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RESISTANCE JET MICROTHRUSTER SYSTEM DEVELOPMENT

W. N. Neiman
General Electric Company

TECHNICAL REPORT AFAPL-TR-69-22

March 1968

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AFAPL-TR-69-22

RESISTANCE JET MICROTHRUSTER SYSTEM DEVELOPMENT

W. N. Neiman

FORWARD

This report summarizes the work performed by General Electric Company, Evendale, Ohio under Air Force Contract F33(615)-67-C-1163 on the development of a system of ammonia propellant, resistance-heated microthrusters for station keeping and station changing duties for a satellite. The period of effort extended from October 1966 to April 1968.

The contractor's number for this report is GESP-184. It was submitted to the Air Force for review and approval on January 24, 1969. The author is W. N. Neiman.

The primary portion of the program was monitored for the Air Force by Mr. A. T. Molisse (APIE-2) of the Air Force Aero Propulsion Laboratory. Mr. Jack Geis (APIE-2) of the same group monitored the program in its concluding phase.

Principal contributors to the program were Dr. M. L. Bromberg, Messrs. B. A. Free, J. Holowach, J. I. Kamin, B. C. Merten, C. C. Schnell, W. R. Young, and W. F. Zimmerman, all of General Electric Company, Evendale, Ohio.

Publication of this report does not constitute Air Force Approval of the report findings or conclusions. It is published only for the exchange and stimulation of ideas.

Philip E. Stover
PHILIP E. STOVER
Act'g Chief, Propulsion & Power Branch
Aerospace Power Division

ABSTRACT

This report is a summary of the work performed under Air Force Contract F 33615-67-C-1163 in the design, fabrication, and test of resistance jet microthruster systems for East-West station keeping and station changing duties on the Dodge-II Satellite. The thrusters were electrically heated and ammonia gas was used as fuel. For the station keeping function, thrust values of 3, 6, and 9 micropounds were supplied. For station keeping, 100 micropounds of thrust were provided, with a catalyst being used to assist in decomposition of the ammonia.

Included in this work was the design and fabrication of a torsion-wire device that was used for measuring thrust in the required low range.

Parabolic flights of KC-135 aircraft carrying plastic models of the microthruster system's fuel tank proved its suitability for delivering gas to the propulsion system under zero-g conditions. These experiments, showing equilibrium positions of the propellant, were recorded in slow motion on a 900 foot reel of color film.

Electrical designs were completed for a power and signal conditioner to convert satellite electrical power into the forms required by the microthruster system. A portion of this conditioner contained circuitry to condition the various instrumentation signals into a form acceptable to the vehicle telemetry system for earth transmission.

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SECTION I

INTRODUCTION

The performance of work under this contract was directed toward the delivery to Wright-Patterson Air Force Base (AFAPL) of one complete prototype resistance jet microthruster system (Figure 1), two complete flight-type resistance jet microthruster systems (Figure 1) and one partially complete flight-type resistance jet microthruster system (Figure 2). All of these systems made use of ammonia propellant, resistance heated microthrusters designed to perform station keeping and station changing duties for the Dodge-M satellite.

Three primary tasks were involved in this work. One was the preparation of the flight system thrusters. Included in this was the design, fabrication, and checkout of a microthrust measuring system, the design, fabrication and test of preliminary evaluation thrusters, and finally the design, evaluation, and test of the flight thrusters. Another primary task was concerned with the ammonia propellant storage and feed system. This involved the design of the fuel tank and the selection of the components to control and regulate the flow of gas to the thrusters. Parabolic flights of KC-135 aircraft carrying transparent fuel tanks were expected to confirm experimentally the predicted equilibrium position of the liquid under zero-g conditions. The third task consisted of the design and buildup of both prototype and flight-type power and signal conditioners. Such conditioners were to provide power for certain of the electrical components of the system upon command. They were also to provide for operation of certain of the instrumentation and indication devices built into the system.

Associated tasks involved the integration of the systems with the satellite (requiring liaison with The Johns Hopkins University, Applied Physics Laboratory), launch support, and data reduction during flight.

Before any of the systems could be delivered, certain changes occurred in the proposed DOD satellite (Dodge-M) upon which it was to fly. This situation required reassessment of the design and performance requirements by WPAFB personnel. After a period of time, the satellite was redesignated as Dodge-II, and a revised Work Statement (dated 7 Feb 68 and described further in Section VIII of this Report) was prepared by the Air Force. Then a stop work order was issued. When lifted, the contractual effort was redirected to provide for shipment of all residual hardware items to WPAFB, to prepare an Installation and Operating Manual for the Microthrust Measuring System, and to prepare this Summary Report. This Report is a compilation of the progress to date and is based upon the content of previous reports that have been submitted during the tenure of this contract.

Appendix XVII of this Report contains a list of formal drawings prepared under this Program and is indicative of the design effort involved.

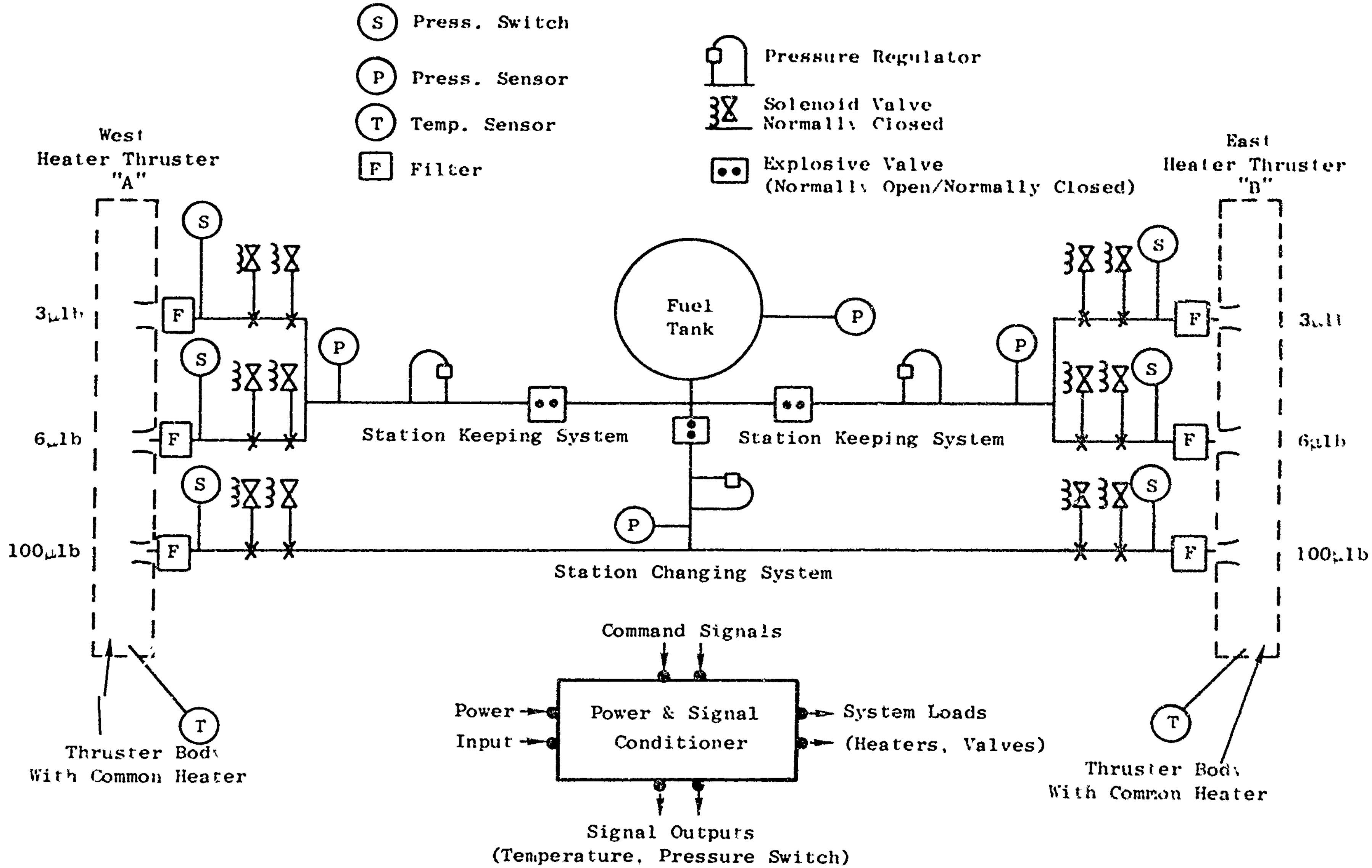


Figure 1. Resistance Jet Microthruster System. Schematic Diagram.

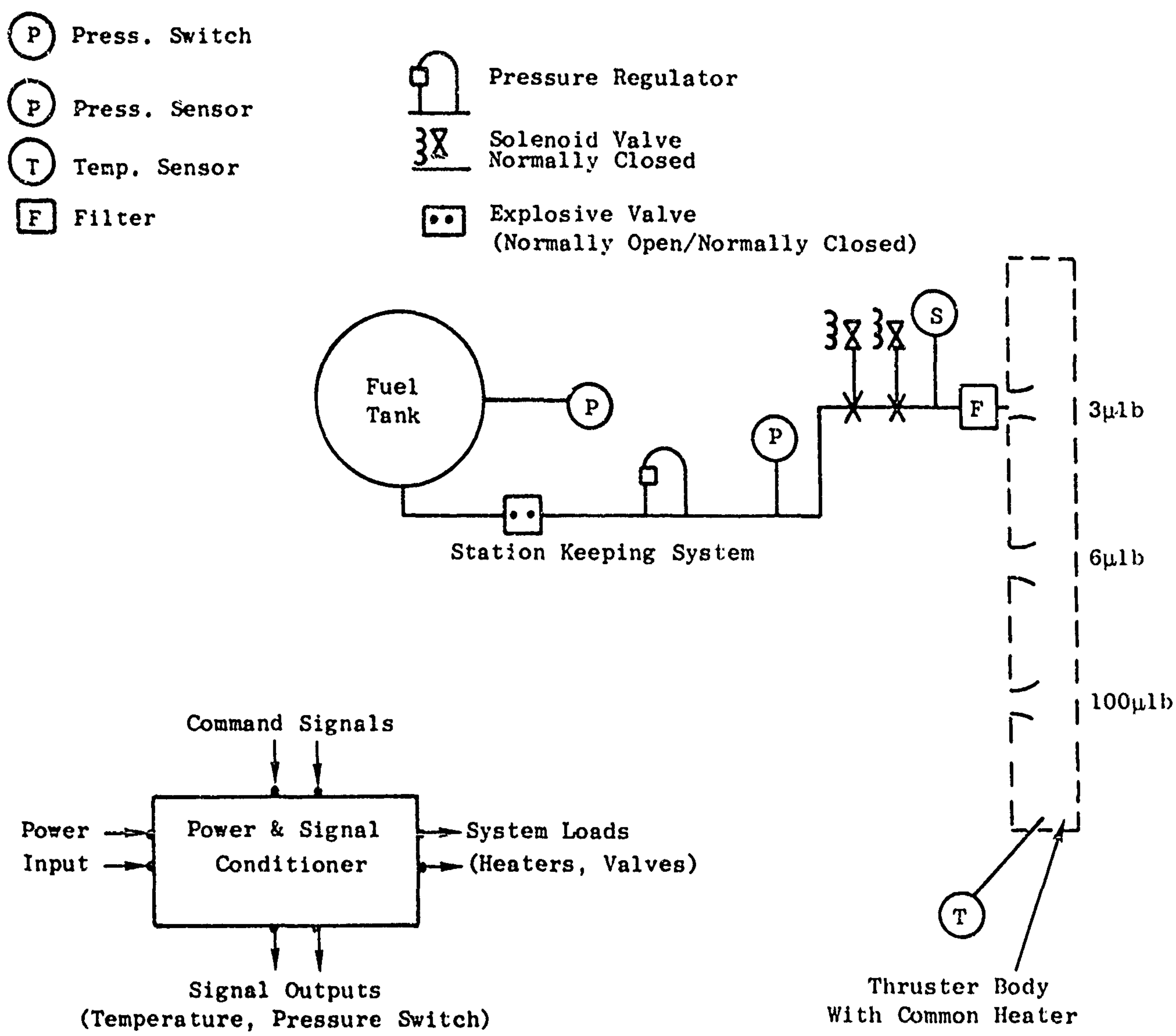


Figure 2. Resistance Jet Microthruster Partial System (For Special Testing), Schematic Diagram.

SECTION II

THE MICROTHRUST MEASURING SYSTEM

This task is concerned with the design, fabrication, and operation of a system to be used for measuring thrust in the 1 to 2000 micropound range as well as determination of thrust vectors.

1. ORIGINAL CONCEPT

It was initially planned that the system would consist of a torsion wire suspended table, the rotational motion of which would be nulled by application of a restoring force generated by an electromagnet. The current in the magnet was to be a measure of the force exerted by the thruster. It was intended that the table would float in a container of mercury, reducing the weight supported by the tension wire. This would permit the use of a small diameter wire, thereby increasing the sensitivity of the system. By virtue of the small clearances and the high surface tension of the mercury, the lateral motion of the table would also be restrained. Electrical power to the table was to be achieved by the use of electrodes fastened to the table, dipping into cups of mercury. Control of the thrust function was to be maintained by means of a photodiode switching arrangement. This would also eliminate lead drag on the table.

The calibrating procedure was to derive a calibration curve, plotted from the current flowing in the restoring force electromagnet winding and the micropou d force indicated by the force applied to a force transducer. Such a calibration was to be made physically separate from the thrust system. A displacement transducer mounted in line with the electromagnet on the thrust system was to provide for the determination of the null position. Thus when the microthruster force moved the thrust table off the null point, the measurement of the current in the electromagnet winding to bring the table back to its null position provided for a determination of the thrust in micropounds.

As an indication of the sensitivity that may be achieved using this type of equipment, consider a 12 inch diameter table with a total effective weight of 5 pounds. Using a piano wire torsion element with an allowable yield of 300,000 psi, the diameter required to support the 5 pounds is 4.6 mils. If a 5 mil wire 20 inches long is selected, the rotation in degrees caused by a twisting movement T of 6 micropound-inches (1 micropound at its periphery of the table) is

$$\theta = \frac{(57.3)^{*}(T)\ (l)}{(G)\ (J)}$$

* 1 radian = 57.3 degrees

Where l is the wire length of 20 inches, G is the shear modulus which, for piano wire, is approximately 12 x 10^6 psi, and J is the polar movement of inertia which, for a 5 mil diameter shaft, is $6.25\times10^{-11}in^4$. The rotation is then approximately 4-1/2 degrees corresponding to approximately 1/2 inch linear displacement at the 6 inch radius. Linear displacement transducers are readily available having outputs of 100 mv/mil. Since a 100 mv signal can easily be resolved, the null position can be identified to within 0.1% of the one inch displacement. Therefore, a high degree of accuracy can be derived for this thrust measurement. With this resolution, the system could also be capable of evaluating thrust vectors with good precision. In practice, it was to be accomplished by placing a thruster so that it pointed in a radial direction. Any non-axial thrust vector would be manifested as a tangential vector. The vector would be completely resolved by determining the tangential vector at two angular positions of the thruster about its axis.

2. THE SYSTEM AS DESIGNED

Figures 3, 4, 5, and 6 show the system as designed. This design incorporates, except for the photodiode switching arrangement to operate the thruster valves, all the significant features of the original concept.

3. INITIAL CALIBRATION OF THE SYSTEM

Initial efforts to calibrate the system revealed several unexpected problem areas:

- The surface tension of the mercury resulted in an unacceptable amount of drag on the thrust table. Therefore, the mercury cups and electrodes that provided the electrical connections between the table and the fixed portion of the system were eliminated, and thin coiled leads were substituted. The mercury bath that partially supported the weight of the thrust table was replaced by a pivot. This was to act as a guide, restricting the lateral motion of the table. Ultimately this pivot, too, was eliminated because of the friction it introduced.

- It was found that the degree of freedom of the thrust table was impeded by the small clearances between the displacement and the restoring force solenoid cores. An improvement was observed when the clearances were increased.

- Tugging of the earth's magnetic field upon the thrust table was eliminated by discarding certain clamps that had been made of magnetic steel and replacing them with ones made of non-magnetic stainless steel.

- The vibration effects of the vacuum tank pumping system were eliminated by suspending the thrust table from a frame which was in turn hung from the interior of the vacuum tank. A damper magnet was used to reduce further v oscillatory motion.

