the fourth quarter of 1964 for the fuel turbopump assembly, thrust chamber, gas generator, controls, engine lines, and engine structures. An updated listing of the M-1 components selected as protoypes was prepared and the Critical Parts List was completed.

A Critical Experiment Review of the S/N 017 M-1 gas generator assembly components and test Stand H-8 was conducted during October 1964. Another Critical Experiment Review for the gaseous nitrogen drive spin testing of the oxidizer turbopump assembly (pumping liquid nitrogen) was conducted on 8 and 9 December 1964.(118)

One Interim Failure Analysis Report was published during this time.

1965

A Critical Experiment Review was conducted during February 1965 for the scheduled testing of the prototype gas generator valves on a gas generator at Test Stand H-8.(119)

A Critical Experiment Review was conducted on 12 March 1965 for the first gas generator test series at Test Stand E-3.(120) Another Critical Experiment Review was conducted on 19 March 1965 for the first test series of the fuel turbopump assembly pumping liquid hydrogen with a gaseous nitrogen drive.(121)

Adjustments were made during the first quarter of 1965 as pertained to the scope and responsibility for accomplishing reliability tasks within the program redirection requirements. The maintenance of problem folders as well as the initiation and publication of failure reports was reassigned to the cognizant engineering managers although Reliability would continue to monitor these functions.

Eight Initial Failure Reports and two Failure Analysis Reports were issued during this first quarter of 1965.

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(118) <u>Critical Experiment Review for First Test of the M-1 OTPA (Mod I), Gaseous Nitrogen Drive, Pumping Liquid Nitrogen</u>, Aerojet-General Report No. RR-CER-2-64, 31 December 1964

(119) <u>Critical Experiment Review for First Test of the M-1 Gas Generator Valves</u>, Aerojet-General Report No. RR-CER-1-65, 12 February 1965

(120) <u>Critical Experiment Review for First Test of M-1 Gas Generator on Test Stand E-3</u>, Aerojet-General Report No. RR-CER-2-65, 2 April 1965

(121) <u>Critical Experiment Review for First Test of the M-1 Fuel Turbopump Assembly</u>, Aerojet-General Report No. RR-CER-3-65, 7 April 1965

A revised Reliability Program Plan was published during the second quarter of 1965 and two Critical Experiment Reviews were conducted.(122)(123) Also, a reliability manual was published and distributed.(124) Five Initial Failure Reports and five Failure Analysis Reports were prepared.

An informal Critical Experiment Review was conducted during July 1965 for the first test of an M-1 gas generator assembly at Test Stand E-1.

The Critical Experiment Review for the first test of the M-1 Fuel turbopump assembly using a gas generator assembly drive was held on 3 August 1965.

Eight Initial Failure Reports and five Failure Analysis Reports were submitted during the third quarter of 1965.

A Formal Critical Experiment Review for initiating uncooled thrust chamber assembly testing at Test Stand H-8 was conducted on 4 November 1965.(125) In this meeting, detailed consideration was given to the actions taken in response to the recommendations made by the M-1 Accident Special Investigation Team regarding the C-9 accident(126) and H-8 activation.(127)

Four Initial Failure Reports and seven Failure Analysis Reports were prepared during the fourth quarter of 1965.

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(122) Informal Critical Experiment Review for Change to Prototype M-1 Gas Generator Valves on First "E" Area Gas Generator Test Series, Aerojet-General Report No. RR-CER-4-65, 4 June 1965

(123) Final Report, Critical Experiment Review for First E-3 Test of M-1 OTPA with GGA Drive Pumping LN_2, Aerojet-General Report No. RR-CER-5-65, 17 June 1965

(124) Engineers Handbook of Procedures for M-1 Reliability Tasks, May 1965

(125) Critical Experiment Review for First Test of M-1 Uncooled Thrust Chamber Assembly on Test Stand H-8, Aerojet-General Report No. RR-CER-8-65, 21 December 1965

(126) M-1 Thrust Chamber Assembly Accident (Test No. 1.1-02-EHM-003), 20 June 1964, M-1 Accident Special Investigation Team, Lt. Col. Joe. E. Heatherly, Chairman, 28 August 1964

(127) Pre-Operational Inspection and Review of M-1 Test Stand H-8, M-1 Accident Special Investigation Team, Lt. Col. Joe E. Heatherly, Chairman, 30 December 1964

X. FACILITIES AND SPECIAL TEST EQUIPMENT (STE)

SUMMARY

Initial emphasis in the program was concentrated upon design. A facility plan was developed, design criteria were established, and preliminary design layouts were initiated.

The basic facility plan was released on 18 February 1962(128). This plan was designed to make maximum use of existing facilities. It was revised throughout the program as facility requirements changed because of hardware, configuration, and development plan iterations as well as to reflect improvements resulting from test and facility experience. Significant facility plan revisions were released in January 1963(129)(130) as well as in February and March 1964.(131)(132)

The responsibility for the management of the basic facility design and construction was assigned to the AETRON Division (Architectural-Engineering) of the Aerojet-General Corporation. The Aerojet-General Sacramento Test Operations was responsible for the establishment of criteria, design review, over-all facility program management, facility activation, and facility operation. This effort has been extensively documented as pertains to design criteria(133) and facility drawings(134) as well as to other technological areas which are approximately referenced throughout this discussion.

Early preliminary design phase considerations were concerned with the identification and procurement of critical long-lead items (i.e., large cryogenic and high-pressure gas storage vessels, valves, pumps, and instrumentation). The schedule of facility availability was predicated upon both the complexity as well as the magnitude of design and construction along with the timing requirements for meeting hardware testing demands. Interim facilities were provided when required (i.e., Test Stand C-6 for interim gas generator

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(128) M-1 Engine Development Program Plan, Aerojet-General Report No. 256, 18 February 1962
(129) Facilities for the M-1 Development Program, Aerojet-General Report No. LR62313B, Volume 4, 31 December 1962
(130) Special Test Equipment for the M-1 Development Program, Aerojet-General Report No. LR63212B, Volume 3, 6 January 1963
(131) M-1 Engine Research and Development (Special Test Equipment), Aerojet-General Proposal No. LR642200, Volume 3, Part 2, 28 February 1964
(132) M-1 Engine Research and Development Facilities, Aerojet-General Proposal No. LR642201, Volume 4, 4 March 1964
(133) M-1 Design Criteria for Liquid Rocket Plant Test Facilities Serving M-1 Engine Development Program, August 1963
(134) Drawing List for M-1 Facilities (By Specification), October 1965

assembly testing). Major M-1 emphasis at the outset of the program was placed upon those test stands considered critical to the test program as well as being most independent of engine configuration changes or contract funding schedules as related to the over-all NASA planning and the assumption of program management by NASA/LeRC. These critical facilities were Test Stands C-9, E-1, E-3, and H-8.

By the end of 1963, the engine configuration as well as the test requirements had been sufficiently established to permit design of the K Zone complex (layout) including Test Stand K-1 and its support facilities. Also, conceptual designs of Test Stands K-2 and K-3 were undertaken. The K-1 facility was approximately half completed when the stop order was received on 5 February 1965.

Table No. I provides a summary of the original facilities plan and identifies appropriate changes leading to the final configuration. It includes utilization, capability, status, stand and nomenclature changes, as well as additions and deletions.

CHRONOLOGY

A. TEST ZONE C

1. Test Stand C-6

1963

This test stand was modified for interim gas generator testing. Essentially, this modification consisted of a thrust mount, piping, and a burnoff stack.

Design of the test stand modification was completed during the first quarter of 1963 and the actual work was completed during the second quarter of 1963.

The first gas generator test at this stand was made on 17 May 1963 and the modifications proved to be satisfactory.

2. Test Stand C-9

1962

The civil engineering, structural drawings, and specifications for this stand were completed during the second quarter of 1962. Also, run vessels VE-30 and VE-31 were released for procurement. The structural contract was awarded during the third quarter of 1962 as well as the civil works contract. By the end of the quarter, pilings were being driven and the lower portion of the concrete foundation had been poured (see Figures No. 39 and No. 40).

ORIGINAL PLAN		FINAL PLAN		
FACILITY	UTILIZATION	FACILITY	UTILIZATION	
E-4	Wedge Chamber, GGA, and TCA	C-6	Interim GGA, Wedge Chamber	Exist (E-4) USAF
		C-9	GGA, TCA, and Wedge Chamber	
E-2	TPA	E-2A and E-2B	TPA	Addit bilit
E-2A	TPA	E-1(1)	FTPA	Chang stand
E-2B	TPA	E-3(1)	OTPA	
H-8	GGA and TCA	H-8	GGA and TCA	No ch
E-1A, 3A, 4A	Engine	J-1, 2, 3, 4	Engine	Reloc Zone capab
J-1, 2, 3, 4	Engine	K-1, 2, 3, 4(2)	Engine	Chang stanc
Cryogenics Laboratory	Components	Cryogenics Laboratory	Components	No ch
D Zone	Scale Pumps and other Components	D-3 D-4 D-8	Scale Pumps and other Components	No ch
Azusa/ G-7	Stage Igniter	J-1A	Stage Igniter and Uni-Element Injector	Trans Azusa Sacrai chang nomen

* All hot tests unless otherwise specified.

(1) Both Test Stand E-1 and E-3 were designed and construc utilizing on-stand tankage only and long duration test testing capability only for the FTPA at Test Stand E-1

(2) Several changes were evolved during the facility progr
Test Stand K-1: Single position 30 sec sea-level run
Test Stand K-2: Two position - 330 sec duration, one conditioning capability was also incl
Test Stand K-3: Two position - full 550 sec duration,

183-1

TABLE I

SUMMARY OF M-1 TEST FACILITIES

REASON FOR CHANGE	FIRST TEST DATE*	FINAL TEST FACILITY CAPABILITY	STATUS
ing Facility required for Test Programs	Wedge: May 1962 GGA: May 1963 GGA: October 1963 TCA: June 1964	Vertical Position, 27 sec GGA duration Vertical Position, 33 sec GGA and 3 sec TCA duration at 1.2M lb thrust	8 cold flows and 1 hot wedge test + 5 cold flows and 11 hot GGA tests. Test capability switched to C-9. 5 cold flows and 7 hot GGA tests + 49 cold flows and 3 hot TCA tests. Test stand damaged - not repaired.
.onal capa- / required › in test nomenclature	Cold: May 1965 Hot: December 1965 Cold: January 1965 Hot: August 1965	TPA tests with both GGA and cold gas drive. Duration: on-stand vessels only - 15 sec; with off-stand vessels - 335 sec. GGA test duration - 146 sec.	17 OTPA tests with gaseous nitrogen drive and 7 tests with GGA drive. 10 FTPA tests with gaseous nitrogen drive and 2 tests with GGA drive. 4 cold flow and 9 hot GGA tests. Test stands are in stand-by "mothball" condition.
ange	GGA: November 1964 TCA: May 1966	Horizontal position, 146 sec GGA duration. Horizontal position, 10 sec uncooled and 15 sec cooled TCA duration. Thrust measurement up to 1.5M lb.	7 cold flows and 18 hot GGA tests. 16 cold flows and 10 hot TCA tests. TCA testing continuing under Large LO_2/LH_2 Injector Test Program.
ated to J and added ility e in test l nomenclature		See (2)	Design was completed for Test Stand K-1, the Control Room, and the area support facilities. Construction terminated prior to completion. Test Stand K-2 and Test Stand K-3 were not started.
ange	November 1963	Operational tests of bearings, seals, valves, etc.	Approximately 430 tests were conducted. Facility is now being utilized for NERVA Program testing.
ange	April 1964 January 1963 September 1962	Scale Pump water flow tests, valves, and other components flow tests under cryogenic conditions.	66 scale pump water flow tests. Approximately 170 tests of valves, seals, controls, instrumentation, etc. Facility has reverted to USAF testing.
'er from to iento and › in test stand :lature	Stage Igniter: July 1964 Uni-Element: October 1964	Ignition system testing and scale injector element hot firing tests.	29 cold flow tests and 1 hot firing of the stage igniter. 51 cold flows and 59 hot firing single element injector tests. (47 stage igniter tests were performed at the Azusa facility prior to moving the test effort to Sacramento.)

:ed to provide for both FTPA and OTPA separate as well as combined test capability in each stand. Short duration tests
› utilizing the large off-stand vessels were also provided for. M-1 Program redirection imposed a short duration
. and the OTPA at Test Stand E-3.
ım as pertained to the number of test stands as well as the configuration. The final plan provided for the following:
vith altitude start only and environmental conditioning.
position vertical and one position horizontal with altitude capability for the full run duration. Environmental
uded.
vertical firing configuration.

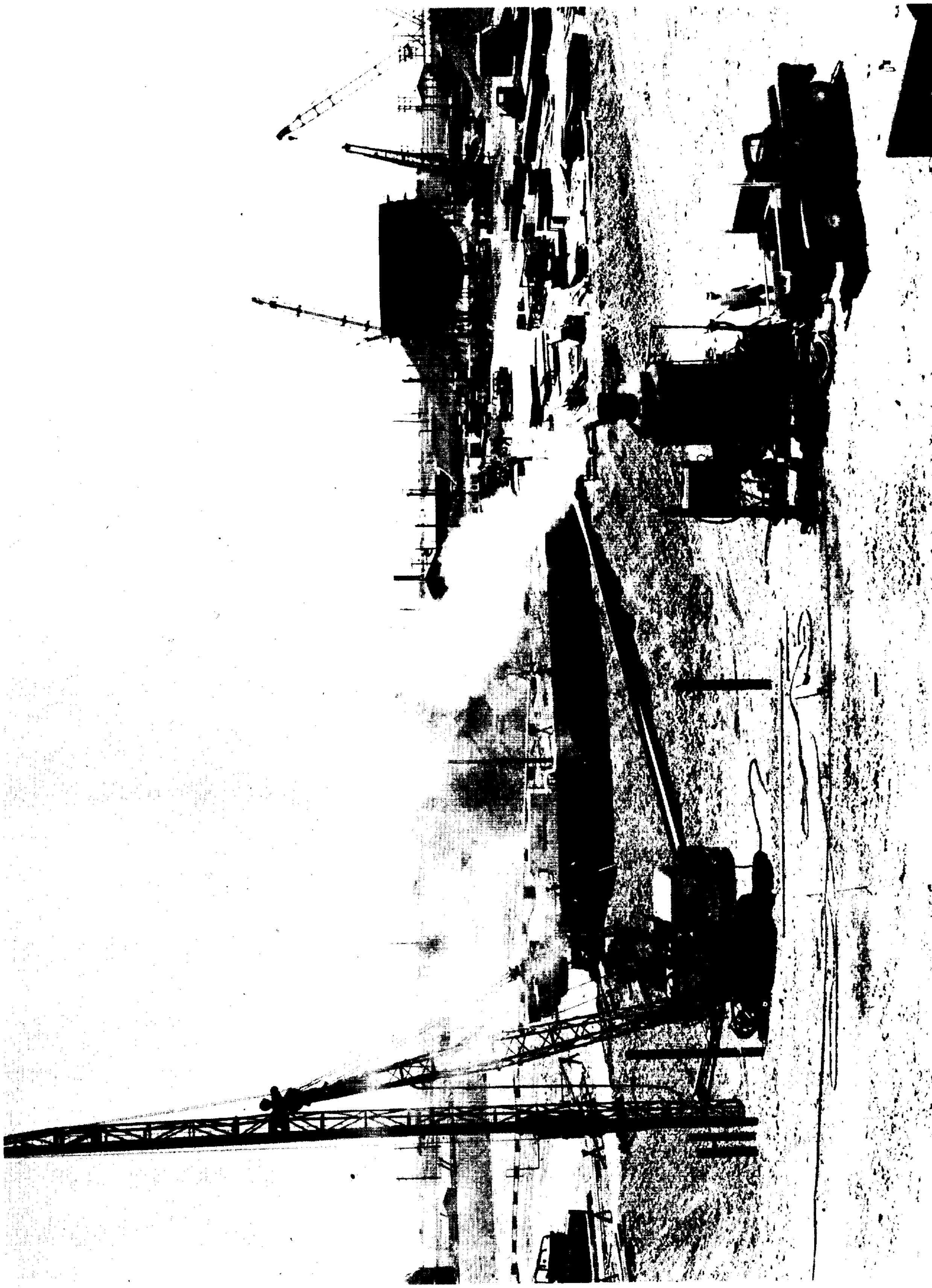

Figure 39. Piling Being Driven for Test Stand C-9

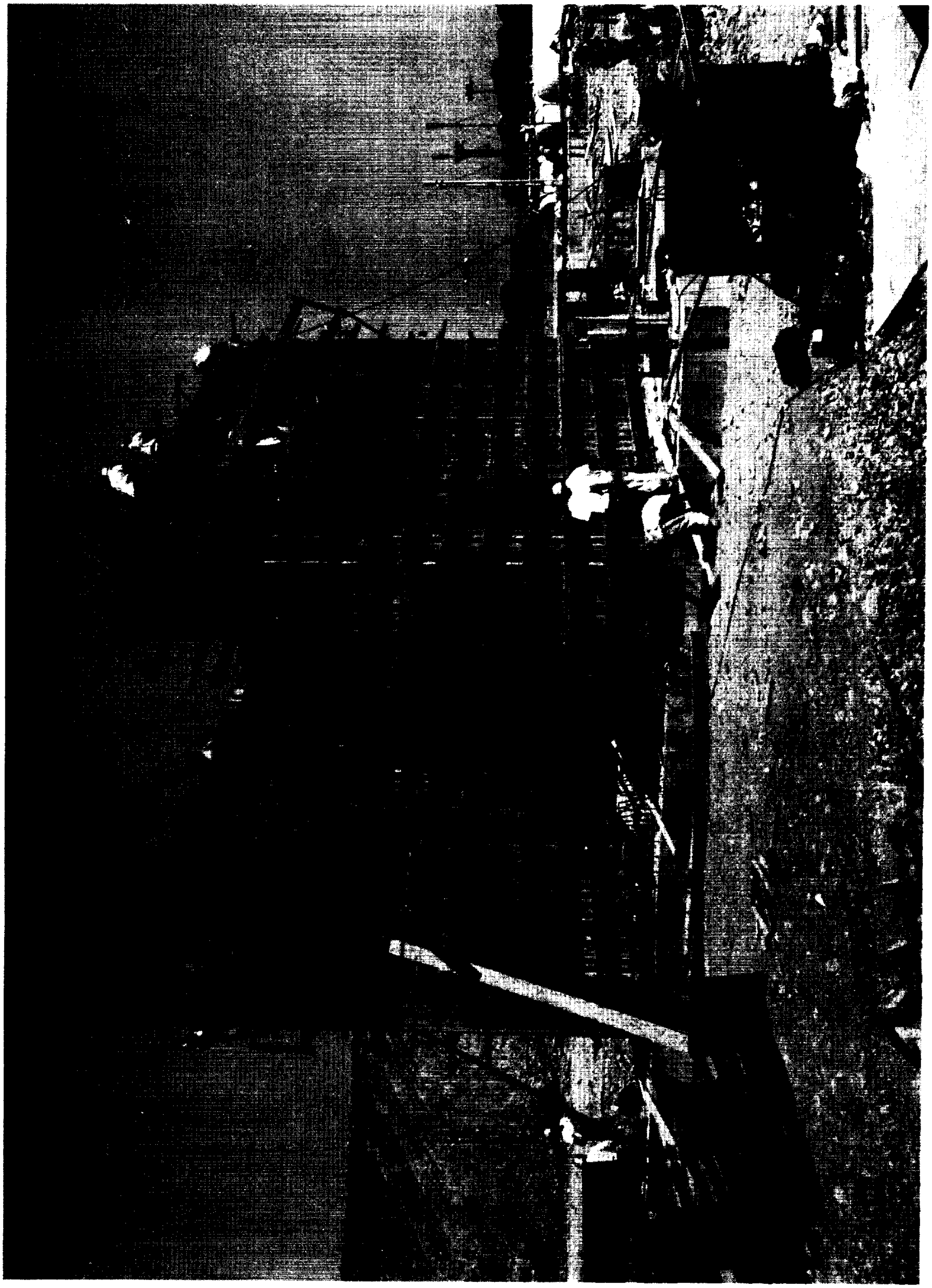

Figure 40. Test Stand C-9 Concrete Foundation

1962

The civil works and steel superstructure was essentially completed during the fourth quarter of 1962, including the rolling platform and jib cranes (Specification 6447). Also, two 1,500-gal-capacity, high pressure water vessels were relocated to Test Stand C-9 from other test areas to provide a water supply for the wedge chamber cooling and work was started on the mechanical installations (Specification 6451).

Installation of the water deluge piping as well as the conduit and light standards on the superstructure were started when work was temporarily suspended because of a need to raise the superstructure approximately five feet to accommodate the increased M-1 thrust chamber assembly over-all length.

1963/1Q

The Test Stand C-9 superstructure was disassembled and five feet were added to the base of all columns during the first quarter of 1963. The structure was re-assembled to vessel support level and the liquid hydrogen run vessel (VE-31) and the liquid oxygen run vessel (VE-30) were installed. Figure No. 41 shows Test Stand C-9 at the time of structural modification.

The C-9 instrumentation and controls package was awarded and the electronic servo-controllers were received.

By the end of March 1963, earth-moving had been started for the road from C Area to Test Stand C-9 and for the revetment between C-9 and E Area. These latter items were included as a portion of the E-Area civil works package (Specification 6473).

1963/2Q

The civil works package (Specification 6447), the concrete wall and earth revetment between C-9 and the Test Zone C propellant storage area, the road between C-9 and the propellant storage area and the earth revetment extension between C-9 and Test Zone E were completed during the second quarter of 1963.

The mechanical package (Specification 6451) works also exhibited considerable progress during this quarter. The 14-in. liquid oxygen and liquid hydrogen main tank safety valves as well as the thrust chamber assembly elevator (see Figure No. 42) and the gas generator burnoff stack were installed. Also, the vacuum-jacketed liquid hydrogen transfer line was essentially completed and the design of the equipment handling package was completed. This latter includes hoists and structures for removing or installing valves as well as heavy equipment on the test stand. In addition, the conversion of Control Room D-3 was started and completed during this quarter.

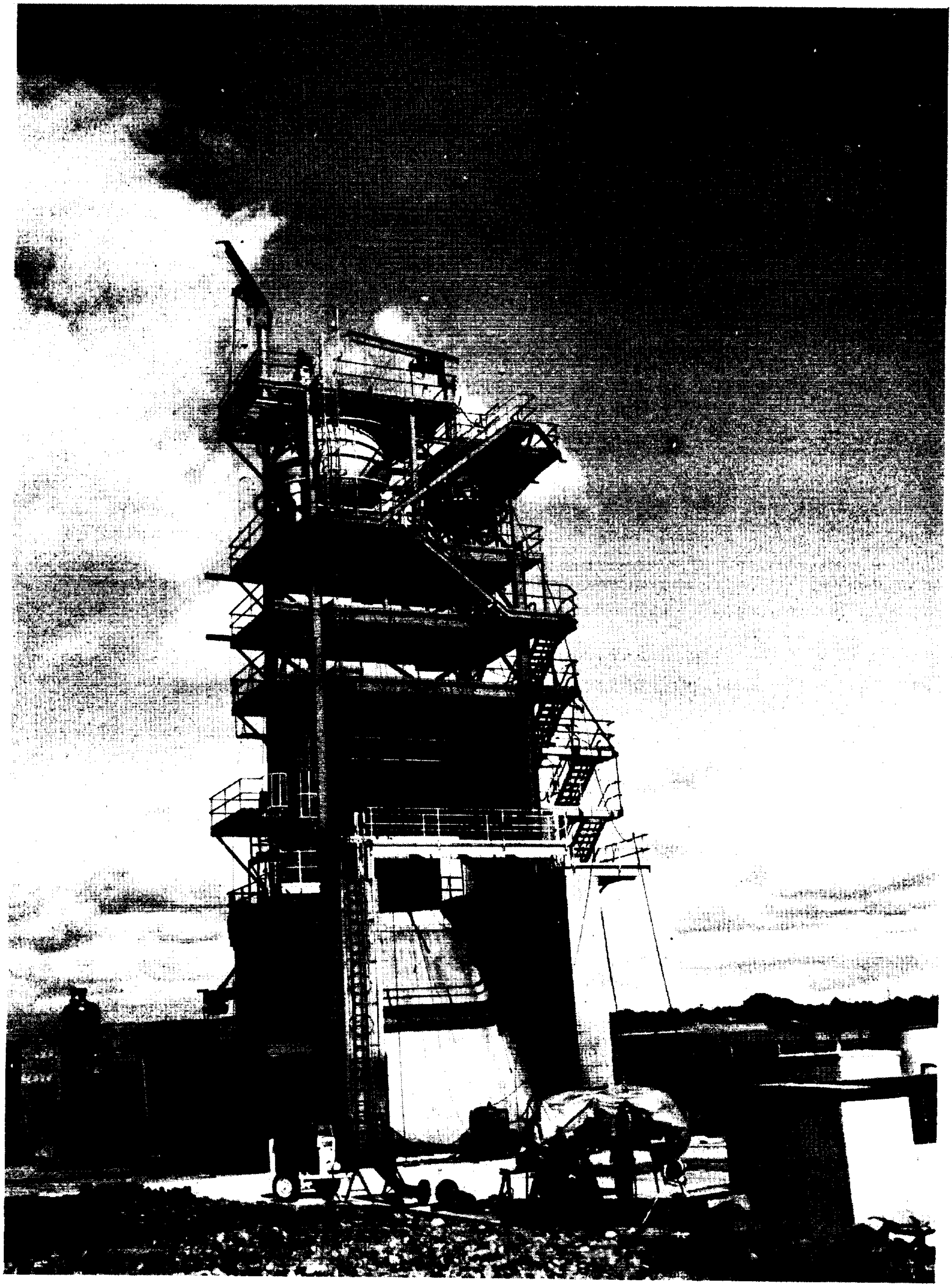

Figure 41. Test Stand C-9 at the Time of Superstructure Modification

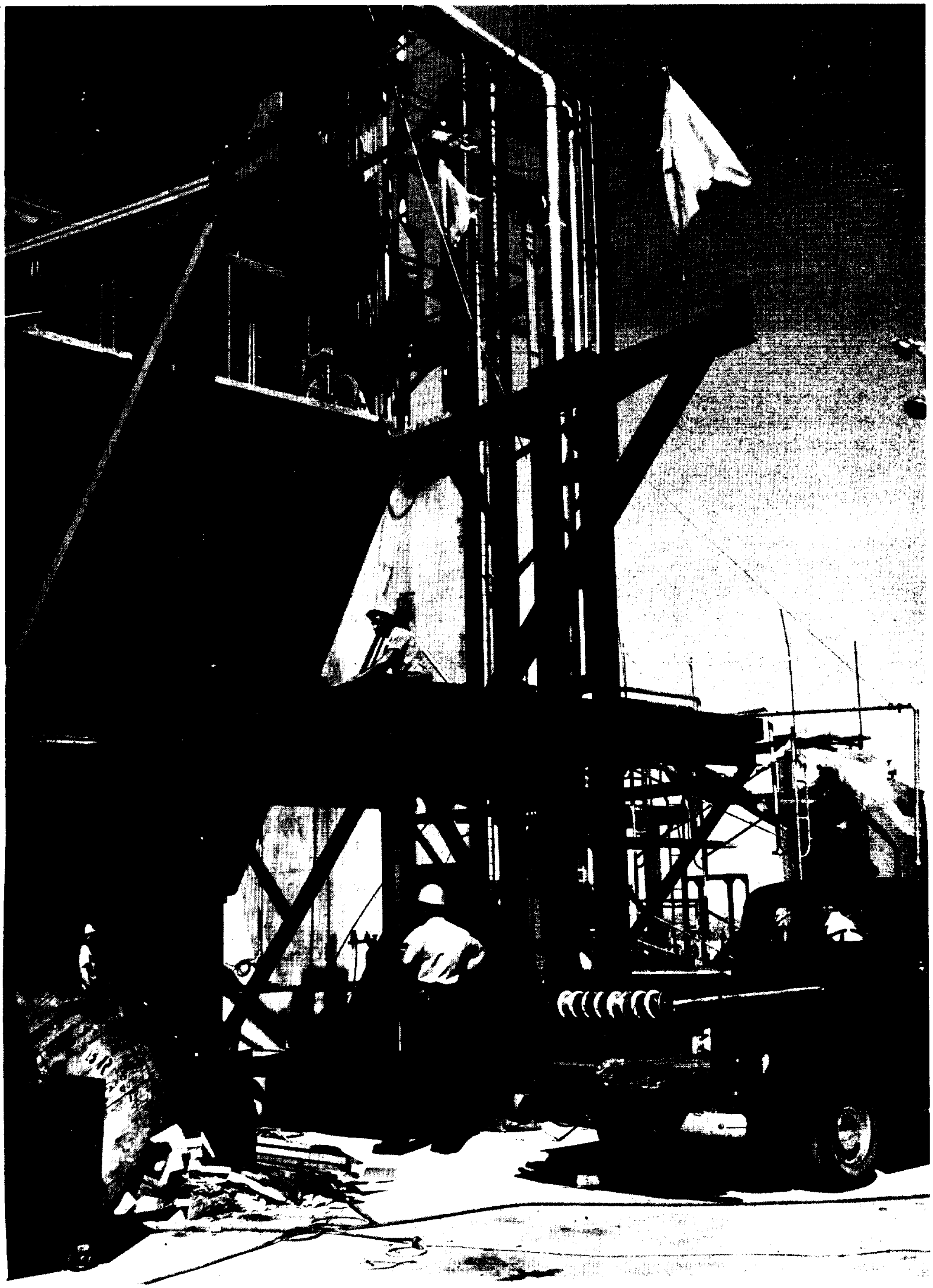

Figure 42. Test Stand C-9 TCA Elevator

1963/3Q

All contractor mechanical work necessary for gas generator tests on Test Stand C-9 was completed in the third quarter of 1963 and activation of the stand was started. The liquid oxygen gas pressurization system and the purge systems were checked out. Installation of gas generator propellant feed lines was completed and they were inspected (see Figure No. 43). Checkout of the liquid oxygen fill system was also completed and the test stand run vessel was filled. Dynamic checkout and calibration of the servo-controlled pressure regulating systems were completed.

By the end of September 1963, the instrumentation and control system installation was essentially completed. Instrumentation and control system activation was started concurrently with mechanical activation.

1963/4Q

The first gas generator test firing was accomplished at Test Stand C-9 on 18 October 1963. Gas generator testing continued through 20 November 1963 when the contractor returned to the test facility to complete construction of the thrust chamber assembly test position. The almost completed stand is shown on Figure No. 44. This construction included the installation of the thrust chamber assembly fuel and oxidizer suction line, high and low point bleeds, propellant line supports, and thrust chamber assembly test instrumentation. Work was also being continued for the gaseous hydrogen injection system.

1964/1Q

The second and final construction phase for Test Stand C-9 was completed during January 1964. The 2:1 area ratio thrust chamber assembly was then installed for final adjustment of the propellant lines and the side thrust restraint structure.

The thrust chamber assembly was removed from the stand during February and the mockup thrust chamber valves were replaced with the valves to be used for the first test. The thrust chamber assembly was reinstalled on the test stand and all test stand systems were then considered to be completed.

1964/1Q/2Q

Pressurant and propellant system calibration flow tests began during March 1964; however, the gas filter elements in the gaseous hydrogen and gaseous nitrogen pressurant systems failed structurally during the initial phase of the cold flow tests. Two courses of corrective action were immediately started. The filter elements were returned to the vendor for redesign and rebuilding. Also design and fabrication of replacement elements with relaxed filtration requirements were started as an interim measure. Calibration and flow tests were resumed in the second quarter of 1964 after the rebuilt filter installation was completed. Excessive pressure drop across the filters and greater than anticipated gas collapse rates in the liquid oxygen tank resulted in marginal conditions for thrust chamber assembly testing.

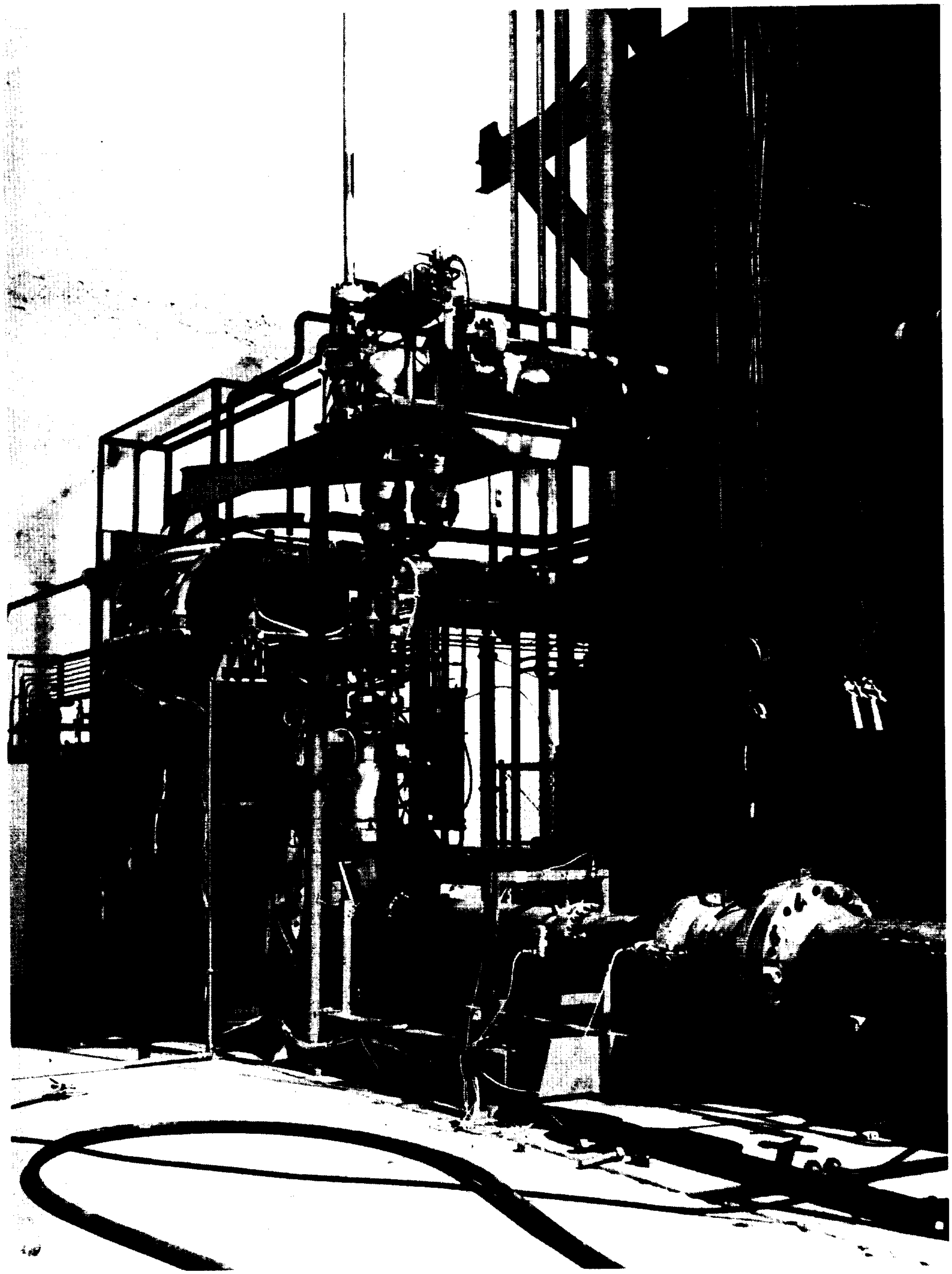

Figure 43. Test Stand C-9 GGA Installation

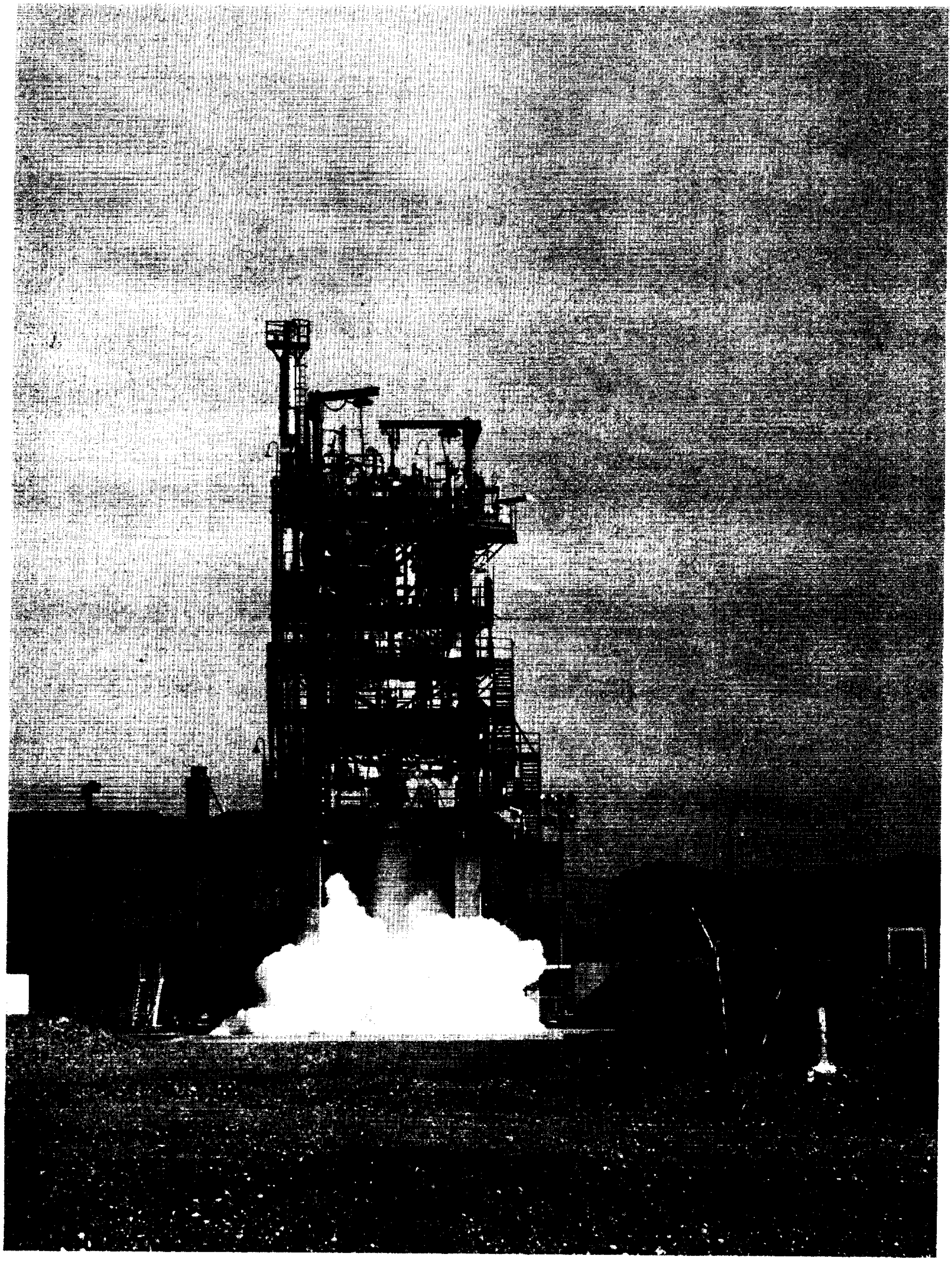

Figure 44. Test Stand C-9

1964/2Q

A design review resulted in a redesigned pressurant diffuser for the liquid oxygen tank and a series of gaseous nitrogen pressurant flow checks conducted in May 1964, and the first week of June verified the adequacy of the system to permit the first thrust chamber assembly firing.