Despite all the improvements described in the above paragraphs, it was still found difficult to calibrate the system with consistent and repeatable results. It was decided, then, to eliminate the position transducers and the restoring force electromagnet, and to use a transducer to measure force exerted by the thrust table as it rotated

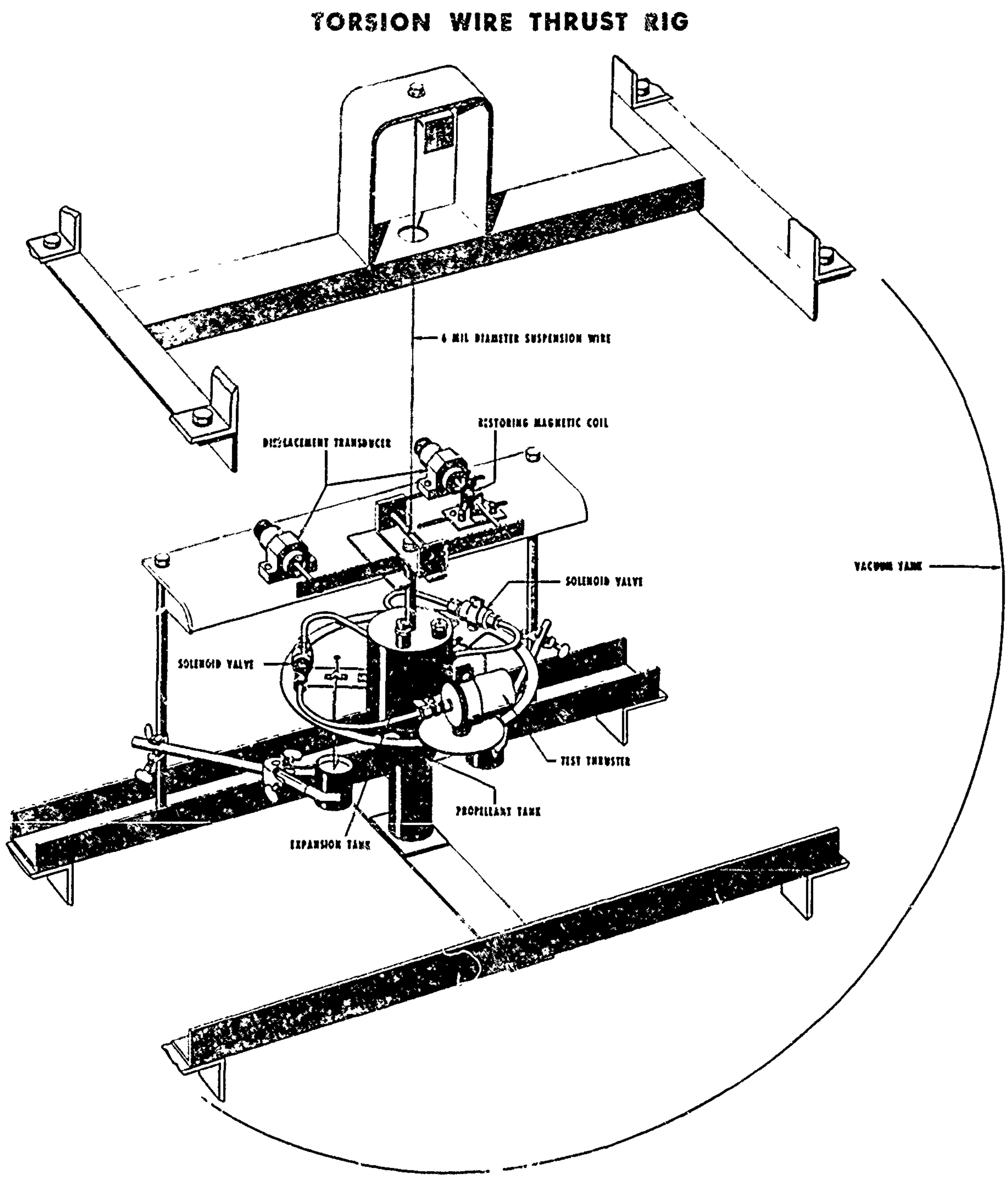


Figure 3. Three Dimensional View Microthrust Measuring System, as Designed.

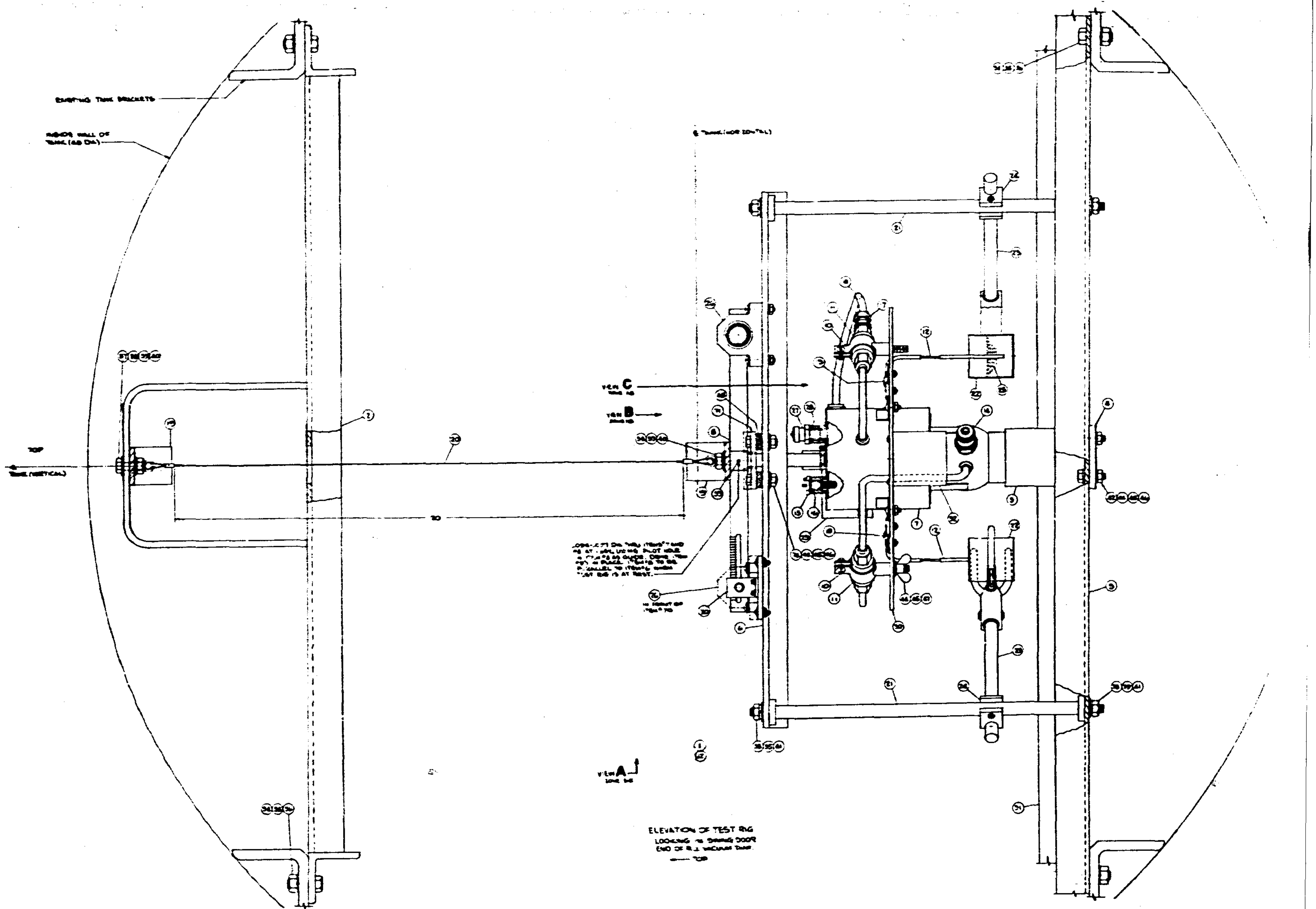


Figure 4. Microthrust Measuring System, as Designed, Front View.

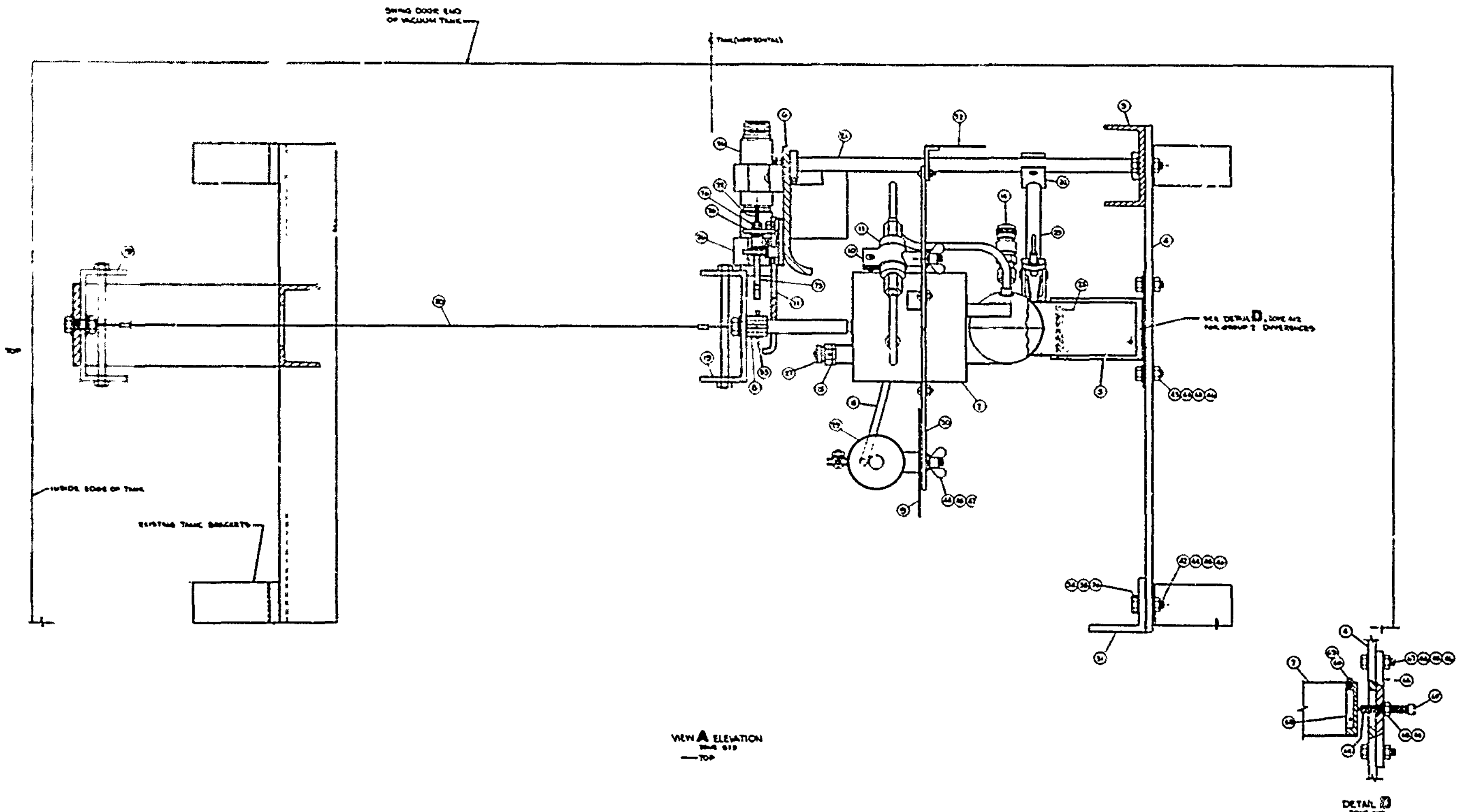


Figure 5. Microthrust Measuring System, as Designed, Side View .

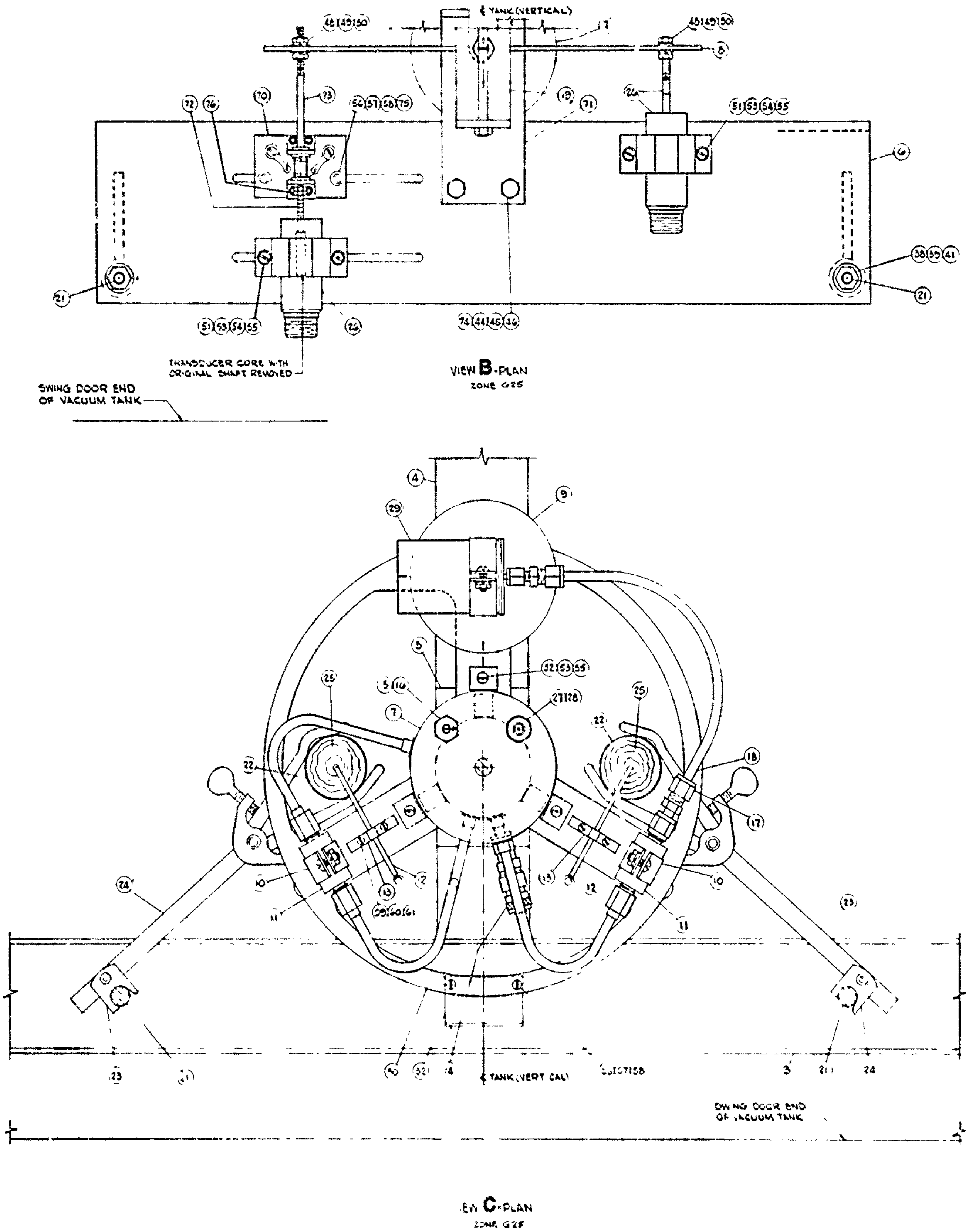


Figure 6. Microthrust Measuring System, as Designed, Top View.

as the result of action of a thruster.

4. INITIAL USE OF THE SYSTEM

After it was determined that the calibration of the system was no longer required and that direct measurement of force with a force transducer was possible, the arrangement of the system appeared as shown in Figure 7, 8, and 9. Note that the pivot on the thrust table is free of contact with its mating part, the restoring force electromagnet has been removed, and the displacement transducers have been dismantled. Thus it was not until the measuring system had been simplified to the degree shown in the schematic diagram, Figure 10, that accurate and repeatable results were obtained. The pressure switch appearing in the schematic was intended to close the regulated fill valve if the pressure in the NH_3 vapor expansion tank approached the 50 psi limit. A damper magnet was used to restrict any vibratory motion of the thrust table.

At this point in time, the thrust measuring system consisted of a flat circular table 12 inches in diameter suspended horizontally by a wire attached at the center. The thruster was mounted near the outer rim so that it imparted its thrust tangentially, tending to rotate the table about the wire axis. A bracket extended from the table below the thruster and rested against a force transducer. The thrust was read directly on this transducer.

Propellant storage and supply components including a pressure transducer were mounted on the table. Instrumentation and power leads were thin, coiled wires strung between the table and the support member holding the wire.

The entire apparatus was built inside an aluminum framework which was suspended by a chain in the vacuum chamber. Eddy current dampeners were used to stabilize the system.

Using displacement transducers, with the force transducer on a drive mechanism to apply a force, the thrust rig was found to have an angular displacement of approximately one mil per micropound force at the thruster mounting radius. The force transducer core displaces one mil per 220 micropounds.

Therefore, thrust measurements could be taken directly on the force transducer without nulling the system since a thrust of 220 micropounds would rotate the rig only one mil. This one mil movement would cause a restoring force in the rig of 1 micropound, resulting in an error of 0.45%, a negligible amount.

Operation of any one of the solenoid valves resulted in an interaction of its solenoid field with the dampening magnet fields and to a lesser degree with the earth's magnetic field. The disturbances introduced took 10 or 20 seconds to dampen out. Consequently, thrust measurements were taken as follows for the small thrusters (3 and 6 micropound) designed to operate at 3.4 psi. The force transducer was pre-loaded by driving it against the table foot. The expansion tank

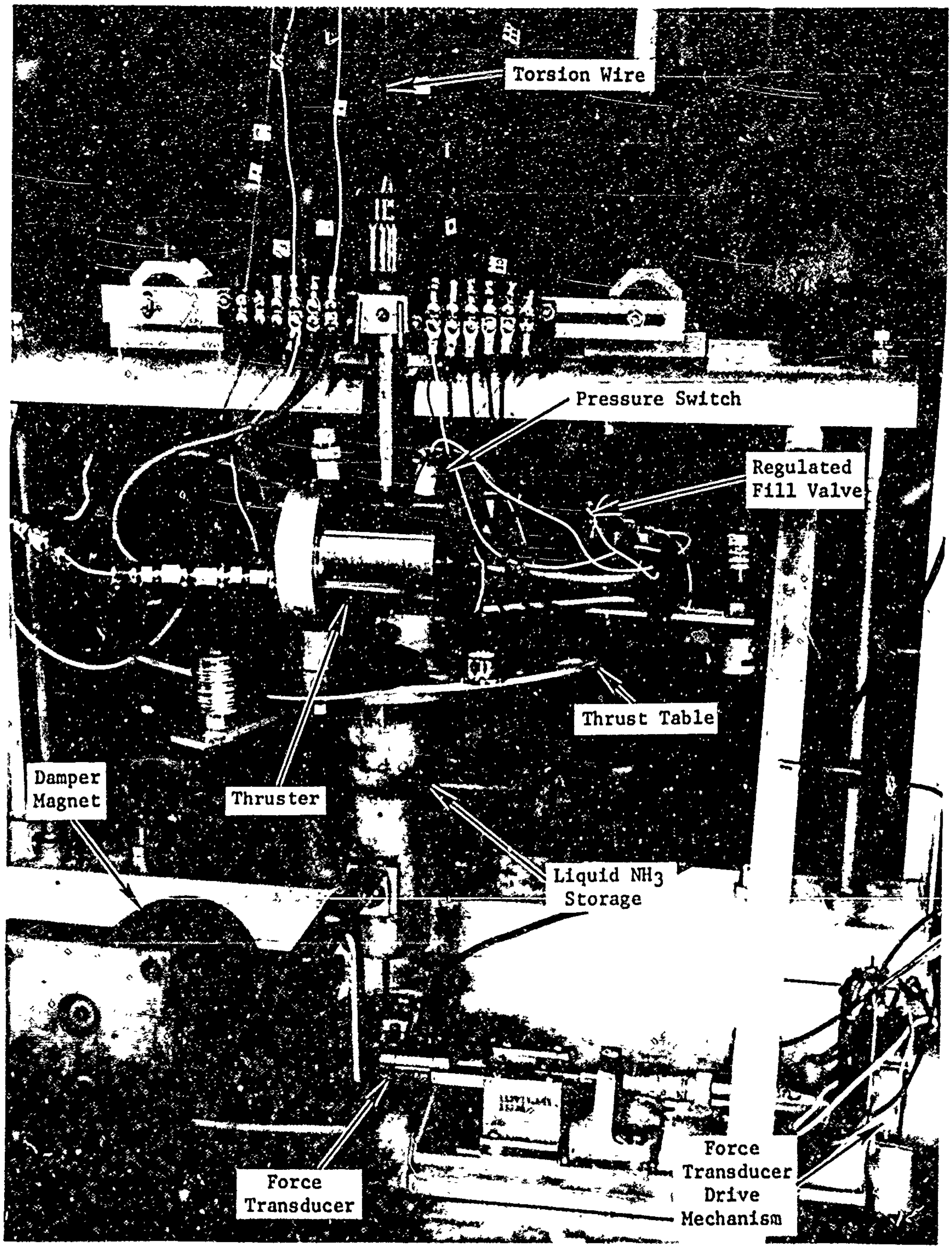


Figure 7. Microthrust Measuring System, Front View, During Initial use.

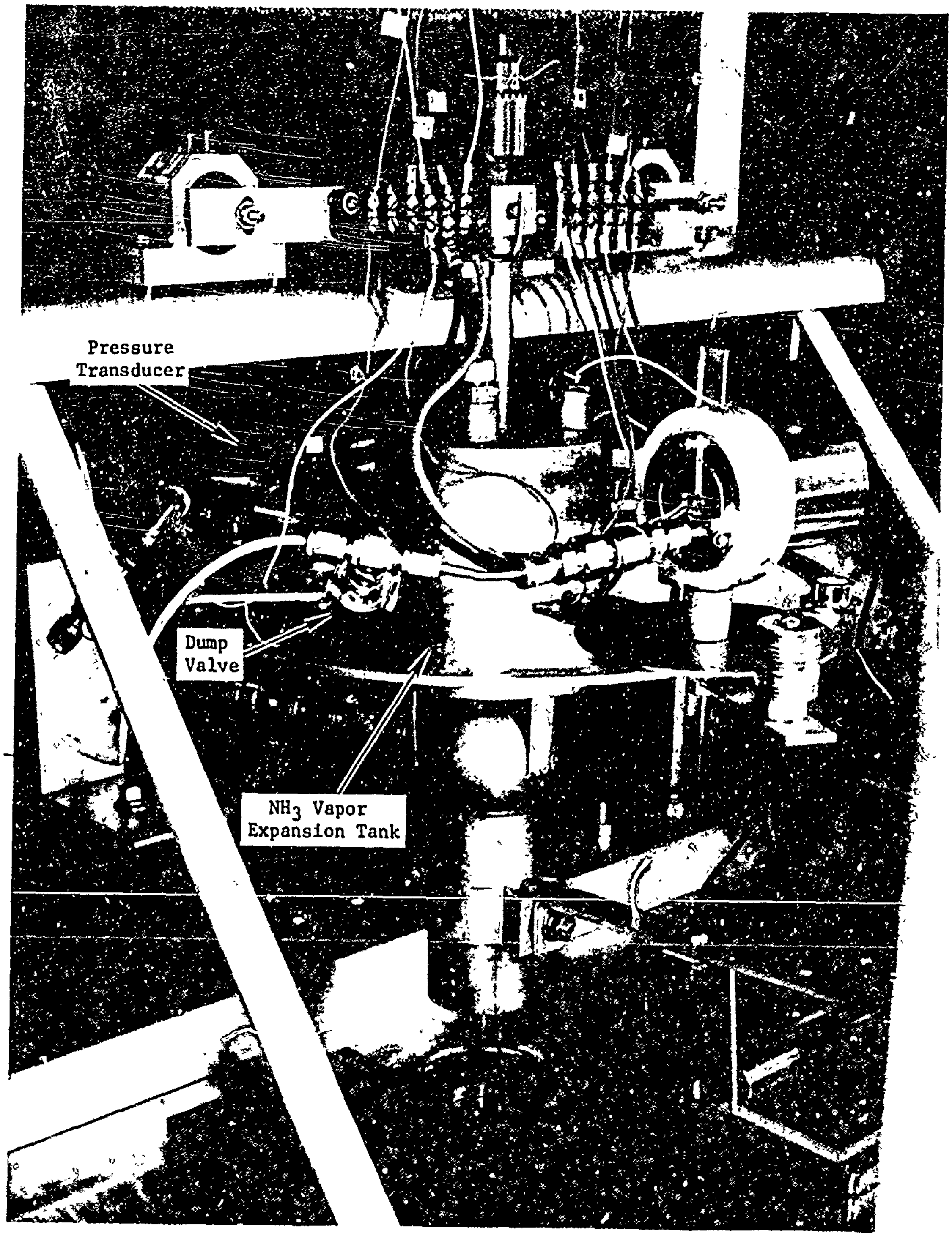


Figure 8. Microthrust Measuring System, Alternate Front View, During Initial use.

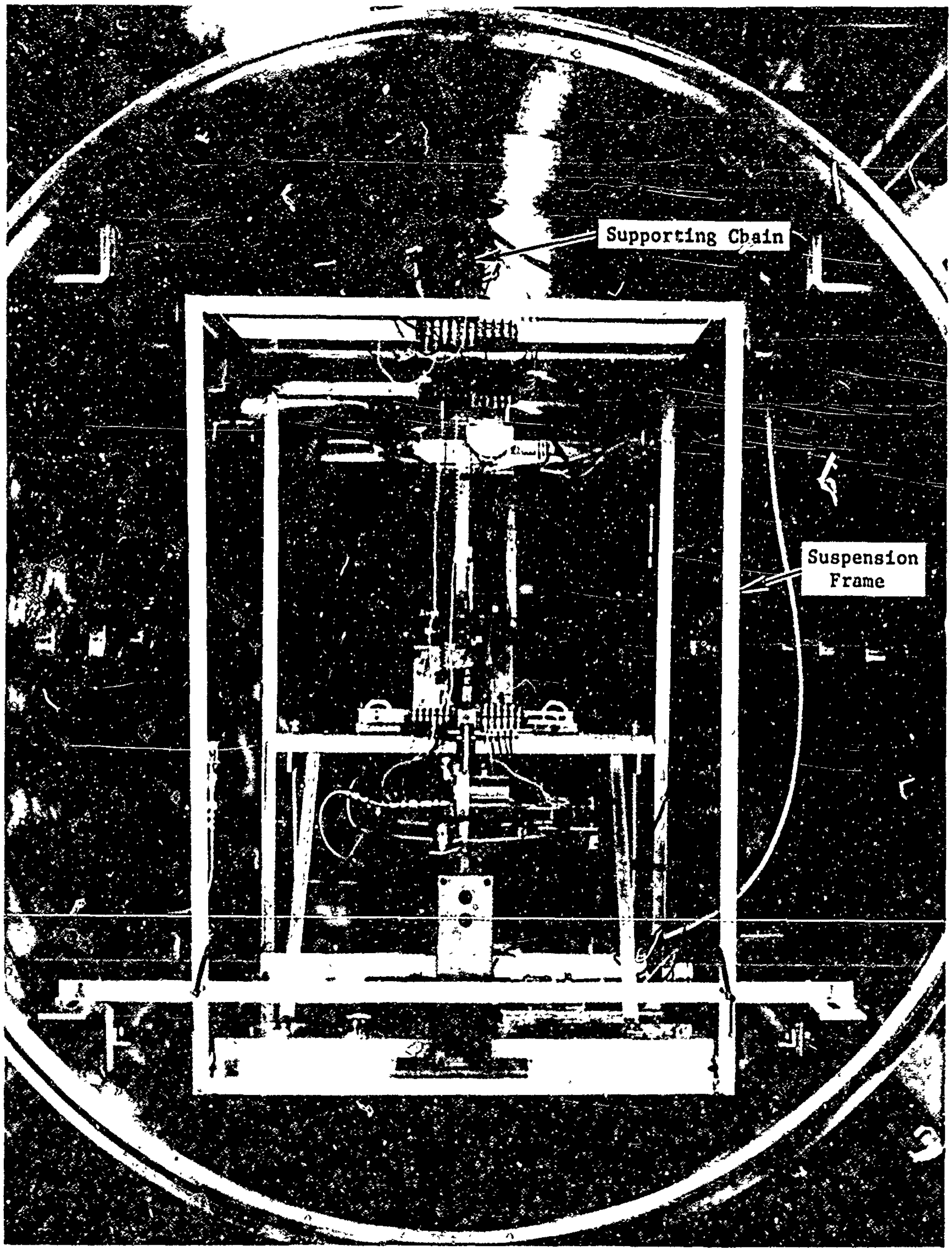


Figure 9. Microthrust Measuring System as Installed in Vacuum Tank During Initial Use.

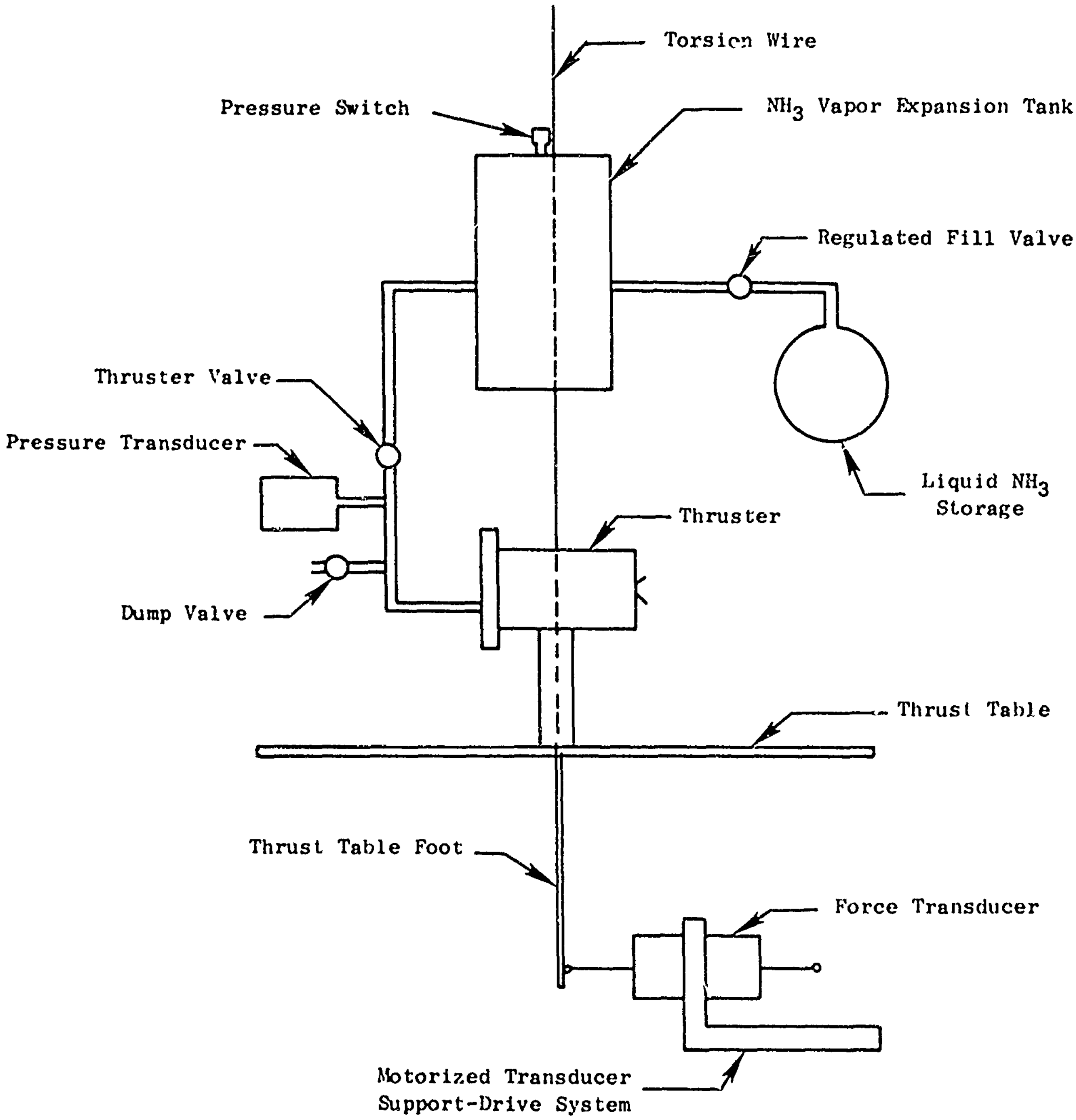


Figure 10. Pneumatic Schematic Diagram, Initial Use of Microthrust Measuring System.

was filled to about 8 psi NH_3. The dump valve was opened and closed to assure that no pressure was on the orifice. The pressure transducer zero was checked at this point and re-set if necessary. The thruster valve was then opened long enough to allow the thruster pressure to reach about 8 psi, and then closed. The thruster pressure decayed through the orifice, and by the time it reached 3.4 psi (the desired pressure), the vibrations caused by actuating the valve had been dampened out and a clear thrust trace was obtained.

For the 200 micropound thruster measurements, the thruster valve was removed. The expansion tank was filled to approximately 20 psi and the pressure of the entire volume was allowed to decay through the thruster nozzle, again giving time for the vibrations to dampen out before the desired pressure is reached. Figure 11 shows a 170 micropound thrust trace. The pressure reading on the chart is only an indicator and is not calibrated. Pressure was read by observing a calibrated digital voltmeter on the pressure transducer output and marking the chart at the appropriate time.

5. CONTINUED USE OF THE SYSTEM

Continued use of the system resulted in several refinements. A pressure regulator and a second pressure transducer were mounted on the thrust table and arranged so that thrust and mass flow rate measurements could be taken simultaneously. This eliminated the need to depend upon a previously-obtained mass flow rate vs. temperature curve in determining specific impulse of a thruster. The thruster test time was thereby cut in half. The calibration of the force transducer was also made a matter of record.

6. THE SYSTEM IN FINAL USE AT GENERAL ELECTRIC COMPANY

The microthrust measuring system during its last use at General Electric appeared as shown in Figures 12, 13, and 14. It is described as follows:

- Description of the System

The microthrust measuring system consists of an assembly which, when properly installed and instrumented, provides for the measurement of the thrust of a suitably-mounted thruster over the ranges tabulated below with an accuracy of ± 1% of full range accuracy:*

Zero to 25 micropounds
" " 125 "
" " 250 "
" " 500 "
" " 2000 "

It consists of a magnetically-damped thrust table suspended by a wire

* With the Vacuum System Pumps operating at the General Electric Co. installation.

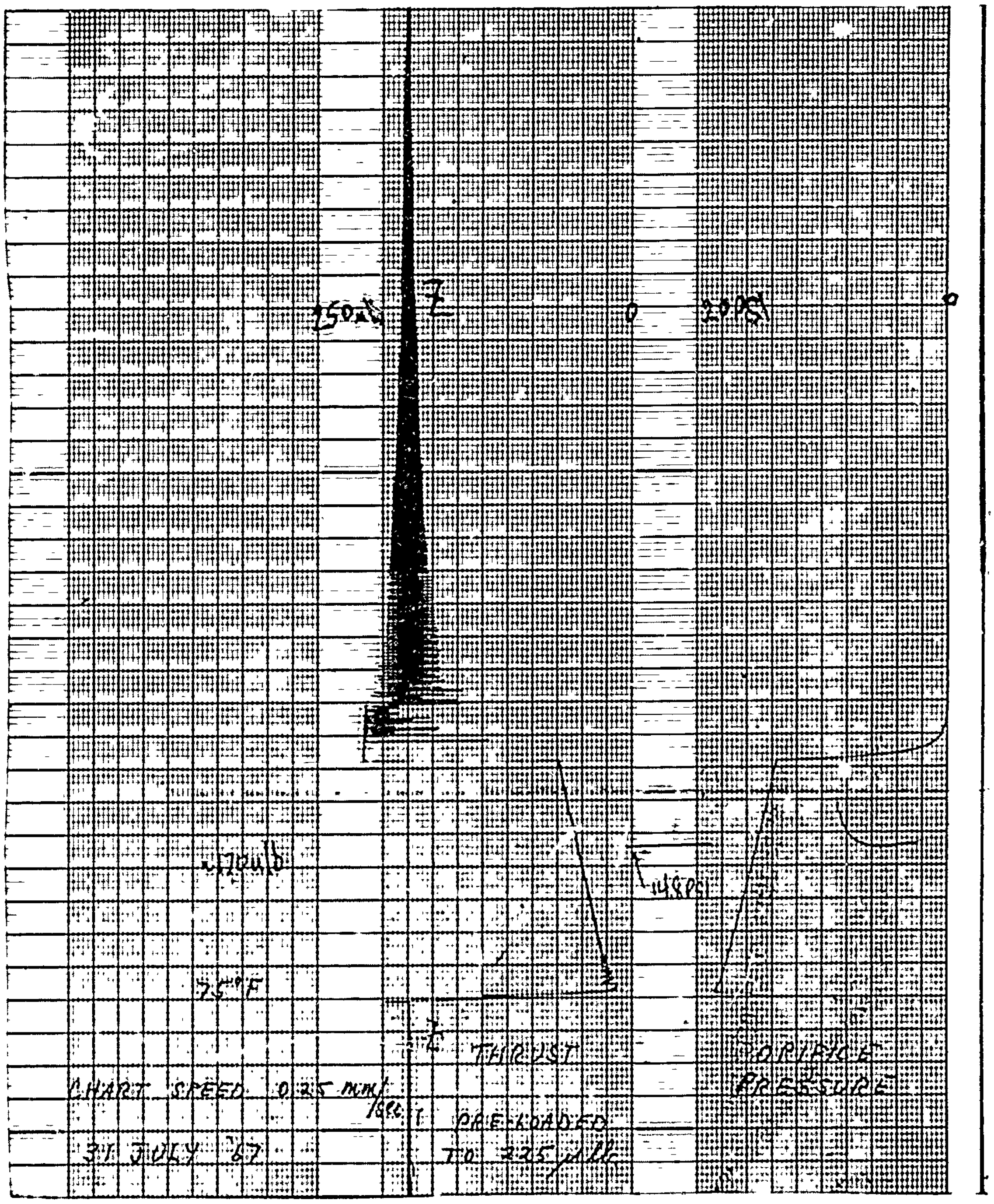


Figure 11. Thrust Trace, 170 Micropounds, as Measured with the Microthrust Measuring System.