The first firing attempt was made on 6 June 1964; however, combustion was not obtained. A slight chamber pressure was recorded in the second firing attempt. All test stand systems apparently functioned satisfactorily. Problems associated with the liquid oxygen system during the third test on 20 June 1964, resulted in a malfunction and extensive hardware and test stand damage was incurred. Evaluation of the damage and creation of a repair task force team was initiated immediately.

An investigation team was initially convened by NASA from 24 June through 27 June 1964, to review test stand damage and summarize findings for cause and corrective action for the benefit of other programs as well as the M-1 Program.

1964/3Q

Demolition of the damaged portions of Test Stand C-9 was completed in July 1964 and a comprehensive contamination survey was accomplished. Test stand restoration design and preprocurement of certain long-lead items were initiated. Also, a contract for the repair of the liquid oxygen propellant vessel, VE-30, was awarded. Figure No. 45 shows the stand after the accident.

The final report of the Test Stand C-9 accident was published on 28 August 1964.(135)

During August, the liquid hydrogen propellant vessel, VE-31, was inspected, pressure tested, and found to be in satisfactory condition for re-activation. The restoration design for all structural work and piping, except vacuum-insulated pipe, was completed, released for bid, and the bids were received.

Award of the restoration contract was withheld pending NASA/LeRC review of alternatives for the installation of facility thrust chamber valves and the program requirements for the Test Stand C-9 thrust chamber assembly test position. In september 1964, a decision was reached between the Aerojet-General Corporation and NASA/LeRC to defer the reactivation of Test Stand C-9 and to initiate a standby plan.

1964/4Q

The design and specification for restoration of Test Stand C-9 for gas generator assembly testing were completed during the fourth quarter of 1964. Design, procurement, restoration, and operational activities for Test Stand C-9 were then suspended and never reactivated during the remaining course of the program.

(135) M-1 Thrust Chamber Assembly Accident (Test No. 1.1-02-EHM-003), 20 June 1964, M-1 Accident Investigation Team, Lt. Col. Joe E. Heatherly, Chairman,

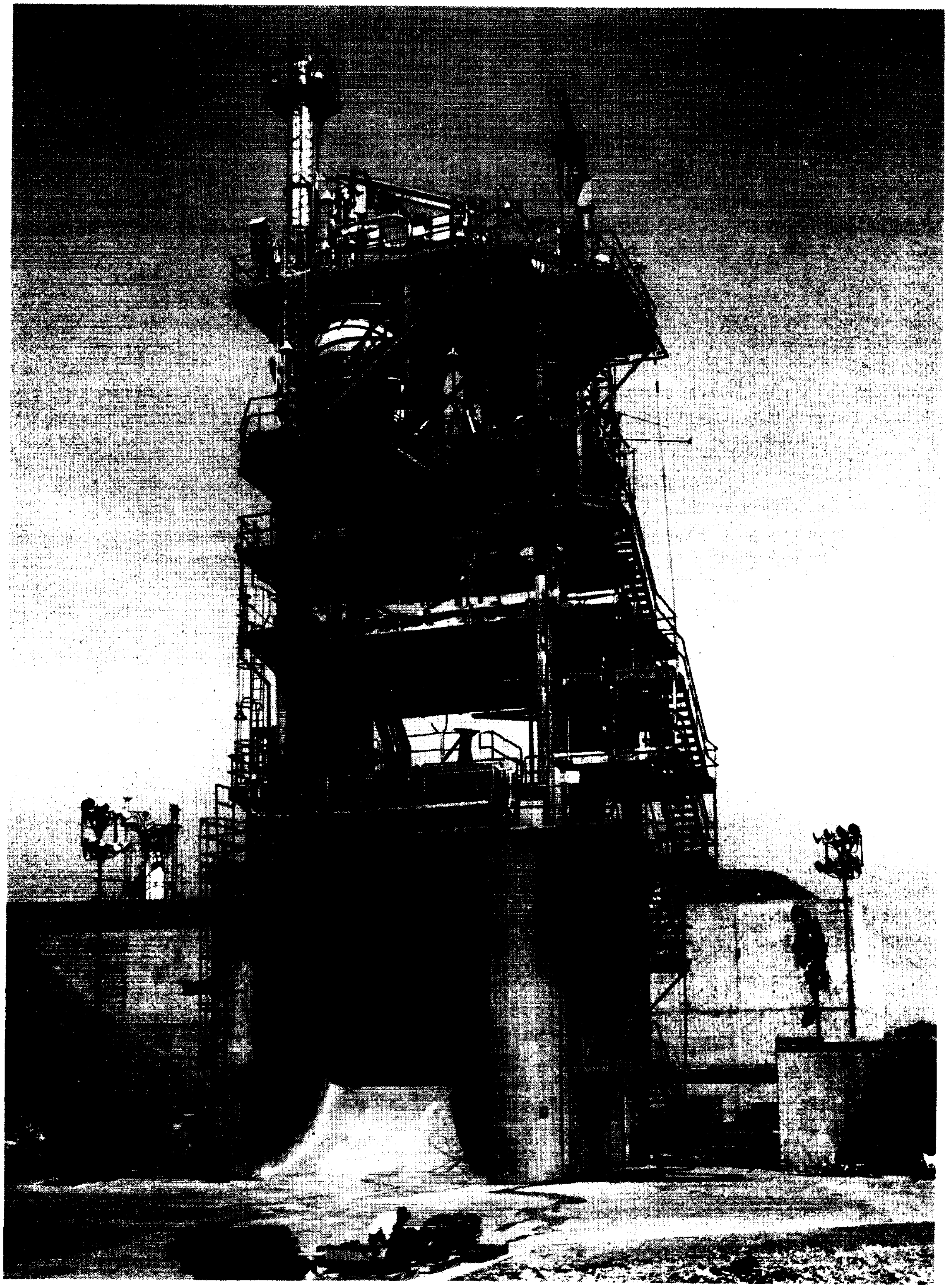

Figure 45. Test Stand C-9 after Accident

B. TEST ZONE D

Three existing test bays in this zone were used for the M-1 Program; the D-3 scale pump test facility, the D-4 bearing and seal test facility, and the D-8 valve test facility.

The design of a one-ton capacity crane was completed during the third quarter of 1963. This crane was for use in Test Bay D-3 for handling and positioning M-1 scale pump hardware. It was installed during the fourth quarter of 1963.

The Test Bay D-3 flow test facility modification was essentially completed and ready for the initial test series by the end of the first quarter of 1964. A scale model fuel pump was installed during the ensuing quarter and the first phase of testing began on 15 April 1964. The test systems performed satisfactorily and the test facility was considered activated.

The first scale model oxidizer pump was installed in Test Bay D-3 during the third quarter of 1964 and the first test was successfully accomplished. The activation of Test Bay D-3 was then considered to be completed. Sixty-six scale pump tests were made in this facility.

One-hundred-and-seventy component tests were conducted during the course of the Program in Test Stands D-4 and D-8, which required only minor special test equipment modifications.

C. TEST ZONE E

A number of changes were made as regards the number of stands, their identification, and their capability during the course of developing this test complex. The final plan provided for turbopump assembly testing capabilities at Test Stands E-1 and E-3. In view of this, the following chronological summary of the development of Test Zone E is limited to developments affecting Test Stands E-1 and E-3 only.

1962

Emphasis during the second quarter of 1962 was placed upon the design of the Zone E support facilities, piping, and mechanical equipment. The procurement of the long-lead large propellant vessels was initiated. The existing test stands (positions E-1, E-2, and E-3) were demolished early in the third quarter of 1962 as can be seen in Figure No. 46. Also, the procurement of critical long-lead 18-in. ball valves was initiated. The final design of the civil works package was nearing completion including roads, revetments, and liquid hydrogen unloading stations.

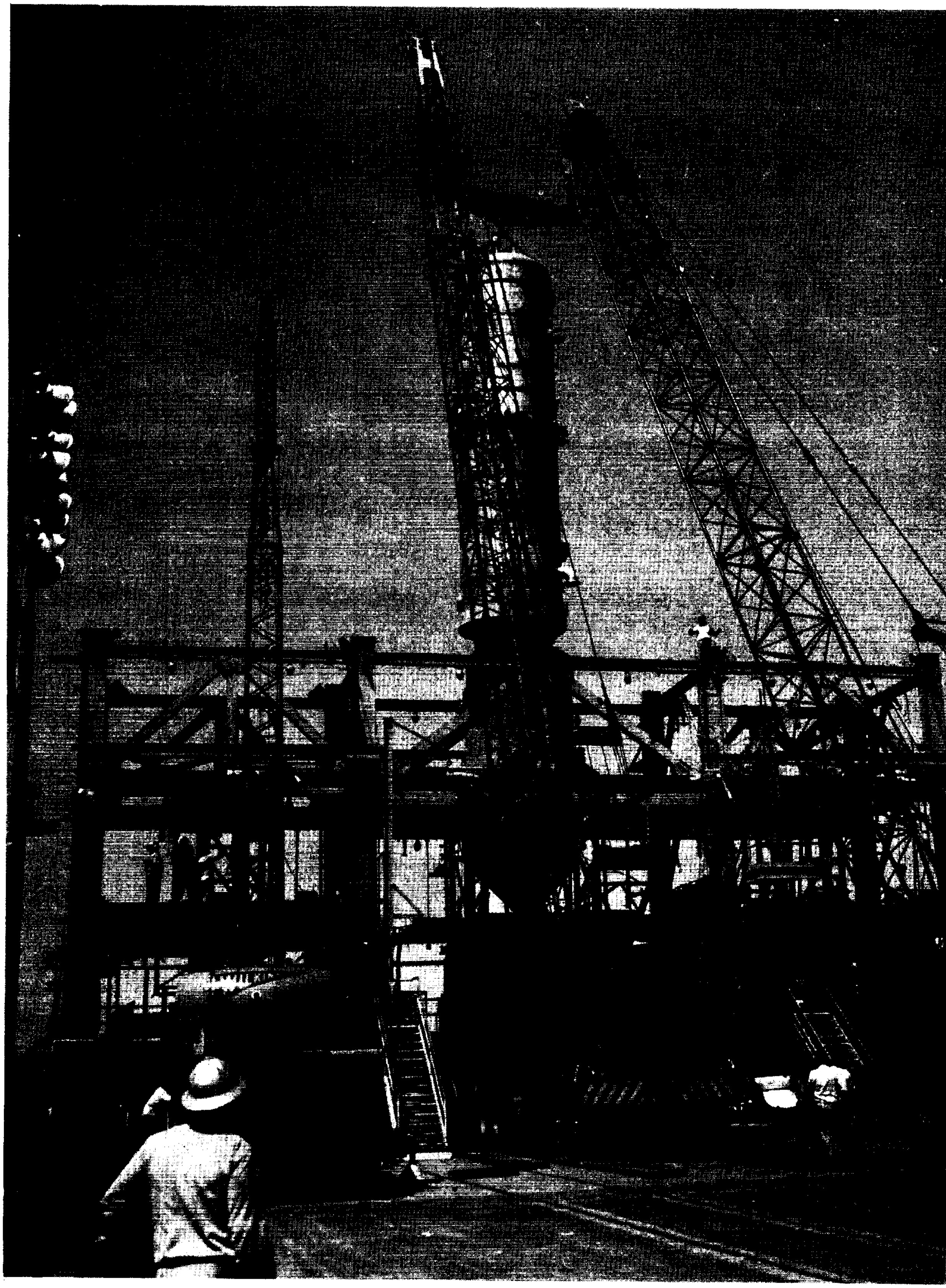

Figure 46. Demolition of Test Stands E-1, E-2, and E-3

1962

By the end of September 1962, the existing gas receivers and foundations had been removed. The new foundations for propellant vessels VE-1 (liquid hydrogen run vessel), VE-2 (liquid hydrogen catch vessel), VE-10 (liquid oxygen run vessel), and VE-11 (liquid oxygen catch vessel) were poured and field erection of these vessels was started as shown in Figure No. 47.

1963/1Q

That portion of Zone E Civil Works Package (Specification 6473) delineating foundation for high pressure gas receivers became a separate bid item during the first quarter of 1963 to expedite the construction of these foundations. A contract was awarded and the foundations were completed. Eight new high pressure gas receivers and three existing receivers were installed on these foundations (see Figure No. 48). The design of concrete blast walls to protect the liquid hydrogen and liquid oxygen run and storage vessels was also completed. A contract was awarded for the blast walls and construction began (see Figures No. 49 and No. 50).

The design of the Test Zone E Mechanical Package (Specification 6367) was completed and released for bidding in March 1963. Also, a contract was awarded for long-lead piping (other than vacuum-jacketed) for installation as part of the mechanical package.

Design of pipe supports and equipment foundations was completed (Specification 6620) (see Figure No. 51). This work, which involved relocation of the liquid oxygen vessels and suction lines, modification of the blast screens, and the addition of work platforms was started immediately.

Fabrication of instrumentation drop boxes was also initiated during March 1963 and a valve programing control panel for use in the E Zone servo-control system was being designed. This servo-control system and the analog simulation in E Zone were detailed in a separate report(136); therefore, reference to them in this report is limited to highlights only.

1963/2Q

All start transient and gas generator propellant vessels were received during the second quarter of 1963 and installed on Test Stands E-1 and E-3 as shown on Figure 52. The propellant storage and catch vessels VE-1, VE-2, VE-10, and VE-11 were essentially completed and the liquid hydrogen storage vessel, VE-34, was received and installed. The E Zone blast walls (Specification 6615) were also completed and the mechanical package was awarded. Construction was started with the installation of the liquid oxygen transfer piping between the unloading area and the liquid oxygen storage area.

(136) Garcia, L. W., Friedland, H., Lehmburg, A. E., Servo-Control System and Analog Simulation in E-Zone, Aerojet-General Report No. 8800-61, 10 March 1966

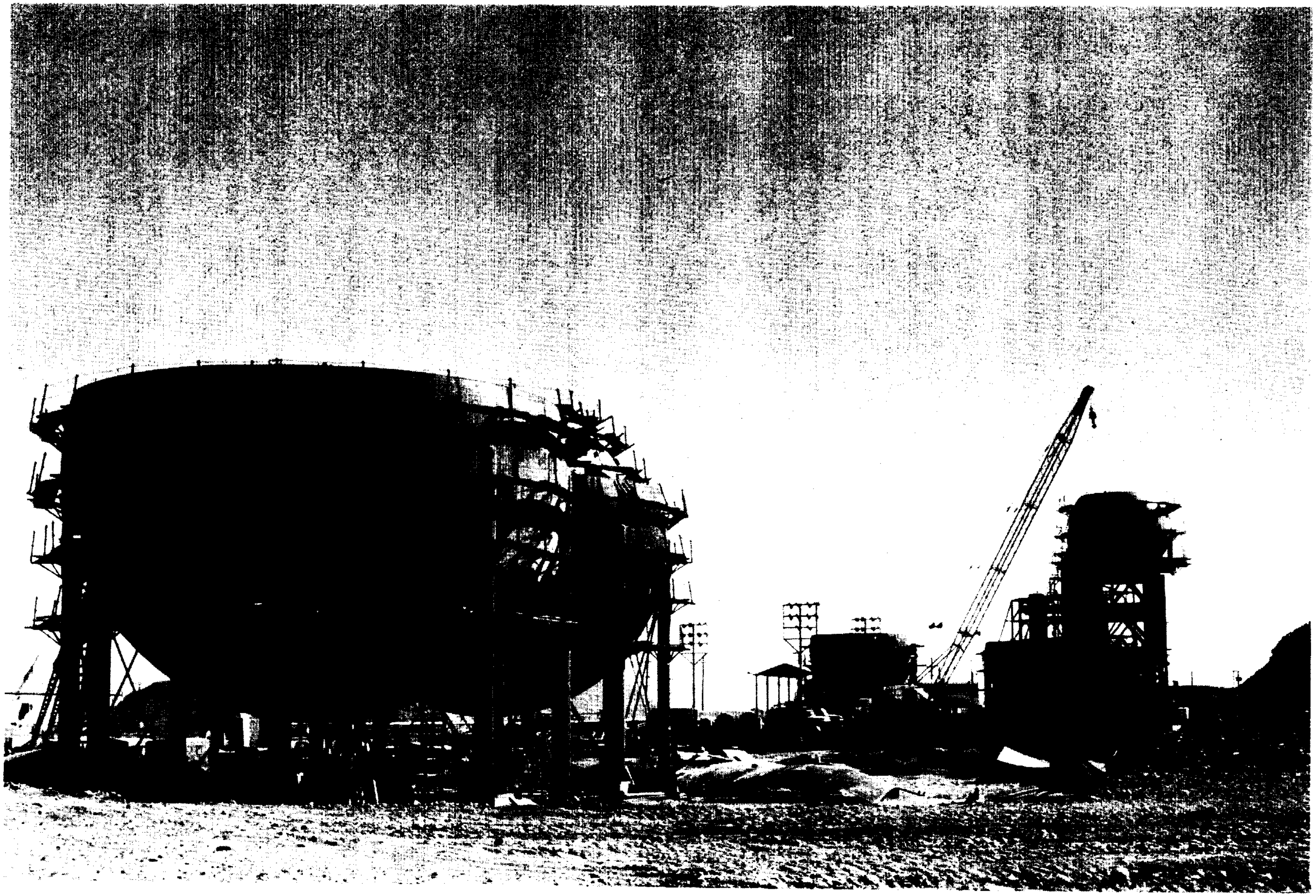

Figure 47. Field Erection of Zone E Propellant Vessels

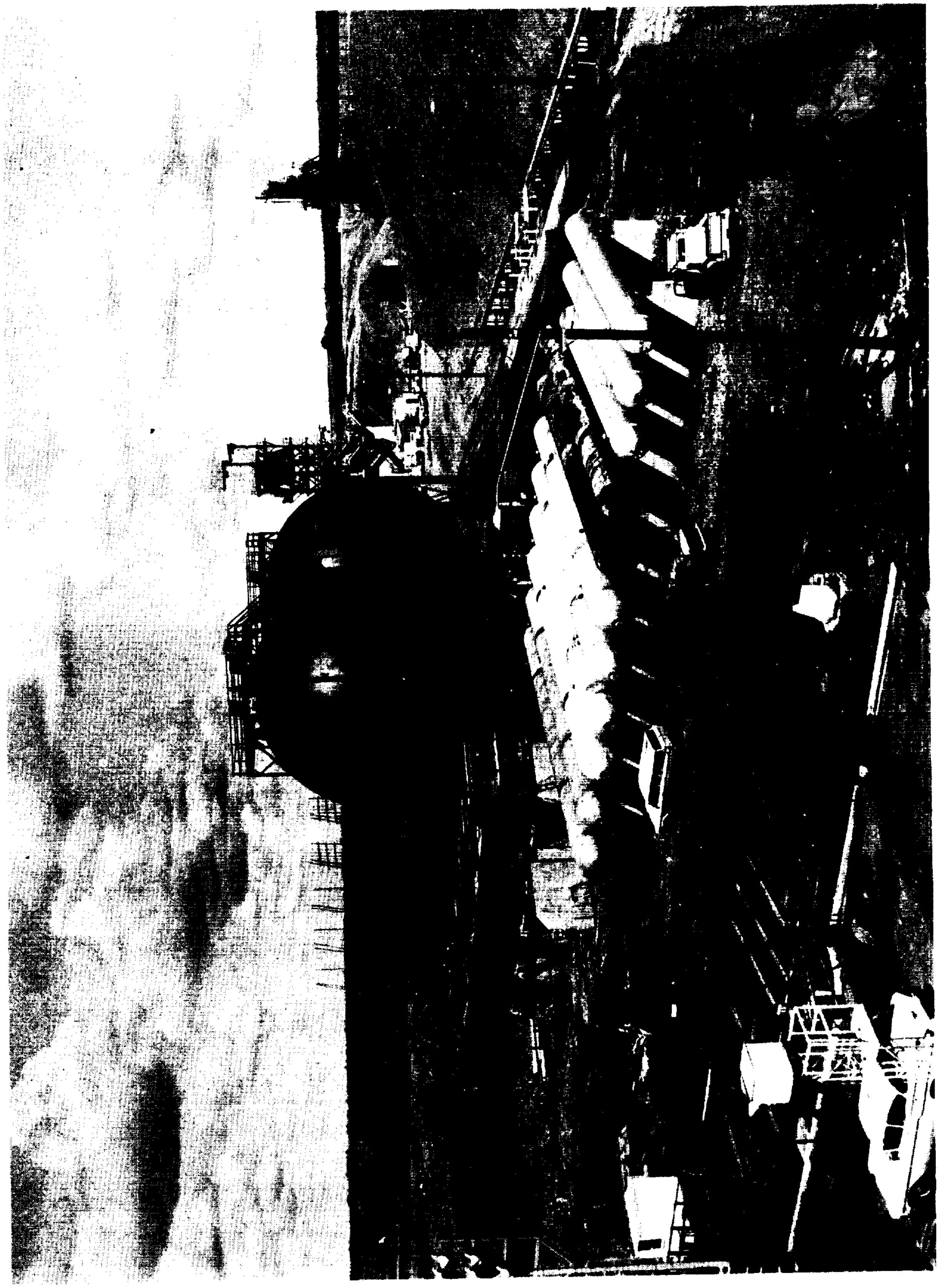

Figure 48. High Pressure Gas Receivers (Zone E)

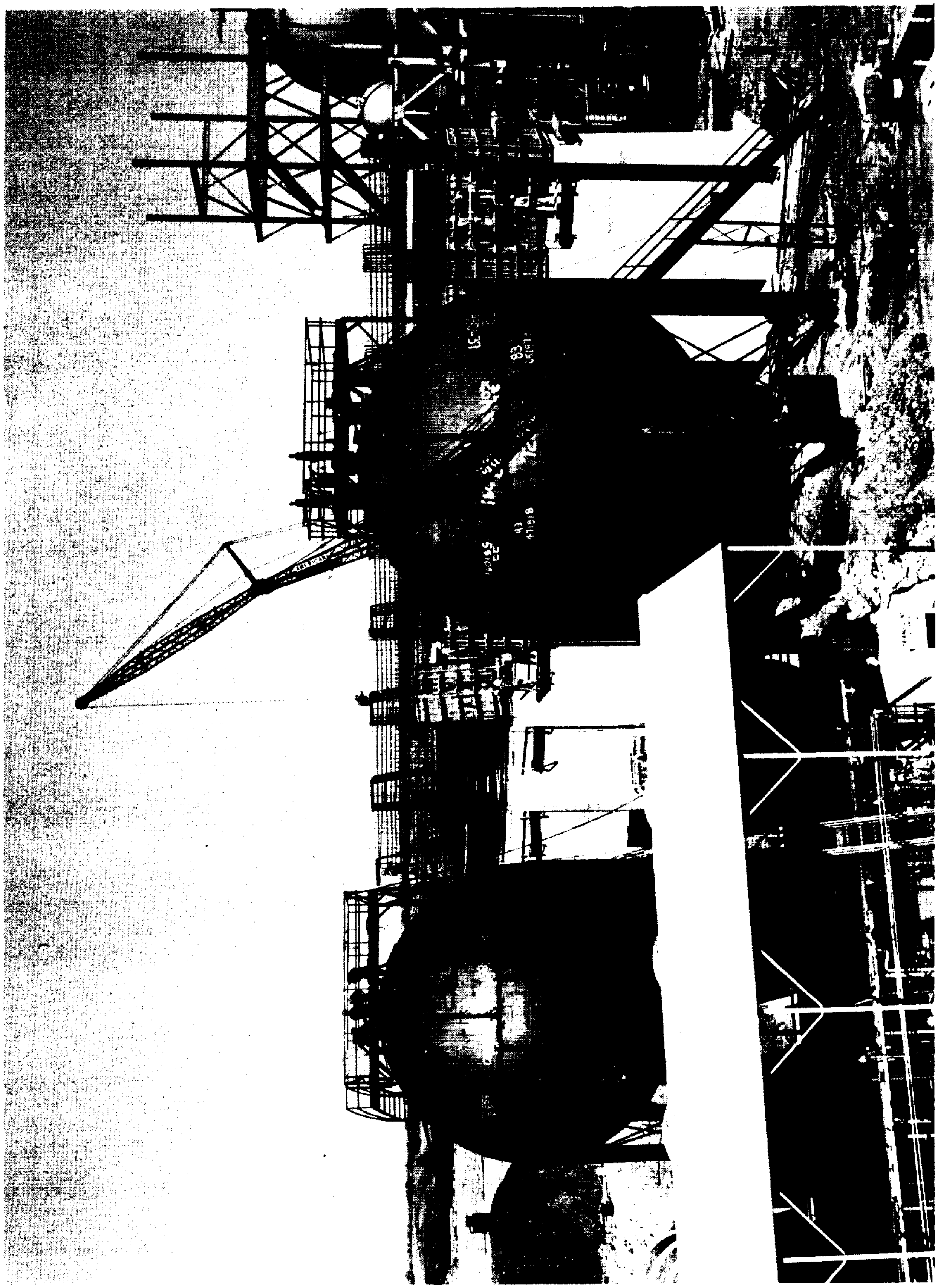

Figure 50. Blast Wall for Liquid Oxygen Vessels (Zone E)

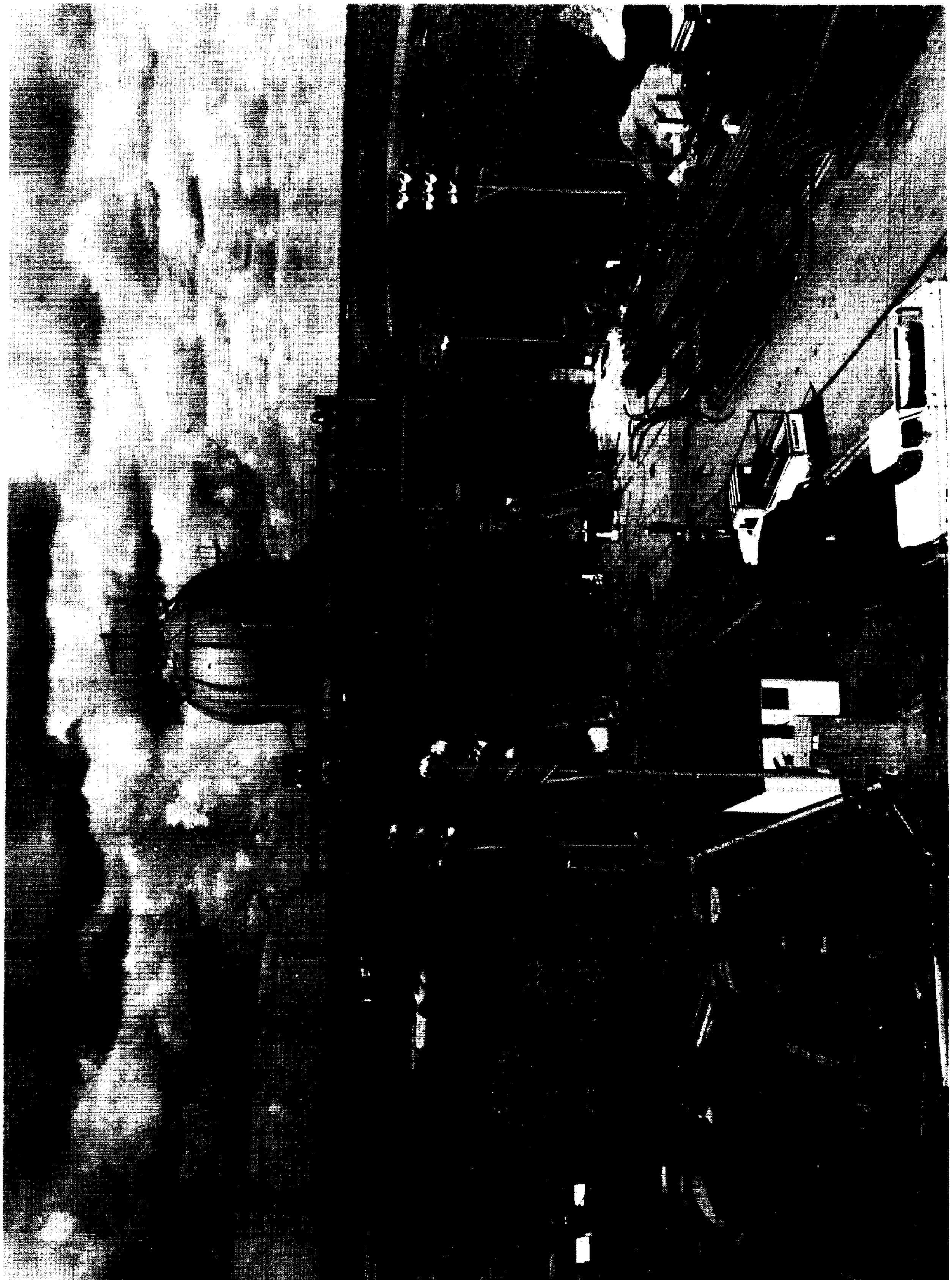

Figure 51. Test Stands E-1 and E-3

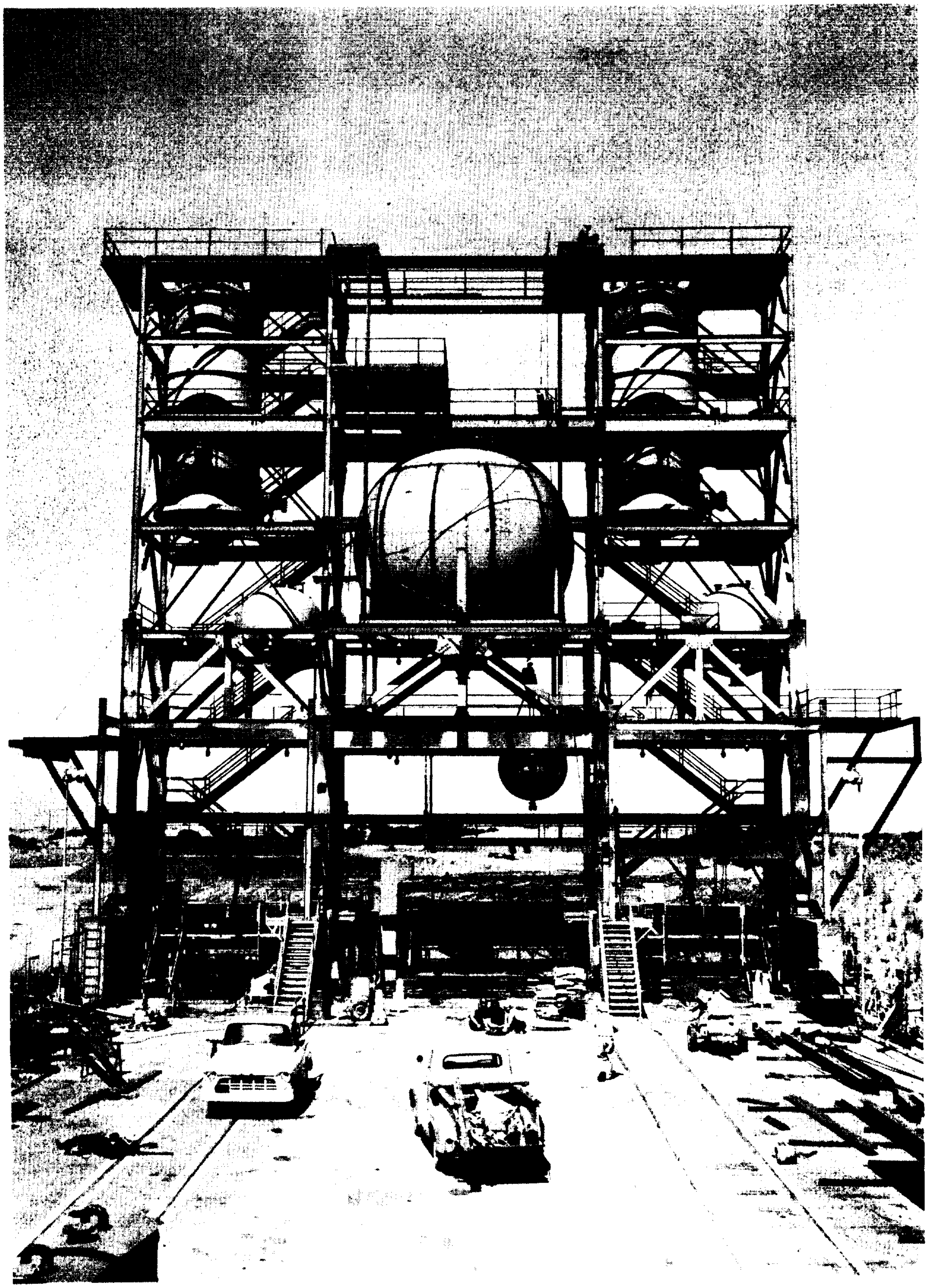

Figure 52. Start Transient and Gas Generator Propellant Vessels Installed on Test Stands E-1 and E-3

1963/3Q

A contract was awarded during the third quarter of 1963 for the installation of the flare stack supports. Three flare stacks for Test Stands E-1 and E-3 were installed. These stacks were located on the north blast wall (see Figure No. 53) and another flare stack was erected for propellant vessel VE-24 (see Figure No. 54).

During this same quarter, the installation of Perlite insulation in vessels VE-1, VE-2, and VE-24 was completed. Installation of vacuum pumps and vacuum systems for all Zone E vessels was completed and the systems were made operational. The installation of the liquid hydrogen vacuum-jacketed transfer piping was also started (see Figure No. 55) and the liquid hydrogen rail tank car unloading stations were installed (see Figure No. 56).

The instrumentation and control package contract (Specification 6546) was awarded on 1 July 1963 and work commenced immediately.

1963/4Q

By the end of 1963, risers had been installed for the catch tanks on vessels VE-1 and VE-2, the vent line from VE-2 to the triple flare stacks was completed, and a gaseous nitrogen header was installed from the nitrogen storage area to VE-10. Also, cleaning of the high pressure nitrogen vessels in the cascade area had been completed.

Sand blasting and prime coat painting of the test stand was accomplished during the fourth quarter of 1963. Also, deluge risers and valves were installed on the south side of the test stand. The upper three levels of test stand lighting was completed and work platforms, blast screens, and hoists were installed on both stands. The terminal room pressurization system was installed and water calibration tests of the liquid hydrogen start transient tanks were started.

1964/1Q

Installation of the turbopump assembly exhaust system was started in March 1964. During that same month, the Hammel-Dahl flow control valves and the 18-in. ball valves as well as the purge and actuation systems were installed in the gas storage cascade area. A contract for the construction clean-up package (Specification 6819) was also awarded.

1964/2Q

The installation of equipment and components (Specification 6367) was completed during the second quarter of 1964 except for the foam insulation of some components in the liquid hydrogen lines, which would be accomplished during activation. This package included the installation of hydrogen flare stacks, diffuser vents, and vacuum-jacketed piping.

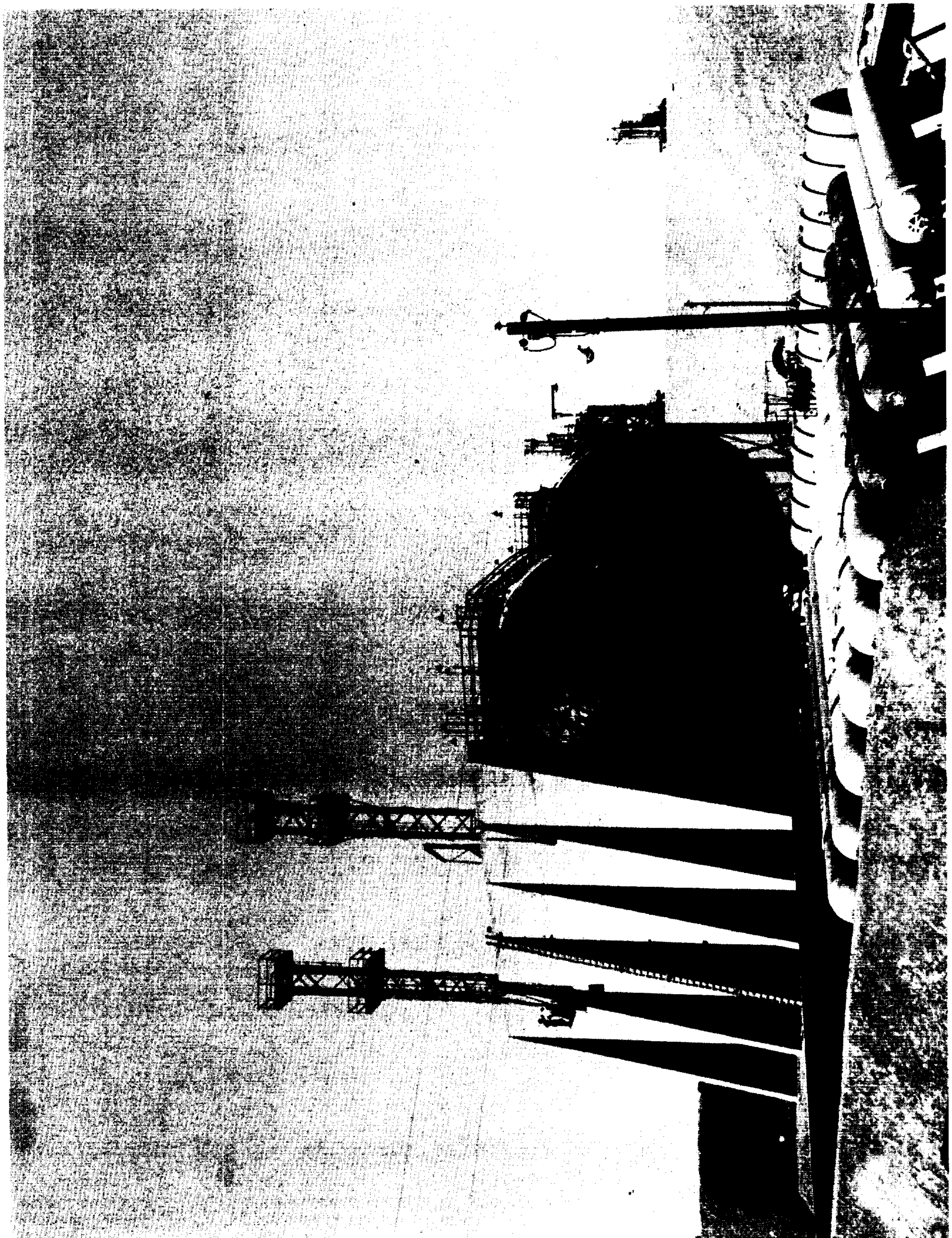

Figure 53. Liquid Hydrogen Storage Area (Zone E)

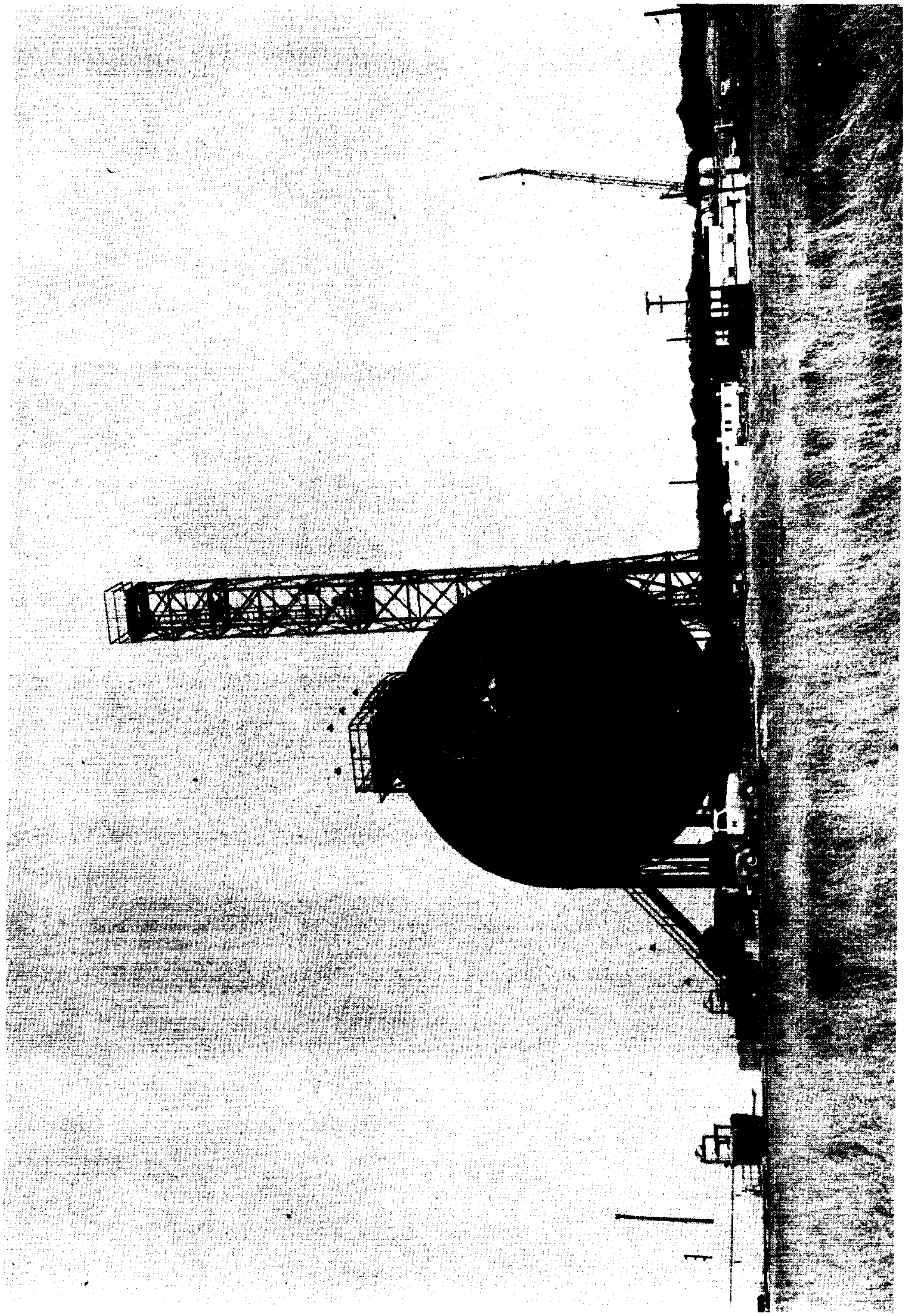

Figure 54. Liquid Hydrogen Storage Vessel VE-24 (Zone E)

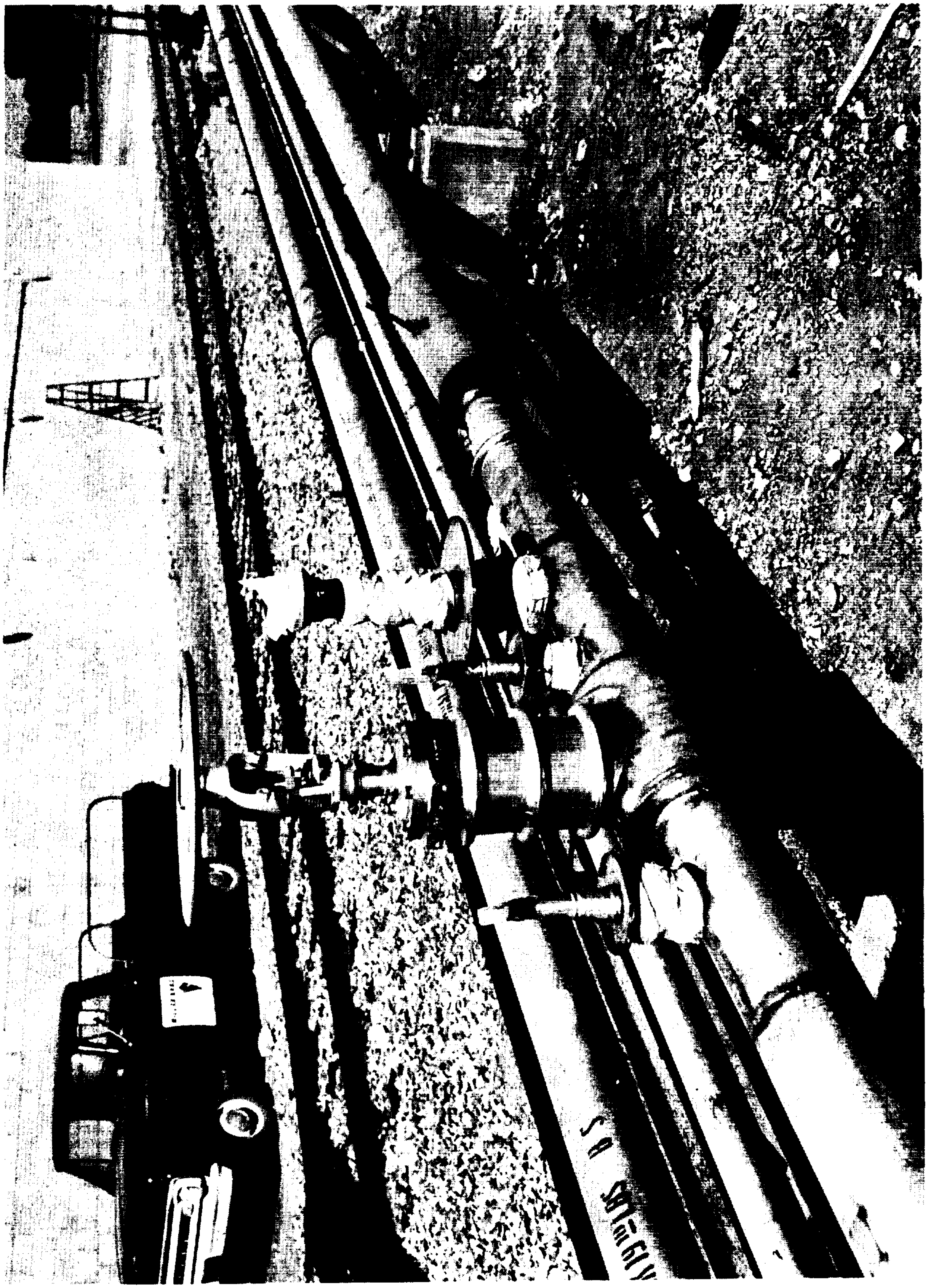

Figure 55. Liquid Hydrogen Vacuum-Jacketed Piping (Zone E)

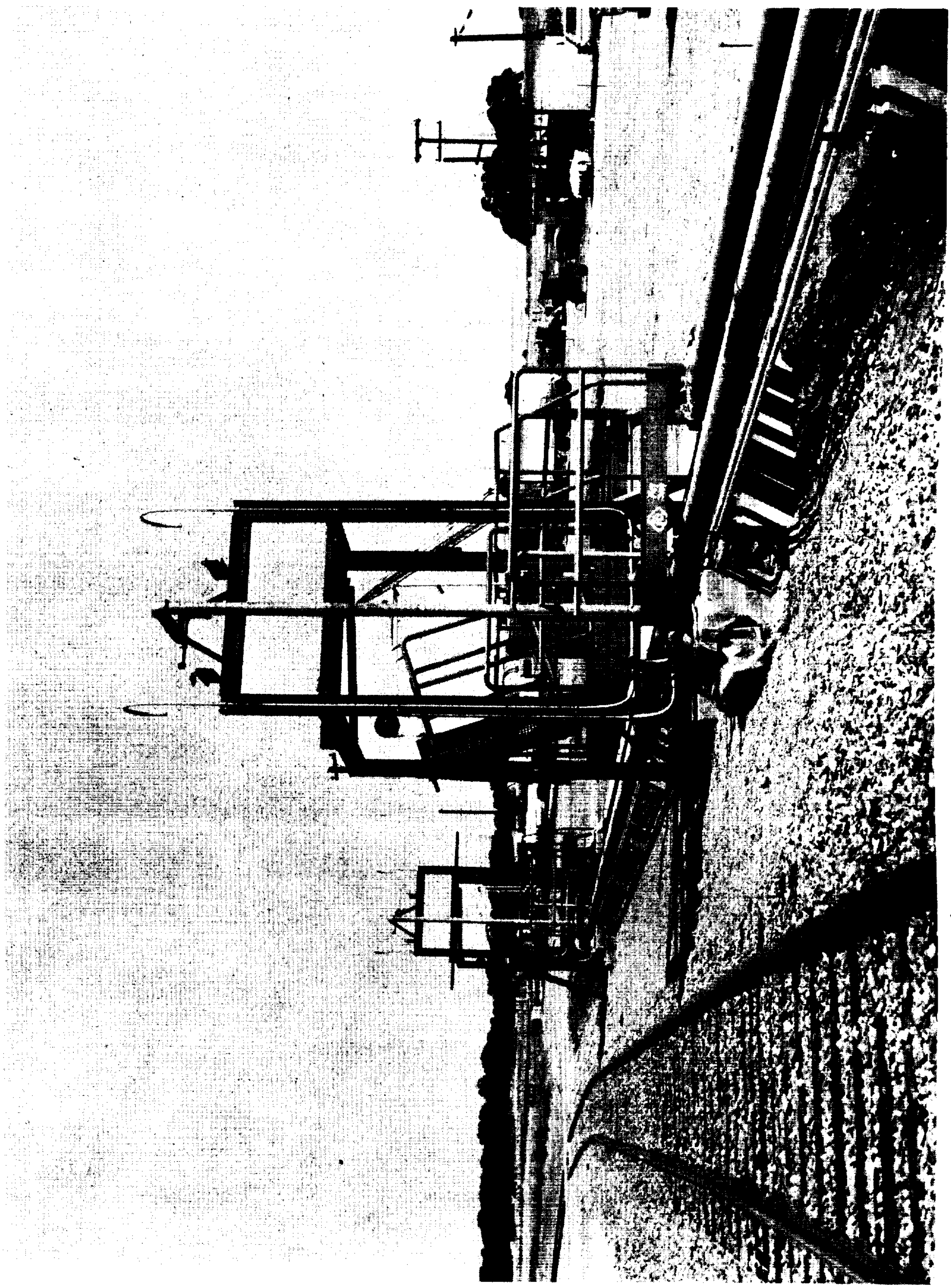

Figure 56. Liquid Hydrogen Railroad Unloading Stations (Zone E)

1964/2Q

Installation of the piping and structural portions of Specification 6778 permitted completion of final leak checks, turbopump assembly hoist testing, pipe identification, and installation of the turbopump assembly exhaust support system. In addition, the painting of the vacuum-jacketed vessels and the fire-proofing of the support columns of the gaseous hydrogen vacuum-jacketed vessels was completed during this quarter.

Installation of miscellaneous structural work, supports and anchors, cool-down piping and deluge tie in (Specification 6819) was also completed except for the cool-down piping, (137) which was to be installed as part of a later contract during activation. Installation of the electro-hydraulic system package (Specification 6779) was completed.

The cleaning and leak checking of the existing liquid oxygen storage area was completed during this same quarter.

Installation of the electrical/instrumentation package (Specification 6546 was completed by the end of June 1964, including the flare stack ignition system. The test stand was turned over to operations personnel.

Mechanical activation of Test Stand E-3 and partial activation of Test Stand E-1 began on 15 June 1964.

A contract was awarded on 29 June 1964 to perform pre-activation work appropriate to trades labor assignment, including the clean-up of those items not included in previous subcontracts. This contract proceeded concurrently with activation.

1964/3Q

The hydraulic control system was activated during the third quarter of 1964. Also, the thrust chamber simulator and oxidizer turbopump housing were installed in the stand with mockup suction and discharge lines. The pump housing was then removed from the stand and returned to fabrication for completion of the turbopump assembly.

The pre-activation construction contract was completed by the end of September 1964. Figure No. 57 shows Test Stands E-1 and E-3 as they appeared at that time.

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(137) Commander, J. C. and Schwartz, M. H., <u>Cool-Down of Large Diameter Liquid Hydrogen and Liquid Oxygen Lines</u>, NASA Report No. CR 54809, 20 April 1966

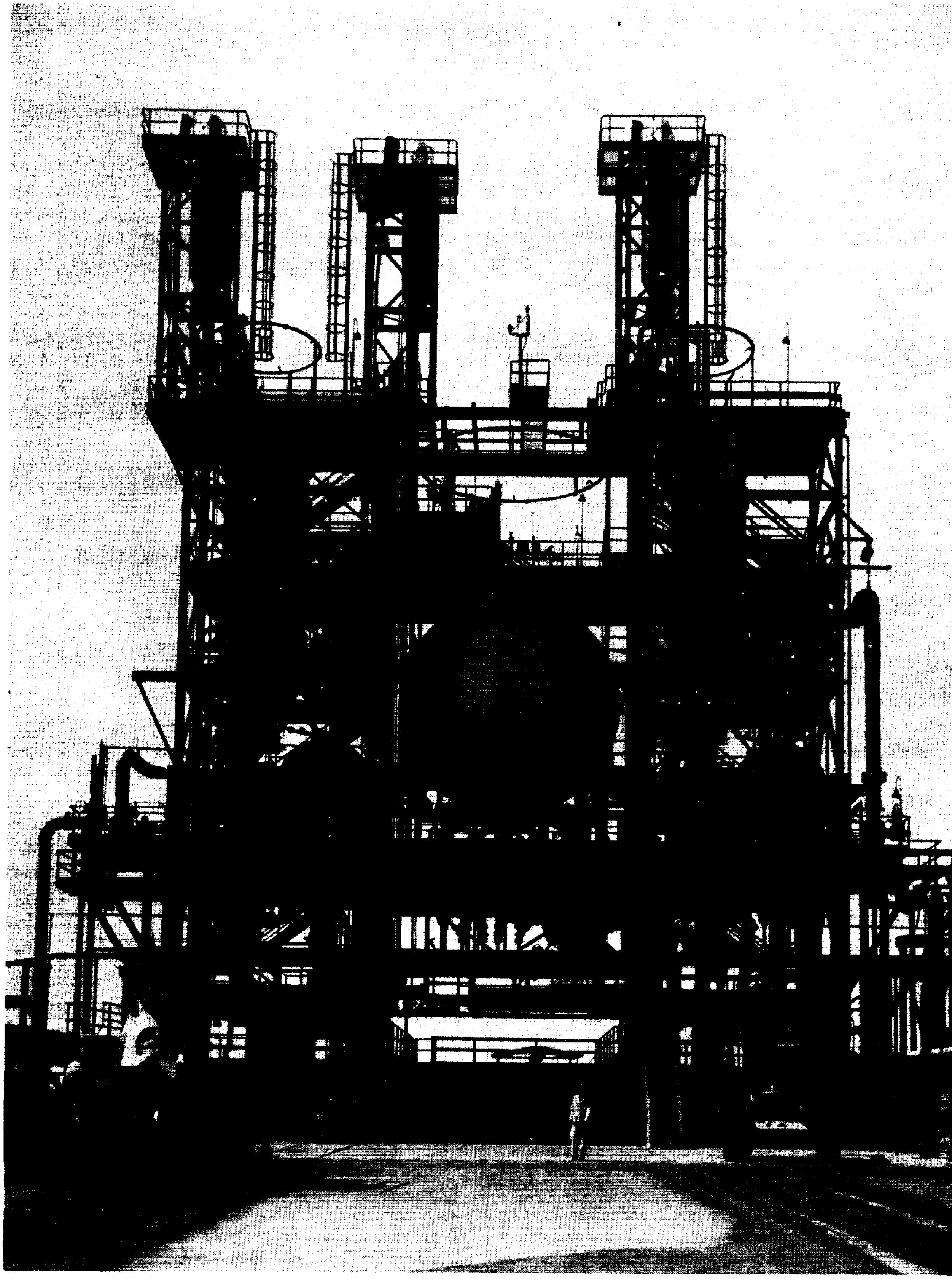

Figure 57. Test Stands E-1 and E-3, Third Quarter, 1964

Mechanical, electrical, and instrumentation activation of Test Stands E-1 and E-3 facilities continued throughout the fourth quarter of 1964. Major emphasis was placed upon completing the cleaning contract and upon the installation of valve and propellant line tie-downs and supports. Test Stand E-3 was also being prepared for oxidizer turbopump testing. In this connection, the dehydration of the liquid oxygen flow system as well as initial cooldown of the Test Stand E-3 liquid oxygen start transient vessel and the 18-in. catch line to the off-stand catch vessel, VE-11, were completed. The large run and catch vessels, VE-10 and VE-11, were dehydrated, chilled-down, and loaded with liquid nitrogen. A preliminary liquid nitrogen facility flow test was conducted to evaluate system components and operability. Also, a contract for the fabrication and installation of supports and tie-downs for major liquid oxygen systems was awarded. Installation of the gaseous nitrogen start by-pass system was completed and both the on-stand and off-stand facilities associated with the first liquid nitrogen/gaseous nitrogen spin tests were checked out with gaseous nitrogen during the initial system chilldown.

The Model I oxidizer turbopump assembly was delivered to the test stand in December 1964. Installation began as shown on Figure No. 58 in preparation for the first gaseous nitrogen drive test.

During the same quarter, activation of Test Stand E-1 for fuel turbopump assembly testing was also under way. The mock-up of the thrust chamber assembly simulator was completed. Activation of the liquid hydrogen auxiliary truck unloading station was initiated concurrent with activation of the liquid hydrogen off-stand catch and storage vessel, the E-1 liquid hydrogen start transient vessel, the 18-in. catch lines, and the liquid-to-gas converters. At the end of December 1964, the entire system and the tanks were dehydrated, the annular space on each tank was evacuated, and preparation for system purging was started.

Activation of the gas generator assembly tanks and propellant systems had also been started. The liquid hydrogen and liquid oxygen gas generator run tanks were both dehydrated and the gas generator liquid hydrogen tank annulus was evacuated. Overpressure switches and liquid level probe electronic package purges were installed. Liquid level probe and meter calibrations for the off-stand liquid oxygen catch vessel and the on-stand E-3 liquid oxygen start transient tank were completed. Initial transducer and cabling installation on the test hardware as well as controller circuitry to the liquid oxygen system Hammel-Dahl pressure regulating valves and liquid flow control valves were completed. Pressure/flow control was successfully demonstrated during the facility liquid nitrogen flow test.

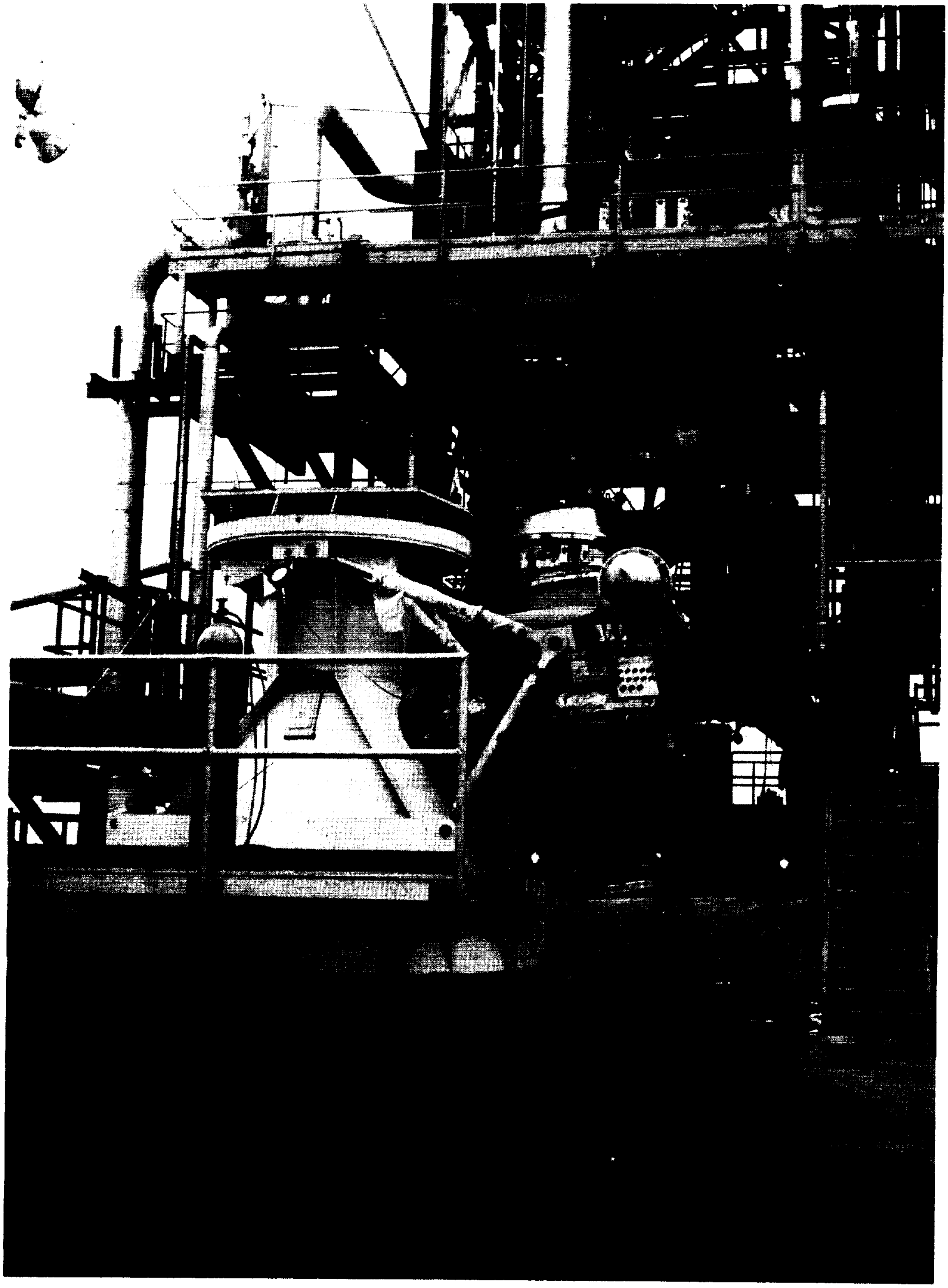

Figure 58. Model I Oxidizer Turbopump Assembly Being Installed into Test Stand E-1

Activation of Test Stand E-3 for oxidizer turbopump testing was completed during the first quarter of 1965. The turbopump assembly was installed and a series of ten tests of the oxidizer turbopump assembly utilizing gaseous nitrogen turbine drive was successfully completed. The turbopump was then removed from the stand and the activation of Test Stand E-3 for hot gas generator testing continued. The exhaust carry-off duct was moved from the right to left hand side of the E-3 position to permit alignment with the gas generator. Fabrication and installation of the gas generator-to-exhaust-duct adapter spool were completed. Also, fabrication and installation of additional valve supports were completed. By the end of the quarter, the installation of gas generator S/N 017A for exhaust duct mock-up and the installation of the interim gas generator valves for propellant feed line mock-up were completed.

The gas generator liquid oxygen run vessel (VE-12) and the liquid hydrogen run vessel (VE-9) were subjected to initial chilldown with liquid nitrogen and then leak and proof tested under cryogenic temperatures.

The gas generator valve Freon actuation system was removed from Test Stand H-8 and installed in Test Stand E-3. The entire system was then successfully leak checked.

Mechanical, electrical, and instrumentation activation at Test Stand E-1 and support facilities for fuel turbopump assembly testing continued during this quarter. Activation of the liquid hydrogen auxiliary truck unloading station with liquid nitrogen was completed. Checkout of the system control valves as well as the liquid hydrogen makeup and discharge flow control valves was completed.

The Model I fuel turbopump assembly was received and installed in Test Stand E-1 during March 1965. Field fitting of the 24-in. exhaust duct system to the fuel turbopump assembly turbine was completed. Final field fitting and welding of the fuel turbopump assembly suction and discharge lines were also completed. Blowdown tests of the gaseous nitrogen drive system, bypassing the fuel turbopump assembly, were successfully accomplished. A gaseous helium purge system for the liquid hydrogen storage and unloading areas was installed. Checkout of the off-stand gaseous hydrogen flare stacks with gaseous hydrogen was completed and the liquid hydrogen flowmeter was installed.

Liquid hydrogen was loaded into VE-34, the liquid-to-gas converter storage vessel, and activation of the hydrogen liquid-to-gas converters was started in preparation for pump-up of the gaseous hydrogen receivers.

The gaseous nitrogen turbine drive system for the first series of fuel turbopump assembly tests was installed and the interface plumbing for the fuel turbopump assembly purge systems was completed.

1965/2Q

Activation of Test Stand E-1 for fuel turbopump assembly testing was completed and a series of ten successful gaseous nitrogen turbine drive pump tests was conducted in May 1965. Preparations for gas generator assembly testing were then begun with the installation of mock-up gas generator S/N 0017A. Fabrication and installation of gas generator assembly propellant lines, restraints, and purge panel interfaces were also started. The activation of Test Stand E-3 for gas generator assembly testing culminated in a series of nine successful gas generator tests during the second quarter of 1965.

Installation of the liquid level point system in the on-stand liquid hydrogen start transient vessel, VE-21, was also completed during this second quarter of 1965.

1965/3Q

Facility activation of Test Stand E-1 for gas generator testing continued throughout the third quarter of 1965.

Significant accomplishments at Test Stand E-1 included fabrication and installation of the gas generator propellant feed lines and associated supports/restraints as well as the gaseous nitrogen start line and turbine simulator. The installation and checkout of the gas generator valve hydraulic control system were completed. Following removal of the mock-up generator, S/N 017A, gas generator S/N 026 with prototype valves was installed. Cryogenic leak and proof tests of propellant systems were then conducted. Gas generator assembly S/N 026 was replaced with gas generator S/N 025 and a successful liquid nitrogen flow test of the gas generator assembly system was conducted.

After a series of gas generator checkout tests at Test Stand E-3, the oxidizer turbopump was reinstalled and a series of gaseous nitrogen turbine drive pump tests were conducted. The initial test of the oxidizer turbopump assembly, utilizing the gas generator for turbine drive was conducted on 11 August 1965. After a series of five tests, two additional gaseous nitrogen turbine drive tests were conducted. Gas generator assembly S/N 022 was then removed and gas generator S/N 025 installed on 8 September 1965. An oxidizer pump hot-turbine drive test was then successfully conducted on 15 September. Gas generator S/N 025 was then replaced with gas generator S/N 022 and the final oxidizer turbopump assembly test conducted on 24 September 1965. However, several modifications were accomplished prior to the initial oxidizer turbopump hot drive tests. This included fabrication and installation of a modified oxidizer turbopump turbine exhaust cone bellows and a 24-in. hot gas duct to carry off the gas generator exhaust during actuation of the quick-dump blow-off cap.

On 30 August 1965, the off-stand liquid hydrogen storage vessel for the liquid-to-gas converters (VE-34) was found to have a crack in the outer shell around the vent-tree nozzle. Repairs were accomplished and 0.312-in. thick doublers were installed over the affected areas. Additional spring hangers and braces were installed to support the piping connected to the vessel nozzle..

1965/3Q

By the end of September 1965, the liquid level point system was installed in the on-stand liquid oxygen start transient vessel, VE-32, and installation as well as checkout of the EAI PC-12 analog computer was completed. The infrared television system was also installed and checked out.

1965/4Q

Contractor effort involving modifications to Test Stand E-1 for hot drive fuel turbopump assembly tests was completed early in the fourth quarter of 1965. Major items included modification of an existing 24-in. facility exhaust bellows; support of the fuel turbopump exhaust bellows; fabrication and installation of a 24-in. blowoff duct; and relocation and support of an existing 10-in. hot gas bypass line. Subsequent to this effort, installation of check valves in two of the flare stack feed lines was completed. The oxidizer turbopump assembly was removed from Test Stand E-3 and the stand was then placed in a standby status.

Reactivation of the liquid-to-gas converter liquid hydrogen storage vessel, VE-34, was completed and the vessel placed back in service during October 1965. However, on 15 November 1965, an investigation of vacuum loss in the annulus revealed a recurrence of the crack in the carbon steel nozzle closure and adjacent shell of the vacuum jacket. The vessel was then warmed and purged in preparation for a thorough investigation of the problem. Through the remainder of the test program, the liquid-to-gas converters were supplied with hydrogen by cross-over lines from the liquid hydrogen catch vessel, VE-2, and on-stand gas generator run vessel, VE-9.

The facility activation of Test Stand E-1 for gas generator testing was successfully completed in October 1965 and a series of gaseous nitrogen drive fuel turbopump assembly tests were conducted in November 1965. The initial hot gas generator drive fuel turbopump assembly test was successfully completed on 6 December 1965 after installation of a hydrogen bleed bypass system for fuel turbopump assembly thrust balance control. Three additional fuel turbopump assembly tests were conducted with the last one completed on 22 December 1965. Facility de-activation was then started and plans for early transfer of liquid hydrogen inventory in VE-2 to VH-1 in H-Zone were prepared.

1966

Zone E facility deactivation was completed during the first quarter of 1966. It was then maintained in a condition of short-term, standby preservation.

In the second quarter of 1966, appropriate documents were being processed to transfer Test Stands E-1 and E-3 to the NERVA Program for turbopump testing.

Because of the indicated reduction of scope in the M-1 Program, the full capability of Zone E was not demonstrated. Specifically, the long duration run capability utilizing the off-stand propellant run vessels and the combined oxidizer turbopump assembly-fuel turbopump assembly capability in each stand. Basic facility capability for providing efficient turbopump assembly testing was amply demonstrated during the short duration, cold gas and gas generator drive tests of the oxidizer turbopump assembly at Test Stand E-3 and the fuel turbopump assembly at Test Stand E-1. Of particular significance in this connection was the successful operation of the automated servo-control valve-tank pressurization and back-pressure control systems.

1962

D. TEST ZONE H

The structural and mechanical systems as well as the instrumentation and control systems for Test Stand H-8 were designed by the end of the second quarter of 1962. The liquid oxygen propellant vessel VH-71 was released for procurement. Procurement of the long-lead liquid hydrogen run vessel VH-70 was also initiated.

During this same quarter, the Aerojet-General-funded civil works modification to Test Stand H-8 was under way. The concrete thrust-takeout structure had been poured. Also, the Aerojet-General-funded liquid hydrogen storage and support facilities, which would be shared by both the M-1 Program and the NERVA Program, were in the advanced stages of construction.

The civil works contract (Specification 6477) was awarded during the third quarter of 1962 and field erection of vessel VH-70 had been started as shown on Figure No. 59.

The concrete piers for the traveling crane, the footings for the gas receivers, the concrete blast wash apron, the safety shower pad, and the camera pads were poured. The mechanical package (Specification 6369) was released by the end of 1962. Also, three high-pressure gas receivers had been delivered. These receivers were set on foundations during the ensuing quarter, during which time the mechanical installation package (Specification 6369) was also awarded.

1963

Steel thrust plates were installed in the thrust wall during the first quarter of 1963 and the remaining portion of the concrete thrust wall was poured. Also, concrete was poured for the cable tray supports, purge panel pads, gas generator assembly exhaust stack pad, and drain line pipe supports. This work essentially completed the civil works site preparation contract (Specification 6477).

The liquid hydrogen run vessel (VH-70) was successfully proof and leak tested. The annular space was vacuum-tested and filled with perlite insulation and the instrumentation installation package (Specification 6541) was released during this same time period.

The liquid oxygen run vessel VH-71 was received and installed during the second quarter of 1963. The liquid hydrogen run vessel VH-70 was completed, proof tested, and chemically cleaned (see Figure No. 60). Also, the civil works (Specification 6477) were completed.

Erection of a steel superstructure began for valve platforms over the propellant vessel and the crane way footings were enlarged to accommodate

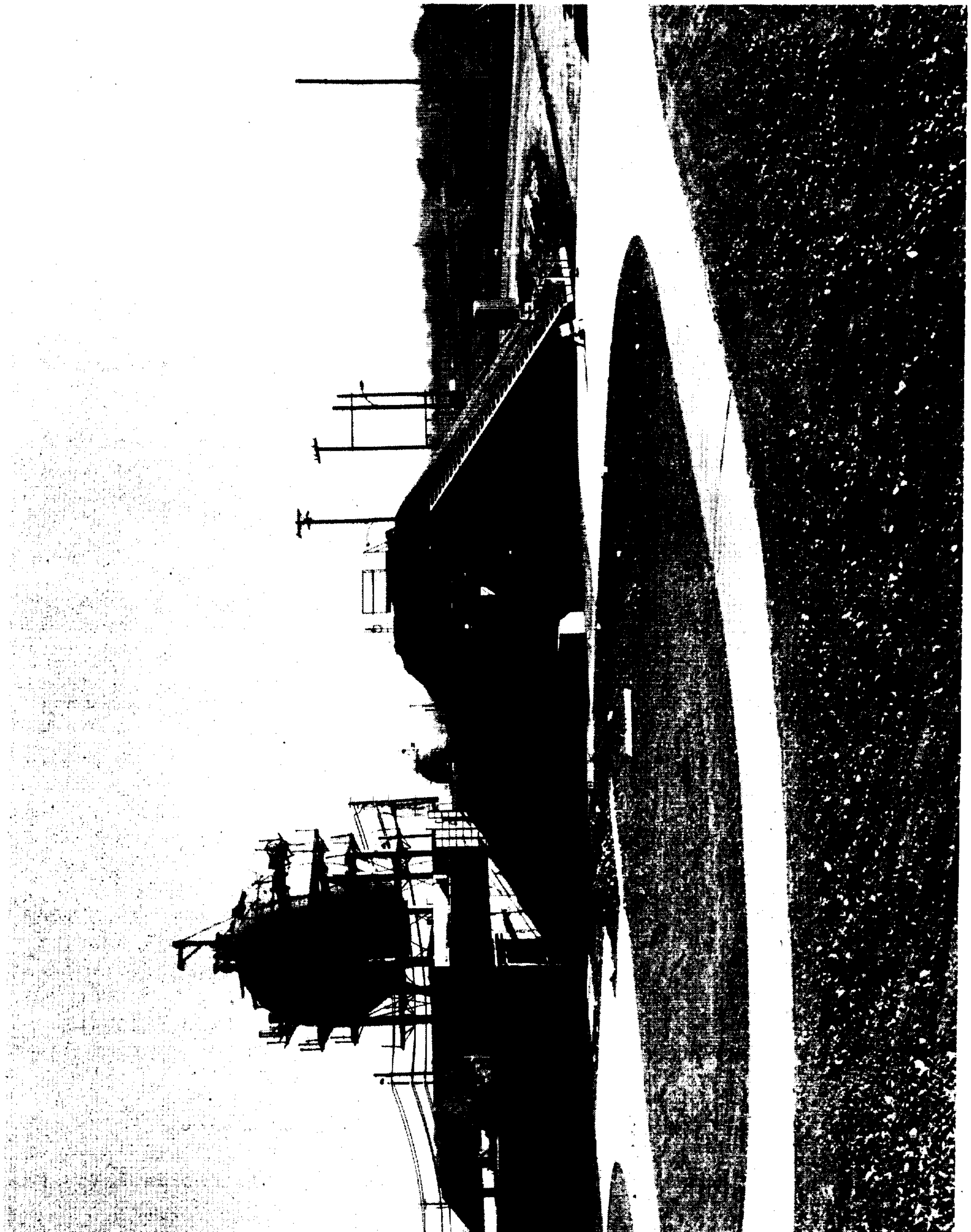

Figure 59. Liquid Hydrogen Run Vessel VE-70 (Zone H)

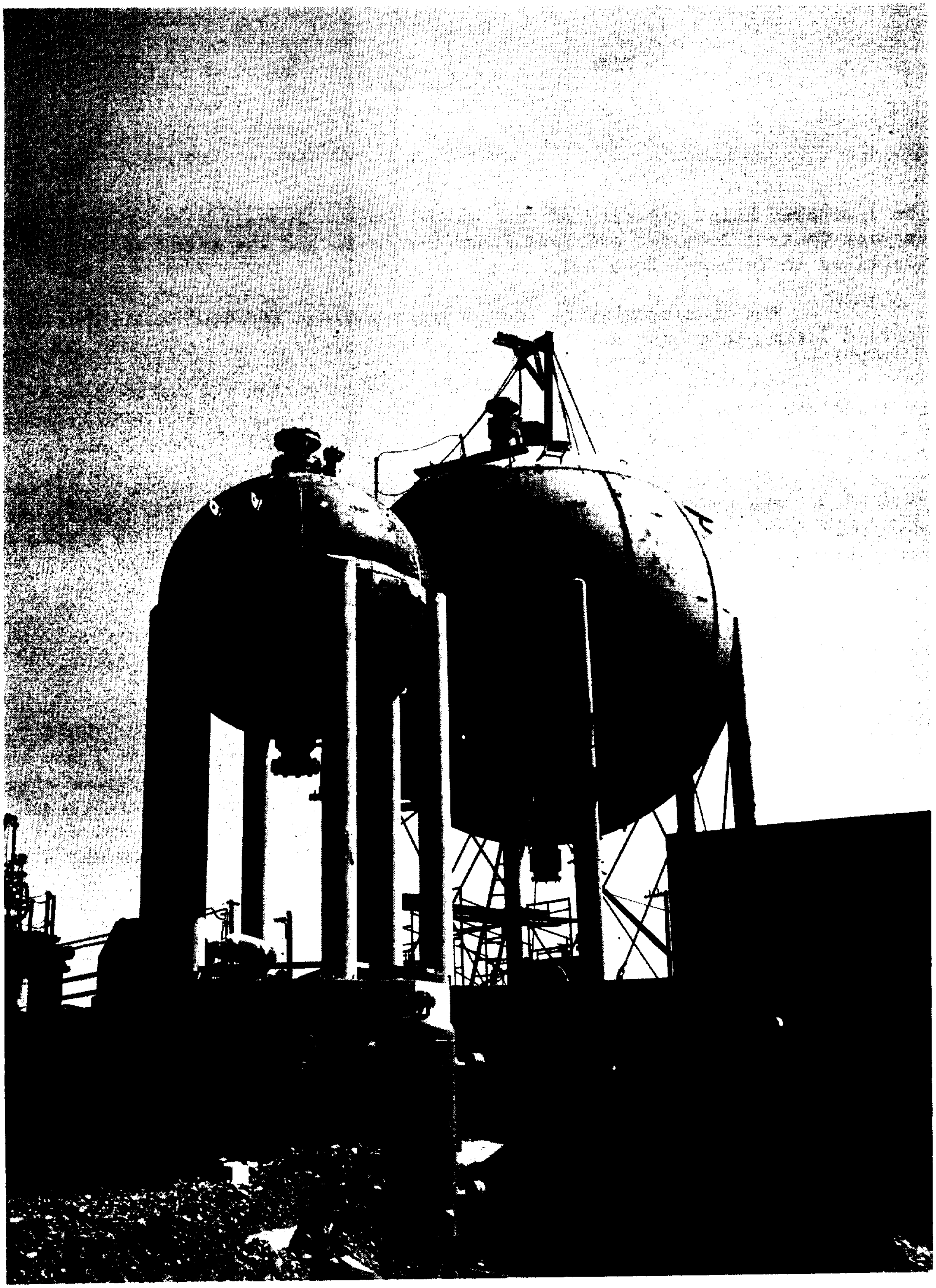

Figure 60. Liquid Oxygen (VH-71) (left) and Liquid Hydrogen (VE-70) (right) Run Vessels (Test Stand H-8)

the increased hoist capacity of the bridge crane. Installation of cable trays between Control Room H-1 and Test Stand H-8 began and the terminal racks were installed in Terminal Room H-8.

The instrumentation and controls package for Test Stand H-8 was awarded during this quarter.

The deluge booster pump was installed during the third quarter of 1963 and the piping was connected. Also, the two bridge cranes were received and installed on the support structure. The new data patch system was installed.