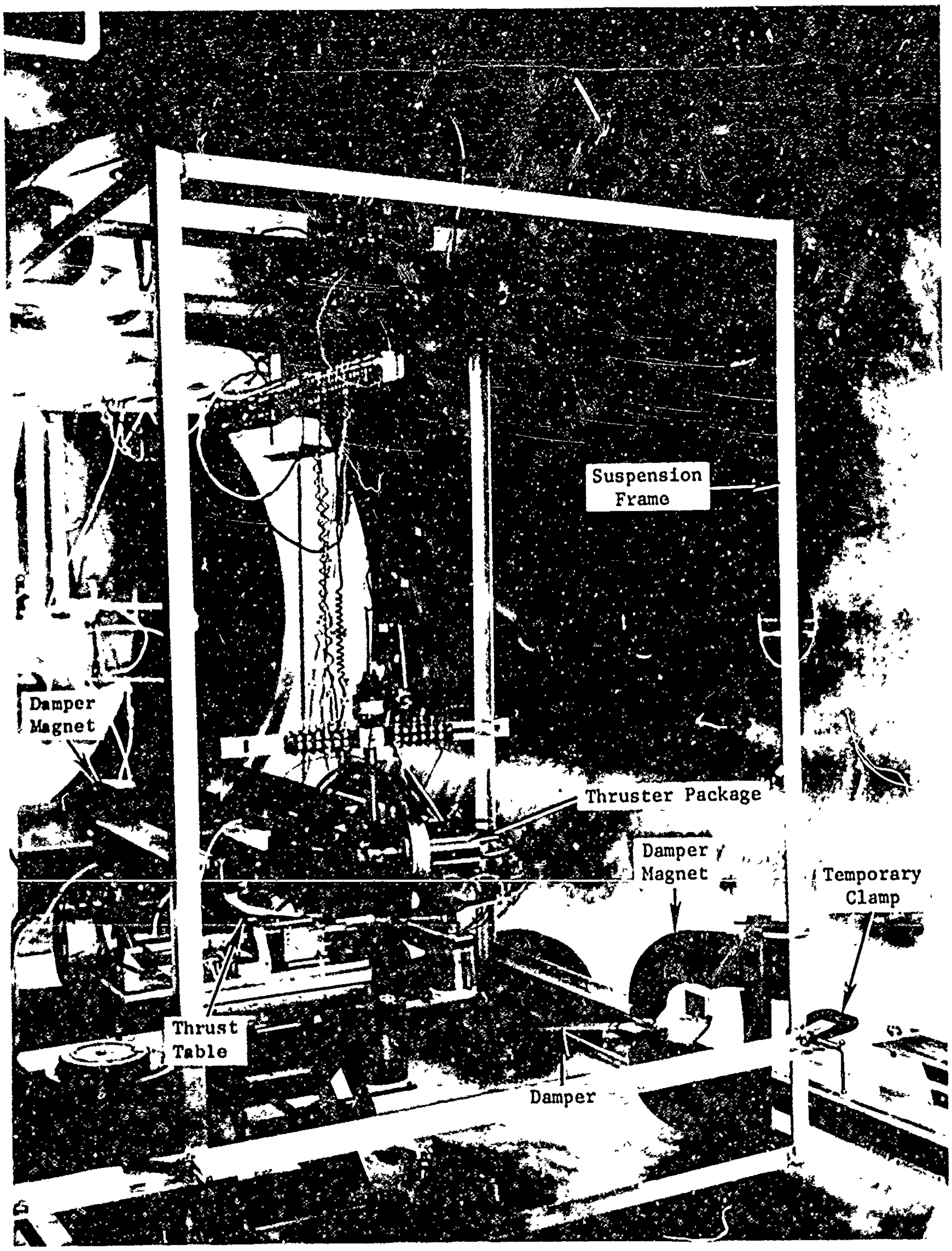


Figure 12. Front View, Microthrust Measuring System as Last Installed in a Vacuum Facility at General Electric Co.

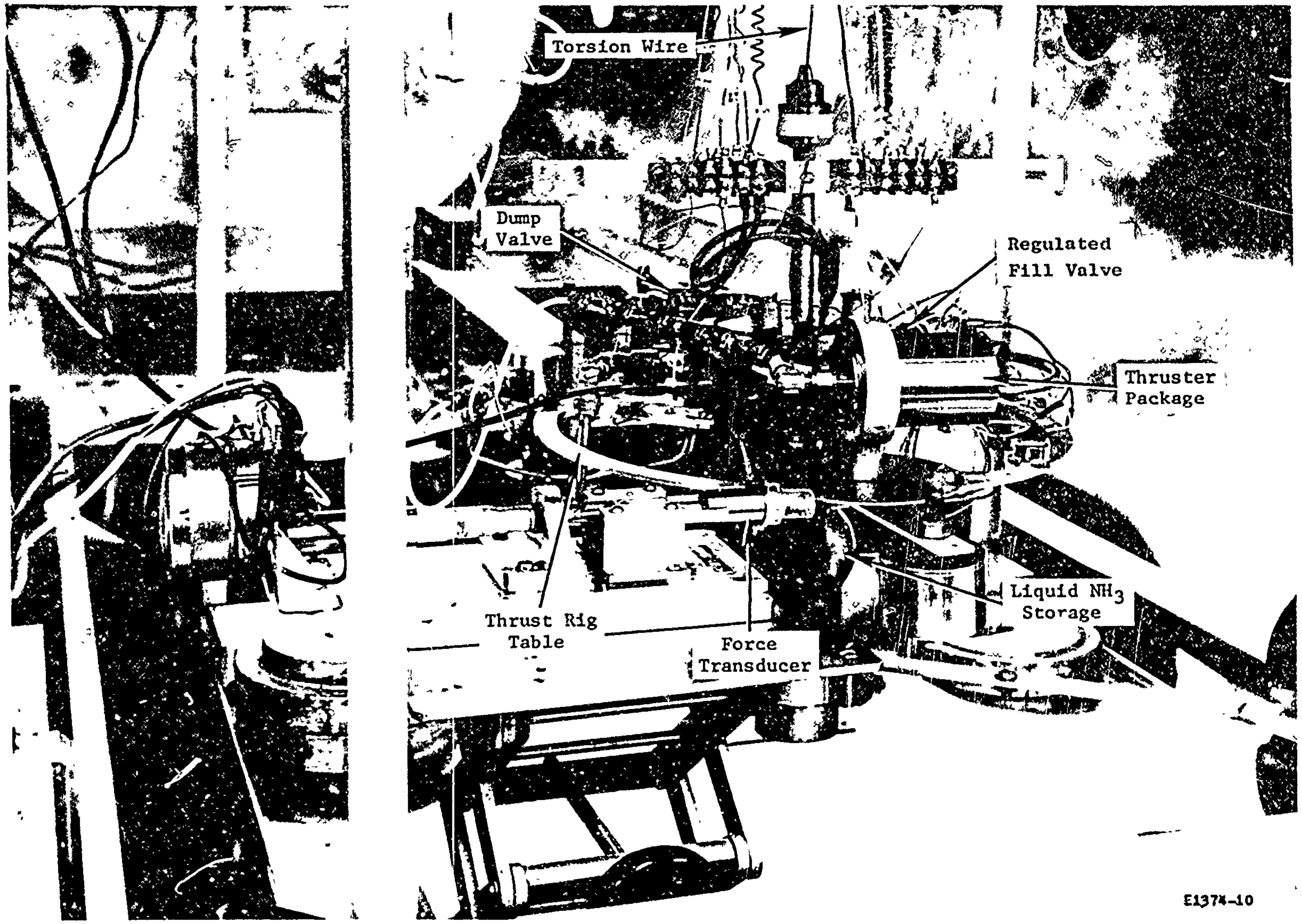


Figure 13. Front View, Thrust Table of Microthrust Measuring System as Last Installed in a Vacuum Facility at General Electric Company.

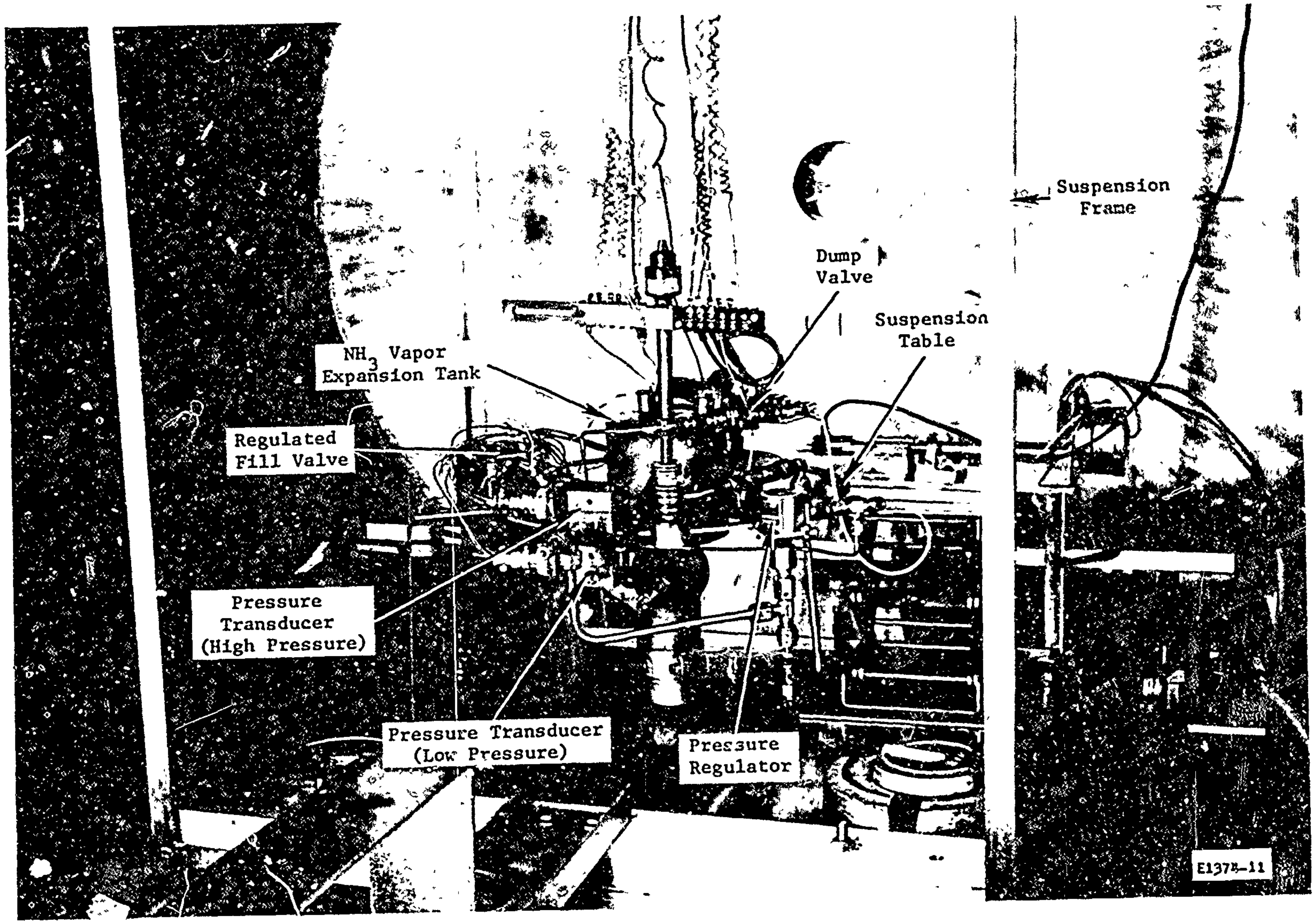


Figure 14. Rear View, Thrust Table of Microthrust Measuring System as Last Installed in a Vacuum Facility at General Electric Company.

from the top of an aluminum angle suspension frame (also magnetically-damped). This frame, in turn, is suspended by a chain within the interior of a vacuum chamber.

Schematically, the system appears as shown in Figure 15. The fill valve and dump valve are 28 volt DC normally-closed solenoid valves. Both the high pressure and low pressure differential pressure, are variable reluctance transducers. The high pressure unit is designed to indicate pressures up to 50 psid, the low pressure unit indicates pressures up to 25 psid. For readout, however, each requires a carrier demodulator. This unit, operating off 115V, 60 cps provides 5 K cps excitation to a bridge which includes two inductance ratio arms of the transducer. This bridge output is converted to DC. Readout may be obtained using any DC recorder. Neither the carrier demodulators nor the recorder is supplied as a part of the system. The force transducer (rated at $\pm$ 1 gram force maximum) operates directly into a carrier preamplifier and then into a recorder (Figure 16). Neither the carrier preamplifier nor the recorder is supplied as a part of the system.

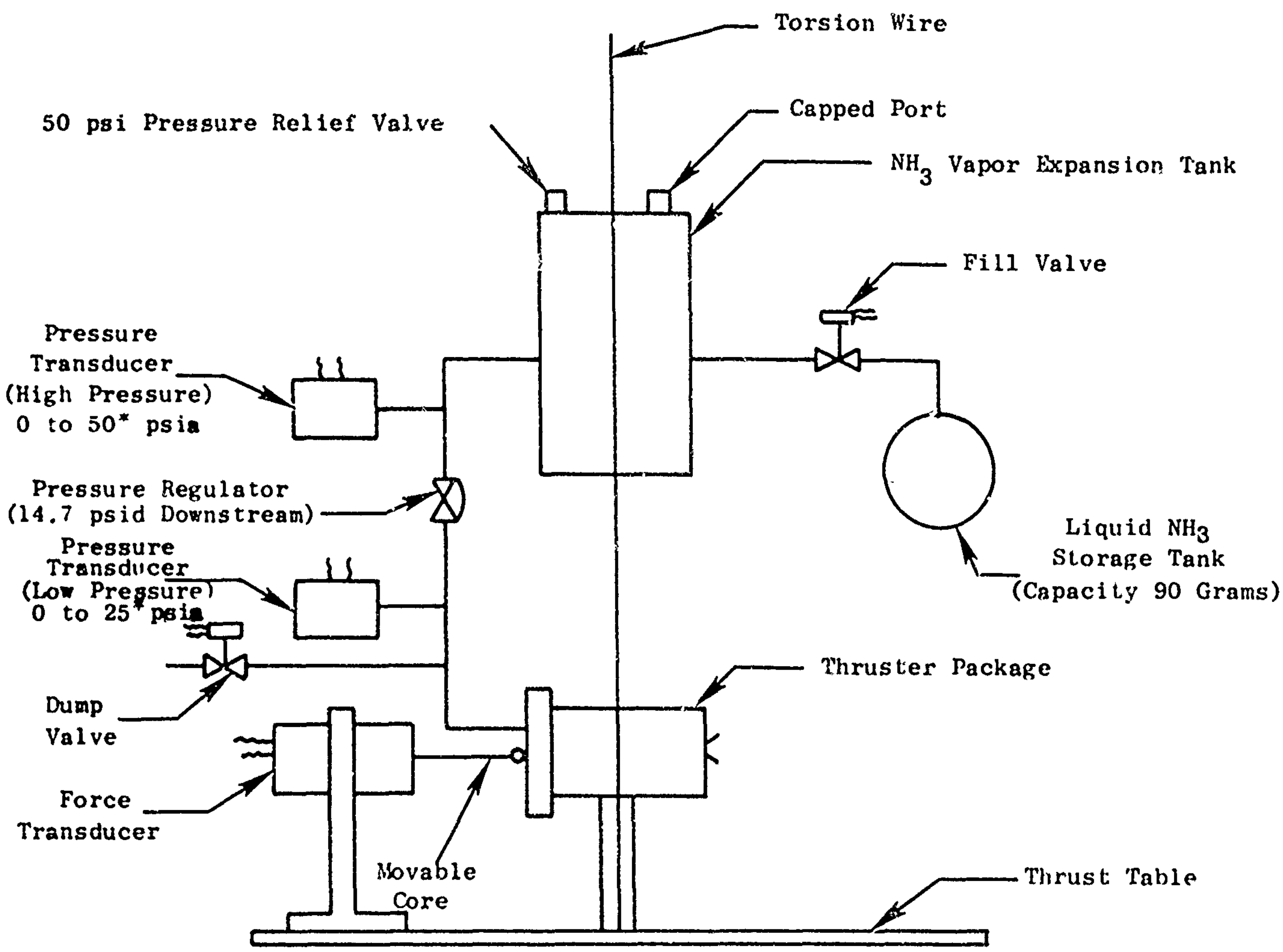
It may be observed in Figures 12, 13, and 14 that the thruster to be tested is mounted near the outer rim of the 12 inch diameter thrust table. Thrust is imparted tangentially, tending to rotate the table about the vertical wire axis. The force transducer movable core, as indicated in Figure 15, contacts the thruster package along its center line, indicating thrust directly. Note that the propellant storage and supply components are also mounted on the table. The instrumentation and power leads are thin, coiled, insulated wires strung between the table and the suspension frame that holds the supporting wire.