The structural portion of the thrust measurement system(138) was completed during the fourth quarter of 1963. Also, the new digital system for the H-1 control room was installed.

1964/1Q

The hydraulic calibration cylinder received during February 1964 was installed during March. The liquid level probes were also installed in the liquid oxygen and liquid hydrogen run tanks.

1964/2Q

Initial construction of Test Stand H-8 was completed on 1 May 1964 with the exceptions of the piping system tie-in to the logistic systems shared with the NERVA Program, and the addition of several area warning lights as well as to complete control console wiring. The design, bid, and award of the gaseous nitrogen pressurization system and vent system modification was also completed.

The activation of Test Stand H-8 began on 4 May 1964. The turbine back-pressure simulator required for gas generator assembly firing was installed. Also, modification and checkout of the gas generator assembly deluge system as well as checkout of the thrust chamber assembly and propellant tanks deluge system were completed.

Figures No. 61 and No. 62 are front and side views of the completed stand.

Tie-in to the gaseous nitrogen and gaseous hydrogen logistics systems shared with the NERVA Program was completed during the third quarter of 1964. The gas generator assembly, gas generator valves, and propellant spools were received and installed along with the actuation system for the interim gas generator valves.

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(138) Hoy, W. A., Close, J. R., Vernon, K. A., 1.5 Million Pound Load Cell Calibration and H-Area Thrust Measuring System, NASA Report No. CR 54793, 27 June 1966

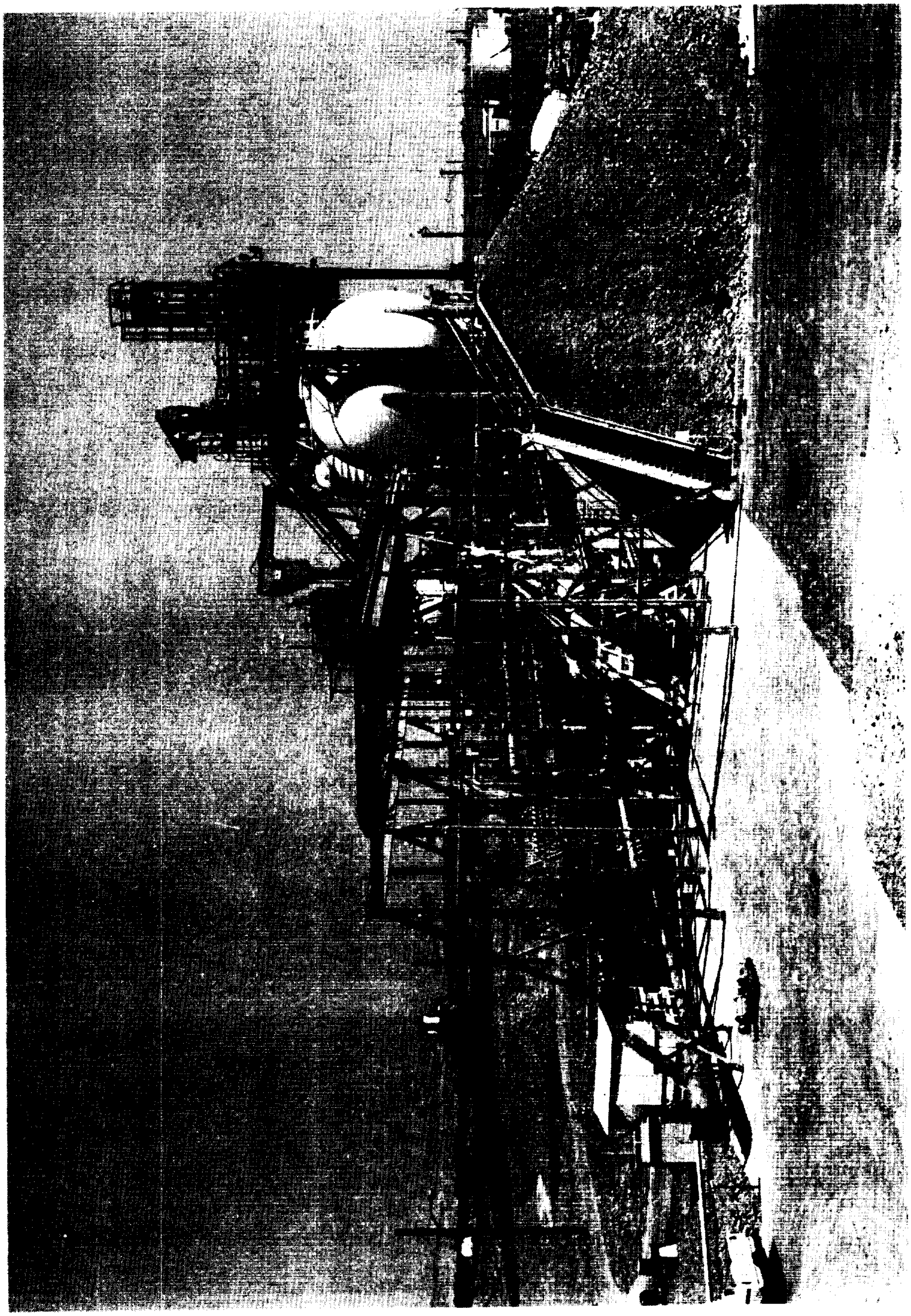

Figure 61. Test Stand H-8 (Front View)

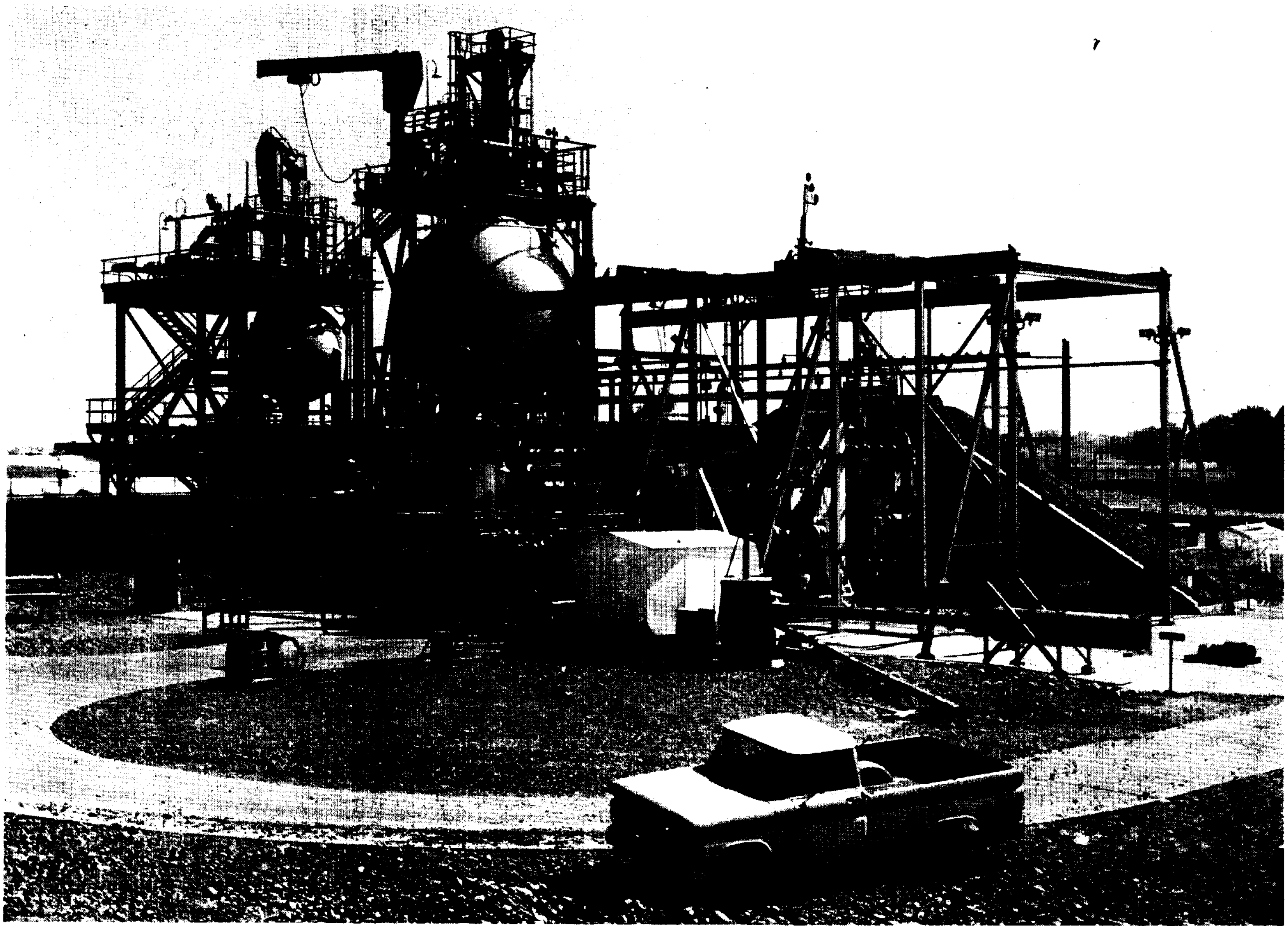

Figure 62. Test Stand H-8 (Side View)

1964/3Q

The gas generator assembly purge and leak check systems as well as additional gaseous nitrogen line anchors were installed. Vent system modifications were completed and the contractor started repair of a leak in the gaseous hydrogen receiver, VH-23.

Cleaning of gaseous nitrogen and liquid oxygen systems as well as installation of the instrumentation on the gas generator and turbine simulator were completed.

Activation of the area warning system was completed and the liquid oxygen gas generator assembly run line restraints were installed. Modifications to the liquid oxygen fill line tie-downs were completed.

1964/4Q

Activation for gas generator testing culminated in a successful gas generator firing on 14 November 1964. Prior to this test, the gas generator valve actuation system was converted to utilize Freon FT rather than RP-1 as the actuation fluid.

The repair of the gaseous hydrogen receiver, VH-73, was completed by the contractor and the vessel was placed back in service during the fourth quarter of 1964.

Gas generator testing continued after the first test and eight successful gas generator hot firings were made. During this series of gas generator tests, the bonnet-to-body seal and packing failed on the liquid hydrogen tank fill valve, the gas generator liquid hydrogen safety valve, and the liquid hydrogen gas generator high point bleed valve. Necessary repairs were made.

Design effort for facility modifications(139) for uncooled thrust chamber testing was completed and the request for bids was issued on 8 December 1964.

1965/1Q

Gas generator testing continued during January and February of 1965.

A contract was awarded for facility modifications to accommodate uncooled thrust chamber assembly testing and on-stand construction effort was initiated on 1 March 1965. This included the awarding of a contract to the Annin Company for fabrication of special hydraulic facility thrust chamber valve actuators. The off-site fabrication of the liquid hydrogen and liquid oxygen propellant run lines was completed. Concrete work for the facility

(139) Vernon, K. A., Test Stand H-8 Systems Analysis and Design, Aerojet-General Report No. 8800-73, 6 May 1966

thrust chamber valve supports, helium receiver foundation, and relocated hydraulic pump was completed. Fabrication of an improved performance gaseous nitrogen diffuser for the liquid oxygen run vessels (VH-71) was completed.

Conversion of the gas generator test set-up to use gaseous-helium-actuated prototype gas generator valves (for actual engine use, these valves actuated by hydrogen) was completed and four liquid nitrogen cold flow tests were conducted.

Performance of the A. O. Smith 1300-cubic-ft gaseous hydrogen receivers was unsatisfactory because of a succession of cracks and leaks at high pressures.(140) As a result, these receivers were limited to a working pressure of 3500 psi, while 4500 psi is required for the first uncooled thrust chamber assembly test. Technical and contractual discussions were held with A. O. Smith, NASA/LeRC, and others as to the probable cause and corrective action. Because hydrogen embrittlement appeared to be involved, repair or rehabilitation of these receivers on-site did not appear feasible in time for the thrust chamber assembly test. While maximum repair effort continued, a backup plan was developed to use the idle K-Zone Struthers-Wells vessels.

1965/2Q

Contractor effort involving facility modification to accommodate uncooled thrust chamber assembly testing continued through 28 May 1965. The facility thrust chamber valves were installed and VE-213, a helium gas receiver, had been relocated from C-Area to H-Area. Also, the liquid hydrogen and oxygen propellant lines were installed, including the gaseous hydrogen mixer, for temperature control of the liquid hydrogen propellant. The high capacity gaseous nitrogen and helium purge system had been fabricated and installed. A contract for the installation of the three-gimbal propellant line dampers and restraints was awarded.

Design of a liquid point level system for the on-stand liquid hydrogen vessel, VH-70, was completed during the second quarter of 1965. Contractor effort associated with the augmentation of the H-Zone gaseous hydrogen capacity also began. Accomplishments included the movement of a K-Area Struthers-Wells gas receiver, VK-11, to H-Zone to replace VH-74, as well as site preparation and foundations for three additional receivers.

(140) Laws, J. S., Hydrogen Gas Storage Problems, Aerojet-General Report No. 8800-67, 15 April 1966

On 1 June 1965, a serious leak developed in the newly-relocated gaseous hydrogen receiver, VK-11. This leak occurred at 3,900 psi during one of the initial pressurization cycles with gaseous hydrogen. A manway was cut into the end of the vessel and subsequent investigation revealed three cracks in the inner wall, in longitudinal seam welds. The cause of this failure was attributed to either one or both of the following: accelerated hydrogen embrittlement; or cracks initiated during fabrication and/or hydrotesting because of faulty welding or other techniques to which the T-1 steel is known to be sensitive.

Further investigation of the T-1 steel, Struthers-Wells vessels was initiated, including the cutting of access ports and inspection of new, unused vessels VK-12 and VK-13. An interim operational plan was prepared which involved moving three E-Zone gaseous hydrogen receivers to H-Zone to supplement that facility.

Activation of the facility for uncooled thrust chamber testing on a full-time beneficial-occupancy basis began on 17 May 1965. This activation effort included: installation of the mock-up thrust chamber assembly in the test stand; purge and instrumentation system checkout; preliminary checkout of the thrust calibration system; activation of the hydraulic system and response testing of pressure regulating valves; fabrication and installation of console lids; and completion of the welding of the three-gimbal liquid oxygen propellant line.

All contractor effort was completed including: augmentation of the Zone H gaseous hydrogen capacity by installation of three high-pressure receivers removed from Zone E; installation of propellant line restraints and dampers; completion of the liquid oxygen run-vessel pressurization system; completion of the 6-in. and 8-in. gaseous hydrogen mixer systems; all system final cleaning and leak check operations; and installation of the facility thrust chamber valve actuation system.

Installation of the liquid point level system in the on-stand liquid hydrogen run vessel, VH-70, was also completed. Modification of the television system for infrared capability was completed and the system checked out. Installation of the PC-12 analog computer was completed.

Activation of the facility for uncooled thrust chamber testing continued. The liquid hydrogen three-gimbal propellant line between the facility thrust chamber valve and the uncooled thrust chamber assembly was fabricated and installed. The facility thrust chamber valve Freon actuation system was activated. All major systems were pneumatically proof and leak checked. The liquid oxygen and liquid hydrogen on-stand run tanks and annulus were dehydrated. A series of ramp rise-rate tests was conducted utilizing liquid nitrogen in both the liquid oxygen and liquid hydrogen on-stand run tanks.

1965/2Q

Flow tests utilizing liquid nitrogen were also initiated and sequence tests of facility valves, thrust chamber valves, and the control system were completed. The gaseous-helium and gaseous nitrogen purge systems were activated.

1965/3Q

Contamination, consisting primarily of ferrous chloride powder, found in the gaseous nitrogen filter in August, necessitated recleaning of the nitrogen pressurization and liquid oxygen propellant systems. The cleaning was satisfactorily completed prior to initiating the ramp rise-rate tests.

1965/4Q

Activation for uncooled thrust chamber assembly testing continued throughout the fourth quarter of 1965. System flow tests were made utilizing liquid nitrogen. Ramp rise tests of the on-stand liquid oxygen and liquid hydrogen vessels were conducted.

The mock-up thrust chamber assembly was removed and the fireable thrust chamber assembly with the 1100 element injector was installed. This was followed by activation of the gaseous helium and nitrogen purge systems, liquid oxygen outflow tests, thrust measurement system static and dynamic calibrations, liquid hydrogen thrust chamber valve functionals at liquid hydrogen temperature, a final ramp rise test on VH-70, the on-stand liquid hydrogen run vessel, and a gaseous hydrogen mixer flow test.

Erratic operation of the facility thrust chamber valves during the initial flow tests in October 1965 was partially corrected by changing the actuator fluid from Freon to Pydraul. This substitution eliminated two significant difficulties experienced with Freon; poor lubrication qualities and inability to properly bleed-in the system because of low Freon vapor pressure. Recurrence of inconsistent valve opening and closing times during subsequent flow tests resulted in rework of the pilot valves. Continued investigation into system bleed characteristics resulted in the addition of several more bleed bosses.

The liquid oxygen propellant line dump valve and burst disc carry-off system failed during a cold flow test in November 1965. This failure was attributed to a pre-test failure of the burst disc which permitted excessive flow through the carry-off system upon closure of the thrust chamber valve. The system was repaired, incorporating safeguards to prevent a recurrence of the problem.

Internal structural failure of a gaseous helium pressure regulator occurred in November 1965 during a helium purge system calibration with nitrogen. The downstream propellant line was then disassembled and inspected for regulator fragments, but a majority of the fragments were apparently in the 1100 element injector thrust chamber assembly torus or injector. As a result, it was decided to conduct the first firing utilizing the 3300 element injector thrust chamber assembly. This eliminated the need for a separate back-flush of the 1100 element thrust chamber assembly. Installation of a redesigned and rebuilt pressure regulator was then completed.

The gaseous hydrogen mixer filter element failed during a mixer flow test in December 1965. Collapse of the element was attributed to excessive pressure drop through the element resulting from clogging by iron oxide. Thorough investigation revealed that the iron oxide came from the carbon steel gaseous hydrogen line immediately upstream of the filter. All sections of the fuel system that were exposed to contamination from the filter element failure were disassembled, thoroughly cleaned, and reassembled. Special attention was given to the carbon steel line to ensure the removal of all iron oxide particles and a new, improved filter element was installed.

The thrust measurement load cell exhibited low internal insulation resistance during checkout on the test stand. This was caused by moisture within the cell as a result of a seal failure. The cell was satisfactorily repaired in the Transducer Laboratory and returned to service. Thrust calibrations conducted with the repaired load cell verified that the thrust measurement system response was quite satisfactory. Calibration tests were run with the propellant lines chilled to liquid nitrogen temperature and pressurized to 1500 psig as well as at ambient temperature and unpressurized. As a result of the excellent data obtained from these tests, no further calibration tests were planned. The stand hysteresis was measured and found to be minimal. As a result, no further static calibrations with dynamic excitation were planned.

Review of the plans and specification for repair of the A. O. Smith high pressure gaseous hydrogen receivers was completed by the end of 1965. Review with the State of California Code Committee and the NASA resulted in a decision to install the static seal compression plugs in place of the welded nozzles.

Tie-in of the up-graded NERVA gaseous hydrogen and gaseous nitrogen systems to the gaseous hydrogen and gaseous nitrogen cascade system was completed during December 1965.

1966/1Q

The first quarter of 1966 was also devoted to activation for uncooled thrust chamber assembly testing. The facility thrust chamber valve actuation system was modified and functional tests were conducted at both ambient and cryogenic temperatures. Valve actuation was verified by means of three low pressure liquid nitrogen flow tests. The valve timing and purge sequencing were verified by conducting a combined systems liquid nitrogen flow test. Injector manifold fill characteristics were evaluated during a liquid oxygen system flow test using liquid oxygen.

The 1100 element thrust chamber assembly was removed and the 3300 element thrust chamber assembly was installed. A flow nozzle and high-pressure shut-off valve were installed in the gaseous helium purge system to replace a helium pressure regulator that had failed previously.

NASA/LeRC had requested a change in the thrust chamber assembly start system and by the end of March 1966, planning and design were completed for the necessary facility modifications to provide a fluorine ignition system for the initial thrust chamber assembly test. The fabrication and installation of the system was also completed. Pre-procured components and NASA-furnished liquid and gas vessels, flanges, and valves were used. Preliminary operational checkout was accomplished, including valve micro-adjustments. The system was subjected to leak tests and a proof test followed by a final cleaning of the ignition system. The gaseous fluorine manifold was fabricated and then installed on the thrust chamber assembly.

During on-stand checkout, the thrust measurement calibration load cell exhibited high internal resistance. The cell was repaired in the Transducer Laboratory and reinstalled on the stand. Satisfactory load cell and rod flexure calibrations were then completed and this calibration load cell was removed from the stand leaving the previously-installed measurement cell in place.

The 14-in. liquid oxygen flowmeter was calibrated at MSFC using liquid oxygen as the calibration fluid.

Coordination of the repair of the high-pressure A. O. Smith gaseous hydrogen receivers was undertaken by the NERVA Program during the first quarter of 1966.

The investigation of the Struthers-Wells high-pressure gas receivers relative to the condition and failure of VK-11 and the condition of VK-12 and VK-13 was continued.(141)

1966/2Q

Activation of the Test Stand H-8 facility for the initial uncooled thrust chamber assembly test was completed early in May 1966. The gaseous

(141) ibid

fluorine ignition system had been both activated and passivated, including both oxygen and fluorine flow tests. A helium dryer along with associated systems was installed. The liquid oxygen propellant bleed systems line, flange, and burst disc damaged by fire during a high pressure leak check on 29 April 1966 was repaired and modified.

The Struthers-Wells gas receiver repair proposal of 4 March 1966 was reviewed. Receiver VK-11 was shipped to the Struthers-Wells Corporation for dismantling and inspection in accordance with the repair proposal. Disposition of vessels VK-12 and VK-13 awaited the results of the VK-11 investigation.

The first Test Stand H-8 uncooled thrust chamber assembly test was performed on 20 May 1966. A satisfactory short duration, low chamber pressure, ignition checkout test was made. This test revealed excessive gaseous hydrogen flow in the hydrogen mixer system with the fuel inlet temperature and the mixture ratio being other than the intended values. A revised controls sequence for subsequent tests was developed.

A second test was conducted on 27 May 1966. This test was terminated by an automatic shutdown at 0.635 sec as a result of sensing low pressure in the actuation system for the thrust chamber fuel valve. The actuation system low pressure shutdown limit was adjusted from 2000 psi to 1200 psi.

The third thrust chamber test was performed on 1 June 1966. This test was automatically shut down at 0.846 sec as the temperature at the inlet of the fuel thrust chamber valve reached the -200°F limit indicating excessive mixer gas flow. No hardware or facility damage was experienced.

The gaseous hydrogen/liquid hydrogen mixer section was removed for modifications. A hole was cut into the end of the gas diffuser to decrease the gas velocity, particularly in a radial direction to provide improved mixing. The mixer section was reassembled, cleaned, and reinstalled.

A mechanical stop for the fuel safety valve was set to restrict the full-open position to provide a pressure drop of 140 psi hardening the system to prevent backflow from the mixer section.

Changes in the start sequence involving re-programing of the gaseous hydrogen mixer flow were made. The COMCOR computer was set up in the control room to verify the sequence revisions.

Switchover to Control Room H-3 was initiated during the down-time for the mixer modifications. This was done to alleviate control room interference problems. Relocation of instruments, control panels, cabling, and equipment was completed.

1966/2Q

After a test abortion on 27 June 1966 which was caused by leakage at the liquid hydrogen flowmeter flange, a loss in system pressure was detected while condensing gaseous fluorine during post-fire operations for the fourth test attempt. Inspection revealed that one fluorine bottle valve had burned-out. Two additional actuator motors and several control pigtails on adjacent bottles were damaged by the fire. This damage was repaired and preparation continued for the next test.

During countdown for the fourth test on 30 June 1966, leakage and a fire occurred during the pre-test pressurization phase of the liquid hydrogen system. Post-inspection revealed that the leakage was at the packing gland and bonnet on the liquid hydrogen low point bleed valve. Minor fire damage to insulation and electrical controls wiring resulted. This damage was repaired and preparation continued for resumption of thrust chamber assembly testing.

1966/3Q

A 2.528 sec duration thrust chamber assembly test was conducted on 5 July 1966 at the 50% thrust level. An early shutdown was experienced because of warming hydrogen temperature at the thrust chamber valve. After repair of the hydrogen mixer pressure regulating valve leaks as well as modification of the mixer program and fuel tank pressure settings, a satisfactory, full duration (3.248 sec) test at the 50% thrust level was conducted on 12 July 1966.

The initial full-thrust, thrust chamber assembly test was conducted on 20 July 1966. All test stand systems functioned satisfactorily. The second and third full-thrust tests were satisfactorily conducted on 28 July 1966. The final two tests, also full-thrust, scheduled for the M-1 Phaseout Program were satisfactorily completed on 4 August 1966.

Test Stand H-8 was utilized for two distinct operational phases. Initially, it was used for gas generator assembly testing. This was followed by the extensive facility modifications needed to provide the uncooled thrust chamber assembly test capability. It should be noted that the criteria for these modifications evolved from the testing experience gained at Test Stand C-9. Of particular significance was the requirement for facility-type thrust chamber valves and the gas-liquid mixer system. Besides these changes, modifications were made to improve the dynamic characteristics and the liquid oxygen compatibility of the propellant systems. Following modification, a comprehensive test stand activation-checkout program was accomplished to identify existing problems. These problems, which were primarily connected with equipment failures and system cleanliness, were then corrected.

In the second operational phase of Test Stand H-8, which was concerned with the thrust chamber assembly testing, several tests were required to properly adjust the sequencing and timing of the facility valves to satisfy the thrust chamber assembly testing requirements. This was done without incident. The diligence exercised during modification and the extensive activation and checkout permitted a series of 10 thrust chamber assembly tests (five tests during a one week period in August 1966) to be conducted without any significant test stand problems.

The facility thrust chamber valves, which were subjected to more than 2000 actuation cycles under varying conditions, performed excellently as did the servo-controlled gas-liquid mixer system, which provided hydrogen temperature conditioning during the start, steady-state, and shutdown transients to within ± 3°F in the -300°F temperature range. Also, the servo-controlled run vessel pressurization system provided intricate ramping sequences and mixture ratio control to within one percent of the tank pressure requirements. The performance of the 1.5-million lb thrust measuring system verified its design and calibration by providing thrust measurement accuracies of 0.25%.

E. TEST ZONE J

This area was originally designated as Zone G. It was redesignated as Zone J at the outset of the second quarter of 1964 and all reference to the facility in this discussion designates it as Zone J.

1962

During the third quarter of 1962, a contract was awarded for applying insulation to the liquid hydrogen run tank in Zone J. This insulation was completed by the end of the quarter and Test Stand J-1A was activated for liquid hydrogen bearing and seal testing, which began immediately.

1963

The J-1 control room was modified during the second quarter of 1963 and electrical contractor effort was initiated on Test Stand J-1A. During the ensuing quarter, the J-1 control room instrumentation was modified and the control room was reactivated.

Preliminary design effort for a stage igniter facility was under way by the fourth quarter of 1963.

1964

A contract was awarded during February 1964 for vacuum-jacketed liquid hydrogen and liquid oxygen vessels. Fabrication of these vessels started during the following month.

1964

The test facility design was completed during February 1964 and the contract was awarded in March 1964. Modifications to the stand were completed early in May 1964, including installation of the liquid hydrogen and liquid oxygen vessels. Activation for igniter testing started on 18 May 1964. The cold flow testing phase commenced on 2 June 1964 and the first article stage igniter was received and installed early in July 1964. Activation was completed and the first test firing was conducted on 14 July 1964.

The necessary basic planning, design, and fabrication of modifications to the test facility to accommodate the thrust chamber assembly coaxial injector unit were completed in August 1964. During September, a gaseous hydrogen/liquid hydrogen mixer was incorporated into the system to provide the 143°R hydrogen required to simulate cooled thrust chamber assembly firings. Active testing was initiated in the form of a mixer flow study to assure adequate hydrogen temperature and flow control prior to subjecting the test hardware to hot firings.

A parallel study was made of the proposed fluorine hypergolic ignition phase of this program. Low pressure gaseous fluorine was selected for the first hypergolic tests. Facility modification to incorporate this capability was also started.

Hot firings of the thrust chamber assembly coaxial injector unit began on 7 October 1964 and continued into November. Following this series of tests, the gaseous hydrogen mixer section was removed and a straight line section was inserted into the hydrogen system to provide cold hydrogen for gas generator assembly injector unit tests. Gas generator assembly coaxial injector unit testing then continued with satisfactory results.

In October 1964, the digital system was replaced with a more modern unit from Test Zone G. Also, system modifications began to provide for gaseous fluorine hypergolic ignition in future tests and major items were procured.

1965

Testing continued throughout the first quarter of 1965. The system modifications to provide for gaseous fluorine hypergolic ignition testing of injector units were also completed. This system was then successfully checked out. In addition, a flow-calibrated water coolant system for the thrust chamber uni-element test system was installed and checked out.

Testing was completed during the second quarter of 1965 and the Test Stand J-1A facility was deactivated.

F. TEST ZONE K

This area was originally designated as Test Zone J. It was redesignated as Test Zone K at the outset of the first quarter of 1963 and all reference to the facility in this discussion designates it as Zone K.

1962

The original test stand configuration for Test Zone K called for one engine stand and one thrust chamber assembly stand. This was changed during the second quarter of 1962 to two engine stands with extended separation distances. The initial layout and basic criteria for the Test Zone K facilities were also being formulated. The test stand concepts (see Figures No. 63, No. 64, and No. 65) and the basic criteria were completed by the end of 1962.

1963/1Q

Test stand locations were established during the first quarter of 1963 and a contract was awarded for soil investigation. Preliminary reports from the soil survey were received and detailed designs were started for site preparation and civil works for Test Stand K-1, the control room, and the shop building. In addition, the design of high pressure gas receivers was completed and cost studies for an alternative configuration of Test Stand K-2 with two horizontal positions rather than the proposed one horizontal and one vertical position.

1963/2Q

Propellant storage vessels, primary switch-gear, deflector water pumps, and gas receivers were released for bid and quotations received during the second quarter of 1963. The basic civil works package for access to Zone K and utilities was released for bid.

An excavation approximately 50 ft deep was made at the site for Test Stand K-1. Small concrete pads were poured at the bottom of the excavation and vibrators installed (see Figure No. 66). A vibration study was started to determine soil characteristics for design of test stand foundations.

1963/3Q

The study to determine the dynamic characteristics of earth at the Test Stand K-1 site was completed during the third quarter of 1963. A summary of the dynamic analysis development for Test Stand K-1 has been prepared as a separate report.(142) Results indicated that the soil was

(142) Rotter, L. and Rotz, J. V., Summary of Test Stand K-1 Dynamic Analysis Development, Aerojet-General Report No. 8800-17, 8 October 1965

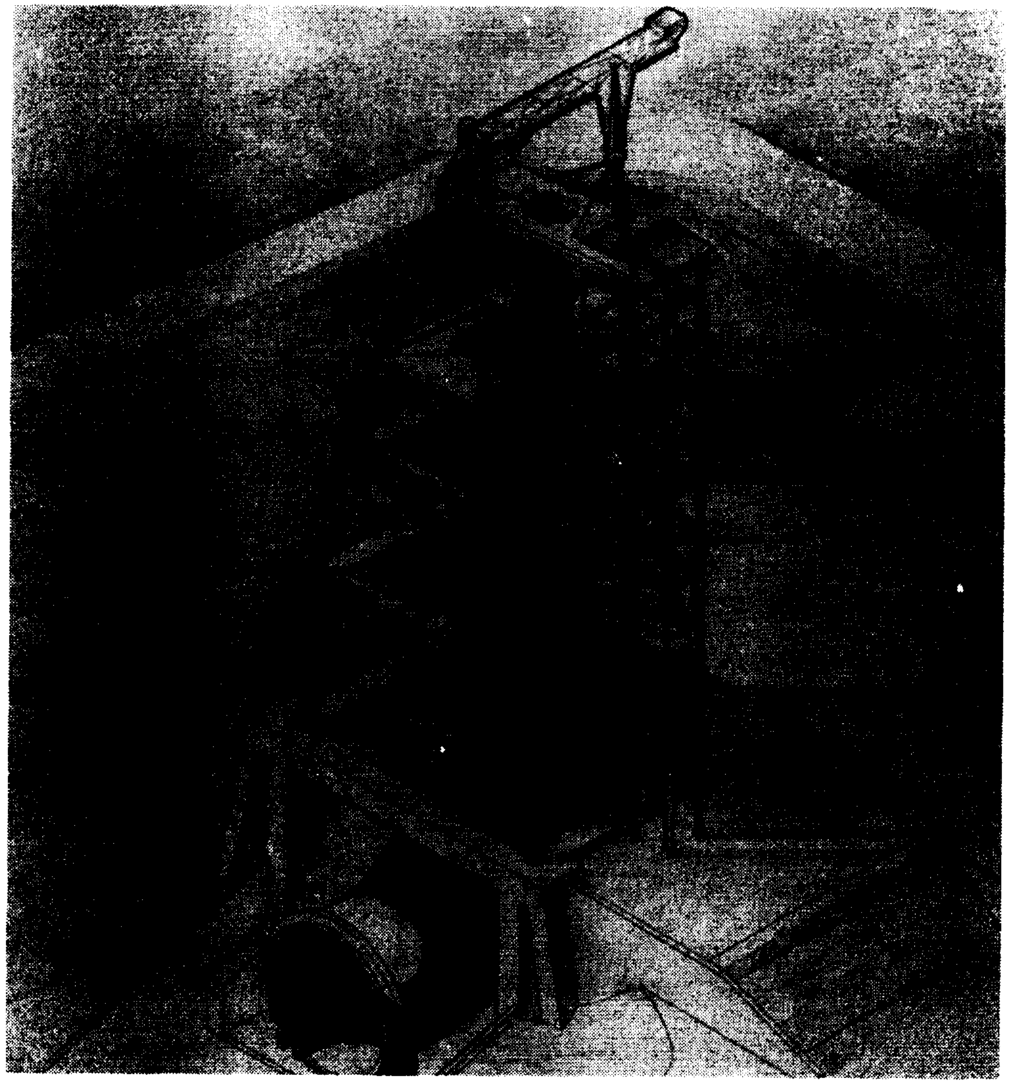

Figure 63. Engine Test Stand Concept, Test Stand K-1

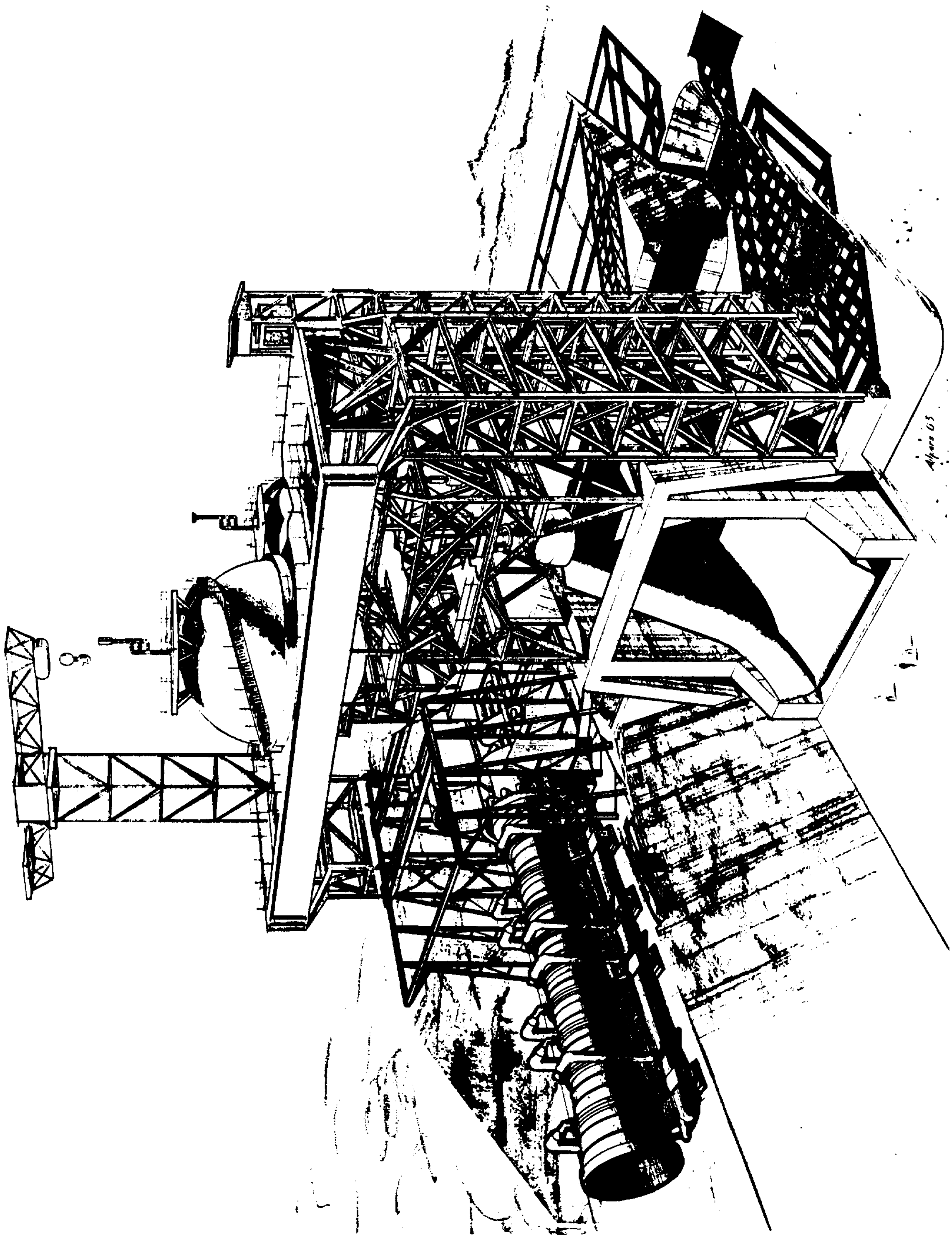

Figure 64. Test Stand K-2 Concept

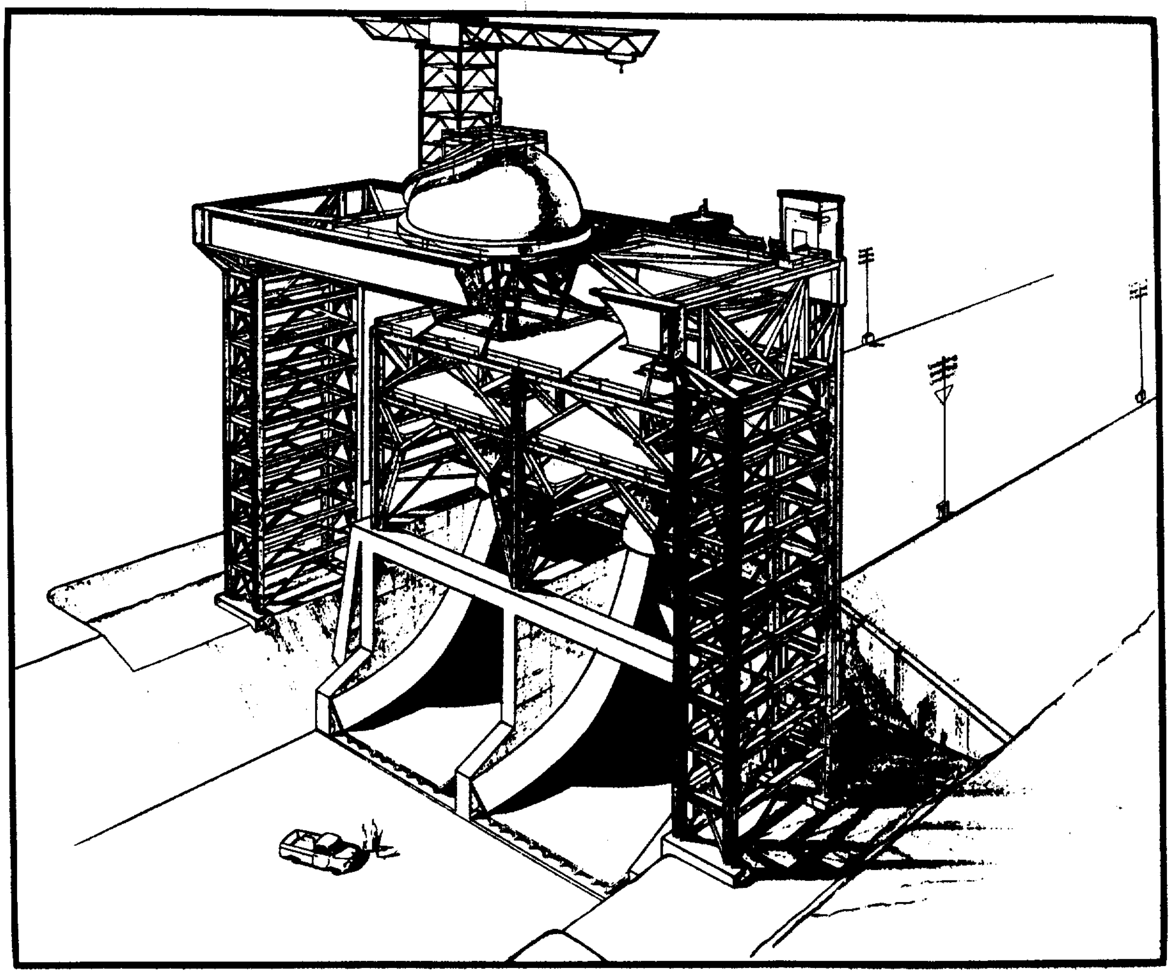

Figure 65. Engine Test Stand Concept, Test Stand K-3

Figure 66. Test Stand K-1 Vibration Study Pads

satisfactory for the M-1 engine test stands, with an estimated natural rocking frequency of the soil-foundation system of 3.1 cps. Emphasis during this time was placed on criteria established and review, design, and construction planning. A contract was awarded for the fabrication of four high pressure gas receivers. The specification for the digital computer was approved and released for bid. Also, the control room and signal conditioning center floor plans were made final.

1963/4Q

The civil site preparation package (Specification 6648) was awarded on 11 October 1963 and work began on 14 October 1963. This effort included the initial road work and the basic area water supply.

A basic facility plan for Test Zone K was prepared during the fourth quarter of 1963. The plan provided for incremental construction of the test stands upon the basis of then-current NASA budgetary projections.

1964/1Q

Proposals LR 642200/642201, dated 4 March 1964, were submitted at the end of the first quarter of 1964. They provided for incremental construction of Test Zone K according to program need. The high-pressure gas receiver foundation (Specification 6829) was awarded. The major civil works package (Specification 6650), which was Aerojet-General funded, was awarded on 25 March 1964.

1964/2Q

The site preparation (Specification 6648), which was also Aerojet-General funded, was completed in April 1964. It included construction of the Test Zone K access road from the main highway, the water storage vessel (see Figure No. 67) and connecting pipeline, and the water headworks (see Figure No. 68), which included the reservoir, pump, and filtration plant.

Construction activities associated with Test Stand K-1 and area support civil works, the Aerojet-General funded package (Specification 6650) developed to full stride during the second quarter of 1964. Major progress was made in all areas. The excavation for the Test Stand K-1 substructure was completed and three major concrete pours were made. Figure No. 69 shows the status of this work at the end of the quarter. Concrete pours for the first floor of the control room were also completed and forming for the second floor was started (see Figure No. 70). Excavation and grading in the liquid oxygen, liquid nitrogen, and helium unloading and storage areas was completed. Concrete pours for the unloading pads and road crossing pipeways were also made. The liquid oxygen propellant control building was erected.

Figure 67. Test Zone K Water Storage Vessel

Figure 68. Test Zone K Water Headworks

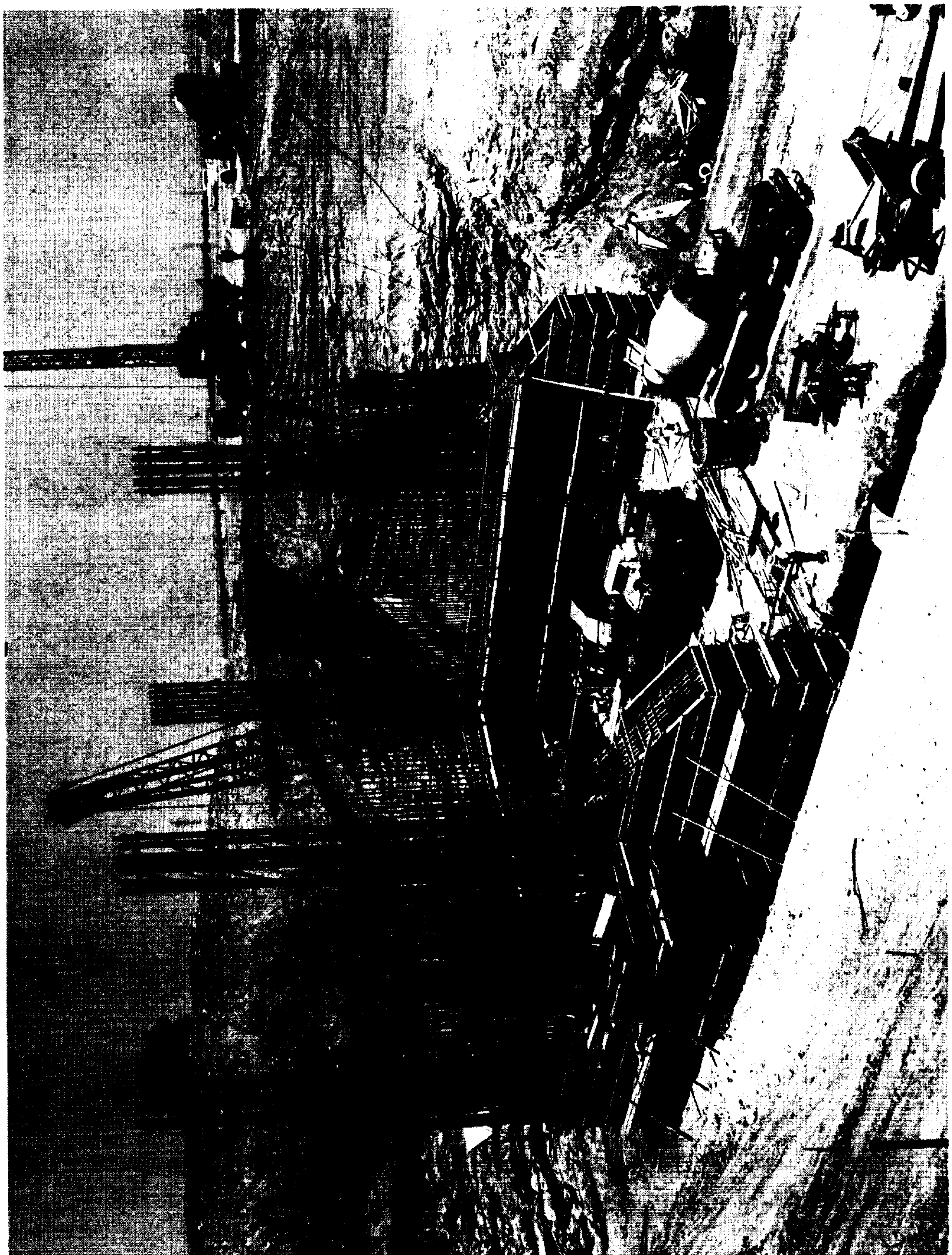

Figure 69. Test Stand K-1 Foundation (Second Quarter, 1964)

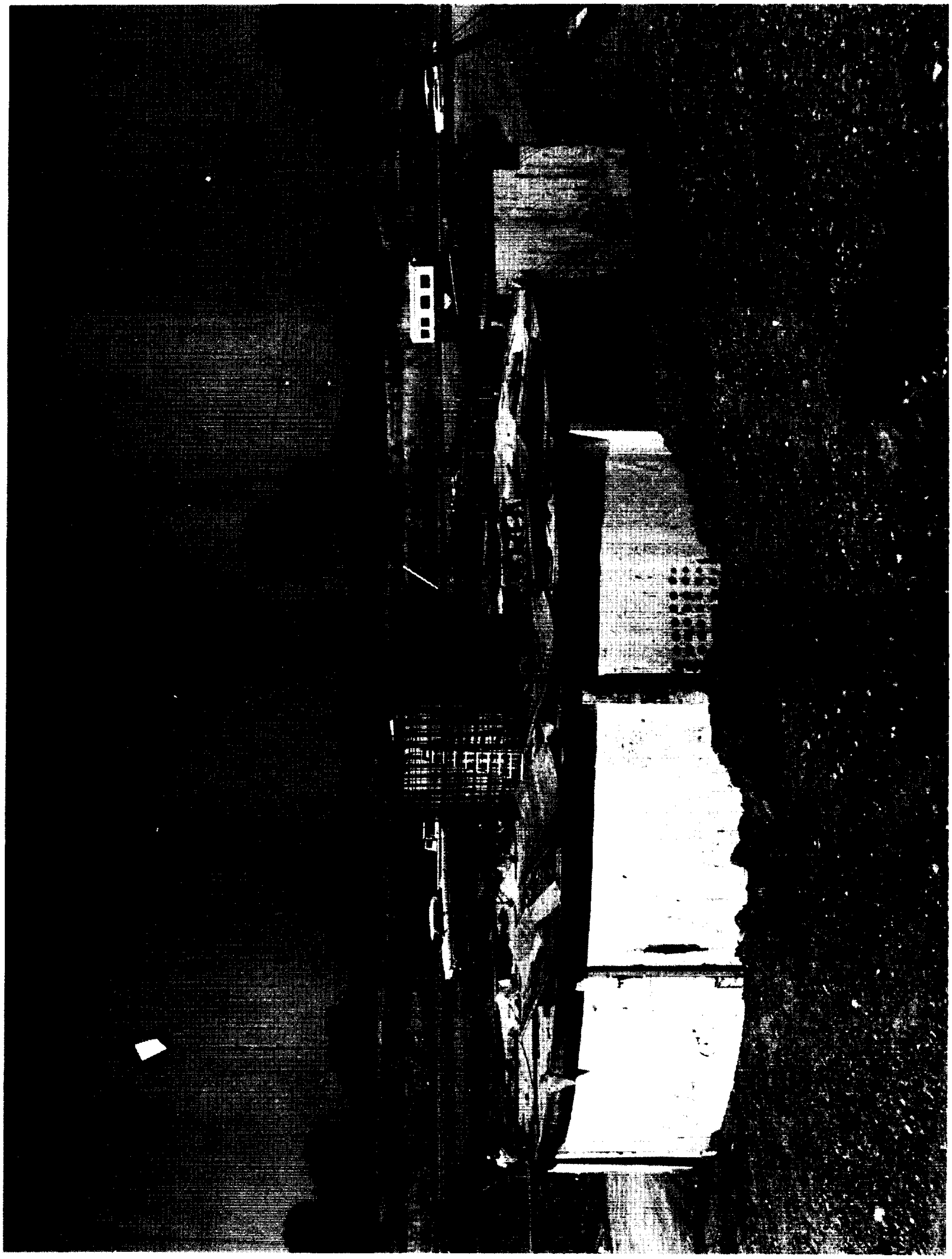

Figure 70. Test Zone K Control Room (Second Quarter, 1964)

Excavation and grading in the auxiliary liquid hydrogen unloading area and the VK-1, liquid hydrogen storage area was completed and the concrete unloading pad poured. Fabrication and field installation of the high pressure receivers (two gaseous nitrogen, one gaseous hydrogen, one helium; Specification 6637) were completed.

Bids were received, evaluated, and award was made on 25 May 1964 for the Test Stand K-1 liquid hydrogen Run Vessel VK-37 (Specification 6846). The award was based on an alternative to utilize Quilted Super Insulation in lieu of perlite to improve evaporation loss characteristics and prevent the possible problem of perlite compaction.(143)

Award was made on 6 April 1964 for the procurement of six 20,000 gpm water pumps for deflector cooling (Specification 6869). Another award was made on 10 June 1964 for the procurement of one 370,000 gallon liquid nitrogen storage vessel utilizing Quilted Super Insulation and one 115,000 gallon liquid oxygen vessel utilizing perlite insulation.

A rearrangement of the major Test Stand K-1 design-construction package (Specification 6653) was made in compliance with a NASA/LeRC request. It provided an over-all schedule improvement. The single major package was split into three separate packages: On-Stand Mechanical (Specification 6653); Environmental Altitude Chamber(144) and Deflector Tube (Specification 6845); and Test Stand K-1 Superstructure (Specification 6907).

1964/3Q

Concrete was poured for the basic portion of the Test Stand K-1 substructure during the third quarter of 1964. The concrete "wing" retaining walls were also poured. Earth backfill around the stand was completed up to the elevation of the terminal room floor and forming for the terminal room was started. The concrete storage pad adjacent to the stand for the environmental chamber was poured and excavation, plumbing, and the concrete floor for the Test Shop were completed.

The concrete dome roof of the Control Building was gunited and the forms were removed. The entryway structure concrete was also poured (see Figure No. 71).

(143) Commander, J. C. and Rotter, L., Economic Analysis of Perlite Versus Super Insulation in Liquid Hydrogen Storage and Run Vessels for the M-1 Program, NASA Report No. CR 54720, 15 September 1965

(144) Rotter, L. L., Description of Test Stand K-1 Altitude and Environmental System, Aerojet-General Report No. 8800-30, 25 November 1965

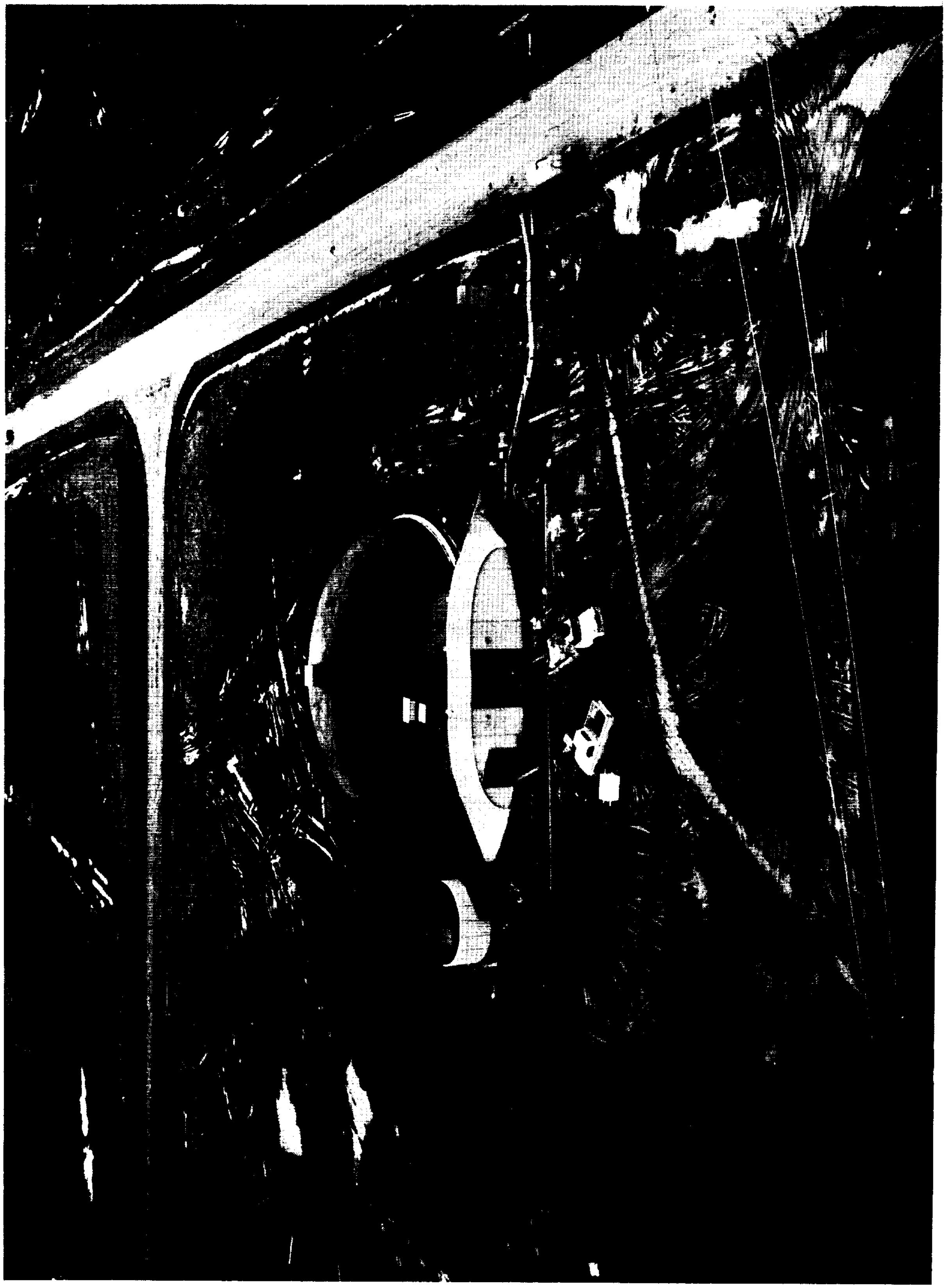

Figure 71. Aerial View of Test Zone K Control Building and Entry Structure (Third Quarter, 1964)

All civil work construction activities associated with the liquid hydrogen, liquid oxygen, liquid nitrogen, and helium unloading and storage areas were completed. Construction activities associated with the installation of the basic power, road lighting, and domestic water utility system were also essentially completed.

Bid, award, and the start of construction activities associated with the off-stand mechanical package (Specification 6652) were accomplished by the end of September 1964. The contractor moved on-site, started the procurement of materials, and began installing the foundation for the large, 2.5-million gallon deflector water storage tank, TK-2, and the deflector pump foundations. The foundations for propellant storage vessels, VK-1 and VK-3, were also poured.

Award of the contract for fabrication and erection of the Test Stand K-1 superstructure (Specification 6907) was made during the third quarter of 1964. Shop drawings were received and approved, steel orders placed, and procurement of other components initiated.

Also, award for the 30,800 gallon, QSI insulated liquid hydrogen storage vessel, VK-10 (Specification 6756), supplying liquid hydrogen to the liquid-to-gas converters, was made.

1964/4Q

All design work associated with Test Stand K-1 and the K-1 Area support facilities was essentially completed during the fourth quarter of 1964 except for activities associated with changes required during the construction phase.

On-site construction activities associated with the Test Stand K-1 and area support civil works (Specification 6650) were also essentially completed. The basic Test Stand K-1 substructure and the earth backfill at the test stand were completed. The concrete approach apron was poured and asphalt paving around the apron was completed (see Figure No. 72).

Activities in connection with the off-stand mechanical package (Specification 6652) included: installation of the instrumentation duct banks; installation of pipe supports along the main pipeways; removal of the liquid nitrogen storage vessels, VE-102 and VE-104, from E-Zone and then installation in the liquid oxygen/liquid nitrogen storage area; the erection of structural steel for platforms and supports at the auxiliary liquid hydrogen unloading area; pouring of the foundation for VK-10 (liquid hydrogen liquid-to-gas converter supply vessel); and the installation of piping for the gaseous nitrogen and gaseous helium distribution system.

Figure 72. Test Stand K-1 View from Apron Side (Fourth Quarter, 1964)

By tne end of 1964, the outer-carbon steel shells of VK-1 (liquid hydrogen) and VK-3 (liquid oxygen)(Specification 6836) were essentially completed. Assembly of the inner and outer vessel of VK-37 (Specification 6846) was completed. All six of the 20,000 gpm pumps were delivered and installed (Specification 6869). Off-site pre-fabrication of the test stand superstructure (Specification 6907) was essentially complete.

1965

A stop work order was placed on all M-1 engine test facilities in Test Zone K on 5 February 1965 as a result of program redirection. Figures No. 73 through No. 81 show the status of all field work at the time of this stop order.

All on-site as well as off-site special test equipment work was stopped, with in-process work and equipment left in a condition that would not create a hazard or incur damage.

Because the Aerojet-General funded civil works package (Specification 6650) was almost completed, it was decided to complete this work. Final painting, installation of trim hardware, and lighting fixtures for the control building was completed. Asphalt paving of the parking area was completed and final checkout of the domestic water pumping system was accomplished. All work associated with the civil works package was completed on 31 March 1965.

Prefabrication of all steel for the basic stand structure (Specification 6907) was completed and the steel was delivered to the site. Also, fabrication of the elevator and rolling platform was 90% completed with some parts on-site. The installation of the elevator equipment had started. Erection of the superstructure was terminated at the first level (an elevation of 292 ft 6-in.). Welding of corner columns and diagonal braces continued after the stop work order to secure the structure from possible wind and seismic load damage. Prefabricated steel on-site was rearranged for storage and the 150-ton crane was dismantled and moved off-site.

On-site installation of the off-stand mechanical package (Specification 6652) had reached a peak and off-site fabrication of piping, equipment and components, with a few exceptions, was in the final stages. Pipeways, pipe sleepers, foundations for the deflector water pumps and vessels, installation of electrical/instrumentation duct banks and man holes were approximately 90% completed. Erection of the 2.5-million-gallon deflector water storage tank was essentially complete, except for a minor amount of welding, testing, and cleaning. As a securing measure, work on TK-2, the 2.5-million-gallon deflector water storage tank, continued for several days after the stop work order to complete closure on the roof and two side nozzles. Vessel VK-32 in this system was delivered and installed.

Figure 73. Test Stand K-1 at Time of Stop Order

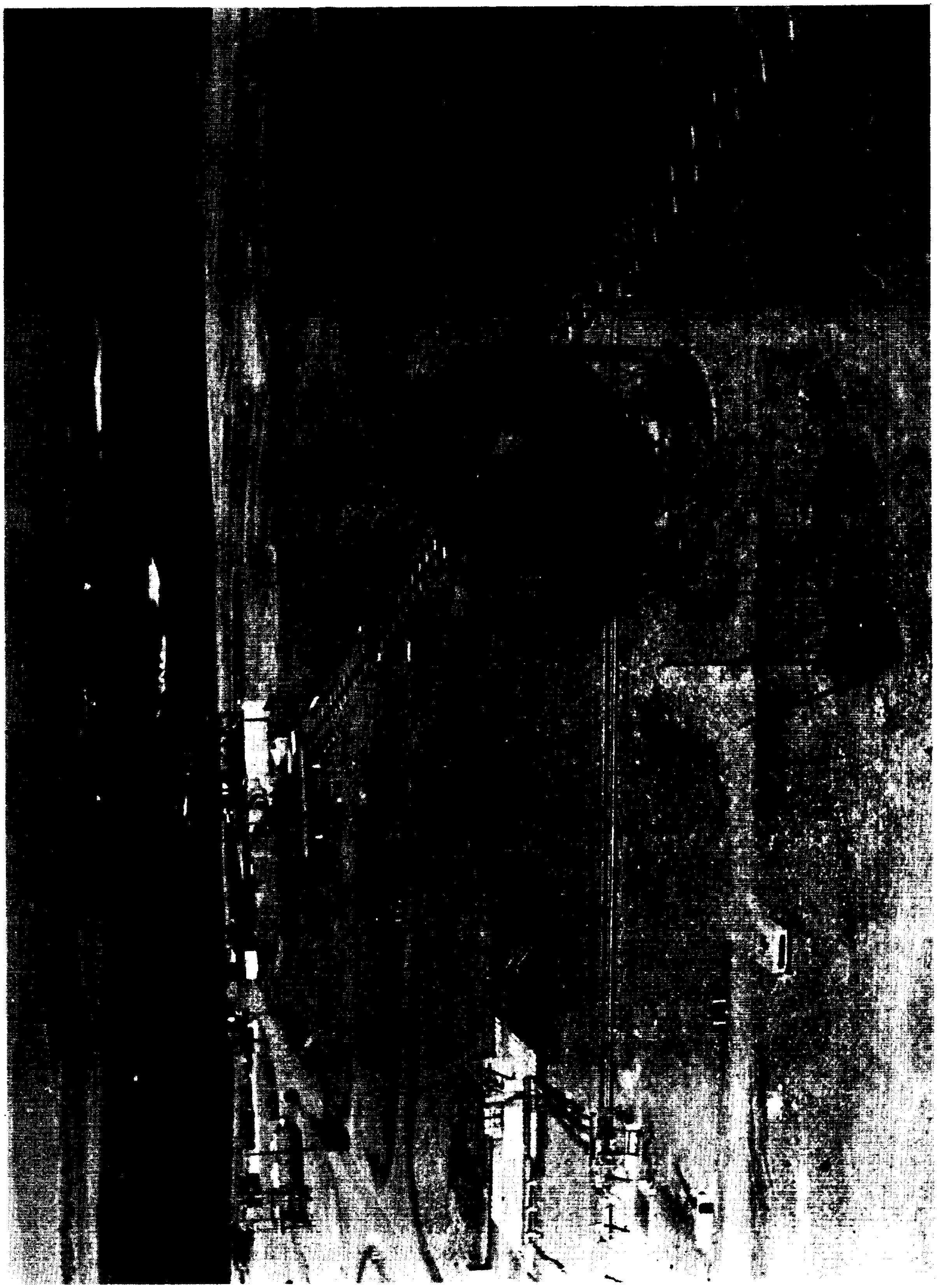

Figure 74. Liquid Hydrogen Storage Vessel VK-1 at Time of Stop Order

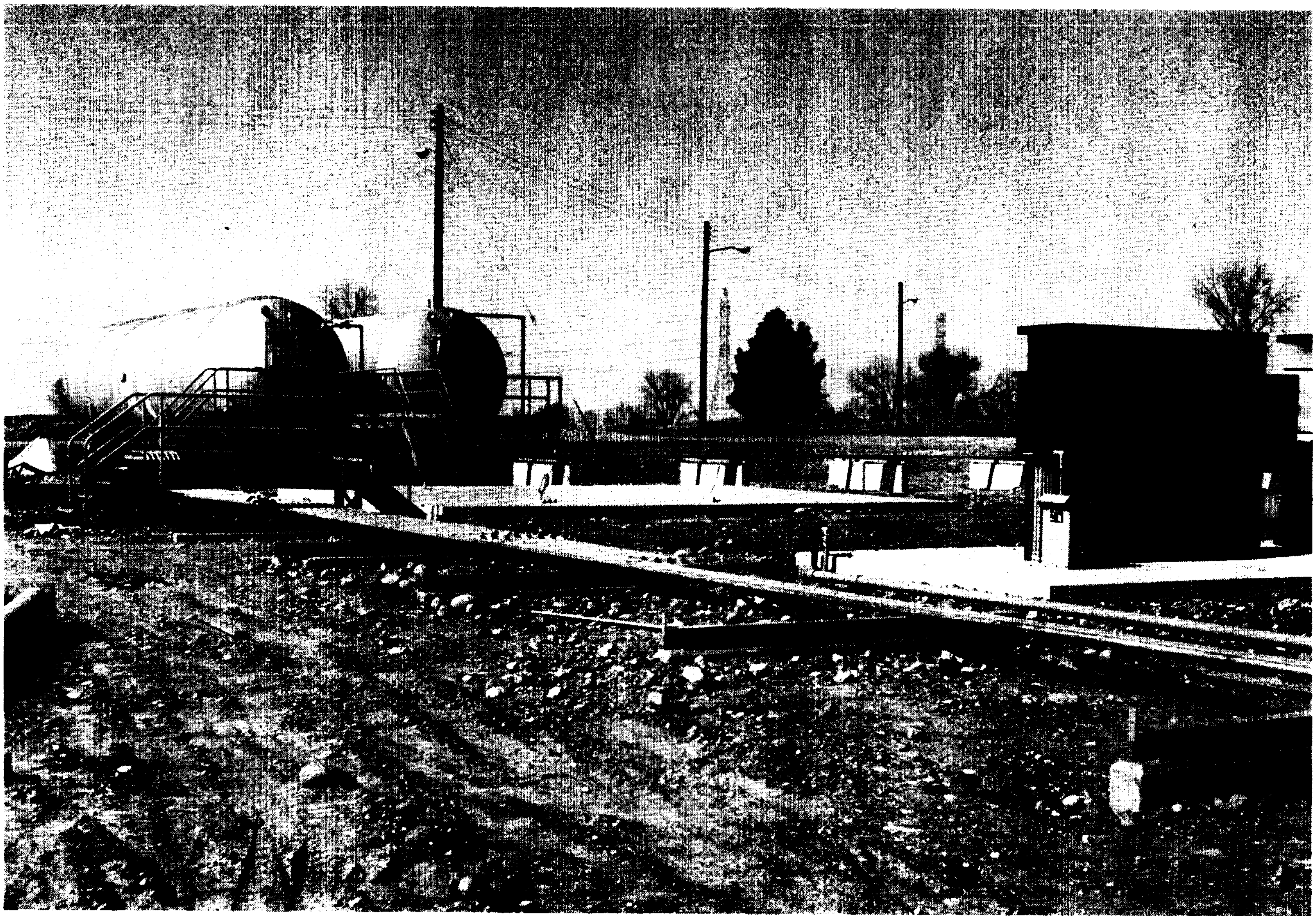

Figure 75. Liquid Nitrogen Storage Vessels VE-102 and VE-104
at Time of Stop Order

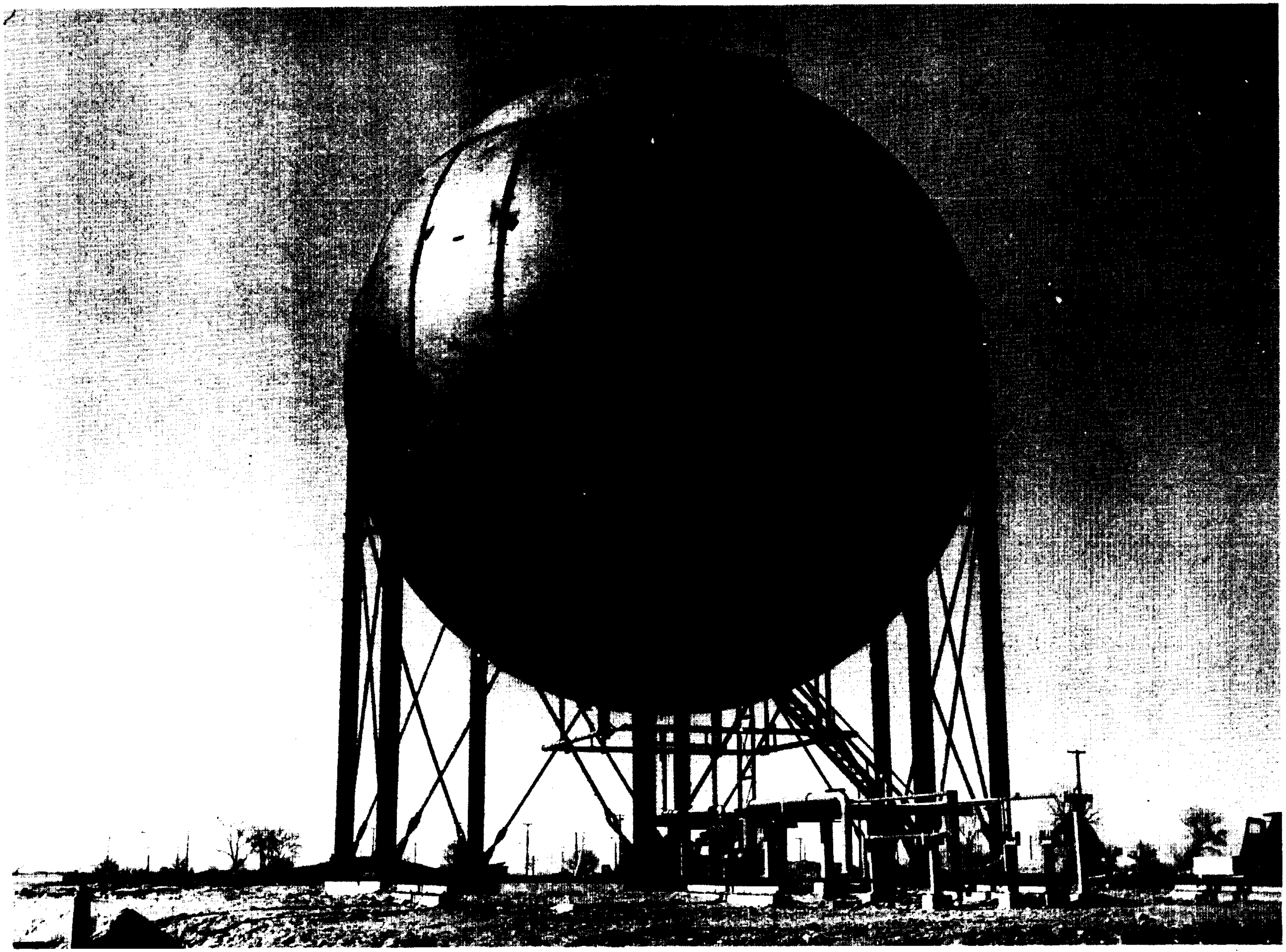

Figure 76. Liquid Oxygen Storage Vessel VK-3 at Time of Stop Order

Figure 77. Pipeway Between VK-1 and Liquid Hydrogen Unloading Station at Time of Stop Order

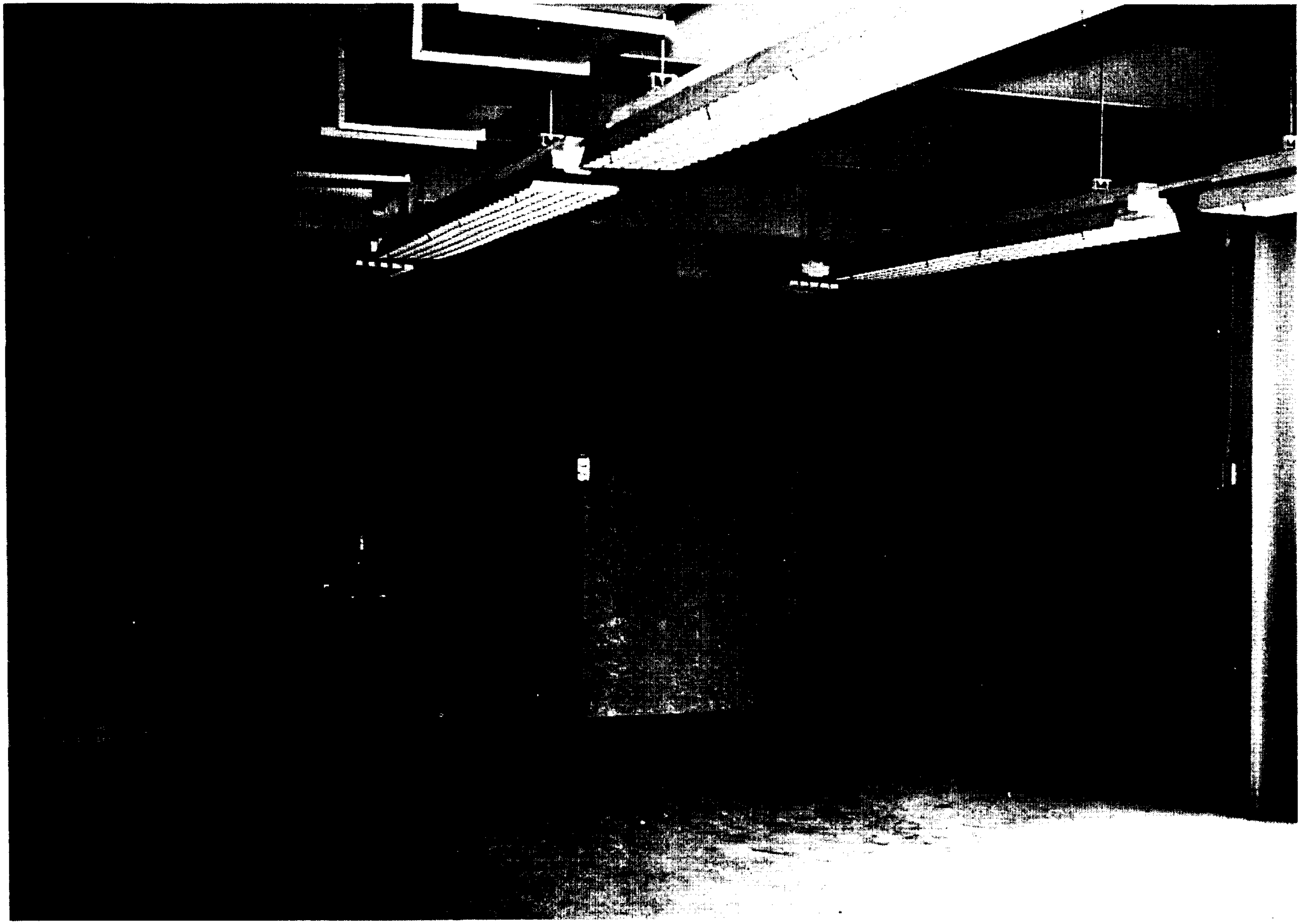

Figure 78. Interior View of Test Stand K-1 Service Building at Time of Stop Order