The force transducer itself may be used to determine the sensitivity of the system. If the micrometer drive is turned through some distance, such as 10 mils, a corresponding force reading will be obtained. The thrust system sensitivity in terms of mils* deflection/micropound force is simply 10/force reading and has been determined to be approximately 1 mil/micropound. This sensitivity evaluation may be performed with the vacuum tank open.

- Thrust Measurements

The first step in making thrust measurements was the calibration of the force transducer. With the transducer connected into the readout equipment as indicated in Figure 16, an electrical balance was first performed. A "known load" calibration was then conducted to determine the transducer linearity and then whenever desired, if it is believed that some change in characteristic may have occurred. This had to do with the application of a known load and adjusting the readout controls for a convenient deflection. The first step in this calibration consisted of mounting the transducer vertically so that weights could be suspended from the transducer core. In this position the weight of the core (approximately 525 micropounds) appeared on the

* Linear deflection at the 6 inch radius point.



*0 to 10 psia and 0 to 200 psia Diaphrams are Included with the System Hardware as Accessary Equipment

Figure 15. Pneumatic Schematic Diagram, Final Use of Microthrust Measuring System at GE.

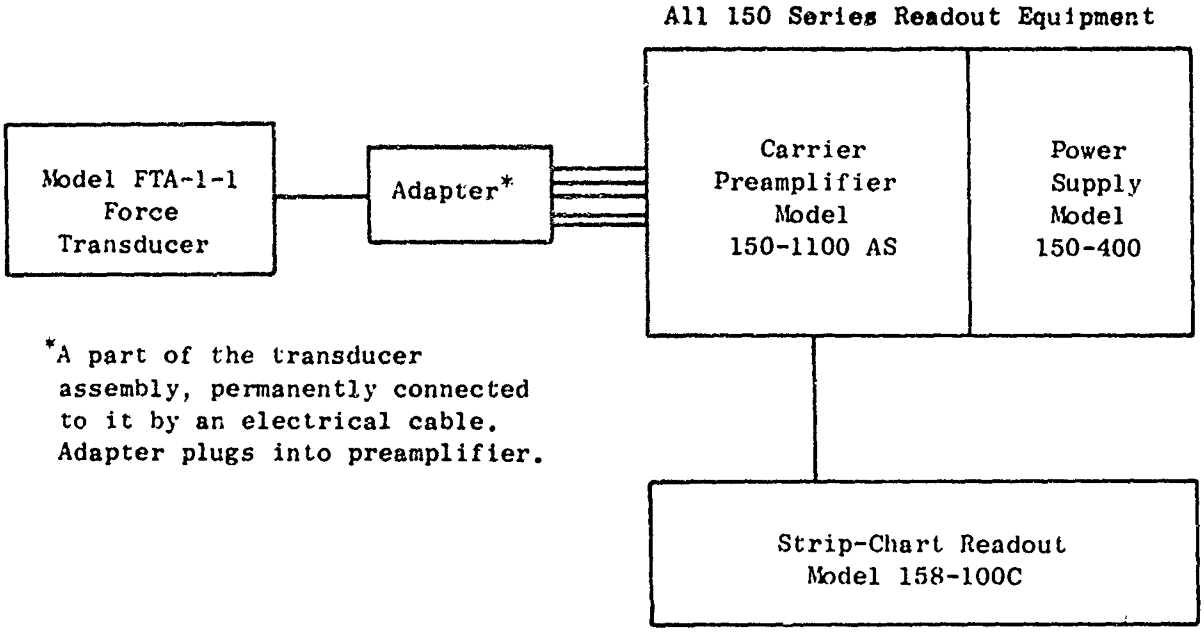


Figure 16. Schematic Diagram Force Transducer Readout, Microthrust Measuring System.

read-out chart. Calibration simply consisted of hanging weights on the transducer core and observing the force indicated on the readout chart. As an example of this work, data from calibrating the force transducer over the ranges of zero to 25 micropounds and zero to 250 micropounds appear on Table I. These data appear in curve form in Figure 17. Examination of these data show that the maximum recorded error on the 0 to 50 micropound scale was 1.7% of full scale (0.7 micropounds at 8.6 micropounds). This could be readily reduced 0.74% by averaging these readings, recalibrating the electronic readout equipment between readings. On the zero to 250 micropound scale, the maximum error was 3.3% of full scale (8.2 micropounds at 209 micropounds). This could also be reduced to less than 1% by averaging, as described above.

With the fuel tank filled with ammonia and all power and instrumentation checked out, attention was then directed towards the mounting of the thruster.

The thruster mounting bracket presently in position on the thrust table was designed to support a particular thruster known as the Preliminary Evaluation Thruster (an electrically heated, ammonia fueled, resistance jet thruster). When properly installed, the alignment mark on the bracket matched the radius mark on the thrust table. Bolting the thruster flange to the bracket automatically aligned the thruster axis with a tangent to the table. The force transducer had been mounted so that its axis is tangent to a circle through the mounting bracket center with its core contacting the center of the thruster (Figure 18).

It should be noted that the operation of any solenoid valve on the thrust table results in an interaction of its solenoid magnetic field with the damping magnet fields and to a lesser degree, with the earth's magnetic field. The disturbances introduced may take 10 to 20 seconds to dampen out. Consequently, thrust measurements were taken as follows:

Thrust Readings Less than 50 Micropounds:

For thrust readings less than 50 micropounds, the Microthrust Measuring System shown in Figure 15 was changed by the addition of a solenoid valve (the "thruster valve") in the line between the NH_3 vapor expansion tank and the high pressure pressure transducer. The force transducer was pre-loaded by driving it against the thruster to a value in excess of the micropounds to be measured. With the thruster valve open, the fill valve was then pulsed until the high pressure transducer readout indicated a pressure somewhat in excess of 30 psia within the NH_3 vapor expansion tank. With the thruster valve closed, the dump valve was opened momentarily to make certain that no gas under pressure was being supplied to the thruster. The low pressure pressure transducer zero was checked at this point and reset if necessary. The thruster valve was then opened long enough to allow the thruster pressure to reach a pressure in excess of its normal operating pressure and then closed. The pressure decayed through the orifice and by the time it reached the desired operating pressure of the thruster, the vibrations caused by actuating the thruster valve

TABLE I

FORCE TRANSDUCER
CALIBRATION DATA

MICROTHRUST MEASURING SYSTEM

Date	Calibration Weight (μ lb)	Force Reading (μ lb)	Readout Span (μ lb)	Error (Reading) (μ lb)	Error (Reading) %	Error % FS.
9-19-67	15.3	14.6	0-50	-0.7	4.6	1.4
	30.4	30.5	"	+0.1	0.3	0.2
	43.8	43.5	"	-0.3	0.7	0.6
	67.1	66	0-250	-1.1	1.7	0.4
	205.5	210	"	+4.5	2.2	1.8
9-28-67	8.60	8.1	0-50	-0.5	5.8	1.0
	8.60	8.7	"	+0.1	1.2	0.2
	8.60	7.9	"	-0.7	8.1	1.4
	18.74	18.4	"	-0.34	1.8	0.7
	25.13	25	"	-0.13	0.5	0.3
	42.33	42	0-250	-0.33	0.8	0.1
	75.28	76	"	+0.82	1.1	0.3
	109.60	112	"	+2.4	2.2	1.0
	148.80	153	"	+4.20	2.8	1.7
	30.42	31	"	+0.58	1.9	0.2
	43.87	43	"	-0.58	2.0	0.3
	66.14	67	"	+0.86	1.3	0.3
	67.02	70	"	+2.98	4.4	1.2
	208.8	217	"	+8.2	3.9	3.3

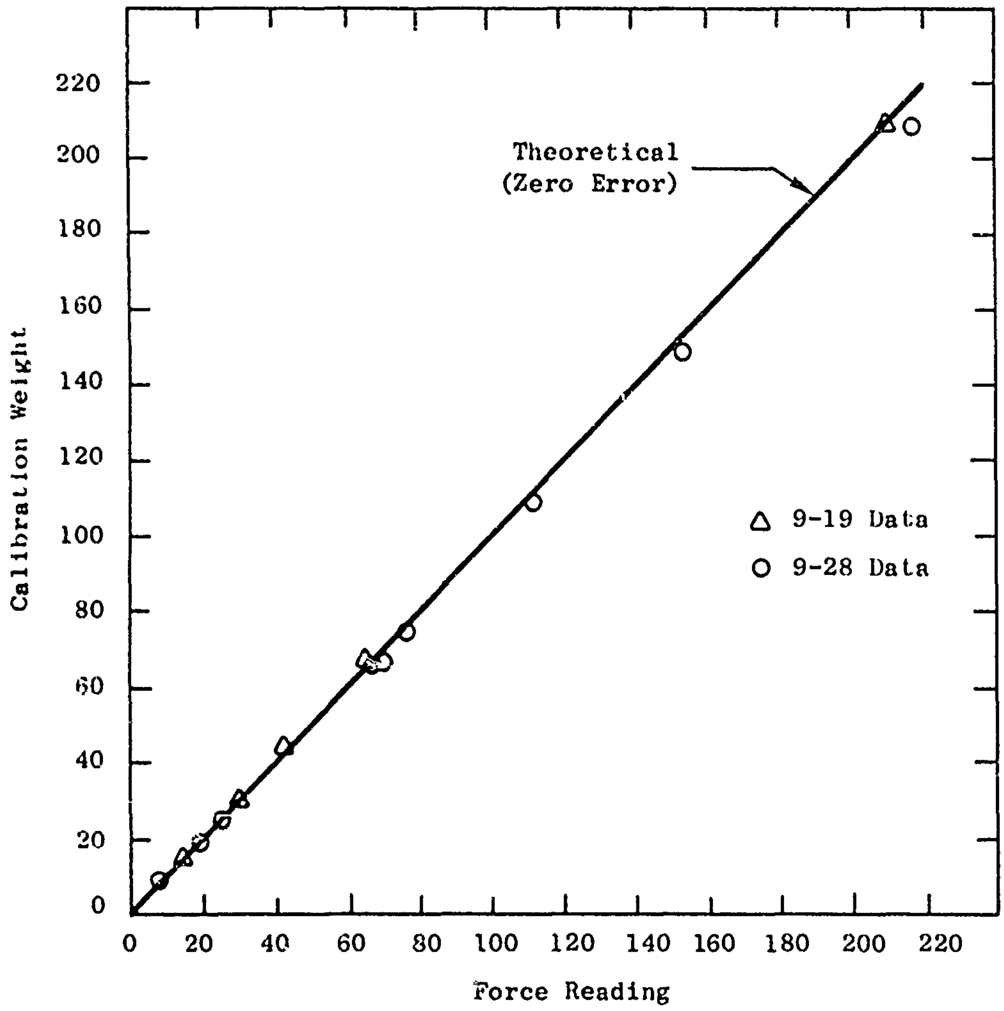


Figure 17. Force Transducer Calibration Characteristic Microthrust Measuring System.

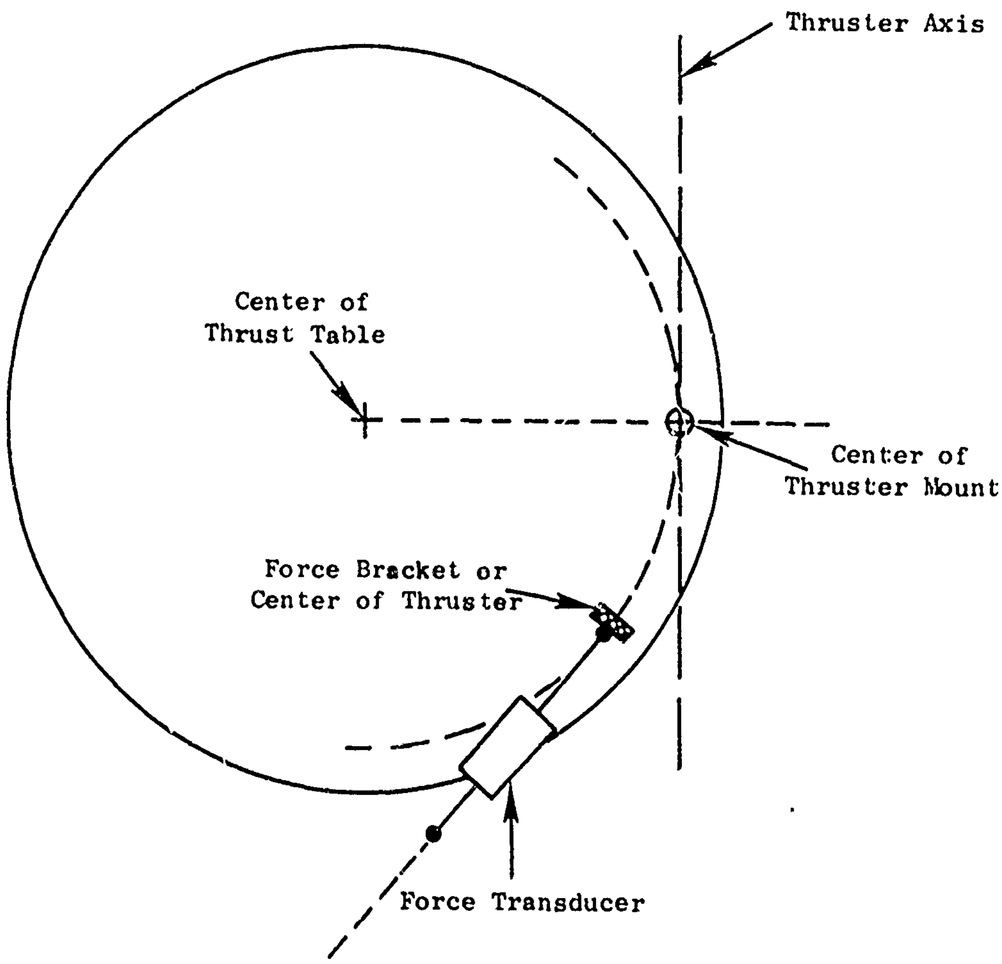


Figure 18. Force Transducer Alignment, Microthrust Measuring System.

had been dampened out and a clear thrust trace obtained. The thrust at this operating pressure point was determined by examining the readout chart. The dump valve was then operated to discharge the remaining ammonia gas from the lines. All zeroes were checked and the test repeated.

An example of such a thrust measurement appears in Figure 19. This shows a 3.2 micropound thrust trace. The pressure reading appearing on the chart was only an indication and was not calibrated. Pressure was generally read by observing a calibrated digital volt meter on the (low pressure) pressure transducer output and marking the chart at the appropriate time. The linearity of the thrust trace had already been determined from the Force Transducer Calibration previously performed. With the vacuum chamber pressure down to a desired value, the pressure transducer zeroes were checked out and reset if necessary.

Thrust Readings of 50 Micropounds and Up:

With the Microthrust Measuring System set up as shown in Figure 15, the force transducer was pre-loaded by driving it against the thruster to a value in excess of the micropounds to be measured. With the vacuum chamber pressure down to a desired value, the pressure transducer zeroes were checked out and reset if necessary.

The fill valve was then pulsed until the high pressure transducer readout indicated a pressure somewhat in excess of 30 psia within the NH_3 vapor expansion tank. The pressure of the entire volume was allowed to decay through the thruster nozzle, giving time for the vibration caused by the fill valve to dampen out before the desired operating pressure of the thruster was reached. The thrust at this operating pressure point was determined by examining the readout chart. The dump valve was then operated to discharge the remaining ammonia gas from the lines. All zeroes were checked and the test repeated.

An example of such a thrust measurement appears in Figure 11. This shows a 170 micropound thrust trace. The pressure reading appearing on the chart is only an indication and is not calibrated. Pressure is generally read by observing a calibrated digital voltmeter on the (low pressure) pressure transducer output and marking the chart at the appropriate time.

The linearity of the thrust trace has already been determined from the Force Transducer Calibration previously performed. With the vacuum chamber pressure down to a desired value, the pressure transducer zeroes were checked out and reset if necessary.

- Vector Tests

The thruster bracket was rotated 90° so that the thruster nozzle or orifice pointed outward from the thrust table axis. Alignment marks are provided for this on the thruster mounting bracket and the thrust table. The rotational orientation of the thruster on its own axis should be noted. The transducer was then preloaded to approximately 20 micropounds with the transducer core contacting the side of the thruster,

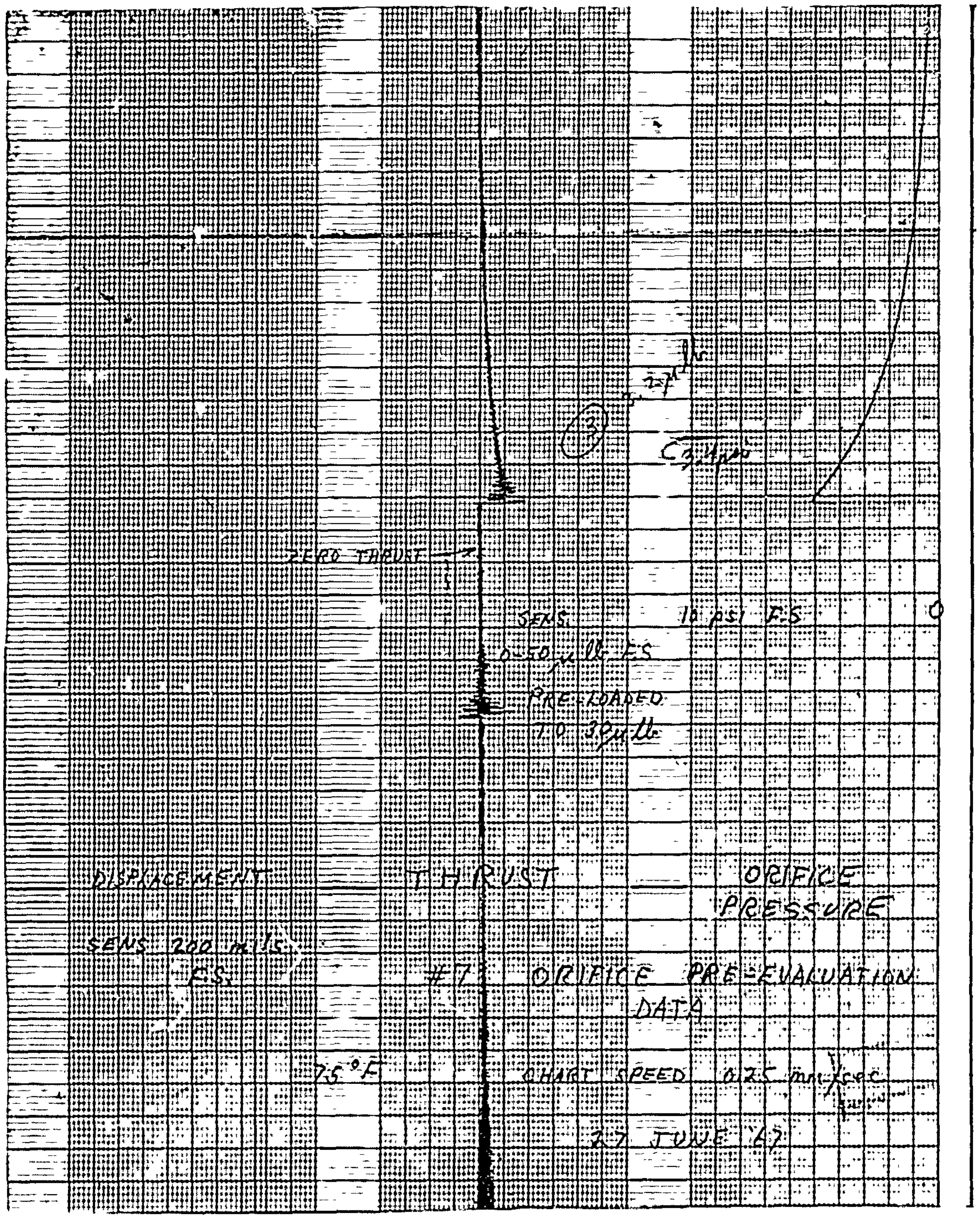


Figure 19. Thrust Trace, 3.2 Micropounds, as Measured with the Microthrust Measuring System.

using the care previously described in making thrust measurements. The thruster was then operated as previously described. The only thrust detected was that of a horizontal vector perpendicular to the thruster axis. With the thruster rotated 90° on its axis, the thrust measurement was repeated. Again, the rotational orientation of the thruster was noted. Using the results from each of these thrust measurements, the resultant thrust vector that is perpendicular to the thruster axis was calculated. The nozzle misalignment angle could also be determined from these data. Using this angle, the thickness of a shim could be calculated which, when inserted between the flange and mounting bracket, would correct the misalignment. With the shim in position, the vector tests should be repeated to verify the correction.

7. SUMMARY

As explained on the preceding pages, continued use of the Microthrust Measuring System resulted in simplification and refinement to such a degree that its performance was repeatable and its results accurate. For such a sensitive device, however, repeated checking and recalibration is recommended. For this system, an Installation and Operating Manual* has been prepared and is intended to be used as a guide for its use.

*Reference No. 7

SECTION III

PRELIMINARY EVALUATION THRUSTERS

The object of designing and fabricating the preliminary evaluation thrusters was that they would serve as a means of procuring thruster performance data that would be directly applicable to the design of the flight thrusters. The flight thrusters were required to produce a thrust of 3, 6, or 9 micropounds (station keeping application) with a specific impulse of 140 seconds, minimum, as well as a thrust of 200 micropounds (station changing application) with a specific impulse of 200 seconds, minimum.

1. DESIGN, FABRICATION, AND BUILDUP

The design (Figure 20) of these thrusters was such as to permit the testing of the largest number of orifices and nozzles with the least number of thruster bodies, heat shielding packages, and heaters. The plan was that the orifice and nozzle plates could be readily fastened one at a time to a thruster body by welding and then cut off when the evaluation was completed, being replaced with another for the next test. It was intended that three thrusters would be built up for test, one with a catalyst bed and two without.

To arrive at the required orifice/nozzle dimensions (Figures 21 and 22) and the operating pressures for the necessary thrust levels, numerous computer solutions were obtained. The results of this effort indicated that for the station keeping thrust level, the orifice diameter should be very nearly one mil; whereas, the station changing thruster would require a nozzle with a throat diameter of about 4 mils. It was also expected that the required chamber pressures for the operation of the station keeping and station changing thrusters should be of the order of 1 atmosphere. Accordingly, a total of 10 orifices and 4 nozzles were procured and prepared for test.

An exploded view of the thruster appears in Figure 23. An orifice plate is shown in Figure 24. The completely assembled thruster is shown in Figures 7 and 8.

2. PERFORMANCE TESTING OF ORIFICES AND NOZZLES

- Orifice Testing

Orifices finally selected for 3 and 6 micropounds thrust at 3.4 psia ammonia pressure were sharp-edged and varied between 1.0 and 1.5 mils in diameter. A second set of orifices of 3.9 mil diameter were also selected to provide for 200 micropounds thrust at a pressure of 15.2 psia of ammonia. Some of these latter were straight, sharp-edged orifices similar to the smaller ones and the remainder were convergent-divergent nozzles.

As a preliminary check, the orifices were sized using a wet test meter and helium at constant flow rate. The pressures used were 50 and

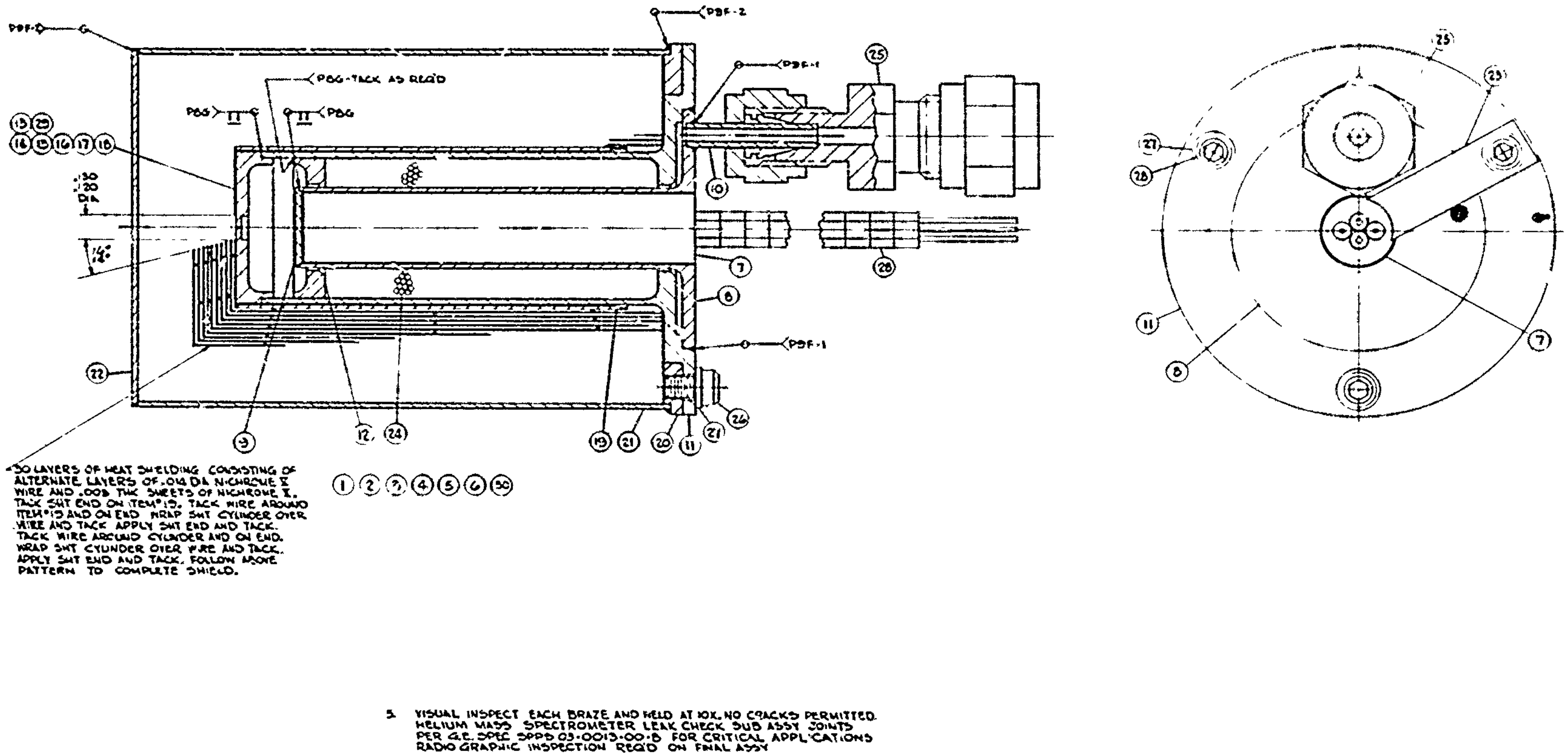


Figure 20. Assembly, Preliminary Evaluation Assembly.

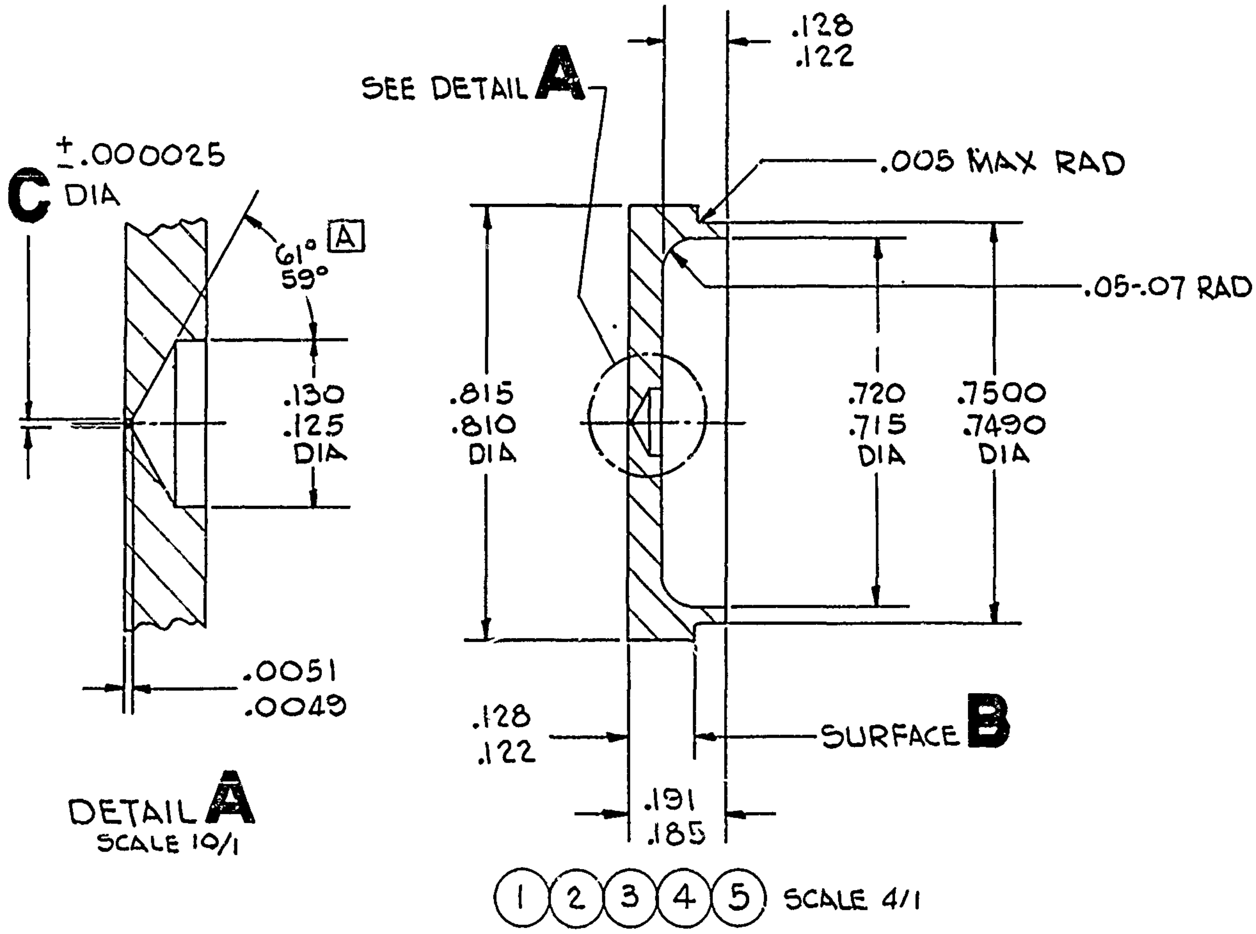


Figure 21. Orifice Design, Preliminary Evaluation Assembly.

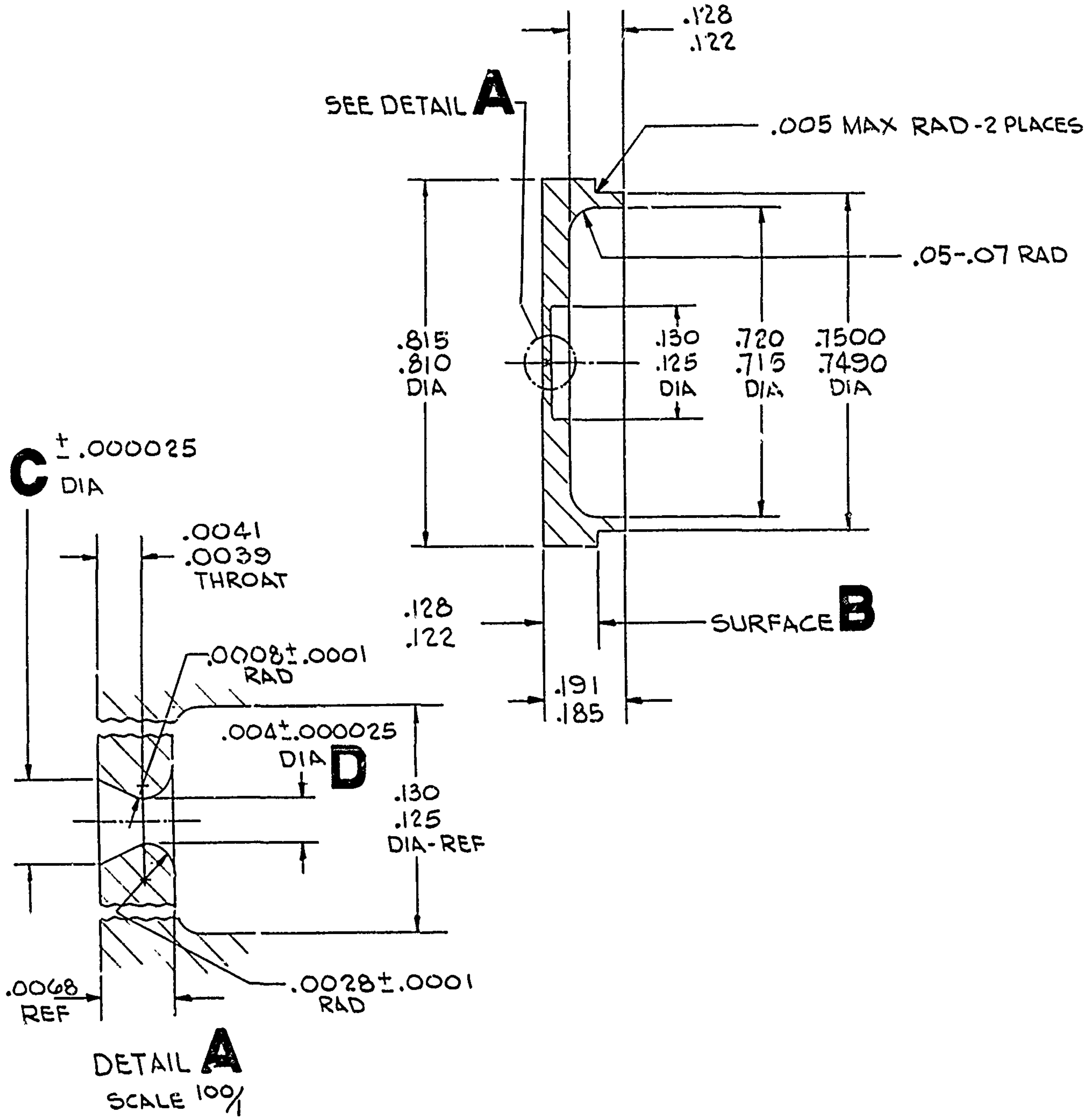


Figure 22. Nozzle Design, Preliminary Evaluation Thruster.