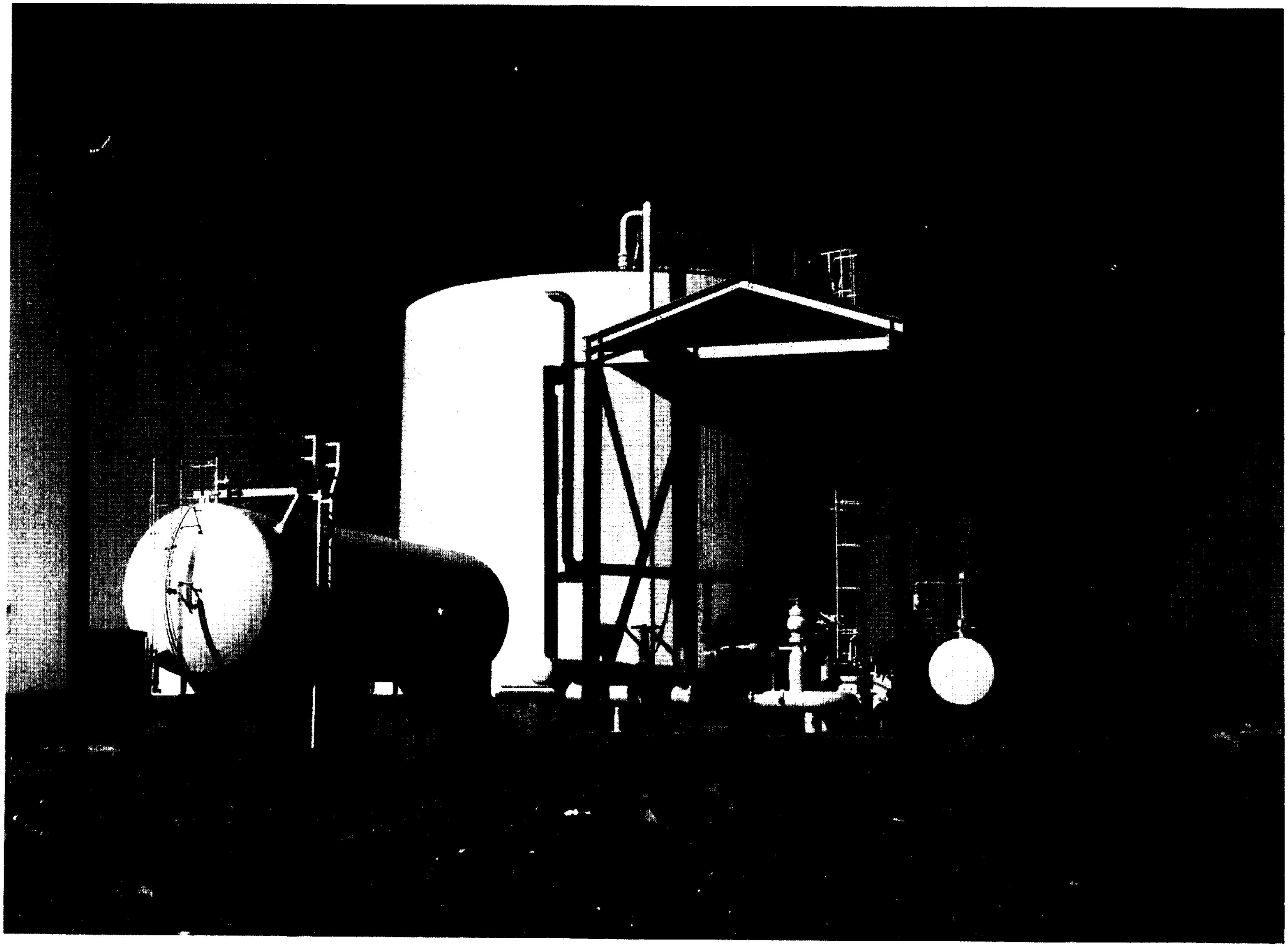

Figure 79. Test Zone K Domestic Water Facility at Time of Stop Order

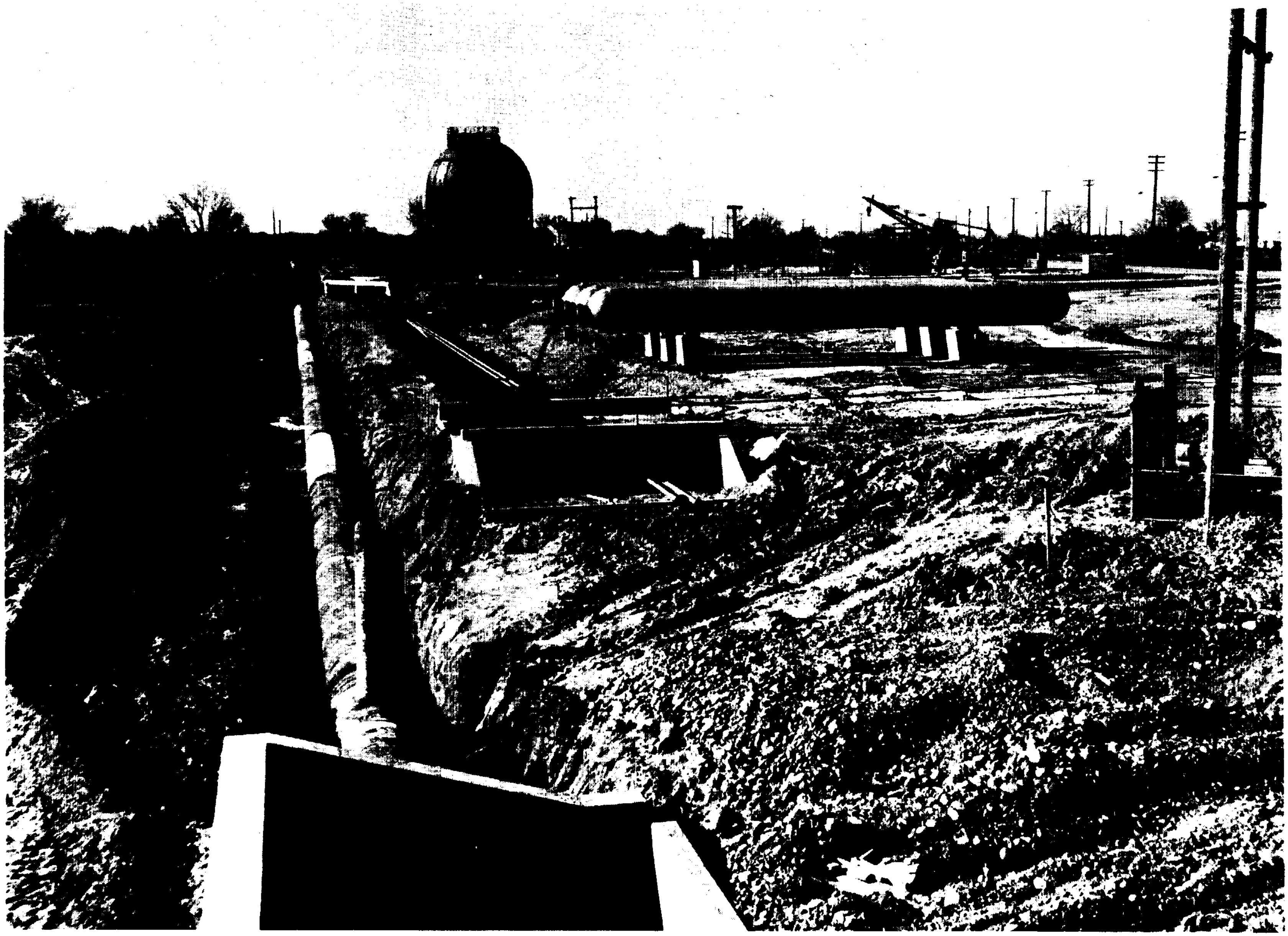

Figure 80. 48-in. Deflector Water Line Installation at Time of Stop Order

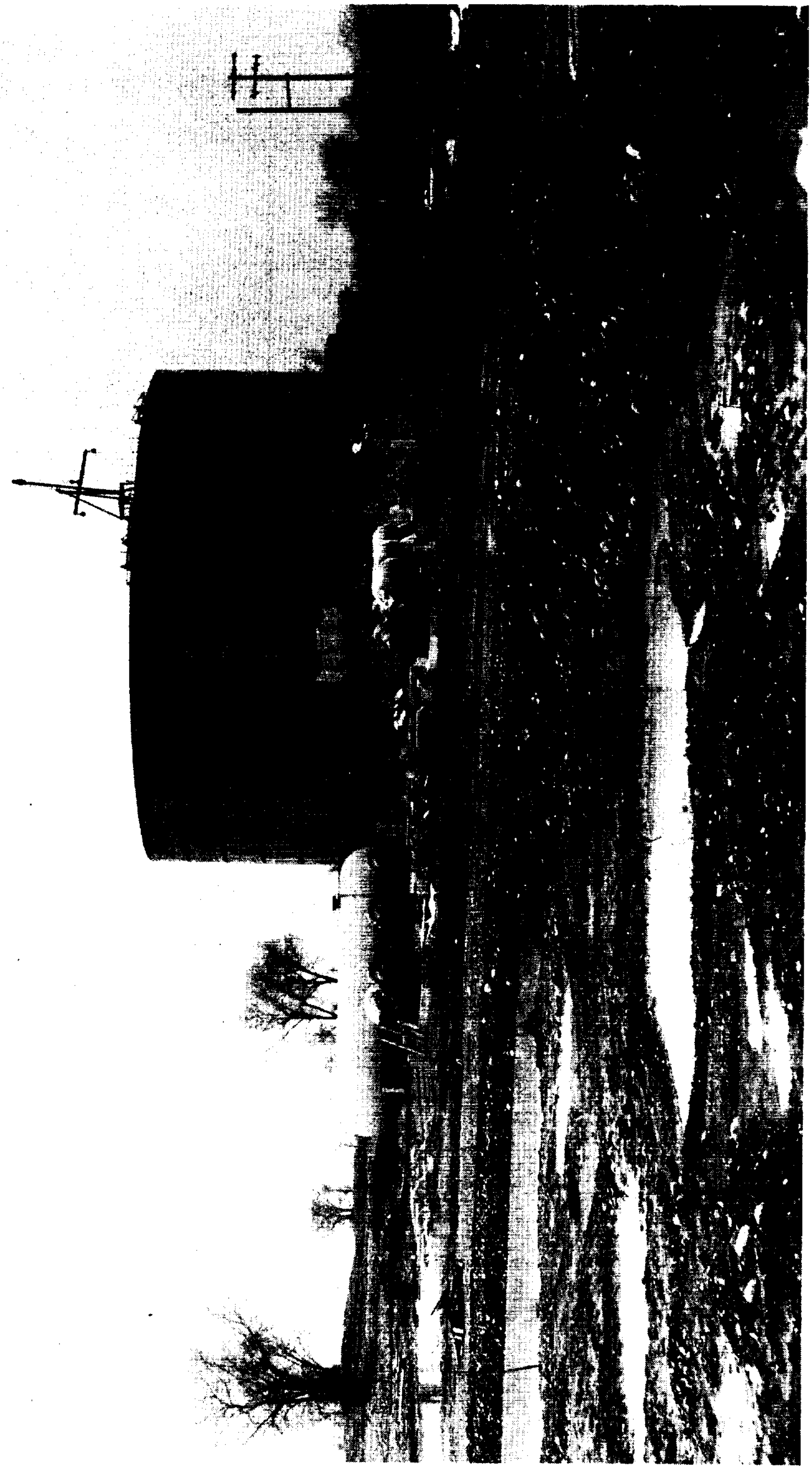

Figure 81. Deflector Water System (Zone K) at Time of Stop Order

1965

Approximately 75% of the steel structure for VK-17 was erected prior to the stop work order; however, the vessel was not installed. Installation and cleaning of gaseous nitrogen, gaseous hydrogen, and gaseous helium piping and tubing was in process in all areas and was approximately 75% complete at the time of the stop work order. Also, installation of 1840 ft of 48-in. deflector water supply piping was complete at that time.

Approximately 95% of the welding on the inner and outer vessel shells of VK-1 and VK-3 (Specification 6836) was completed. All of the scaffolding was removed from the inner and outer vessels, the electrical and pneumatic equipment dismantled, and these items were shipped off-site. However, it was decided to complete the welding on both vessels as well as the installation of the QSI insulation in VK-1.

Final cleaning and inspection of VK-37 (Specification 6846) was completed.

All material for the Test Stand K-1 deflector tube (Specification 6845) had been procured and shop fabrication was started.

The components and equipment status under subcontract for the mechanical off-stand package (Specification 6652) were evaluated and disposition as to whether to cancel or complete the item was made.

Two plans for storage and preservation of Test Zone K were prepared. The first was a short-term plan and the second, a long-term one.

The storage and protection of equipment from the elements by both the off-stand works contractor and Aerojet-General was initiated in November 1965 and continued through December 1965. In lieu of a formal plan for long-term storage and preservation, termination proceedings were being accomplished in accordance with standard Government policies.

1966

Appropriate closure work on storage vessels VK-1 and VK-3 was initiated during the first quarter of 1966. Also, hydrotesting of VK-3 was completed and the vessel cleaned.

The closure work on VK-1 and VK-3 was completed during the second quarter of 1966 and the annular space was filled with gaseous nitrogen. Vacuum testing of VK-1 was completed during July 1966. Also, the QSI insulation originally intended for VK-1 was shipped from the factory to NASA/LeRC.

G. TEST AREA INSTRUMENTATION AND CONTROLS

Effort was initiated at the outset of the program to provide appropriate instrumentation support to all of the M-1 test zones. This effort was conducted in conjunction with the mechanical construction and modifications accomplished in the various test zones throughout the course of the program. Therefore, the work accomplished in connection with instrumentation and controls is briefly summarized without regard to chronological presentation.

1. Instrumentation

Control Room D-2 was modified to support the gas generator and thrust chamber assembly testing at Test Stand C-9. This included the up-dating of the digital system, the purchase of dc amplifiers, and the installation of a high frequency data acquisition system.

Control Room E-1 was modified to incorporate new control systems, new patching, closed-circuit television, dc amplifiers, temperature measuring systems, a digitally-operated turbine overspeed malfunction detection device (developed by Aerojet-General), as well as new test stand cables and outlet boxes. This control room was used in support of Test Stands E-1 and E-3. A noteworthy accomplishment with this control room was the successful use of infrared closed-circuit television utilizing a wide-range vidicon in connection with rocket component testing.

Control Room H-1 was modified to support gas generator assembly and thrust chamber assembly testing at Test Stand H-8. A new digital data acquisition system was designed and constructed by Aerojet-General as was a combustion stability monitor. Additional equipment incorporated included amplifiers, patchings, temperature measurement systems, and closed-circuit television.

Instrumentation design support for Test Zone K had been initiated and the signal conditioning modules were ready for fabrication when the stop order was imposed. The K Zone digital data system, which incorporated an on-line digital computer for test control, had been procured and acceptance tested.(145)

The transducer laboratory modifications included the procurement and installation of an extension to the existing 60,000 lb dead weight load cell calibration system. It was activated during November 1964. The

(145) Anderson, J. D., M-1 Engine Test Complex Data Acquisition Systems, NASA Report No. CR 54811, 25 April 1966

extension consisted of a system of flexures and levers that allowed for calibration up to 1.5M lb force. This system was used successfully for routine calibrations and in the resolution of load cell problems related to testing at Test Stand H-8.(146)

In November 1963, flowmeters were ordered for all phases of M-1 testing except Zone K. Although some problems were encountered with calibration facility capabilities and straightening vane fabrication, all meters were delivered in time to satisfy testing requirements.(147)

The additions to the vibration laboratory included a test console and an exciter. These additions were activated during December 1963.

2. Controls

Two major efforts were undertaken to satisfy the M-1 Program test requirements for a sophisticated electro-hydraulic control system.

The first of these efforts included the subcontracting of three engineering studies for a Zone E facility design approach and the subcontracting of a back-pressure study for closed-loop control of the pump discharge conditions. The facility design studies were to incorporate recommendations for control valve locations, methods of closed-loop control, and component sizing.

The second major effort involved the design and construction of servo valve controller units, which would integrate the transducer power, calibration circuits, readout, controls, and operational amplifiers. These were successfully used in testing at C-9, E-1, E-3, and H-8. The use of on-line analog computers for testing at Test Stands E-1, E-3, and H-8 permitted rapid test condition changes as well as the centralization of control and malfunction detection/shutdown circuits.(148)

A sound level and atmospheric condition measuring plan was prepared during 1963. It related the noise generated during thrust chamber assembly and engine testing to a need for local zones of personnel control as well as community reaction to the large energy low frequency sound levels.

(146) NASA Report No. CR 54793, op. cit.
(147) Deppe, G. R., Large Size Cryogenic Turbine Type Flowmeter Technology, NASA Report No. CR 54810, 1 June 1966
(148) Aerojet-General Report No. 8800-61, op. cit.

1966

Much of this effort was based upon sound level studies conducted by MSFC. Because of the reduction in M-1 Program scope, work in this area was limited to making local and remote sound level recordings during thrust chamber testing at Test Stand H-8 without any atmospheric sounding. Therefore, any refractive reinforcement at remote sites could not be related to precisely known atmospheric conditions. This limited correlation of far field sound data with varying atmospheric conditions in the Sacramento area.

Near and far field measurements, projected to the source, indicated a source level of approximately 204 db at steady-state, which was within 0.5 db of the calculated value (over-all sound pressure level). The combustion of a large amount of hydrogen-lead propellants at the start gives an instantaneous source level of approximately 212 db; however, it is for an extremely short duration.

H. CRYOGENICS LABORATORY

SUMMARY

The Cryogenics Laboratory provided the capability for component development tests especially in the areas of high pressure-high flow rate hydrogen, bearing and power transmission test fixtures, vibration-structural testing, and a materials testing capability in a hydrogen environment. A clean room environment was also available at this facility and permitted rapid assembly-disassembly of components for immediate evaluation. Over 400 tests of various components were accomplished at this laboratory without any significant facility problems.

CHRONOLOGY

1962

The design of the cryogenic laboratory vessels was completed during the second quarter of 1962 and bidding procedures were initiated. The mechanical system design was also under way at this time. In the third quarter of 1962, the environmental vacuum chamber was released for procurement and bids were solicited for the mechanical package. Building construction had also started.

The structural portion of the addition to the Cryogenic Laboratory building was completed during the fourth quarter of 1962 along with the test bay blast walls. The structures test bay including the overhead crane supports, was also completed. Fabrication of the liquid oxygen run vessel was also completed.

1962

The 2-ton and the 10-ton bridge cranes were received and installed by the end of 1962. Also, evaluation was continuing for the leak detection system, the analog tape system, the helium start valve test facility, the environmental chamber, as well as the sampling and print-out system. In addition, the gaseous oxygen storage vessel had been received.

1963/1Q

Award of the installation contract for the process piping, equipment, instrumentation (Specification 6408A), the space simulation chamber (Specification 6422A), and the steam power plant (Specification 6591) was completed on 20 March 1963.

The liquid oxygen and liquid hydrogen run vessels were received during the first quarter of 1963 as well as the liquid oxygen storage vessel. Also, the procurement of the chamber VAC-A1 (Specification 6420), valve and cold trap (Specification 6421) was completed.

The procurement of valves (Specification 6417) for the mechanical (STE) and distribution (Facilities) piping systems was almost completed by the end of March 1963. To that date, 106 of the total 143 procured valves to be installed according to Specification 6408A had been delivered.

Test Bays 1, 2, 9, 10 and 11 control consoles (Specification 6408A) were nearing completion at the AETRON Division. Also, the operating console for the space simulation chamber was delivered to the Cryogenics Laboratory for contractor acceptance on 22 March 1963.

Installation contract bids for the cryogenic laboratory Clean Rooms (Specification 6599) were opened on 4 March 1963, the contract was awarded, and the contractor started work on 25 March 1963.

The 750 KVA sub-station transformer for the vibration test system and the motor generator set went out to bid on 22 March as Specification 6611.

1963/2Q

The procurement of all valves (Specification 6417 and 6451) was completed during the second quarter of 1963. Also, the control consoles for Test Bays 1, 2, 9, 10, and 11 were completed at the AETRON Division and these consoles were received at the Sacramento facility. The vacuum dehydrator (Vessel T-1) was received and a partial shipment of the vacuum-jacketed piping for the liquid oxygen test bay was received on 24 June 1963.

The installation contract for the data acquisition system (Specification 6497) was awarded in April 1963. The reference junction bay, storage bay, leak alarm patch bays, and miscellaneous cables were received in

May 1963. The range and filter bay as well as the analog tape recorder rack (without the tape recorder) were also received. Fabrication was continuing for other instrumentation bay groups.

The construction of clean room areas (Specification 6599) in the Cryogenics Laboratory was completed in June 1963 while the water distribution piping (Specification 6606) from Test Zone C to the Cryogenics Laboratory was completed in May 1963.

The design for the miscellaneous installation contract package (Specification 6665, Aerojet-General funded) was completed in May 1963 and the contract was awarded in June 1963. The design for the NASA-funded miscellaneous installation contract package (Specification 6680) was also completed in May 1963 and awarded in July 1963.

1963/3Q

Oxidizer test bays 1 and 2 were completed and inspected. Occupancy of the Phase I facility was obtained during the third quarter of 1963. A performance test and operational procedures manual for the space simulation chamber installation (Specification 6422A) was prepared and approved.

Installation by the contractor of the steam power plant and transmission tester foundations (Specification 6591) was almost completed during this same quarter. The test bay jib cranes and hoists were delivered to the site and erected, as were the bay blast walls and chevron-type steel walls and roof.

Fabrication of the data acquisition instrumentation system (Specification 6597) bay group assemblies was completed at the AETRON Division Foothill Facility. The assemblies were delivered to the site and set. Terminations, electrical testing and checkout were then conducted.

The installation contract and the Amendment I package to Specification 6640 (provided for installation of the electric drive bearing testers and the AC drive motors) were awarded in July 1963. The motor generator set, the 750 KVA substation transformer, the liquid oxygen power transmission tester and load simulator, the power transmission console, the lube system, the AC drive motors, and electric drive bearing testers were received and installed. The bearing tester foundations, piping, and instrumentation trenches were poured.

Both of the miscellaneous installation packages (Specifications 6665 and 6680) as well as the power equipment room and primary power line (Specification 6695) were completed by the end of September 1963.

1963/4Q

The basic mechanical piping, equipment, and instrumentation package (Specification 6408A) was completed in October 1963 and activated during November and December 1963. Liquid oxygen Test Bays No. 1 and 2, and liquid hydrogen Test Bays No. 9, 10, and 11 were operational, as were the liquid oxygen, liquid hydrogen, gaseous oxygen, gaseous hydrogen, and helium supply systems. Also, the gaseous nitrogen pressurization system, and the process control instrumentation system became operational.

The vacuum dehydrator vessel and attendant piping (installed as part of Specification 6408A) was undergoing operational checkout and the basic data acquisition instrumentation system (Specification 6597) was completed in October 1963. Calibrations, checkouts, and system activations were conducted and an operational status for the basic systems was achieved by mid-November 1963.

Contractor installations of the steam power plant (Specification 6591) and the power transmission tester, bearing testers and support equipment, (Specification 6640) were completed in October 1963 and activation was accomplished in November 1963.

Activation and test buildup of the liquid oxygen bearing tester was completed. The initial liquid oxygen bearing test was conducted at the Cryogenics Laboratory Test Bay No. 1 facility on 26 November 1963. Three test runs were made, each at full load.

1964/1Q

Three intermediate program mileposts were accomplished during the first quarter of 1964 with the initiation of fuel seals testing, oxidizer power transmission assembly testing, and liquid hydrogen bearing testing.

The initial fuel seals test was conducted at the Cryogenics Laboratory (Test Bay No. 8) during January 1964.

Activation and test buildup of the oxidizer power transmission system was conducted and completed during January 1964. Oxidizer power transmission assembly testing was initiated in Test Bay No. 2 on 26 January 1964. Four subsequent oxidizer power transmission assembly liquid nitrogen tests of varying loads and durations were accomplished with this unit. The initial assembly unit was returned for development evaluation and preparations are being made for the installation and test of subsequent units.

Buildup and activation of the liquid hydrogen bearing test system proceeded concurrently with the oxidizer power transmission assembly system throughout January 1964. System activation and checkout was concluded on 3 February and the initial liquid hydrogen bearing test was run 7 February in Test Bay No. 10. Subsequent liquid hydrogen bearing tests were conducted on 18 and 26 February 1964.

1964/1Q

The transmission line check system fabricated by the AETRON Division and the analog tape recording system, purchased from Ampex, were received during March 1964.

The space simulation chamber installation in Test Bay No. 6 (Specification 6422A) was completed on 24 January 1964.

Final machining and testing of the helium start valve tank was completed at the vendor location in late January 1964. The start tank was delivered to Aerojet-General on 27 January 1964. Installation of the tank and attendant piping was immediately initiated and continued throughout February. Instrumentation installation and checkout was undertaken parallel to activation of the helium start system and test facility. With the exception of the catch vessel, facility completion was achieved in March 1964 and the first helium start valve test was conducted on 20 March 1964.

The installation of the vacuum dehydrator was completed by the end of the first quarter of 1964.

1964/2Q

Another intermediate program milepost was accomplished during April 1964 with the initiation of gas generator valve leak and actuation testing. Test buildup was completed and the initial test conducted on 2 April 1964. Eleven subsequent gas generator valve leak and actuation tests were conducted during April at the Cryogenics Laboratory, Test Bay No. 4. Other tests conducted during this same month included one liquid hydrogen seal evaluation, five liquid oxygen shaft riding seal evaluations, six liquid oxygen power transmission assembly accelerations, seven liquid hydrogen Conoseal evaluations, and two liquid hydrogen thrust chamber valve tests.

A total of 37 M-1 tests were conducted during May 1964. Of these tests, five were with the oxidizer power transmission assembly. Twelve fuel thrust bearing evaluation accelerations, 16 oxidizer face riding seal static cyclings and leak checks, and four liquid hydrogen static Conoseal evaluations were also conducted that same month.

Twenty M-1 tests were conducted during June 1964, three tests of the liquid oxygen seals, two of the liquid hydrogen bearings, six of the liquid hydrogen Conoseals, five of the thrust chamber fuel valve, and four of the heat transfer tube bundle.

Installation and checkout of the data acquisition system was initiated with the receipt of the transmission line check system and the analog tape system on 6 April 1964. The analog-to-digital data conversion system and the time code generator and tape search system were received on 15 and 29 April 1964, respectively. Installation and checkout of these items was completed by the end of June 1964.

1964/2Q

A study, initiated in March 1964, to evaluate spin test requirements and to determine modifications or additional capabilities needed to support greater requirements of the fuel and oxidizer turbopump assembly development programs was completed in April 1964. Design data and drawings were reviewed and cost estimates prepared on 24 April. Procurement specifications were also prepared for vendor quotations for the new drive turbine, slip ring assembly, heating device, and speed indicator. The design criteria were established and drawings started for the bottom damping device.

1964/3Q

The ADC range and filter system was delivered 17 July 1964. Installation and checkout of the equipment was completed by the end of the month.

The closed circuit television system arrived on 21 August 1964 and the factory lens changes and adjustments were incorporated. It was placed into operation during the fourth quarter of 1964 and right angle connectors were installed to reduce maintenance during the ensuing quarter.

During the third quarter of 1964, 162 tests were conducted with M-1 hardware at the Cryogenics Laboratory. A summary of these tests follows:

Test	July	August	September
Liquid Hydrogen Environment-Roller Bearing	13	14	2
Liquid Hydrogen Cold Cycle Leak-Conical Seal	18	9	5
Helium Start Valve	1	0	3
Heat Transfer and Tube Bundle Leak Test	8	12	0
Thrust Chamber Fuel Valve - Leak and Actuation	2	0	0
Liquid Oxygen Seals - Leak Test	0	14	6
Thread Sealant Program	0	0	28 (28)
Monthly Total	42	76	44

NOTE: Control System Laboratory Tests were not included in the above summary.

1964/4Q

Assembly and installation of the supplemental data equipment was completed during the fourth quarter of 1964.

1962

XI. FABRICATION FACILITIES

CHRONOLOGY

A special "industrial-pool" search list was prepared at the outset of the program for development-fabrication-facility items. Detailed justification forms were also prepared for each item. In addition, a plant layout for reorganizing existing facilities and integrating new facilities into the Development Shops was completed.

By the third quarter of 1962, purchasing negotiations for firm prices and delivery dates were in process for all items originally proposed except for the decontamination area. Only a lathe and a balancing machine were available from the industrial pool at that time. This activity completed Aerojet-General's search of the production equipment redistribution group facilities.

Appropriate PERT charts were completed at the end of the third quarter of 1962. These charts were used to coordinate the on-site date required for each new facility item and the tooling date with the hardware PERT charts.

Three of the 47 development fabrication items originally proposed were received by the end of 1962. These were the stake welder, the basic engine lathe, and the balancing machine. The basic acquisition schedule remained compatible with a revised program schedule which resulted from a redefinition of the NOVA requirement and schedule at that time.

The acquisition of equipment for the fabrication activities became a two-fold effort. Purchase orders for machine tools and equipment were submitted to NASA for approval and lists of industrial machinery and equipment were prepared for NASA/LeRC screening of the industrial reserves for surplus items.

1963

In the third quarter of 1963, all facility items had either been delivered, were on order, or purchase requisitions had been prepared except for the nozzle extension segment brazing furnace. Also, installation plans had been completed for all facility items. Gant charts had been prepared showing the schedule of the installation of the M-1 facilities as well as the relocation of existing facilities. Installation of items as they were received proceeded on schedule.

1962-1963

A. FABRICATION SHOPS

1. Sundstrand OM-3 Omnimil

A requisition and a purchase order for this equipment was prepared during the second quarter of 1962. The unit was scheduled for delivery during August of 1962. Also, three programers were sent to the Sundstrand Corporation for initial training with the Omnimil.

In the ensuing quarter, work orders were released for preparation of the Omnimil foundation. Delivery of the unit was re-scheduled to 15 October 1962 and maintenance personnel completed a training course for Sundstrand Omnimils.

By the end of 1962, the Omnimil, which was to be used for machining M-1 impellers, had been received and installed. Checkout was also initiated. Production personnel as well as additional programers received Omnimil training at the Sundstrand Corporation.

The checkout of the Omnimil was completed during the first quarter of 1963 and the machine was programed for operation.

Two problems with the machine developed during the fourth quarter of 1963. Investigation showed that either the tape reader was not functioning correctly or the electronics system was not digesting the tape readings correctly. Also, when the large cutters were used, there was a tendency for these cutters to pull out of the spindle. A thorough check of the reader and the electronics was made and units were changed as necessary. Also, a new gripper arm was installed on the spindle.

2. MacIntosh-Hemphill 246-in. Lathe

1962-1964

Aerojet-General personnel inspected this lathe in its original 60-in. swing condition during the second quarter of 1962. It was found suitable for modification to a 200-in. swing (subsequently made 246-in.) and formal construction drawings were prepared by the manufacturer for the necessary modification. This lathe was obtained from the Navy "pool."

The machine was received from the Industrial Reserve during the third quarter of 1962 and was placed on its foundation during the ensuing quarter. Installation of this lathe was completed early in 1964, the machine was rehabilitated, and successfully underwent proof tests prior to being considered fully operational.

3. Fabrication Building Addition

The steel framing and outer walls of this addition were completed during the fourth quarter of 1962 and occupancy commenced during mid-1963.

4. High Bay Fabrication and Assembly Area

1962-1963

Construction of a high bay addition to the Fabrication Shops was nearing completion by the end of 1962. In January 1963, there was a fire in an adjacent warehouse and the new high bay assembly area incurred some structural damage. As a result, occupancy was delayed approximately three months although this did not affect the fabrication program schedule.

By the end of the first quarter of 1963, it was decided to replace the stores and office building that was destroyed by fire with an additional high bay and office complex adjacent to the fabrication shop high bay then under construction.

Beneficial occupancy of both high bay structures was accomplished by the third quarter of 1963.

5. Balancing Machine

1962

A balancing machine was received from the Industrial Reserve Pool during the third quarter of 1962 and installed.

6. Pump and Valve Assembly Area

1962-1963

Construction of this area commenced during the fourth quarter of 1962 as part of the facilities realignment program. The area was completed during the ensuing quarter; however, occupancy was delayed approximately one month because it was used by personnel displaced by the fire discussed under the high bay.

7. Brazing Furnace

1963

At the outset of the program, the use of a quartz-lamp brazing furnace to satisfy thrust chamber requirements was considered. However, at the outset of 1963, brazing furnace studies of the vacuum brazing techniques were initiated. These studies were completed during the second quarter of 1963 and resulted in a purchase order with detailed specifications

1963

being prepared for a hot wall, bottom loading furnace for the thrust chamber. Also, specifications were completed for an integrally-heated ceramic mold brazing furnace for the nozzle extension segments.

The Ipsen furnace selected is discussed as part of the Welding Shop equipment.

8. Decontamination Facility

1963-1964

The specifications for this facility were completed early in 1963 and the AETRON Division had consolidated the procurement and installation into a complete subcontractor package by the second quarter of 1963. At the beginning of 1964, the facility design had been completed and an ultrasonic cleaner ordered. Construction continued throughout 1964 and by the end of the year, the construction was completed. An ultrasonic cleaner and vapor degreaser was installed, the facility was checked out, activated, and was considered operational.

9. Giddings and Lewis 120-in. Boring Machine

1963

This machine was installed and made operational during the fourth quarter of 1963. A 10,000 lb rotary positioner was also installed and made operational during this same period.

10. King Vertical Boring Mill

1963

This machine was installed and made operational during the fourth quarter of 1963.

11. Cincinnati Hypro 120-in. Milling Machine

1963-1964

Installation of this machine began at the end of 1963 and was completed early in 1964. The machine was rehabilitated, a fine feed attachment was installed, and the unit became operational during the third quarter of 1964.

12. DeVlieg Milling Machine

1963-1964

This machine was installed during the fourth quarter of 1963 and made operational during the ensuing quarter.

1963

13. Elox Drill

This equipment was modified, new components installed, and rewired during the fourth quarter of 1963.

14. Cincinnati No. 5 Vertical Mill

1963-1964

This machine was received during the fourth quarter of 1963. It was installed and made operational during the ensuing quarter.

15. Special Numerically-Controlled Thrust Chamber Drilling Machine

1963-1964

The components for this machine were received at the end of 1963. It was assembled and moved to its permanent location during the first quarter of 1964. Installation and activation were completed by the end of the second quarter of 1964.

16. Jones and Lamson Numerically-Controlled Turret Lathe

1964-1965

The foundation for this machine was completed early in 1964. A test piece was run using this machine at the manufacturers during the second quarter of 1964. The machine was then made ready for shipment to Aerojet-General, Sacramento. The lathe was received during the third quarter of 1964. It was installed and activation as well as checkout were initiated. The machine was considered operational during the first quarter of 1965.

17. Pratt and Whitney Numerically-Controlled Jig Borer

1964

The foundation for this machine was completed during the first quarter of 1964, during which time the unit was delivered in a damaged condition. Appropriate repairs were made and the machine was operational during the ensuing quarter.

18. Turbopump Assembly Room

1964

This area is located in the second high bay addition. Its construction was under way by the second quarter of 1964 and it was completed during the third quarter of 1964.

1964

19. Pacific Shear

The foundation for this equipment was completed during the third quarter of 1964 and installation commenced. The machine was installed and made operational by the end of 1964.

B. WELDING SHOP

All of the major items for the welding shop were on order and being fabricated by the end of 1963. Installation of the welding room had been started early in 1964 and it was in use by the end of the year.

1. Electron-Beam Welder

1962-1964

The development of electron-beam welding techniques began at the outset of the program. A purchase order for an electron-beam welder was placed with the Alloyd General Corporation. This company encountered extensive manufacturing problems and the purchase order to them was cancelled during the fourth quarter of 1964. Preliminary quotations and technical proposals were received from the Sciaky and the Hamilton Standard corporations during the ensuing quarter; however, it was decided not to pursue the purchase of this equipment.

2. Ipsen Brazing Furnace

1963-1965

Construction of the pit for the brazing furnace was under way by the end of 1963. All basic components for the brazing furnace, except for the retort, were received during the first quarter of 1964. Installation began immediately and checkout tests began during the second quarter of 1964 when the retort was received. This checkout was progressing satisfactorily when the retort collapsed during a test run in September 1964.

An analysis of the brazing furnace failure was completed by the end of 1964 and an informal report was received from Ipsen, the manufacturer, on 29 December 1964 wherein they delineated their plan to provide a braze furnace in complete accord with the provisions of the original purchase order.

The purchase order for a replacement retort was placed during the first quarter of 1965. This replacement was received during the third quarter of 1965 and installation was undertaken immediately. At the same time, base car rework was accomplished. This consisted of replacing the brick and thermocouple material. The pressure differential system was also reworked and a back-up system installed.

1964

3. Weld Boom Manipulator

This equipment was received during the third quarter of 1964. By the end of the ensuing quarter, it had been installed, checked out, and activated.

4. Induction Brazing

1964-1965

This equipment was ordered during the third quarter of 1964. The induction brazing unit generator was received and installed during the ensuing quarter. By the end of the first quarter of 1965, all of the equipment had been received, checked out, and activated.

C. FOUNDRY

1963

Five items were approved for the Development Shop Foundry. These were: core oven, core roll-over machine, skip loader, sandblast unit, and forklift truck.

The core roll-over machine was received during the second quarter of 1963. Three items were received during the ensuing quarter and the remaining one, the sandblast unit, was received during the fourth quarter of 1963. All of the items had been installed by the end of 1963.

D. HYDRAULICS TEST LABORATORY

1963

The facility design requirements as well as the testing criteria were established early in the program. Enlargement of the existing laboratory was started during the first quarter of 1963. An increased capacity water pump and modified piping system was constructed to accommodate the greater testing capabilities needed. Bids were also received for a 9000 gpm pump and motor.

During the second quarter of 1963, the water flow system was modified to establish testing stations under a 19-ft-6-in. crane hook height. These new test stations, located in Bay C, were activated within the limitations of the existing 8000 gpm system during the ensuing quarter. The 9000 gpm system was redesigned to eliminate all elements that were superfluous to the existing program schedules.

1963

By the end of 1963, the Water and Test Console had been installed and was operating. Also, the specifications for the redesigned 9000 gpm system were completed. The Data Monitoring consoles and the Power Supply Consoles were being fabricated and the overhead cranes had been installed.

1964

The Data Monitoring consoles and the Power Supply consoles were installed and operating during the first quarter of 1964. Also, installation of the dehydrator was started.

During the second quarter of 1964, the strainer for the 8000 gpm system was installed to complete the system, which then became operative.

Installation of the dehydrator was completed during the third quarter of 1964.

E. QUALITY CONTROL

Sixty-four items were originally requested for three major Quality Control areas (Receiving and Source Inspection, Development Shops Inspection, and Quality Control Laboratories). Fifty-nine of these items were approved and by the end of the third quarter of 1963, forty-five had been received.

1. Building 2002 Inspection

1963

All concrete work was completed in the quality control portion of High Bay L during the third quarter of 1963. The 10 ft by 12 ft surface plate and 84-in. rotary table had also been installed. The optical tooling was assembled and used to check out the first thrust chamber assembly.

2. Building 2002 Metallurgical Laboratory

1963-1964

Designs for modification of the building and installation of the equipment were completed during the third quarter of 1963. Necessary building modifications were completed during the first quarter of 1964. All of the equipment installations except for the high temperature furnace installation were also completed. The impact testing machine was received and the design of the impact testing facility was initiated.

1963

3. X-Ray Facility

Preliminary design for this new 300 KV facility were completed during the first quarter of 1963. All purchase orders had been placed by the end of the fourth quarter of 1963. The ultrasonic and film processing equipment had also been received.

1964

The Eddy current inspection equipment and the tube handling crane for the 2 MEV X-Ray unit were received during the second quarter of 1964. Also, the construction contract for the X-ray facility was awarded during this same quarter. Actual construction started during May 1964.

Construction of the X-ray facility was completed except for fencing and paving by the end of 1964. Also, installation of the 300 KV unit was accomplished. At the same time, installation of the X-ray film processing equipment began. The ultrasonic equipment installation was completed and activation as well as checkout was initiated.

1965

Activation of the 300 KV X-ray unit was completed during the first quarter of 1965. Activation and checkout of the ultrasonic equipment was also completed during this same time.

The remaining portion of the civil works construction, fencing and paving, was completed during the second quarter of 1965.

4. Receiving-Inspection

1963-1964

A design package for expanding receiving-inspection facilities in Building 2022 was released for bid on 6 August 1963. Bids were received and evaluated by the end of the third quarter of 1963. Necessary construction and relocation were under way in the fourth quarter of 1963. Modification was completed during the first quarter of 1964. All of the equipment installations had also been completed including surface plates, rotab, and the Pratt and Whitney measuring machine. This machine was being checked out for dimensional accuracy and the design as well as procurement of a temperature-controlled enclosure for it were proceeding on schedule.

The jig borer enclosure design was completed during the third quarter of 1964.

1964-1965

5. Liquid Oxygen Impact Facility

The design for this facility was completed, bids were received, and construction was started during the third quarter of 1964. This construction was completed during the fourth quarter of 1964 and the installation of the testing equipment was started. This was completed during the first quarter of 1965.

6. Alternative Immersion Fixture

1964

This fixture was completed during the third quarter of 1964 and installation was accomplished.

7. Fluorescent Penetrant Inspection Facility

1964-1965

This equipment was received during September 1964 along with bids for its installation. This installation was completed during the fourth quarter of 1964 and activation as well as checkout were started. This was completed during the first quarter of 1965.

XII. HYDROGEN HEAT TRANSFER FACILITY

CHRONOLOGY

1963

Design and preprocurement activities for this facility were completed during the second quarter of 1963. M-1 operating facilities as well as special test equipment items were designed during the third quarter of 1963. During the ensuing quarter, a comprehensive test program was prepared to investigate the parameters required for the design of the nozzle and thrust chamber.

By the end of 1963, the instrumentation trench had been completed and work started to install the high pressure hydrogen transfer line as well as the electrical system leading from the power supply to the test apparatus.

1964

In January 1964, the gaseous hydrogen receiver, the liquid hydrogen run vessel, and the test section chamber were installed. Also, instrumentation lines were laid in the instrumentation trench connecting the test apparatus with the control panel.

The four 50 KW power supplies were located in the power supply shed and the bus bars, which connect the test apparatus with the power supply, were completed in February 1964. Also, piping which connected the test apparatus with the high pressure gaseous hydrogen supply in the Cryogenics Laboratory was completed. The majority of the high pressure valves were installed and the hydrogen vent line was completed.

The facility itself was completed during March 1964 and became operational during the second quarter of 1964. At the same time, a program was prepared for using this facility to obtain data relative to cooling with turbine exhaust gases.

XIII. GENERAL SUPPORT FACILITIES

CHRONOLOGY

A. RECEIVING WAREHOUSE

1962/1963

Construction expansion of this facility (Building 2022) was undertaken at the outset of the program. It was completed on 21 March 1963.

B. RECEIVING INSPECTION

1963/1964

The expansion of this facility in Building 2022 was started during March 1963 and was completed during the first quarter of 1964.

C. ENGINEERING BUILDING

1962/1963

Construction of this additional facility (Building 2025) was under way when the M-1 Program was initiated. This 50,000 sq ft building included accommodations for M-1 Program design and engineering personnel. Beneficial occupancy of this building was accomplished during April 1963.

XIV. MATERIALS

SUMMARY

The materials technology developed in the M-1 Program has been summarized in a separate report.(149) In addition, specific aspects of the materials research effort have been individually documented as applicably noted in this report; therefore, this discussion is limited to brief descriptions of the specific efforts and the technical detail presented in these other reports is omitted.

CHRONOLOGY

A. ENGINE SYSTEMS

1. Localized Aging of Inconel 718(150) Lines

1963/1964

This task started during the fourth quarter of 1963 and continued through the second quarter of 1964. Its objective was to develop a selective heating device and to establish a procedure for locally-aging Inconel 718 weld joints. Accomplishments are delineated in the referenced report.

2. Hard Surfacing 18% Nickel Maraging Steel for Ball and Socket Assembly

This task was performed to develop wear and corrosion-resistant coatings to be applied on 18% nickel maraging steel for use as M-1 ball and socket assemblies. It spanned the period from the fourth quarter of 1963 through the fourth quarter of 1964, at which time it was discontinued because of a material change in the ball and socket assembly. No significant results were obtained.

3. Investigation of Brazed(151) and Welded Separable Tube Connectors

1963/1965

This effort commenced during the fourth quarter of 1963 and was completed during the first quarter of 1965. A program plan covering the

(149) Janser, G.R., Summary of Materials Technology of M-1 Engine, NASA Report No. CR 54961, 22 July 1966

(150) Inouye, F.T., Hunt, V., Janser, G.R., and Frick, V., Application of Alloy 718 in M-1 Engine Components, NASA Report No. CR-788, June 1967

(151) Hunt, V., Induction Processed Separable Tubular Brazed Connectors, NASA Report No. CR-515, July 1966

brazing and welding of separable tube connectors was written. Part I of this plan consisted of investigating the feasibility of brazing, testing, disassembly, and re-brazing as well as retesting small diameter (1/4-in. to 1-in.) tubes a number of times on stainless steel material. Induction heating was to be used as the heat source. Part II of this plan consisted of welding large diameter separable tube connectors (up to 15-in.) disassembling, and re-welding them a number of times.

In summary, methods for producing separable tube joints, other than mechanical, were investigated for application of these joints to the M-1 engine lines. The induction process was selected for brazing the separable tube connectors. Special plier-type induction brazing tools were made to permit the brazing of the M-1 thrust chamber transition tube joints to the fuel torus. An 82% gold-18% nickel brazing alloy was used with a brazing temperature range of 1900°F to 1950°F. The tubes and sleeves in this application were of 0.032-in. wall AISI, Type 347 stainless steel.

4. AISI 9310 Alloy Steel Weldments for Cryogenic Applications

1964

This task was performed to obtain data regarding the cryogenic toughness of AISI 9310 weld joints. It was initiated at the outset of 1964 and lasted into the third quarter of the same year.

Because of the combination of low cost and acceptable cryogenic toughness, AISI 9310 alloy was selected as a material for use in the fabrication of support struts for various M-1 component test applications(152). The material was not made in wire form and there was little information regarding filler rods that could provide the required low temperature toughness. Available information indicated that there was a ductile-brittle transition below a temperature of -200°F, which was the limit for the Class E, 8018 C-2 (ASTM-A 316-58T) electrode to be used with the AISI 9310 weld specimens.

5. Mechanical Properties of 18% Nickel Maraging Steel Ball and Socket Bearings for Discharge Lines

This task was performed during the fourth quarter of 1964 to determine the mechanical properties of 18% nickel maraging steel balls, P/N 279974, at room temperature, -323°F, and -423°F.

(152) Silvus, J.A., Strength of AISI 4130 and AISI 9310 for M-1 Engine Tripod, Aerojet-General Report No. PVR 64-207, 13 March 1964

1964

Tensile specimens were machined from the longitudinal and transverse directions of two balls and tested at room temperature, -320°F, and -423°F. The room temperature test results showed that, although this material was not purchased in accordance with Specification AGC-44092A, the mechanical properties exceed the requirements of this specification in every respect except reduction in area, which was slightly under the required level. The cryogenic test results verified that this material had high strengths and good ductility at -320°F, but at -423°F the ductility was inadequate.

6. Notch-Toughness of As-Welded Heat-Treated Inconel 718 (153)

1964/1965

This task was started during the fourth quarter of 1964 and only limited work had been accomplished when effort was stopped during the ensuing quarter. Its objective was to determine the mechanical properties of as-welded, heat-treated Inconel 718 at room temperatures, -320°F, and -423°F. The material was heat-treated in accordance with AGC-46604.

Because Inconel 718 discharge line welds were not to be locally aged, tests were conducted to determine the mechanical properties that could be expected from as-welded heat-treated Inconel 718 (i.e., aged material was welded and tested in the as-welded condition).

7. Gas Generator 3-1/2-In. Line Bellows Assembly and Valve Bolts Failure Analysis

1965

The objective of this investigation, which was conducted during the second and third quarters of 1965, was to determine the effect, if any, of metallurgical factors in the failure of tie-bolts (P/N 250469-11) and tripods (P/N 280039-3) during proof testing of a 3-1/2-in. M-1 line with liquid nitrogen.

Three 3-1/2-in. diameter flexible couplings were assembled into a "U" shaped line for testing with liquid nitrogen. During pressurization, the test assembly failed severing the line in several places. Examination of the failed assembly showed all eight tie-bolts broken, at least one of each set of tripods torn from its tube assembly, two of the bellows ripped, and the convolutions unfolded from all three bellows.

(153) NASA Report CR-788, op.cit.

1965

Pieces of the line assembly were examined relative to the failure of the bellows tripods. Metallographic examination of the 17-4 PH stainless steel bolts and Inconel 718 tripods indicated both materials to be normal. Marginal weld quality (poor penetration at the bellows liner to tripod by joint) was found and could have contributed to the failure. Hardness surveys of all parts indicated the heat treatment and attendant strength requirements were met. It appeared that the use of 17-4 PH stainless steel at liquid nitrogen temperatures was also a contributing factor to the failure. This material was inherently brittle at cryogenic temperatures.

Visual examination of the valve bolts and clincher nuts showed severe distortion of the bolts and nuts which occurred through a combination of over-torque and the use of an oversize bolt clearance hole on a flange during assembly. It appeared that this condition was the primary cause of failure.

B. THRUST CHAMBER ASSEMBLY

1. Development of Brazing Process for the Coaxial Injector

1963/1964

The objective of this task was to develop the technique for brazing the swirler-element assembly and then to braze this assembly into the injector face. The effort was initiated toward the end of 1963 and continued through the third quarter of 1964.

The braze material, O.F.H.C. copper was plated into the element (approximately 0.0008-in. in radius). The element was then placed into the injector body with additional copper-wire preform placed at the injector element junction. The brazing cycle was 2030°F to 2050°F for 10 to 15 minutes at temperature in a dry-hydrogen atmosphere.

This process was used to prepare and braze joints simulating the production part. The simulated injector was visually inspected, pressure-tested, and cross-sectioned. Visual inspection revealed good filleting on either side of the element-injector joint. Pressure tests revealed no leaks at 300 psig using liquid nitrogen as the pressure media. Metallographic inspection revealed good braze coverage, nominal diffusion, and negligible porosity.

1963/1964

2. Braze Alloy Development(154)

This task started during the fourth quarter of 1963 and continued into the third quarter of 1964. Its objective was to develop a low cost, low vapor pressure braze alloy to meet the fabrication and service requirements for the M-1 combustion chamber. The selected braze materials (copper and Nicoro) had inherent disadvantages in that copper had a high brazing temperature and was sensitive to joint clearance while Nicoro was expensive.

Eight copper-base braze alloys were developed by Aerojet-General for this program. Those eight braze alloy systems, plus an additional four commercial braze alloy systems underwent preliminary testing. Mechanical property data for these 12 braze alloys were obtained for ultimate-shear strength and ultimate-tensile strength at room temperature and -423°F.

Upon the basis of the results of these tests, candidate braze alloys were reduced to three alloys: AGC 200, AGC 201, and Anaconda 651. Additional effort with these alloys was undertaken as part of the thrust chamber assembly braze alloy and process development task.

3. Thrust Chamber Assembly Braze Alloy and Process Development

1963/1965

The purpose of this task was to select a suitable braze alloy and to establish a furnace-brazing procedure for the M-1 thrust chamber. It was based upon the results of the materials research efforts undertaken as part of the braze alloy development task as well as work in connection with the selection of a suitable, commercially-available alloy.(155) Therefore, in effect this was an effort that began towards the end of 1963(156) and continued into early 1965.

Because the production furnace was not available, wet and flow studies of AGC-200, AGC-201, and Anaconda 651 alloys were conducted using 10-in. long tubular specimens instead of the full-scale 14:1 tube sample as

(154) Gustafson, K.L., Development and Evaluation of Braze Alloys for Vacuum Furnace Brazing, NASA Report No. CR-514, July 1966

(155) Ibid.

(156) Gustafson, K., Program to Evaluate Braze Alloys for Furnace Brazing the M-1 Exit Nozzle in Dry Hydrogen, Dry Argon, or Combination of Both, Aerojet-General Report No. DVR 63-549, 10 October 1963

originally planned. Particular attention was to be directed towards determining the consistency with which the braze alloys responded to controlled conditions of heating rate, temperature, time at temperature, and atmosphere. The investigation was also to include a study of the effects of controlled volumes of braze alloy upon erosion and undercutting of tube walls after flowing down measured tube lengths. However, these wet and flow studies were stopped during the first quarter of 1965 as a result of M-1 Program redirection.

4. Inconel 718(157) Plate for the M-1 Thrust Chamber Jacket

1964

This task was conducted during the first quarter of 1964 to establish the optimum heat treatment for the thrust chamber jacket by heat treating and testing Inconel 718 plate that was used to fabricate the jacket. The material, which involved several heats, was supplied by two mill sources.(158)

5. M-1 Thrust Chamber Assembly Tube-Bundle and Jacket Fit-Up Problems

The objective of this task, which was started during the second quarter of 1964, was to investigate and determine nonmetallic materials suitable for filling the space between the thrust-chamber tube bundle and the thrust chamber jacket if fit-up problems should occur.

An epoxy adhesive, X-Epon 99-105-1, which had been made flexible appeared to be the most promising of the four materials investigated. The lap-shear strength of aluminum-to-aluminum bonds approached 1500 psi at room temperature and exceeded 2200 psi at -320°F. Compressive strength of the 300°F-cured material was measured at 8700 psi, at room temperature, and at 63,000 psi at -320°F.

Tensile shear testing of X-Epon 99-105-1 resulted in cohesive failure at 1200 psi for room temperature pulled specimens (68°F) and adhesive failure at 2400 psi for specimens tested in liquid nitrogen (-320°F). These specimens were prepared from abraded and solvent-cleaned Inconel 718 coupons bonded with X-Epon 99-105-1 cured for 30 minutes at 160°F plus one hour at 300°F.

(157) Aerojet-General Report No. 8800-37, op. cit.
(158) Silvus, J.A., Inconel 718 Plate for the M-1 Jacket, P/N 293904-9, Aerojet-General Report No. DVR 64-198, 11 March 1964

Work in connection with the evaluation and bond strength of Teflon FEP and Mylar/aluminum foil composites was discontinued by the end of 1964 as a result of a jacket design change and no additional effort was expended in this task.(159)

6. Glass-Fiber Reinforcement of the Thrust Chamber Tube Bundle

The objective of this task, which was started during the second quarter of 1964, was to investigate the methods for reinforcing the thrust chamber tube bundle with glass-fiber compositions. The task was limited to a literature survey and subsequent study of information. Also, property data was collected.

The concept of glass-fiber reinforcement of thrust chambers had already been proven in the Titan I and Atlas Programs. Advantages (strength to weight ratio, ability to match nonuniformities, etc.), disadvantages (difficulty in varying wrap pattern, strength lower than metal strength, etc.), anticipated problems (full scale testing of glass-wrapped thrust chambers, nondestructive quality assurance testing of production hardware, etc.) and other aspects with respect to M-1 engines were studied. Initial results of the study indicated that glass-fiber reinforcement could provide both axial and circumferential strength. A literature survey for property data of the various candidate glasses (E-HTS, N-672, S-994, YM 31A, X-37B, and X-815) revealed that cryogenic property data was not then available (only extrapolations). However, in view of the favorable results with Titan I and Atlas, it may not have been necessary to actually perform material property tests at cryogenic temperatures.

The literature survey and collection of available property data were completed. However, in accordance with the decision to utilize wire wrapping to reinforce the tube bundle, further work under this task was cancelled during the fourth quarter of 1964.

7. Stress-Rupture Properties of High Temperature Alloys

1964/1965

This task was performed to establish stress-rupture properties for thin-gage materials for thrust chamber application. The stress-rupture strength of 0.010 gage Hastelloy X and C was to be determined at 1800°F to 2000°F. Work began during the third quarter of 1964 and was terminated during

(159) Kertulla, Jr., E.F., Epoxy Adhesive Development for Thrust Chamber Assembly Jacket Fit-Up and Wire-Wrap Support, Aerojet-General Report No. DVR 65-124, 4 March 1965

the first quarter of 1965 as a result of M-1 Program redirection. No results were obtained.

8. Induction Brazing of Fuel Torus Combustion Chamber Tube Connectors (160)

1964

The purpose of this task, which was accomplished during the second quarter of 1964, was to evaluate the induction brazing process for joining fuel torus tubes and to coordinate the tooling acceptance of procured equipment.

A special brazing tool, designed and built by the Aeroquip Corporation, Jackson, Michigan, was purchased to braze the M-1 torus to the chamber tubes. Ten tube joints, representative of the chamber, were brazed consecutively and tested by the vendor. Three additional joints were brazed for evaluation at the Aerojet-General Sacramento Plant. (161)

9. Heat Treatment of Inconel 718

The objective of this task, accomplished during the fourth quarter of 1964, was to determine the minimum cooling rate from a solution-annealing temperature that would not adversely affect final mechanical properties of Inconel 718.

10. Investigation of Transpiration-Cooled Injector Faceplates and Baffles (Rigimesh

1965

The objective of this task was to determine the effect of shop processes on the flow characteristics of Rigimesh fabricated into injector face plates and baffles. It was started during the second quarter of 1965 and terminated during the ensuing quarter as a result of M-1 Program redirection.

(160) NASA Report No. CR-515, op.cit.
(161) Hunt, V., Evaluation of Induction Brazed Tube Joints, Aerojet-General Report No. DVR 65-206, 11 June 1965

11. Electron-Beam Welding of Rigimesh Baffle Test Specimens

This investigation was conducted during the second quarter of 1965 to evaluate the feasibility of fabricating Rigimesh baffles using electron-beam welding process.

It was concluded from this investigation that additional development of electron-beam welding of Rigimesh was required to produce sound joints.

12. Carburized Injector Investigation

This investigation was conducted during the third quarter of 1965 to determine the extent and seriousness of the carburization of M-1 injectors S/N 012 and S/N 020, as well as to attempt to decarburize the units.

The source of carburization was investigated and traced to a graphite block which had been used as a base for the injector during dry hydrogen brazing and annealing cycles performed at a vendor's facility.

Bend tests were performed using 1/2-in. thick, pack carburized CRES 347, bend specimens at -320°F to determine the effect upon ductility. Bend ductility was seriously reduced, varying from 3 degrees to 20-degrees at -320°F depending upon the extent of carburization.

Decarburization and diffusion was accomplished in laboratory tests utilizing relatively dry hydrogen (-40°F to -60°F dew point). The extent of decarburization and diffusion was dependent upon the time held at 1950°F and the carbon concentration in the austenite matrix. Pack carburized 1/2-in. bend and 0.040-in. tensile specimens subjected to 1950°F in dry hydrogen for various periods up to 20 hours exhibited improved bend and tensile ductility. Metallographic specimens exhibited surface decarburization, removal of a definitive case, and diffusion evidenced by light network carbides to depths of up to 0.100-in. below the surface.

The S/N 012 injector was subjected to a 1950°F/20 hour decarburization treatment in dry hydrogen at the vendor's plant. The graphite support plate was encapsulated in a CRES 304 stainless steel welded container to preclude further carbon contamination. The injector was significantly improved. More heavily pack carburized bend and tensile specimens exhibited minor decarburization with a significant improvement of a tensile elongation and bend ductility.

The S/N 020 injector, which had already been brazed, was decarburized at 1930°F followed by a 2035°F copper rebraze cycle for a total exposure time of 20 hours. Additional copper was added to the coaxial elements to assure adequate copper at the braze joints. Again, ductility, as evidenced by controlled bend and tensile tests, showed significant improvement.

13. Ablative Liner for the Workhorse Thrust Chamber

The objective of this effort, which was initiated during the third quarter of 1965, was to develop an ablative chamber liner capable of sustaining multiple firings which would be utilized during the M-1 injector performance evaluation.

The ablative liner material incorporated high silica fabric reinforcement with an advanced "high char strength" phenolic resin system. The planned duty cycle, requiring a refire capability, imposed conditions that demanded high reliability of the liner. The material designation was WBC-2230.

The design of the liner assembly consisted of three major subassemblies; the torus section, chamber section, and exit cone. The torus section ablative subassembly began at the injector face and extended the full length of the fuel flange and torus section. Liner wall thickness varied from approximately 1/2-in. to 1-1/2-in. The chamber subassembly consisted of the convergent area and the throat. Beginning at the forward flange, the liner wall was 1-in. thick and extended aft of the throat centerline approximately 1-in. where the liner thickness was 1-3/4-in. The exit cone subassembly started at the chamber subassembly interface and extended to a 2:1 expansion area at the aft end of the steel workhorse chamber. The liner wall thicknesses varied from 1-3/4-in. to 1-1/4-in. at the extreme aft edge.

The entire ablative assembly was of the same material and was bonded at the subassembly interfaces with silicone adhesive/sealant.

14. S/N 025 Gas Generator Injector Failure Analysis

The objective of this task was to evaluate the cause of failure of S/N 025 gas generator during its fourth hot gas firing. It was accomplished during the fourth quarter of 1965.

S/N 025 gas generator failed during test run 1.2-05-EHG-006, its fourth hot gas firing, after accumulating a total run time of 22.4 sec. Failure occurred at the weldment joining the injector liquid oxygen torus to the center support stud. Visual examination revealed that fracturing initiated at an area of poor weld penetration. This area was located next to the torus inlet which limited joint accessibility for the weldor.

There was no evidence that failure was caused by parent metal deficiencies. Secondary parent metal failures were ductile as evidenced by pronounced plastic deformation and shear fractures. Spectrographic analysis of the parent metal showed it to be N-155, the specified material. The microstructure of the parent metal was normal. The cause of failure was attributed to the localized, inadequate, weld penetration.

C. TURBOPUMP ASSEMBLY

1. Notch Sensitivity of Aluminum Alloys

1963/1965

This task was initiated during the fourth quarter of 1963 and continued through the second quarter of 1965. Its objective was to perform an engineering assessment of the notch sensitivity of Aluminum Alloy 7079-T652 at temperatures of 68°F, -320°F, and -420°F.

Large 7079-T652 hand forgings were evaluated for pump impeller and inducer applications. The results of investigating the properties of large 7079 aluminum alloy forgings in a cryogenic environment are detailed in a separate report(162), which includes the results of mechanical property tests, reheat-treatment experiments, and microstructural studies.

2. Investigation of Dimensional Stability of Type 440C Stainless Steel

1963

The objective of this task, which was conducted during the fourth quarter of 1963, was to establish the dimensional stability of 440C bearing material when subjected to liquid hydrogen temperatures.

Roller bearings of type 440C Stainless Steel were dimensionally checked and tested for retained austenite content. They were soaked for two hours at -100°F, 320°F, and 423°F, and dimensionally measured. Retained austenite was determined by X-ray diffraction. The width increased from 0.001184-in. to 0.00197-in. after soaking, but the diameters were unchanged. This was attributed to the transformation of retained austenite to martensite in the heat-treated 440C material and to the orientation of the retained austenite in the rolling direction of the original bar.

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(162) Inouye, F.T., Properties of Large 7079 Aluminum Alloy Forgings in a Cryogenic Environment, NASA Report No. CR-513, July 1966

It was found that stability could be improved by subjecting the bearings to extreme cryogenic temperatures. Literature indicated that this treatment should be applied as soon as possible after quenching during heat treatment.

3. Examination of LW-5 Bonding to Inconel X

The purpose of this task, which was accomplished during the fourth quarter of 1963, was to determine the quality of Linde flame-plated LW-5 wear-resistant coating which had failed in service.

The coating system appeared to have a good metal bond, and the coating-base metal interface did not show signs of rupture. However, the coating system did indicate a high degree of porosity and inter-porosity cracks were noted in the wear area of the rotating sleeve. Failure was attributed to the high degree of porosity which ranged from 12% to 18% porous.

4. Inconel 718 Heat Treatment and Properties

1963/1965

The purpose of this task was to determine the effect of heat-treatment and other variables upon the mechanical properties of Inconel 718. This effort spanned the period from the end of 1963 through the third quarter of 1965.

The experience gained in connection with the use of Inconel 718 on the M-1 Engine has been summarized in a separate report(163).

5. Development of TIG Welding Procedures for the M-1 Fuel Pump Rotor

The purpose of this task was to assist in developing welding procedures for simulated pump rotor joints as well as to evaluate these welds. This work was started during the fourth quarter of 1963, but it was temporarily suspended during the third quarter of 1964 because of the press of higher priority needs. The effort was resumed during the first quarter of 1965 and completed the following quarter.

The TIG welded turbopump shaft joint was evaluated using the ultrasonic inspection process. It was also inspected by means of the dye penetrant and radiographic methods. Weld joints simulating the pump rotor barrel and end forging joint were completed and weld procedures were established.

(163) NASA Report No. CR-788, op. cit.

1963/1965

Although the welds appeared to be sound, radiographic inspection showed indications of internal cracks or lack of fusion. After being sectioned, the welds were found to lack fusion in the second weld weave passes. More joints were welded and welding procedures were modified to overcome the lack-of-fusion condition.

One set of Inconel 718 ring forgings in the solution heat treated condition were TIG welded using the same welding procedure as established for welding the M-1 pump rotor. This weldment was aged in accordance with Specification AGC-46604 after welding.

Metallurgical examination(164) of the first weld sample simulating the rotor joint revealed shrinkage cracks in the center of the weld. A second sample was welded, incorporating schedule modifications intended to eliminate the cracking, and was X-ray and ultrasonically inspected prior to metallurgical investigation.

Metallurgical examination(165) of the second weld sample, which was welded incorporating weld schedule modifications to eliminate shrinkage cracks found in the first weld sample, was completed. Shrinkage cracks were found in the second weld sample; however, the number and severity was reduced to a minimal amount. All defects found were intentionally left in the sample rather than removed through repair so that they could be identified.

Serial No. 4 rotor was successfully welded using the weld schedule developed for the second weld sample.

6. Electron-Beam Weld Study of the M-1 Fuel Turbine Rotor Shaft

1963/1964

The objective of this task, which was initiated during the fourth quarter of 1963 and completed in the third quarter of 1964, was to evaluate the efficiency of the joint (Rene' 41 to Inconel 718) that joined the M-1 turbine stub shaft to disc, and the joints (Inconel 718 to Inconel 718) of the disc-to-vanes. The information obtained was to be used to develop a process specification. That portion of this program pertaining to the disc-to-vane joints was discontinued on 11 November 1963 as these were to be separately evaluated.

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(164) Janser, G.R., Fuel Pump Rotor - Weld Development Task Force Metallurgical Investigation, Aerojet-General Report No. DVR 65-167, 31 March 1965
(165) Ibid

1963/1964

Two electron-beam weld specimens (Rene' 41 to Inconel 718) were received from the vendor for evaluation. Both specimens were inspected by radiographic, dye penetrant, and ultrasonic methods. Specimen "A" was solution-treated and aged in accordance with AGC Specification No. 46604, and standard and notched tensile test specimens were machined from the segments. Hardness readings taken of the Rene' 41, the weld, and the Inconel 718 were in the range of R_c = 39-41 which indicated both materials responded to the thermal treatment.

Specimen "B" was welded by the electron-beam process and the "U" groove was filled by the TIG welding process using Inconel 718 weld wire. Dimensional checks were made before and after the electron-beam and TIG welding to determine linear shrinkage. Both specimens received the same thermal treatment. Specimen "B" was sectioned for further machining into standard and notched test specimens.

The prototype rotor weldment, P/N 286169, S/N 0002, was electron-beam welded by a vendor. The joint that was used to develop the electron-beam welding procedure simulated the prototype joint and was cross-sectioned for metallographic examination. Minute intergranular cracks were detected in the parent metal adjacent to the bottom portion of the "nail head" of the weld nugget. This production-weld joint had a 20-degree angle from normal to the axis of the shaft. The angular joint was necessary to obtain the required focal length because of interference of the disc.

The weld joint used in the development program was normal to the axis of the shaft. Metallographic examination showed these welds to be sound. It was concluded that the angular (20-degree) weld joint imparted greater stresses and thus contributed materially to the cracking.

7. Thermal Expansion of Turbopump Assembly Materials(166)

1963/1965

Discrepancies in expansion values found in reports and general literature indicated a need for a review of published information and the selection of single design values. Therefore, this task was initiated during the fourth quarter of 1963 to obtain additional information about the thermal expansion of Inconel 718, Rene' 41, AISI Type 440C, and Invar 36 between ambient temperature and -423°F. The effort was completed early in 1965.

The thermal expansion values evolved in this investigation were used in the design of the turbopump assembly.

(166) Frick, V., Thermal Expansion of Turbopump Assembly Materials, Aerojet-General Report No. DVR 65-146, 2 March 1965

8. Electron-Beam Welding of the Oxidizer Turbine Rotor

This task was performed to develop parameters for welding Inconel 718 blades to the Inconel 718 turbine disc of the M-1 oxidizer rotor. Work began at the outset of 1964 and continued through the third quarter of 1965. Development was conducted in two phases. In Phase I, various designs for electron-beam weld joints of vane-to-disc and vane-to-shroud were explored. The results of Phase I were used in Phase II during the designing and building of tooling to accommodate the designs of production hardware.

Various design approaches for joining the blades to the disc were followed. The final decision was to slot the disc to the contour of the blade. This allowed the blades to be inserted and then to be fused to the disc by electron-beam welding. Also under consideration were various design approaches to determine a means for joining the outer shroud to the blade ends. The final decision was to slot the shroud to the contour of the blade, plus the blade end, insert the plugged end into the shroud, and then electron-beam weld the joint.

Various blade-to-shroud and blade-to-disc joints were welded by the electron-beam process. The initial simulated-weld joints indicated the need for simplifying a weld-joint design for the oxidizer turbine-rotor assembly. The final approach was to groove the disc, insert the blade, and then weld the blade ends to the disc with the beam oriented normal to the blade surfaces. The blade-to-shroud joint was accomplished by inserting the blade into the shroud slot, then joining by electron-beam welding in a similar manner. The blade-to-disc weldment was supplemented with a braze fillet. Mechanical tests were conducted to establish joint properties.

Phase I was completed during the third quarter of 1964 and the final report for the electron-beam welding of the oxidizer rotor was issued.(167)

Tooling was designed and built to accommodate the detailed parts for electron-beam welding. Also, electron-beam welding schedules to weld the blade-to-disc and blade-to-shroud were established.

A four-bladed test specimen was electron-beam welded to verify the welding parameters established prior to welding the production rotors at Thompson-Ramo-Wooldridge. The test specimen was sectioned by Thompson-Ramo-Wooldridge; half of the specimen was retained for their weld evaluation and the

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(167) Hunt, V., M-1 Oxidizer Turbine Rotor Electron-Beam Welding, Aerojet-General Report No. DVR 64-363, 21 July 1964

remainder of the specimen was forwarded to Aerojet-General for evaluation. Examination of the joints at high magnification revealed sound welds with full braze coverage.

A three-vaned specimen duplicating the M-1 oxidizer turbine nozzle was electron-beam welded and brazed by Thompson-Ramo-Wooldridge. Full weld penetration was not obtained through the shroud which was 3.6-in. thick. Melting of the vane trailing edge occurred during the welding operation. Two electron-beam weld passes were made to produce the wide weld nugget. The specimen was brazed prior to sectioning. Examination of the joints at high magnification revealed sound welds with full braze coverage with the exception of the area of vane ends.

This completed the welding development work in this task.

9. Evaluation of the Rene' 41 Turbine Shaft Blank

1964

This task was conducted during the second quarter of 1964. Its purpose was twofold. First, the mechanical properties of the Rene' 41 turbine shaft blank, P/N 281624, S/N 2, were established and then from the test results, the cause for the discrepancy in mechanical property results reported by the forging vendor (Wyman-Gordon Company) and Aerojet-General Quality Assurance was determined.

As a result of this investigation, final recommendations to minimize property discrepancies were formulated. The recommendations to minimize test variable discrepancies were to test at equivalent strain rates, obtain test specimens from corresponding locations, and finish-grind the specimens to an 8 rms finish. The recommendations to obtain higher properties was to require a finer-grain size product. The desired grain size was ASTM 5 and finer for Rene' 41.

10. Subscale Fuel Inducer, P/N 286456-7, S/N 0000005, Failure Analysis

The purpose of this task, which was accomplished during the second and third quarters of 1964, was to determine the causes leading to the cracking of vanes in the 7079-T652 subscale inducer during Tests No. 1.2-02-EMP-001-009. The material, 7079-T652 aluminum alloy forging, was normal and conformed to specification requirements.

A fracture analysis of Vane No. I was performed. This vane failed at 8000 rpm in deep-cavitating testing. The failure was attributed to mechanical fatigue that started at the high pressure surface and was propagated by predominantly a tensile-shear mechanism to the low pressure surface. Fracture patterns were verified by electron microscopy.

Analysis of Vane No. II identified the cause of its failure to have been mechanical fatigue that started on the high-pressure surface of the vane and propagated to the low-pressure side of the vane. Crack propagation continued under tensile loading.

The inducer was examined after vibration testing and Vane No. III was found to be fractured. Fracture analysis indicated a failure mode similar to that observed in Vanes No. I and II.

This analysis demonstrated that 7079-T652 aluminum alloy did not possess adequate fatigue strength under the indicated testing conditions.

11. Evaluation of First-Stage Fuel Inducer A-110-AT-ELI Wrought Titanium Alloy

1964/1965

The purpose of this task, which was initiated during the second quarter of 1964, was to evaluate the properties of the A-110-AT-ELI titanium forging, procured from the Taylor Forge Company, of a large-section size required for the M-1 fuel inducer (P/N 289997). This was back-up for the aluminum alloy 7079-T652 fuel inducer. However, during the first quarter of 1965, this effort was made part of the fuel pump modification program.

12. Evaluation of Second-Stage Fuel Inducer A-110-AT-ELI Wrought Titanium Alloy

The objective of this task was to perform a metallurgical analysis of the second-stage fuel inducer, P/N 289536, fabricated of A-110-AT-ELI titanium alloy, and to determine, for design analysis, the mechanical properties of the alloy at ambient and cryogenic temperatures. Effort was initiated during the second quarter of 1964 and completed early in 1965.

Smooth and notched (K_t of 6.3) tensile specimens of radial and tangential orientations were machined from the second-stage fuel inducer forging and tested for properties at room temperature and -423°F. The forging conformed to specified minimum tensile property requirements at room temperature as required by the drawing. The forging was found to be notch-tough, at room temperature, based upon notched-tensile and notched-yield criteria. The data indicated that the ultimate and 0.2% offset yield strengths increased with decreasing temperature as expected. At -423°F ductility, as measured by elongation and reduction of area, was low for this part; this forging was notch-sensitive, at notch-acuity of K_t to 6.3, based upon the same notched-tensile and notched-yield criteria. The microstructural examination revealed the presence of titanium hydride (Ti H_4) in the forging at -423°F. The hydrogen content was 230 to 240 ppm in the forging. The maximum recommended hydrogen content for ELI A-110-AT titanium is 150 ppm. The interstitial levels for iron, oxygen, nitrogen, and carbon were 0.13%, 0.103%, 0.017%, and 0.08%, respectively.

13. Evaluation of Cast Titanium Alloy First-Stage Fuel Pump Inducer

The objective of this task, initiated during the second quarter of 1964, was to evaluate the feasibility of using a titanium casting of alloy A110 at ELI for the M-1 fuel inducer. Progress in the field of casting titanium indicated that the then current technology was to the point where large complex parts, centrifugally cast under vacuum, were available on a limited commercial basis. Tests made with several vacuum melted, centrifugally cast test bars of titanium alloy A110 AT ELI, obtained from Oregon Metallurgical Corp., Albany, Oregon, showed room temperature properties which rivaled those of forged material and looked promising for use in the M-1 fuel inducer.

Test Bar No.	Ultimate Str. in ksi	Yield Str. 0.2% Offset	Elongation % in 4D	Reduction of Area, %
25	112.2	101.4	15.5	32.8
26	111.6	100.8	16.0	26.9

Annealed at 1500°F for 1 hour, air cooled to room temperature.

A test plan was developed and a full-size simulated inducer casting of A110 AT ELI alloy ordered from the Oregon Metallurgical Corp. The plan included surveillance of foundry operations to obtain background information regarding the process and extensive mechanical testing of the casting to evaluate its metallurgical characteristics including the fatigue properties of the blade and root sections under cryogenic conditions.

The results of this evaluation have been reported previously(168) and are not detailed herein.

(168) Inouye, F.T., Cast A-110AT-ELI (Ti-5Al-2.5Sn) Titanium Alloy Evaluation (SKD-C291), Aerojet-General Report No. DVR 65-170, 29 March 1965

14. Modified Goodman Diagrams for Turbopump Assembly Materials

The objective of this task, which was accomplished during the third quarter of 164, was to provide modified Goodman diagrams for certain turbopump assembly materials used in the M-1 engine.

Diagrams were constructed for Inconel 718 and 7079-T652 aluminum alloy using the straight-line technique recommended by Soderberg. The 0.2% offset yield strength was the limiting steady stress for both alloys at cryogenic temperatures and room temperature. In the case of Inconel 718 for elevated temperature service, the 0.2% creep strength (in 10 hours) was the limited steady stress above 1170°F with the 0.2% offset yield strength being the strength criterion below 1170°F. The 10 hour stress-rupture strength exceeded the 0.2% creep strength (in 10 hours) upon the basis of a Larson-Miller parametric plot of test date; therefore, it was precluded from the diagram construction.

The diagrams were considered safe for design but were not made optimum. Additional fatigue testing was scheduled to define the curvature of the Soderberg line. This included testing under conditions of steady plus fluctuating stress, and complete flexural-stress reversal at temperatures to -423°F to yield a Gerber diagrams.

15. Application of Corrosion-Resistant and Wear-Resistant Coatings

This task was performed during the third quarter of 1964 to evaluate the coating systems that will provide corrosion-resistance and wear-resistance for the power-transmission shafts. The coatings were needed to protect the surfaces of the shaft bearing diameters during assembly and disassembly to prevent scoring or scratching by the bearing inner races.

The two coating systems investigated were found capable of providing adequate protection. These coatings were: Linde LW-5 and Colmonoy No. 6 (AMS 4775). Samples of Inconel 718 and Rene '41 (coating with each of the hard-facing material) were thermal-shock tested (cyclic tests from -320°F to 200°F) for a period of 50 cycles each. Specimens were visually inspected and dye-penetrant inspected. No evidence of cracking, spalling, or base-metal separation was found.

16. Evaluation of Electron-Beam Welded Fuel Turbine Blade-to-Disc Joints

The objective of this task, accomplished during the third and fourth quarters of 1964, was to evaluate the weld-joint efficiency (at room temperature, 800°F, and 1200°F) of the fuel turbine disc-to-vane joints welded by the electron-beam process.

Two blade-to-disc segments that had been electron-beam welded by Solar Aircraft Corp. were received for weld joint evaluation. The segments consisted of six Chevron platformed blades welded from both sides to help equalize the residual stresses and obtain full weld penetration.

Round blank specimens, 1/2-in. in diameter, were machined from the blade-to-disc segments and then heat-treated to Specification AGC-46604 as follows:

a. Solution-anneal at 1950°F for one hours; air-cool to room temperature.

b. Age at 1350°F for eight hours; furnace cool to 1200°F and hold at 1200°F until a total of 20 hours has elapsed (1350°F plus furnace-cool plus 1200°F); then air-cool to room temperature. Hardness readings taken of the blank specimens after heat treatment were R_c 40-42.

The blank specimens were machined into standard R_3 tensile specimens with the welds centered in the reduced section. Tensile tests were conducted at -320°F, room temperature, 800°F, 1200°F, and 1350°F. The specimens tested at cryogenic and elevated temperatures were allowed to soak for 15 minutes prior to testing. The results of these tests showed that the weld joint strength was equal to the parent material when heat-treated after electron-beam welding.

17. Evaluation of Spot Welds Produced by TIG and Resistance Welding Processes

The purpose of this task, which was accomplished during the third quarter of 1964, was to evaluate TIG and resistance spot welds submitted by a vendor. These spot-welding processes were proposed for joining M-1 fuel-pump stator-blade segments, P/N 710906 and 710907.

The TIG spot-welded specimens consisted of AISI type 347 stainless steel (0.060-in. thick) welded to 3/8-in. thick Inconel 718 material. All of the TIG spot welds were found to have internal cracks leading from the weld-shrinkage crater and extending through the full depth of the nugget to the bottom member.

The resistance spot-welded specimens joined AISI Type 304 stainless steel (0.060-in. thick) to 3/8-in. thick Inconel 718 material. Examination of the spot welds revealed internal voids and cracks and poor heat balance in all welds sectioned.

It was recommended that TIG spot welding not be used for this application and a resistance spot-welding procedure be developed to ascertain that sound welds can be obtained for the joint design. Alternative methods of joining were also being investigated.

18. High Temperature Bonding Agents for M-1 Application

The objective of this task was to investigate and evaluate high temperature bonding agents. The selected materials were to be serviceable from cryogenic temperatures (< -300°F) to +1000°F. Effort was started at the end of 1964 and continued into the second quarter of 1965.

Certain ceramic-type materials were considered serviceable to temperatures approaching 1000°F; however, high strength bonds (> 1000 psi) could not be obtained from ceramic cements unless material cure is above 1000°F. Ceramic cements also result in porous bonds; therefore, limit usage in critical scaling applications.