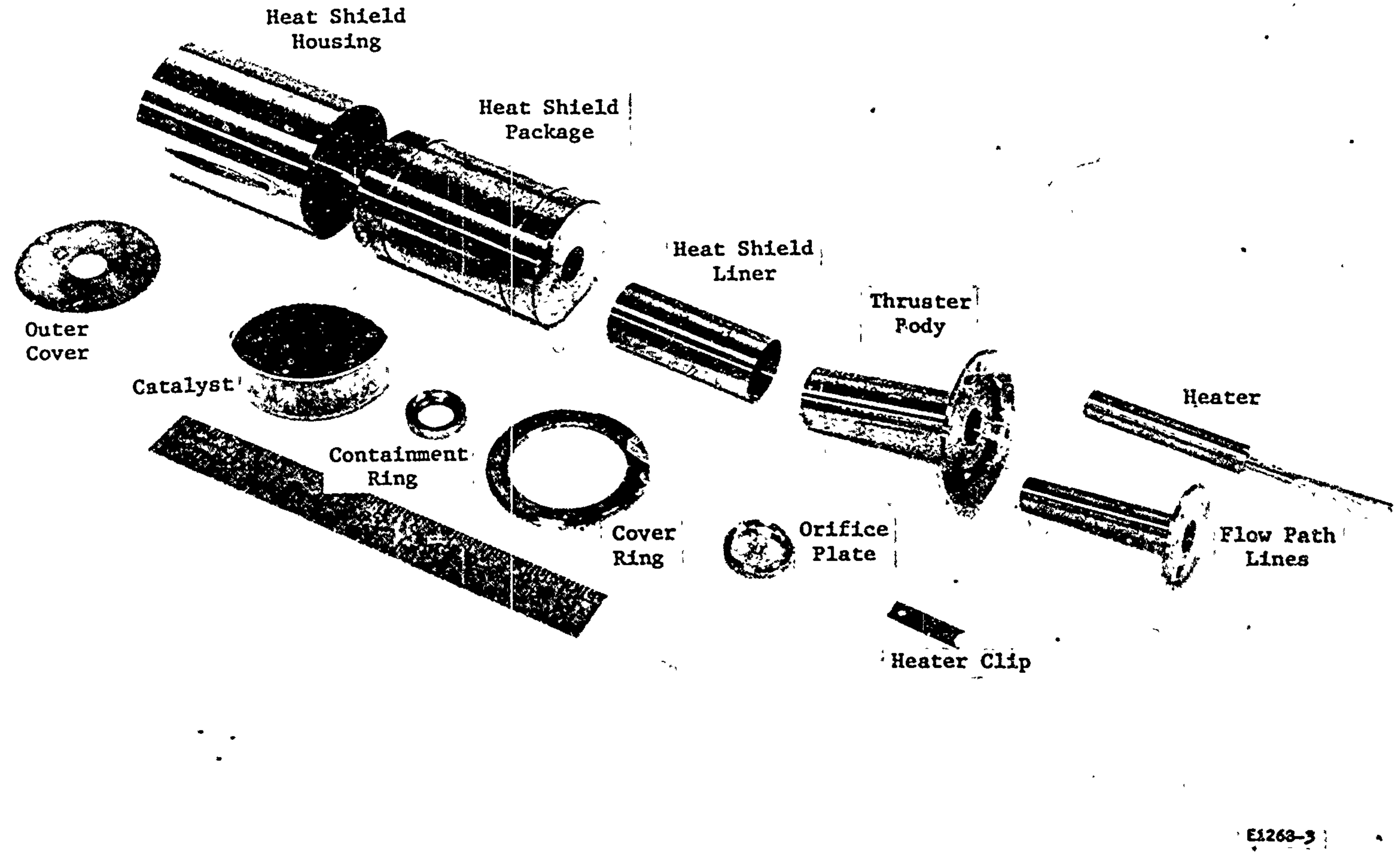


Figure 23. Preliminary Evaluation Thruster (Exploded View Showing Components).

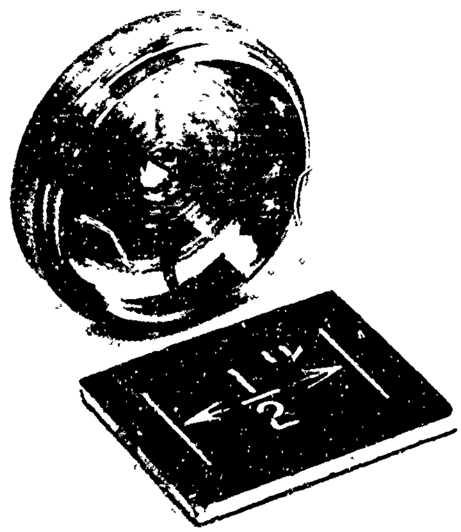


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Figure 24. Test Orifice Used in Preliminary Evaluation Thrusters.

100 psi to assure sonic flow during the tests.

The effective areas determined by this method were slightly smaller than the machined diameter as seen in Table II. However, initial thrust values obtained with orifices No. 2 and No. 7 were considerably less than expected, indicating an even smaller effective area than previously measured.

An apparatus was set up whereby the ammonia mass flow rate through the orifices could be measured under vacuum at actual operating pressure. Data from these tests showed that the effective areas were indeed smaller at the lower pressure (3.4 psi) and corresponded quite closely with thrust values obtained. Later data with the larger orifices at 15.2 psi, however, showed no significant change from the original measurements with helium. This effect at the lower pressures is attributed to a build-up of the boundary layer within the orifice.

A summary of all the data appears in Table II. Detailed characteristics of orifices No. 7 and No. 9 are shown in Figures 25, 26, 27, and 28.

Because of the low thrust obtained with the small orifices, several were rebored to 2.0 and 2.5 mils. Data from tests on these, along with data previously obtained, bracket the 3 to 6 micropound range as shown in Figures 29 and 30. These data provided for the final selection of the orifice sizes.

- Nozzle Testing

Testing was continued on nozzles Number 11 and 13 as described in Table II. Their performance was evaluated over an ammonia gas temperature range extending from room ambient to 2060°F. Thrust values obtained with nozzle Number 11 (3.90 mil throat diameter, 7.24 mil exit diameter) were essentially the same for nozzle Number 13 (3.90 mil throat diameter, 7.81 mil exit diameter) over that temperature range with a chamber pressure of 14.8 psia. These characteristics are shown in Figure 31, Thrust vs. Temperature. Mass flow rates for nozzle Number 11 and nozzle Number 13 were also essentially the same (Figure 32) when measured over an ammonia gas temperature range extending from room ambient to 2060°F with a chamber pressure of 14.8 psia.

It may be pointed out that the mass flow rates in Figure 32 at 70°F did not agree with those published in Table II. Mass flow rate was measured by allowing the pressure of a known volume of gas to decay through the orifice or nozzle under test at a constant pressure using a pre-set pressure regulator. A 15% decrease in mass flow rate was observed after substituting a 14.8 psia regulator for the previously used 15.2 psia regulator. This change in chamber pressure should have resulted in only a 2.6% decrease in mass flow rate. The discrepancy was attributed to either a leak in the system during the earlier tests, or a leaky fill valve to the "known volume" during the later tests. A check of the valve proved it to be leak tight and functioning properly. Therefore, the later data is assumed to be correct, particularly because all checks showed the system to be functioning properly. The

TABLE II

SUMMARY OF ORIFICE PRELIMINARY EVALUATION DATA

Orifice No.	Diameter (mils) Machined	Measured 3-4 atm He	Measured 3.4 psi NH_3	Measured 15.2 psi NH_3	Temp °F	Mass Flow (NH3) at 3.4 psi ($x10^{-8}$ lb/sec)	at 15.2 psi ($x10^{-6}$ lb/sec)	Thrust* 3.4 psi μlb	15.2 psi μlb	Specific Impulse (sec)
2	1.00	0.835	---	---	70	---	---	1.4	---	---
5	1.30	1.124	.970	---	70	5.00	---	---	---	---
7	1.50	1.29	1.09	---	70	5.44	---	3.3	---	61
"	---	---	---	---	1000	2.68	---	2.4	---	90
"	---	---	---	---	2000	1.05	---	1.3	---	124
9	3.90	3.47	3.26	3.46	70	---	2.44	---	152	62
"	---	---	---	---	1000	---	1.61	---	130	81
"	---	---	---	---	2000	---	0.88	---	113	128
11	3.90	---	3.44	3.56	70	---	2.61	---	---	---
13	3.90	---	3.44	3.61	70	---	2.67	---	---	---

*Note: The phenomenom of decreasing micropound thrust with increased temperature for very small orifices or nozzles of fixed mechanical dimensions (and with constant upstream gas pressure) first became evident in this program at this point. It was observed repeatedly throughout the program and appeared to be a characteristic of thrust values in the micropound range for orifices and nozzles with throat diameters of a few thousandths of an inch.

A determination of the cause of this phenomenom was beyond the scope of this program. It would have been investigated under a portion entitled "Associated Technology", but this was deleted at the start of the program by mutual agreement of the Air Force and GE. A formal preposal to evaluate orifice thrust performance was submitted to the Air Force on April 9, 1968, but no award resulted.

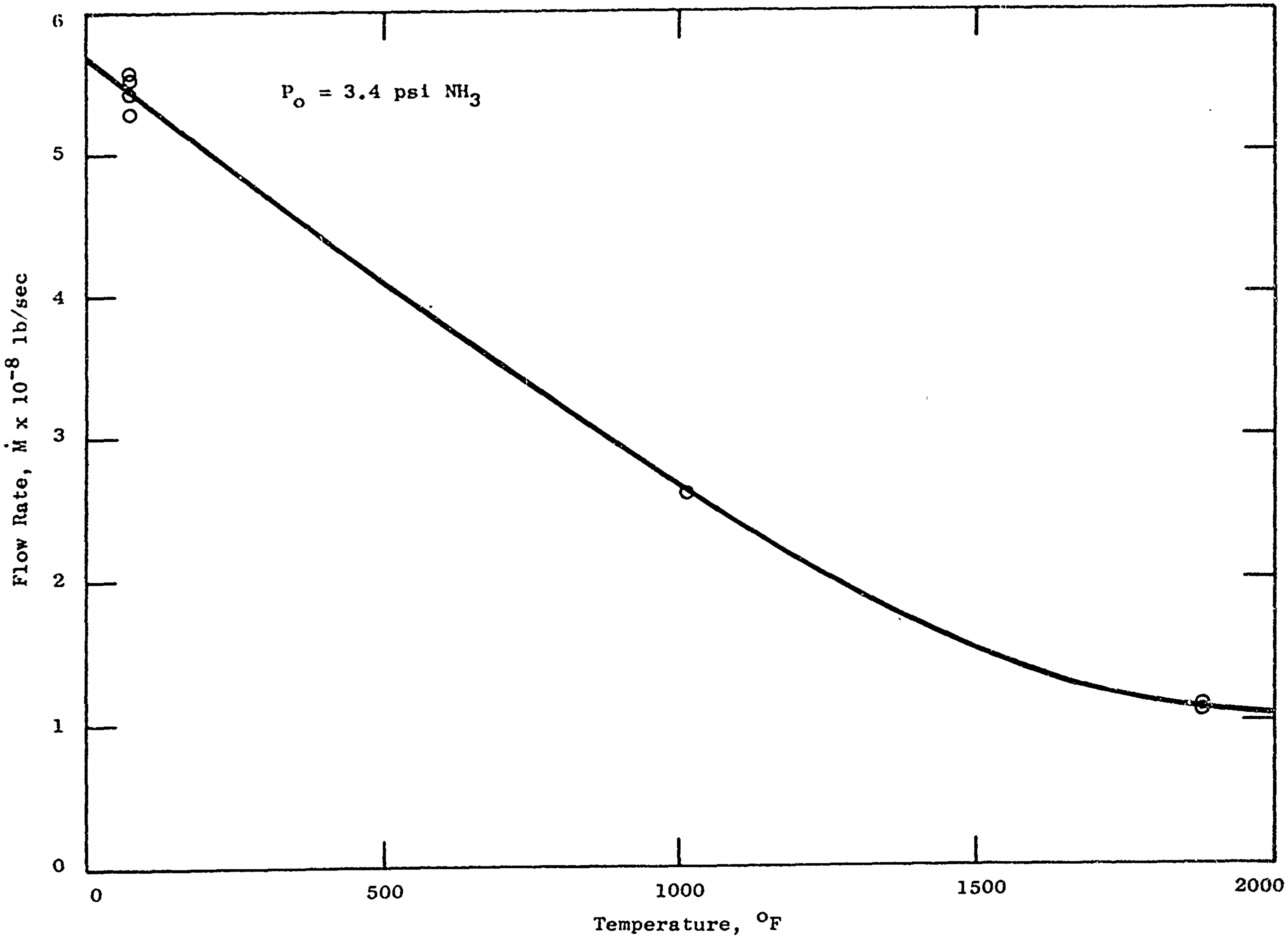


Figure 25. Mass Flow Rate vs Temperature Characteristic Orifice #7 Preliminary Evaluation Thrusters.

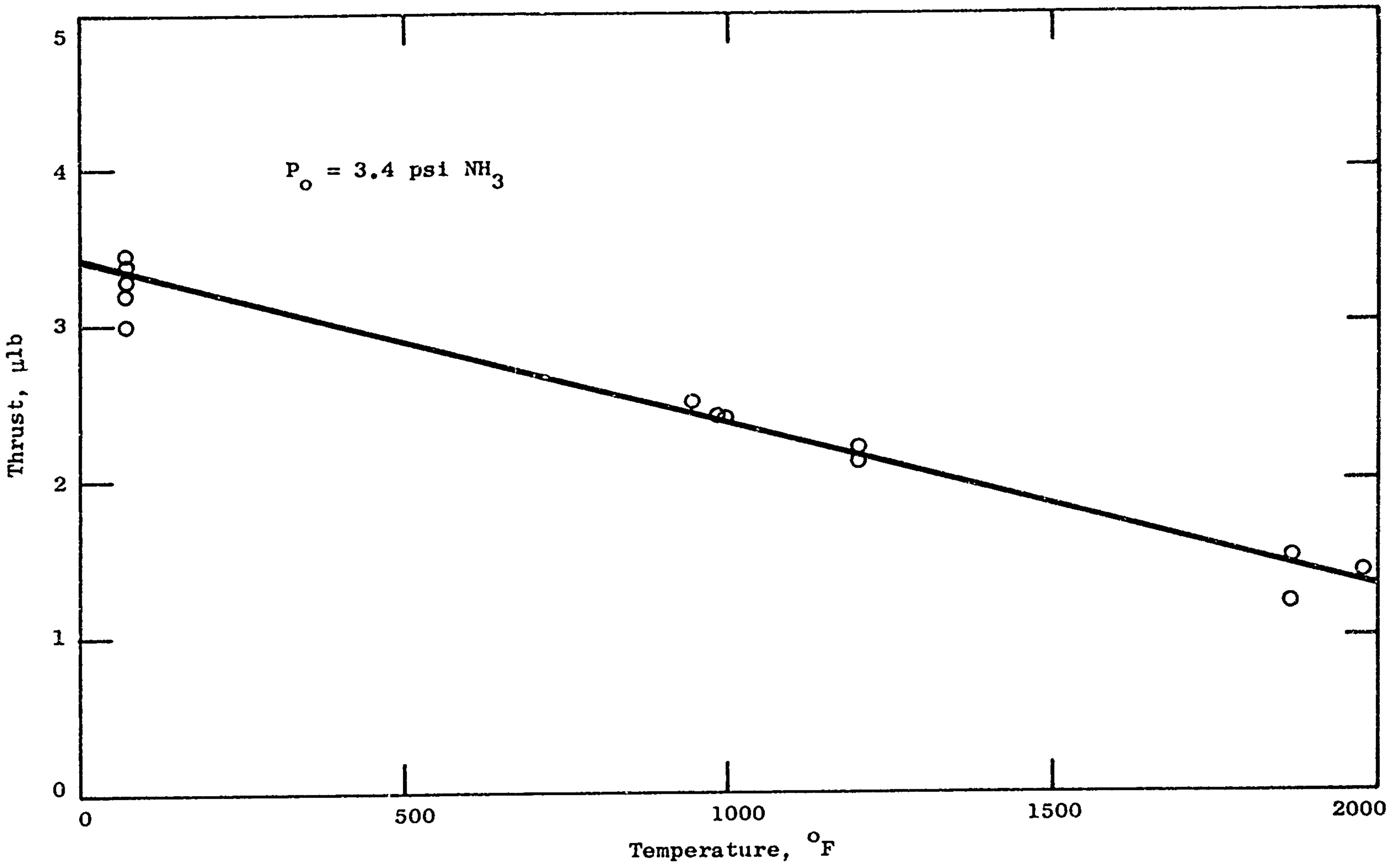


Figure 26. Thrust vs Temperature Characteristic Orifice #7 Preliminary Evaluation Thrusters.