Investigation revealed that the fluorinated ethylene-propylene (FEP) bonding system used on the fuel and oxidizer pump bearing was the most advantageous for these systems.

19. Electron-Beam Welding of the M-1 Fuel Turbine Rotor Assembly

1965

The purpose of this investigation was to establish mechanical properties of electron-beam welded Inconel 718 forgings for the M-1 fuel pump and turbine assembly as well as to coordinate all efforts pertaining to electron-beam welding with vendors. It was started at the outset of 1965 and completed in the third quarter of the same year.

The Indiana Gear Works electron-beam welded joints simulated the Inconel 718 to Rene' 41 stub shaft joint and Inconel 718 first-stage and second-stage disc-to-blade joints.

The weld back-up ring and the external surface were finish machined to aid in radiographic, ultrasonic, and dye penetrant inspection processes.

The electron-beam weld displayed a relatively fine grain structure and sound fusion at both the Rene' 41 and Inconel 718 interfaces.

An electron-beam butt welded Inconel 718 specimen was made to represent the joints of both the first-stage and second-stage fuel turbine blade-to-disc joints. These welds were produced by welding from both sides to maintain uniform weld shrinkage. The specimen was machined at the faces after welding as an aid to radiographic, ultrasonic, and dye penetrant inspection processes.

The Inconel 718 to Rene' 41 stub shaft and the Inconel 718 blade-to-disc specimens were heat treated to the Inconel 718 Specification AGC-46604 prior to sectioning. Although not optimum for the Rene' 41, both alloys responded favorably to heat treatment. The Inconel 718 to Rene' 41 had hardnesses in the range of R_c 41.5 to 45. The blade-to-disc specimen was in the hardness range of R_c 43.8 to 44.8. These hardness readings were taken across parent metal and weld zones.

Appropriate testing was performed to establish electron-beam ranging from -423°F and 1400°F to complete this task.

20. Electron-Beam Welding of M-1 Fuel Pump Rotors

This task was performed to establish electron-beam welding schedules for the repairing of fuel pump rotor No. 2 and for use as an alternative welding process for subsequent fuel pump rotors. Work was started during the first quarter of 1965 and completed during the third quarter.

A series of bead welds on a 1.3-in.-thick Inconel 718 plate was made to establish an optimum weld schedule for butt welding of flat plates and ring forgings. These ring weldments were inspected by the ultrasonic and radiographic processes, which showed them to be satisfactory.

21. Fatigue Properties of Aluminum Alloy 7079-T652

The purpose of this task was to study the fatigue properties of aluminum alloy 7079-T652 from room temperature to -423°F. It began during the first quarter of 1965 and the effort was only partially completed at the time it was suspended during the third quarter of 1965 because of higher priority needs. Fatigue specimens machined from the oxidizer impeller forging (P/N 285202, S/N 0000041) were tested under zero mean stress and alternating tension-compression stress in the 15 to 55 ksi range. The endurance limit (at 107 cycles) was approximately 15 ksi at room temperature. R. R. Moore Rotating Beam fatigue tests conducted with the 7079-T652 aluminum alloy yielded an endurance limit (at 107 cycles) of 25 ksi at the same temperature.

To establish the curvature of the Soderberg line in the Modified Goodman Diagram, room temperature fatigue tests of the 7079 alloy were initiated under the following stress conditions: 30 ksi mean stress and 5 to 15 ksi alternating tension-tension stress; 20 ksi mean stress and 5 to 15 ksi alternating tension-tension stress; and 10 ksi mean stress and 10 to 20 ksi alternating tension-tension stress. Initial results indicated agreement with the predicted theoretical curve based upon the data plot.

22. Low Temperature Mechanical Properties of Large Titanium Forging

The purpose of this investigation was twofold. The low-temperature mechanical properties of the fuel inducer forging of alpha titanium alloy (Ti-5Al-2.5 Sn ELI) were to be established for design information and a comparison was to be made between the wrought titanium alloy properties with those of the 7079-T652 aluminum alloy fuel inducer and the Ti-5Al-2.5 Sn cast titanium simulated inducer. This investigation began during the second quarter of 1965.

The wrought titanium alloy (Ti-5Al-2.5 Sn) inducer forging, with extra-low-interstitial content, exhibited a high degree of notched toughness at ambient temperature and at -423°F. The titanium alloy did not undergo a ductile-to-brittle transition between the temperatures of -320°F and -423°F as experienced with the 7079-T652 aluminum alloy inducer. At -423°F, the subject part had higher smooth bar tensile strength and ductility (elongation and reduction of area) than 7079-T652 and the already evaluated alpha 5Al-2.5 Sn ELI titanium alloy casting. In terms of notch tensile and notch yield criteria, the titanium alloy forging had better notch-toughness than grade 7079-T652 aluminum alloy at -423°F. The toughness of both wrought and cast Ti-5Al-2.5 Sn ELI titanium alloy appeared comparable at ambient temperature and -423°F.

The desired microstructure for optimum toughness, in the titanium alloy, was an equiaxed pattern. This condition was not achieved in the large experimental simulated fuel inducer forging. The structure which was non-equiaxed, dendritic, and coarse appeared to be attendant to the forging operation which either did not sufficiently hot-work the forging or which was carried out at too high a temperature.

The results of the tensile tests at -423°F showed the material to be promising for fuel inducer application. However, laboratory testing to establish the fatigue properties was not performed because of budgetary curtailment during the third quarter of 1965.

23. Lift-Off Seal Bellows Assembly Failure Analysis

The objective of this task, which was conducted during the third quarter of 1965, was to determine the cause of a premature failure of the lift-off seal bellows assembly P/N 285088. This failure occurred during cryogenic cyclic proof-testing of the bellows assembly.

The bellows assembly was a vendor-made part purchased by Aerojet-General to a specification control drawing. The flange, outer bellows, and seal insert were made of Inconel 718. The inner bellows was made of AISI Type 347 stainless steel. Visual examination of the bellows assembly revealed a crack at the outer bellows to flange weld joint. The crack extended approximately 120 degrees at the periphery of the bellows.

The bellows was cross-sectioned and examined. Examination of the failed area indicated that tensile-type failure occurred causing deformation of the weld fractured face. Examination of the outer bellows to flange weld joint of an unfailed area revealed a minute weld root crack approximately 0.003-in. long. Upon examination of the outer bellows weld, it was discovered that seven of nine welds contained non-uniform weld penetration and mis-match at the joints. Two outer bellows welds were uniform and appeared sound.

The outer bellows inner welds and the outer and inner welds of the inner bellows were uniform with the exception of minute weld root cracks approximately 0.002-in. long at some of the joints.

The poor fit-up condition on the Inconel 718 material and inadequate cleaning, resulting in root cracking, caused premature failure. Modification of the tooling and better welding control (e.g., torch angle, cleaning, etc.) were required to produce acceptable hardware.

D. SYSTEMS AND CONTROLS

1. Evaluation of N-155 Propellant Utilization Valve Body Casting

1963

The objective of this task, which was accomplished at the end of 1963, was to evaluate the propellant utilization valve body casting (N-155) to establish its microstructure, soundness, coefficient of thermal expansion, and tensile properties.

Two castings received in early October 1963 were radiographically inspected per Specification MIL-I-6865. Quality level per ASTM E-62T (Reference Radiographic Standards) revealed an exaggerated columnar structure throughout, less dense presence of foreign material (Levels 4 and 5), and presence of shrinkage cavity (Levels 2 through 4) based upon a 3/4-in. thick standard.

Tensile specimens were machined from the two castings and evaluated for mechanical properties in the temperature service operating range of -320°F to 1500°F. The material conformed to AMS 5376 minimum tensile strength and ductility at 1500°F (45,000 psi and 15%, respectively). High strength and low ductility were observed at room temperature and -320°F. Properties at 1000°F were consistently high with the 0.2% yield strength and elongation averaging 38.5 ksi and 6.8%, respectively. Room temperature yield and elongation averaged 56 ksi and 3.8%, respectively. This study showed that "cast N-155" was not suitable for cyogenic applications. This alloy and others of the austenitic class, such as Hastelloy -X and Haynes 25 in the wrought form had better low temperature toughness (accompanied by suitable yield strength). This was attributable to a sound structure as well as a refined microstructure resulting from the rolling or forging operation. With respect to the intended application, the data indicated that cast N-155 should be replaced with a welded sheet on plate assembly. Hastelloy-X was the first choice alloy.

This casting displayed room temperature properties quite inferior to the generally published data for cast N-155.

1963/1964

2. Non-Metallic Sealing of Separable Connectors for Cryogenic Applications

The objective of this task was to determine suitable non-metallic materials and methods for sealing separable connectors to be used at cryogenic temperatures. A seal, capable of withstanding temperatures ranging from -423°F to 300°F while compatible with the fluid environment in the event of leakage, was required. This investigation was conducted during the last quarter of 1963 and the first quarter of 1964.

The application of Narmco 7343 (modified polyurethane) and EC2216 (modified epoxy) cryogenic adhesives were investigated. A Narmco 7343 sealed, bolted-flange assembly did not fail during an immersion test in liquid nitrogen. However, a slight pressure decay was measured after the assembly was subjected to vacuum for 10 hours. Subsequent pressurization of the assembly at 25 psi showed small leakage around the encapsulated bolts but no release of the adhesive-bonded aluminum foil flange strips. Tests in liquid oxygen (per AGC Standard 4003) disclosed both Narmco 7343 and EC2216 adhesives to be sensitive to impact; therefore, they were not to be specified for liquid oxygen systems.

Narmco 7343 was used for sealing of the NERVA propellant feeds system joint simulator test cell in an attempt to monitor leakage under flow conditions. Satisfactory results under vacuum conditions utilizing liquid hydrogen as the fluid media were reported.

Two batches of fluorosilicone sealant, Q-94-002, tested to AGC Standard 4003, were found to be sensitive to liquid oxygen impact at 70 ft-lb. However, two other fluorosilicone sealants, Q-9-0002 and Q-2-0046 (Dow Corning Corp.), were reported to be batch-acceptable materials for liquid oxygen service by MSFC report MTP-P&VE-M-63-14, dated 4 December 1963. These materials were tested to MSFC-SPEC-106 requirements using 0.050-in. thick samples. Thinner samples, 0.025-in. thick, were found to be highly impact sensitive; therefore, they were not acceptable as adhesive sealants.

3. Turbopump Assembly Seal and Sealant Investigation

1963

This task was done during the fourth quarter of 1963 to develop a suitable sealant material for the bonding of carbon seals into steel retainers for operation at temperatures ranging from -423°F to 700°F. The material selected was to provide a leak-free seal and had to be compatible with the environmental fluid.

The existing bonding system (heat sealing with Teflon FEP film per AGC Specification 46464) was investigated for use in sealing the fuel (liquid hydrogen) and oxidizer (liquid oxygen) pump seal assemblies. Purebon

1963

PO3N carbon-graphite specified as the mechanical shaft seal was analyzed for optimum bonding and sealing characteristics.

Several high temperature adhesives were evaluated for possible application as bonding agents for the installation of the Morganite EY105 carbon-graphite mechanical shaft seal into the turbine seal assembly. Because of its contact with the hot gases (steam and hydrogen) and the need for a leak-free seal, this bonded joint might have required a higher temperature adhesive than the specified Teflon FEP film.

Negative results were obtained with the following high temperature bonding agents: Adhesive 4-3, an epoxy-phenolic-silicone manufactured by Westech Plastic and Chemical Company; CA9R cement, an inorganic base adhesive manufactured by Engelhard Industries; and Adhesive N-199, manufactured by the Armstrong Cork Company. The latter is no longer produced for commercial purposes but was investigated because of recommendations from Morganite, Inc.

4. Liquid Oxygen Impact Testing of Material

1963/1965

This task was performed to determine which nonmetallic materials were suitable for liquid oxygen systems. It was an effort that started at the end of 1963 and continued through the third quarter of 1965.

Several materials were initially tested to AGC-STD-4003, "Standard Impact Sensitivity Test," for possible application to the M-1 engine. Because the test results indicated they had impact sensitivity to liquid oxygen, Marmco 7343, a polyurethane adhesive, EC 2216, an epoxy adhesive; Q-2-0046, a fluorosilicone sealant; and Electroplast 202, a Teflon-based plotting compound, could not be specified for liquid oxygen systems.

Purebon P13N, a barium fluoride-impregnated carbon-graphite manufactured by the Pure Carbon Company, was determined to be safe for liquid oxygen service. This material was specified for use with liquid oxygen and liquid hydrogen in the M-1 turbopump assembly seals.

Fluorosilicone sealant Q-94-002 was determined to be marginal for use with liquid oxygen.

Zyglo ZL-4A, a fluorescent type dye penetrant, was evaporated to a syrupy consistency and tested with liquid oxygen. The material was insensitive to 70 ft-lb impact energy per AGC test methods. Previous testing of Zyglo ZL-5A evaporated to dryness by MSFC per ABMA-PD-M-44 had rated the material as unsatisfactory for liquid oxygen. (169)

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(169) Curry, J. E. and Riehl, W. A., Compatibility of Engineering Materials With Liquid Oxygen, Report No. MTP-M-S and M-M-61-7, 21 March 1961

Two dry film lubricants, Fluorocarbon S-122 and Molykote 321S, and one fluorocarbon grease, PD 817, were found to be insensitive to impact at 70 ft-lb when tested to AGC Standard 4003.

Fluorocarbon S-122, Manufactured by Miller-Stephenson Chemical Company, Inc., was an aerosol dispersion of a tetra-fluoroethylene telomer in an inert volatile carrier. The PD 817 fluorocarbon grease, manufactured by Frankford Arsenal, was reported to contain the same solids as the S-122 with an inert perfluorotrialklamine fluid carrier. Both of these lubricants were found acceptable by George C. Marshall Space Flight Center for gaseous and liquid oxygen service in accordance with MSFC-SPEC-106, "Specification for Testing Compatibility of Materials for Liquid Oxygen Systems."

In accordance with AGC-STD-4403, testing indicated that Kynar, a vinylidene fluoride film, was marginal for liquid oxygen service.

One inorganic-bonded Teflon dry-film lubricant, Molykote 523X, manufactured by the Alpha-Molykote Corporation, was found to be impact insensitive at 70 ft-lb when tested to AGC-STD-4003.

Another five materials were tested in accordance with AGC-STD-4003. Results indicated that lithium silicate (AGC-44179); Drilube 702, a phosphoric-acid bonded-molydisulfide thread compound; and the residue from a solution containing a corrosive preventative (75-100 ppm of sodium nitrite in water) were impact-insensitive at a 70 ft-lb energy level. TSI-100 adhesive and Pydraul 60 were found to be impact-sensitive in liquid oxygen; therefore, they were limited to nonoxidizer applications.

Three tubes of dag disperson No. EC-1730, manufactured by the Acheson Colloids Company, were tested and accepted in accordance with AGC-STD-4003. EC-1730 was an anti-seize and sealing compound that was reported to have the same chemical composition as "LOX-Safe" (AGC-44053) manufactured by Redel, Inc. NASA/MSFC reported that EC-1730 was a batch test acceptable material which could be used as a thread lubricant and sealant replacement for AR-IF as specified in ABMA drawing number 10509303.

Molykote Z, a molybdenum disulfide powder lubricant was investigated in the laboratory for report impact sensitivity of preheated (700 to 750°F) samples of MoS_2. No evidence of impact sensitivity was indicated for Molykote Z, preheated from ambient to 1,000°F, when tested in accordance with AGC-STD-4003. Discussion with technical representatives of Alpha-Molykote Corporation (the manufacturers of Molykote Z), indicated that not all MoS_2 powders qualified to MIL-M-7866 were as chemically pure as Molykote Z; therefore they could be impact sensitive. Molykote Z was reported acceptable for liquid oxygen service as indicated by MSFC-SPEC-106 testing.

An acceptable materials listing of nonmetallic materials tested in accordance with AGC-STD-4003 and MSFC-SPEC-106 methods was compiled and forwarded to NASA/LeRC during the fourth quarter of 1964.

A policy regarding the acceptability of materials for oxygen service was enunciated by NASA early in 1965. Materials had to be tested in accordance with MSFC-SPEC-106A and regardless of test facility, final approval by NASA was required.

Dicronite Process L5-47, a Tungsten disulfide (WS_2) dry film lubricant surface preparation by Lubrication Sciences Company, was tested in accordance with AGC-STD-4003. The material met the 70 ft-lb requirement.(170)

A new qualified materials listing for oxygen systems was prepared during the second quarter of 1965 in accordance with the guidelines and criteria set forth by NASA. The list was incorporated into the M-1 Engine Design Information Report (Item M-32). Only those materials listed are acceptable for use in the M-1 liquid oxygen system. Materials could be added to the listing if found acceptable when tested in accordance with MSFC-SPEC-106A and after written approval by NASA.

5. Electron-Beam Welding of Aluminum Valve Body

1963

This program was accomplished during the fourth quarter of 1963 to explore the feasibility of the electron-beam process for welding Type 6061 and Type 7075 aluminum alloy studs to the sleeve valve body.(171) Test specimens of both aluminum alloys were welded and severe cracking occurred in both materials.

Fabrication of the valve body for the M-1 engine by the electron-beam welding process was not considered practical because of a combination of the alloy, the thickness involved, and the joint configuration.

6. Investigation of Distortion in M-1 Valve Bodies

1963/1964

The objective of this task is to investigate the cause leading to the distortion of valve body of Types 6061 and 7075 aluminum alloys during the exposure to low temperature as well as to provide a machining heat treatment sequence and other process data to minimize distortion in these parts. This effort spanned from the fourth quarter of 1963 through the third quarter of 1964.

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(170) Lydon, R.M., LOX Impact Sensitivity of Dicronite, Aerojet-General Report DVR-65-174, 1 April 1965

(171) Hunt, V., Fabrication Method for Valve Sleeve Body by Electron-Beam Welding, Aerojet-General Report No. DVR 63-62P, 23 December 1963

A test program was instituted to determine the cause as well as to establish cures for the distortion problem. Preliminary testing was initiated using forty-three 7075-T6 Titan flanges.

Gas generator valve, P/N 273868, S/N 18, fabricated of 7075 aluminum alloy, was processed as follows:

Heat treated to the T73 condition
Rough machined to within 0.050-in. of the final dimensions
Re-heat treated to the T73 condition
Pressure stabilized at 3000 psig
Finish-machined

The body was thermally cycled from room temperature to -320°F and back to room temperature for 21 complete cycles.

The machining heat treatment process minimized the distortion in the 7075 aluminum alloy valve bodies. The differential measurements were in the order of 0.001-in. to 0.002-in. and were considerably less than the 0.010-in. to 0.012-in. differentials noted in previously tested bodies processed without pressure stabilization or re-heat-treatment after rough machining.

The conclusion reached from this program was that the process treatment developed for this part minimized the distortion in the 7075 alloy in this valve body configuration.

7. Application of Corrosion-Resistant and Wear-Resistant Coatings

1964

This task was performed during the first and second quarters of 1964 to produce corrosion-resistant and wear-resistant coatings for various components through the use of commercial hard-facing alloys.

Inconel X valve gate sleeves were satisfactorily hard-faced using AMS 4775 type alloy. Difficulty originally encountered with this hard-facing system was caused by a high degreee of porosity and consequently, the inability to obtain a required finish of 4 rms. This problem was resolved by improved spraying and torch-fusing techniques.

An Inconel 718 oxidizer shaft was satisfactorily hard-faced with procedures similar to those used for the Inconel X sleeves.

Specimens simulating the Rene' 41 fuel shaft were satisfactorily hard-faced using commercial nickel-base hard-facing materials, (AMS 4775) and Linde coating Lw-5. These specimens were cyclic tested from +200°F to -320°F and their ability to withstand stresses from thermal shock was demonstrated.

1964

8. Sealing of Turnbuckle Cavities in M-1 Sleeve Body, P/N 265328

The purpose of this task was to determine a method and material for sealing the turnbuckle joints and cavities of the thrust chamber valve sleeve body. It was conducted throughout 1964.

Preliminary investigations disclosed that air and fluids could enter and become entrapped in the turnbuckle cavities and it was possible that entrapped media could cause internal icing and/or possible detonation. Pressure-injection, at elevated temperature, of a thermoplastic (Teflon FEP 110) into the cavities was tentatively selected as the most promising method of sealing.

The developed process consisted of three procedures: cleaning the threads and cavities; sealing the threads; and filling the cavities.

Each procedure could be repeated until positive cleanliness, sealing, and filling were accomplished. The cleaning media selected were capable of dissolving the contaminants associated with the fabrication of the sleeve body and, when tested, penetrated through the anodized-thread interface. Lithium silicate, applied by vacuum-impregnation methods, was used to seal the threads. A technique to fill the cavities with tetrafluoroethylene was developed. Sections of a damaged thrust chamber valve were utilized for process development, validation of process effectiveness, and for verification of process reproducibility.

Filling of the turnbuckle cavities with fluorinated ethylene-propylene was also attempted but the tetrafluoroethylene fill proved more effective (i.e., the tetrafluoroethylene filling procedure was reworkable until positive fill can be accomplished).

A Process Specification (AGC-46690) was drafted and called out on AGC Drawing No. 711232. However, the hardware for which the cleaning, sealing, and filling procedures were developed was supplanted.

9. Cryogenic Properties of Thermoplastics

1964/1965

This task was performed to establish material property values suitable for use in the design of specific parts for applications at temperatures to -423°F. It lasted from the beginning of 1964 through the third quarter of 1965; however, it was subjected to intermittent delays because of higher priority requirements.

1964/1965

This program was intended to establish property data for relatively thick, molded cross-sections of candidate materials, mainly fluorocarbon materials, with emphasis upon processing effects, non-destructive material identification tests, and cryogenic test accuracy. A program plan was prepared and a compilation made of the available data and established test methods.

Laboratory testing of CTFE was in progress early in 1965. Tests completed with various thicknesses included infrared, X-ray, diffraction, dimensional stability, surface/profile hardness, specific gravity, solution viscosity, and zero strength time.

Laboratory testing was discontinued in the third quarter of 1965 because of budget restrictions in the M-1 Program.

A review of the incomplete raw laboratory results revealed that properties of thick-molded CTFE were considerably different than the properties commonly published, which were for thin molded sections conditioned and heat aged (macrocrystallinity increased) to simulate thick cross-sections.

Excellent correlations between Knoop hardness, specific gravity, infrared absorption, relative crystallinity, and room temperature tensile properties were also apparent during the review of raw laboratory results. Utilization of these correlations would permit a reduction in acceptance testing of CTFE stock and parts as required in accordance with AGC-44028.

10. Evaluation of Seal Materials for Thrust Chamber Nozzle Extension Flange

1964

The objective of this task, which was accomplished early in 1964, was to evaluate soft seal materials for thrust chamber nozzle extension flange applications. The seal material selected had to withstand temperatures exceeding 800°F for short (7 min cycles) duration.

The following maximum temperature limits were recommended for the materials tested:

600°F for unfilled Teflon compound T-5 (Raybestos-Manhattan, Inc.).
600°F for Viton compound CD-60-133 (Kirkhill Rubber Co.).
700°F for silicone compound 950-B-881 (Kirkhill Rubber Co.).
800°F for filled Teflon compounds No. 116 and No. 134 (Raybestos-Manhatten, Inc.).
800°F for FluoroRay, a filled Teflon compound (Raybestos-Manhattan, Inc.).

11. Bonding of Turbopump Assembly Non-Metallic Face Seals

The purpose of this task, which was accomplished during the first quarter of 1964, was to develop a suitable bonding material for installation of carbon-face seals in the turbopump assembly. The bonding material selected had to provide a leak-free seal for operation in the temperature range of -423°F to 700°F and yet be compatible with the environmental fluids.

Satisfactory bonding with Teflon FEP film was evidenced when the PO3N carbon-face and Invar seal retainer surfaces were prepared by abrading and solvent-cleaning before assembly. Seals for the fuel and oxidizer pumps could be successfully assembled using the amended process specification, AGC-46465B, "Carbon to Metal Bonding for Seal Applications" released 11 February 1964.

12. Thermal Stability of Heat Reflective Coatings for Thrust Chamber Application

The objective of this task, which was accomplished during the first and second quarters of 1964, was to determine the thermal stability of heat-reflective coatings for application to thrust chamber support struts.

Laboratory evaluation of two aluminum-filled silicone-base coatings obtained from the National Lead Company was completed. One of the coatings, 44K49 (qualified to MIL-P-14276B) passed the thermal shock test, but blistered and peeled off the steel coupons during six minute exposure at 1000°F. The other coating, 5542, passed both thermal shock (room temperature to -320°F) and 1000°F temperature exposure tests with negligible effect upon the material.

13. M-1 Gas Generator Fuel Body, P/N 273802, Failure Analysis

The objective of this investigation was to perform a failure analysis of the M-1 gas generator fuel valve body, P/N 273802-4, S/N 3. This was done during the first quarter of 1964.

The failure occurred while the valve body was being cycled at a temperature of -320°F. Damage caused by impact when the frozen actuator was forced from an intermediate position consisted of the following: valve body fracture (broken into two pieces), cracked cylinder walls, and piston lodged in the cylinder. The valve body was examined microscopically and macroscopically. It was also inspected radiographically. Specimens were tensile tested for mechanical properties at ambient and -320°F temperatures in accordance with FED-STD-151. The fracture started in the cylinder wall and propagated to the piston seating area. No indications were noted in the X-ray inspection.

At ambient temperature, the valve body forging had 78.8 ksi ultimate strength, 67.1 ksi 0.2% offset yield strength, 9.7% elongation, 17.2 reduction of area, and 91.03 ksi notched tensile strength at notch-acuity (K_t) of 6.6 to 7.9. Notch-tough at room temperature, the forging met the minimum mechanical property requirements of MIL-A-22771. These requirements were 75 ksi ultimate strength, 65 ksi 0.2% offset yield strength, and 7% elongation (in four diameters) in the forging flow line direction. At -320°F, the forging had high strength and good ductility; average properties were 87.8 ksi ultimate strength, 72.8 ksi 0.2% offset yield strength, 4.75% elongation, 8.9% reduction of area, 89.9 ksi notched tensile strength at notch-acuity (K_t) of 6.1 to 6.9. The forging was notch-tough at -320°F.

The results indicated that the material itself had adequate strength and ductility for the valve body application. Type 6061 aluminum alloy in the T6 temper required evaluation for -423°F service. At -423°F, this alloy had approximately 50 ksi 0.2% offset yield strength, with approximately 25% elongation in sheet (0.073-in.) thickness. However, the properties were dependent upon the thickness of the section. Existing T6 Type 7075 parts required converting to the T73 temper to provide immunity from stress-corrosion cracking. The 7075-T73 alloy appeared satisfactory for -320°F service but its adequacy for -423°F service was dependent upon property data results at -423°F.

14. Evaluation of M-1 Valve Body, P/N 265328, of 6061 Aluminum Alloy

This evaluation was conducted during the second quarter of 1964.

Metallurgical analysis findings for the failed M-1 gas-generator-fuel-valve body, Part 100-273802-S/N 3 showed that the 7075 aluminum alloy in the T6 temper condition was notch-tough at a stress-concentration, K_t of 6.1 to 7.9 at room temperature and -320°F. However, this alloy was not considered suitable for -423°F service upon the basis of the low notch toughness observed in 7079-T652. A similar alloy was used in the fuel inducer. The alloy recommended for -423°F valve body service was 6061 grade (Al - 1 Mg - 0.6 Si - 0.25 Cu - 0.25 Cr) in the T6 temper condition. The reported -423°F service mechanical properties for 6061 - T6 alloy were 50 ksi 0.2% offset yield strength and 25% elongation; associated room temperature properties were 38 ksi 0.2% offset yield strength and 10% elongation. These data were based upon 0.063-in. thick sheet which was not directly applicable for M-1 valve body design.

The properties of 6061-T6 were dependent upon section thickness. Center properties were lower for thicker sections because of the slower cooling rate of the center-section areas during quenching from the solution temperature of 960° to 1010°F. This causes partial hardening response during subsequent aging at 345° to 355°F.

1964

15. A-286 Bolts, P/N AS-4013, Failure Analysis

This analysis was accomplished during the second quarter of 1964.

During installation of high-strength (180 ksi) A-286 bolts (AS-4013) into a M-1 valve body flange, the bolts sheared diametrically at the half-thread root at 650 in.-lb torque. The objective of this task was to determine the cause of bolt failure and to analyze the mechanical properties of stock bolts taken from the same lot as the failed bolts.

The material conformed as AS-4013 standard except for a slightly high nickel content. The structure was normal. The lenticular microcrack present at the half-thread root was caused by the torquing operation. The analysis indicated that bolt failure was caused by mechanism of slow-crack-growth under low applied shear stress caused by stress-concentration.

The unfailed bolt material was in accordance with AS-4013 except for the chemistry variation noted above and the marginal yield strength. The structure was normal.

Several stock bolts were torque-to-tension tested. The bolts were placed in a tensile testing machine which permitted measurement of direct reading of the load while the bolts were torqued in 50-in.-lb increments. The setup was within 2 degrees alignment and the internal thread was a dry 7/16 -20MF3-B. The test failure load exceeded the bolt-installation torque failure load of 650-in.-lb. This test demonstrated that the bolt failures can occur by over-torquing caused by a frictional coefficient so low that the application torque produces a torsional-induced tension stress greater than the minimum yield strength.

16. TCOV Sleeve Lipseals, P/N 265114-5 and P/N 705458-1A, Failure Analysis

1964/1965

The purpose of this investigation was to establish the cause of the thrust chamber oxidizer valve sleeve lipseals failure. It was started during the second quarter of 1964 and was completed early in 1965.

Laboratory testing was completed and preliminary review of the test results indicated that the parts met specified material requirements. The shear failure, indicated by fractographic tests of the three lipseals being analyzed, appeared to have occurred during component operation. This conclusion was also reached after theoretical stress analysis was made.

Significant technological advances, with respect to mode of failure, were developed within the scope of this task. The technique, utilizing electron microscopy for determining the mode of failure, required improvement but its usefulness was definitely established.

17. Thrust Chamber Oxidizer Valve, P/N 705520-39, S/N 003, M-1 Engine Firing, Test Stand C-9, Failure Analysis

1964

This task was performed to investigate and determine the factors contributing to the M-1 thrust chamber assembly failure (Test Stand C-9 incident) on 20 June 1964. It was initiated during the third quarter of 1964 and completed during the fourth quarter.

The investigation consisted primarily of: a chemical analysis of samples taken from various surface areas of the damaged engine hardware; a metallurgical analysis of affected areas; a review of the fabrication history of the involved hardware including chemical analysis of contaminants back-flushed from the fuel and oxidizer systems prior to firing; and a review of related documents and drawings.

Investigation revealed that the drawing callouts did not specify adequate cleanliness nor adequate process control. Under the existing fabrication and assembly procedures (shop, laboratory, and test stand), contaminants, which could contribute to thrust chamber fuel valve and thrust chamber oxidizer valve failure, were introduced. This is evident by the post-incident detection of combustible foreign material within both valves, particularly within the turnbuckle threads and cavities of the valve sleeve body. Isolated hot spots, where metal fusion occurred in the thrust chamber oxidizer valve turnbuckle-thread area, were attributed to the presence of combustible contamination. However, it could not be determined if the contamination in the presence of concentrated oxygen initiated this localized reaction or if it was a secondary effect caused by burning of the contamination in the intense heat associated with the mishap.

Fluorolube-brand compounds used as thrust chamber valve thread lubricants were also detected. As used, the Fluorolube compound was put under high shear forces between the helicoil and the aluminum body during assembly, and still higher shear forces during component operation. Fluorolube compounds were not recommended for use with aluminum when shear loading was involved because detonation could result even without the presence of oxygen.

Upon the basis of the initial findings of this investigation, detailed review of all thrust chamber valve part drawings, processes, and procedures was accomplished. It was deemed essential that this re-review be

1964

completed and all discrepancies and inadequacies corrected before additional thrust chamber valves were fabricated. The cleanliness levels of in-process hardware were raised. Assembled units were cleaned to remove possible contaminants, and certain areas required treatment to prevent re-contamination.

The possibility of contamination in the thrust chamber valve sleeve-body turnbuckle threads and cavities was previously recognized and discussed in this report (Section XIV, D, 8). A process for cleaning, sealing, and filling the turnbuckle threads and cavities was developed as one correction action item to prevent future M-1 firing mishaps. Basically, the process consisted of multiple-step cleaning of the specific areas, sealing the threads with lithium silicate, and filling the cavities with tetrafluoroethylene.

Laboratory test results (infrared, X-ray, and spectrographic) of 47 post-incident samples and 72 pre-incident contamination samples were compiled. A detailed review of the 150 to 200 processing steps involved in the fabrication of the M-1 thrust chamber valve was also made.

18. Control of Hardness of 6061 Aluminum Conical Seals

1964/1965

The purpose of this task, which was started late in 1964 and completed during the first quarter of 1965, was to specify a heat treatment process that would provide a conical seal gasket softer than its flange because 6061 aluminum conical seals in the T-6 condition had scratched the sealing surface of the 6061-T6 valves. The T-4 temper was recommended; however, there was concern that it could age at ambient temperature and eventually become excessively hard. Tests were made to determine the effect of extended time exposure at 150°F in the hardness of 6061-T4 aluminum.

Overaging of 6061-T6 aluminum to produce a material softer than T-6 and more stable than the T-4 temper was investigated. Age cycles using a higher than normal aging temperature were evaluated to determine what age cycle would yield the desired hardness of 26-47 Rb.

Tests were conducted to produce a material softer than T-6 and more stable than T-4. Age cycles using a higher than normal aging temperature were evaluated and indications were that an overaging cycle of 450°F/8 hours would produce the desired hardness; however, this was not necessary because the 6061-T4 condition was found to be satisfactory.

1964/1965

19. Thread Lubricants and Anti-Seize Compounds for Liquid Oxygen Service

A report(172) was issued covering the various thread lubricants investigated for liquid oxygen, gaseous oxygen, and liquid hydrogen service.

E. SPECIFICATIONS

1. Inconel 718 Procurement Specifications

1963/1965

This task encompassed the period from the fourth quarter of 1963 through the second quarter of 1965. Its primary purpose was to upgrade Material Specifications AGC-44098, AGC-44099, AGC-44151, and AGC-44152.

Specifications AGC-44151 and AGC-44152 covering Inconel 718 bars, forgings, sheet, and plate were issued 17 October 1963.

Specification AGC-46604 covering heat-treatment process requirements for Inconel 718 covered by Specification AGC-44151 and AGC-44152 was issued on 8 October 1963.

Four new material procurement specifications, AGC-44192, AGC-44193, AGC-44194, and AGC-44195, were issued in 1964 to cover new designs. These new specifications included the following major requirements: transverse properties for bars and forging stock, revised chemistry, grain size, and higher tensile properties. Additionally, material mill-annealed at 1800°F as stipulated in AGC-44193 and AGC-44194, were required to be tested for true age-hardening response. Melting requirements were modified to minimize Laves phase segregation observed in earlier heats produced by the single-vacuum-induction melting process.

Companion heat treatment process specifications AGC-46604 and AGC-46626 were amended to reference the new material specifications.

2. Hard-Facing of Metals Specification

1964

The purpose of this task early in 1964 was to improve the reliability of the product by upgrading Development Process Specification AGC-46516.

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(172) Thread Lubricants for -423°F to 1000°F Service, Aerojet-General Report No. 64-436, October 1964

The specification was revised to include the newly-developed techniques for controlling the parameters for base metal preparation, spraying, and fusing.

3. Test Method Standard Heat Treatment of Alloy 718 Critical Parts Control (AGC-STD-4861A)

This standard was issued during the first quarter of 1964 to provide additional material and heat treatment process control in the fabrication of critical parts. The basic requirements of the standard were to verify the age-hardening response of alloy 718 as certified by the mill or forging vendor through additional testing by the fabrication vendor and Aerojet-General as well as to provide a heat treatment process control by testing tensile specimens which accompany the parts into the furnace.

4. Specification for Chlorotrifluoroethylene, Tetrafluoroethylene, and Fluorinated Ethylene-Propylene

The objective of this task was to review the appropriate specifications and revise them as necessary. This effort extended throughout 1964.

Revision of Specification AGC-44028 (CTFE) was deemed necessary to incorporate the latest laboratory-verified technology. In particular, the portion pertaining to contamination could be represented in a simpler form. After reviewing the specification with the sole-source resin manufacturer, resin processors-suppliers, and "in-house", it was decided to specify contamination requirements in three levels to expand applications, assist specific part designing, and increase the acceptance level for non-critical areas of parts caused by contamination. Revision E to Specification AGC-44028 (the material specification for polychlorotrifluoroethylene) was issued on 13 May 1964.

Revision of Specifications AGC-44087 and AGC-44113 was also deemed necessary to bring the documents in line with the latest industrial production advancements (i.e., the specification terminology and sample requirements were for small compression-molded parts whereas at that time large parts and/or other methods of processing parts (sample requirement) were acceptable).

Specification AGC-44087 for polytetrafluoroethylene was reviewed and only minor technical changes were needed.

The recommended revision of the fluorinated ethylene-propylene specification (AGC-44113) was postponed because of questions as to whether inconsistencies that resulted in rejection by Aerojet-General were based upon invalid acceptance testing or on substandard material.

1964

5. Inconel 718 Local-Aging Specification

The objective of this task was to write a specification defining the procedures to be followed for local aging Inconel 718 welds where normal furnace-aging methods are not possible.

The specification was issued during the third quarter of 1964 and called for the following aging cycle on the heated side of the part: heating at 1400 $\pm$ 25°F for 8 to 10 hours, cooling to 1250 $\pm$ 25°F, and then holding at 1250 $\pm$ 25°F for a total elapsed aging time of 20 hours.

Preproduction test setups were required to establish processing parameters. Test specimens from this preproduction part were required to have the following minimum room temperature properties:

Ultimate Tensile Strength (psi)	0.2% Yield Strength (psi)	% Elongation (in 1 in.)
160,000	130,000	7.0

Process requirements were defined to control the temperature and time parameters.

F. TEST AREA

1. Gaseous Hydrogen Pressure Vessels

1965

The objective of this investigation was to determine the cause of failure of the 1-in. nozzle on gaseous hydrogen pressure vessel VH-74 in Test Area H-8. This was accomplished during the second quarter of 1965.

Seven receivers in Test Area H-8, four for high pressure hydrogen gas, and three for nitrogen, were designed and built by A. O. Smith Corp. Four failures of the 1-in. nozzle in three of the gaseous hydrogen tanks, VH-3, VH-73, and VH-74 occurred between June 1964 and January 1965. The tanks were of multilayer construction. The 1-in. inner diameter by 2 1/2-in. outer diameter nozzles were inserted through a 4 1/4-in. diameter hole machined through the 23 layers of the vessel shell. The joint design was a 1-in. wide straight-sided joint around the nozzle and was manually arc welded with SW 120 A electrodes. The weld joint was not stress relieved.

1965

A failure occurred on 11 January 1965 in vessel VH-74 after approximately five months of service. It failed after holding for 72 hours at gaseous hydrogen pressure of 4600 psi. A complete failure analysis was made(173).

An additional failure of a Struthers-Wells vessel made of T-1 steel was experienced on 2 June 1965. This vessel failed at 3900 psi gaseous hydrogen pressure after approximately 10 days of pressure cycling at lower levels. Internal examination of the vessel disclosed long cracks at the heat affected zone and partially in the weld of the longitudinal weld joints.

The various failures and problems associated with these vessels is the subject of a separate report(174) and no effort is made to repeat this information herein.

(173) Ishizaka, E. T., GH_2 Pressure Vessel VH-74, Aerojet-General Report No. FA 65-179, 6 April 1965

(174) Laws, J. S., Hydrogen Gas Storage Problems, Aerojet-General Report No. 8800-67, 15 April 1966

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