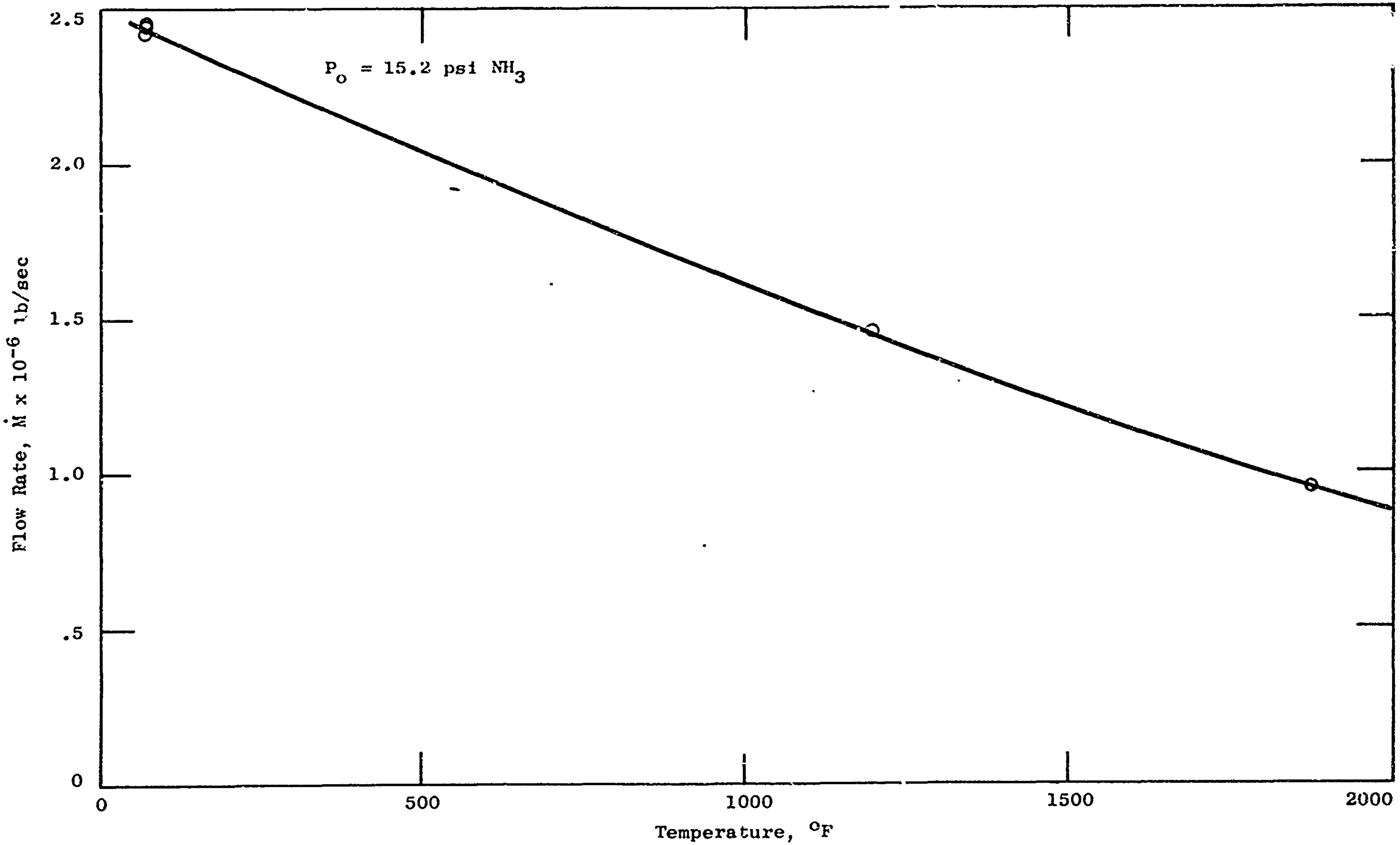


Figure 27. Mass Flow Rate vs Temperature Characteristic Orifice #9 Preliminary Evaluation Thrusters.

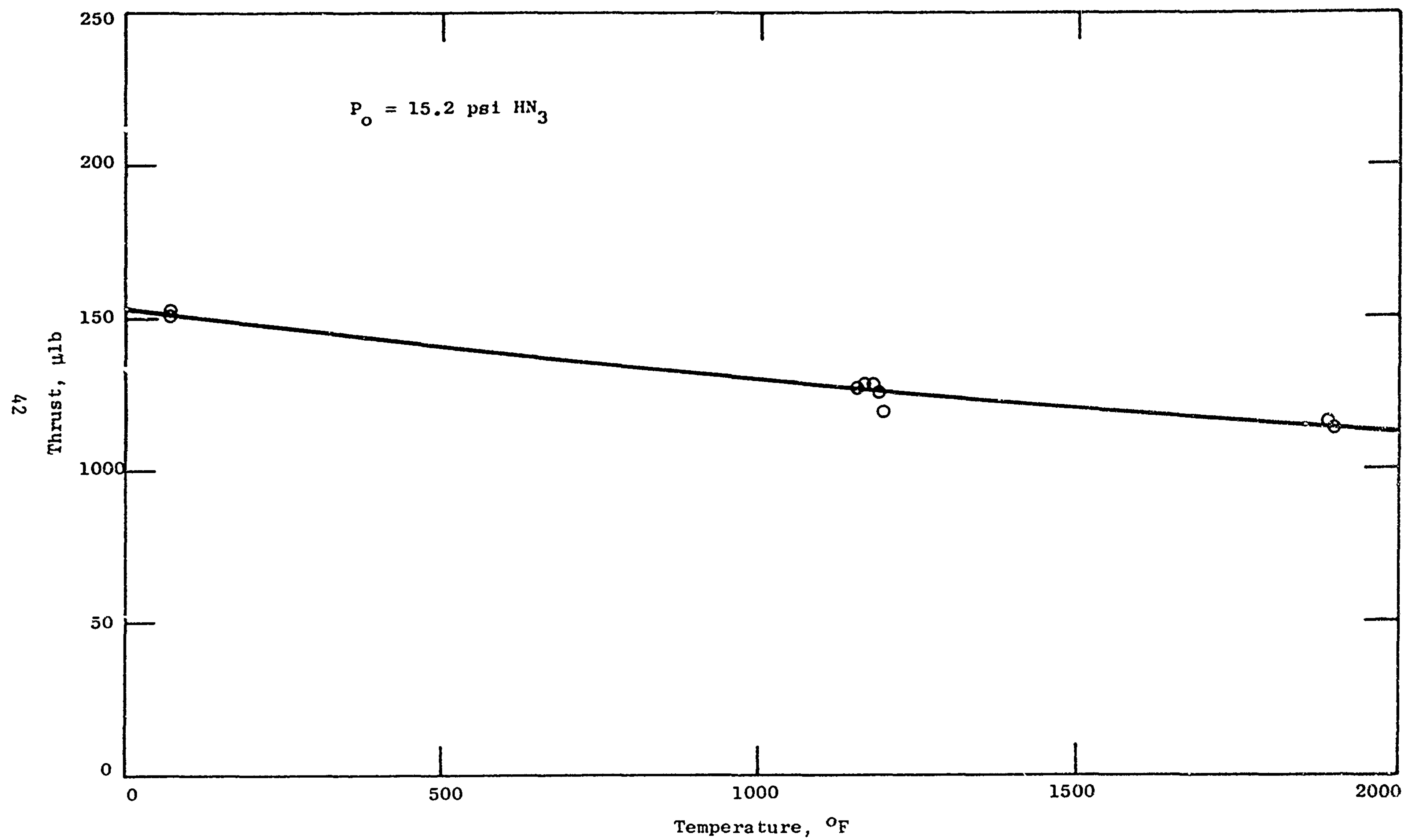


Figure 28. Thrust vs Temperature Characteristic, Orifice #9 Preliminary Evaluation Thrusters.

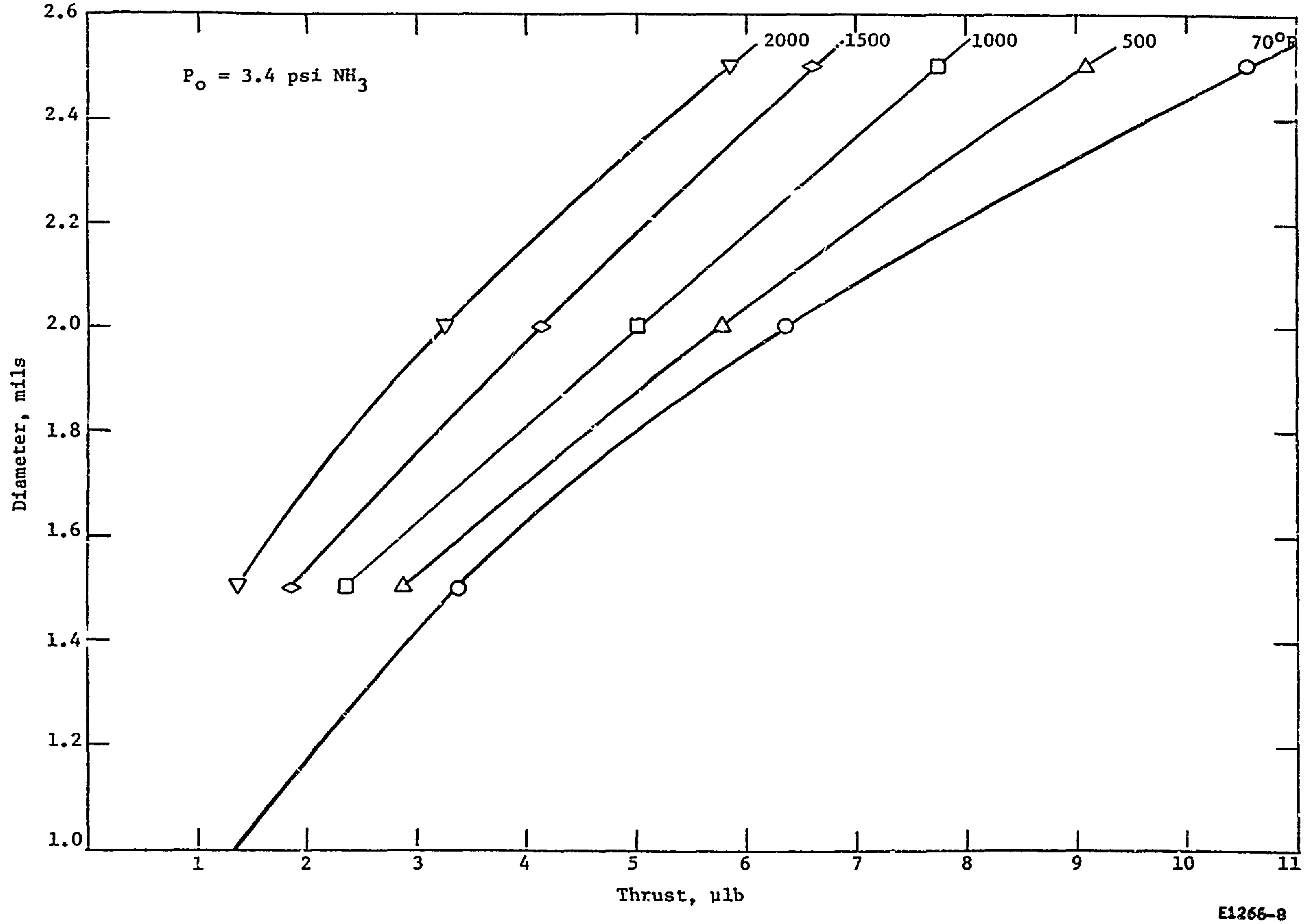


Figure 29. Orifice Machined Diameter vs Thrust, Preliminary Evaluation Thrusters.

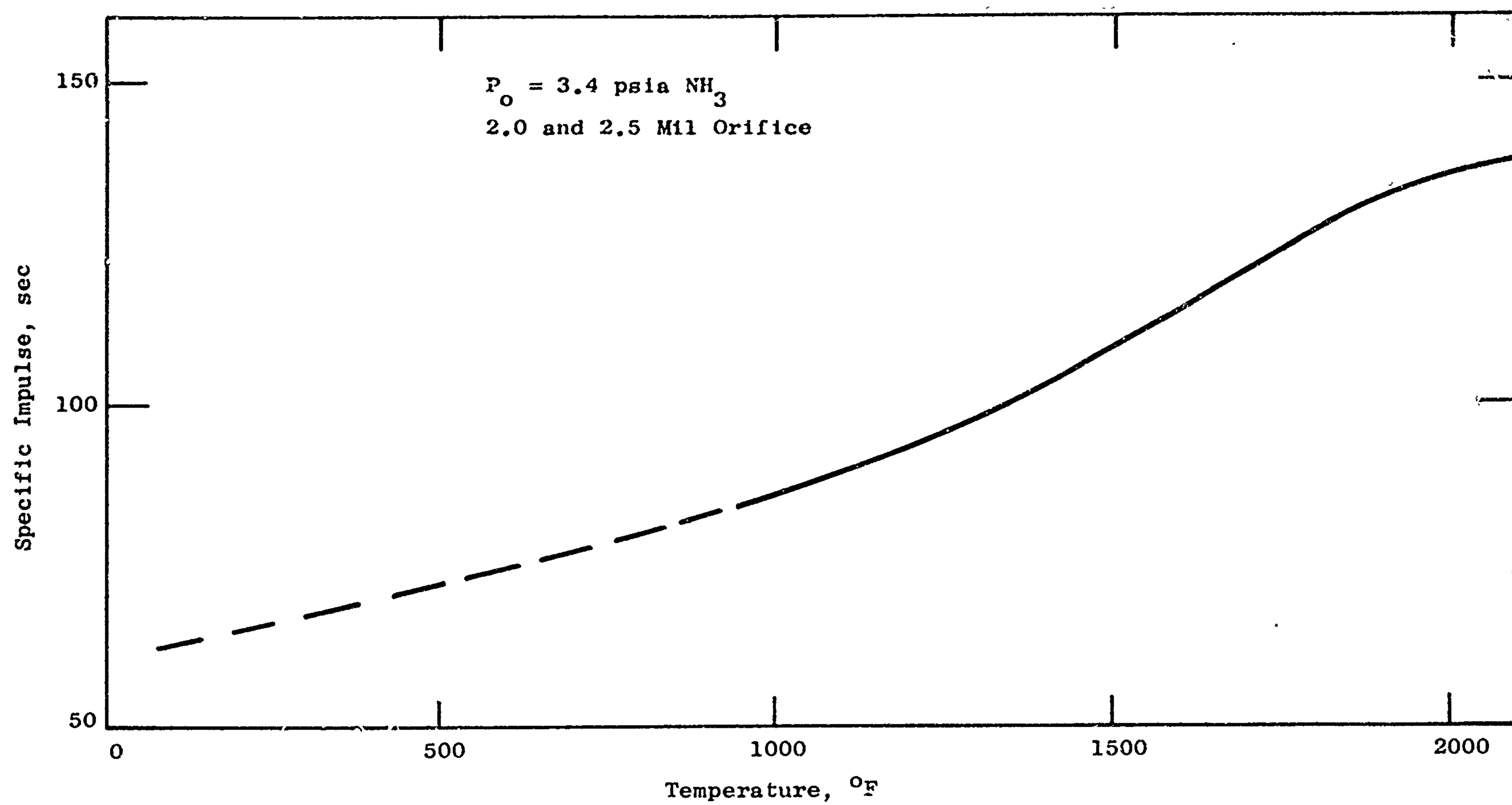


Figure 30. Specific Impulse vs Temperature, 3 and 6 Micropound Orifices (Non-Catalyzed), Preliminary Evaluation Thrusters.

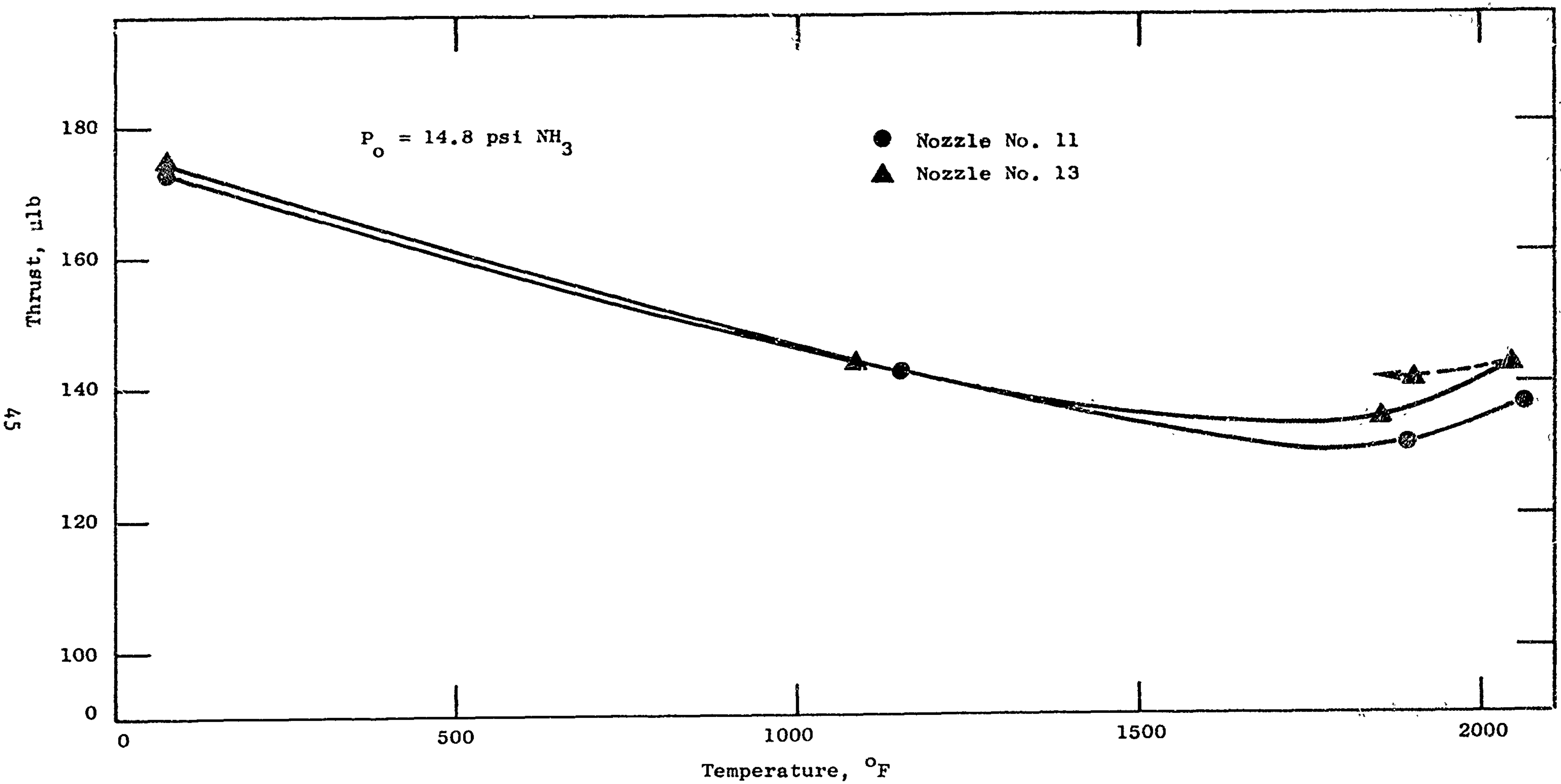


Figure 31. Thrust vs Temperature Characteristic for Convergent-Divergent Nozzles Number 11 and 13, Preliminary Evaluation Thrusters.

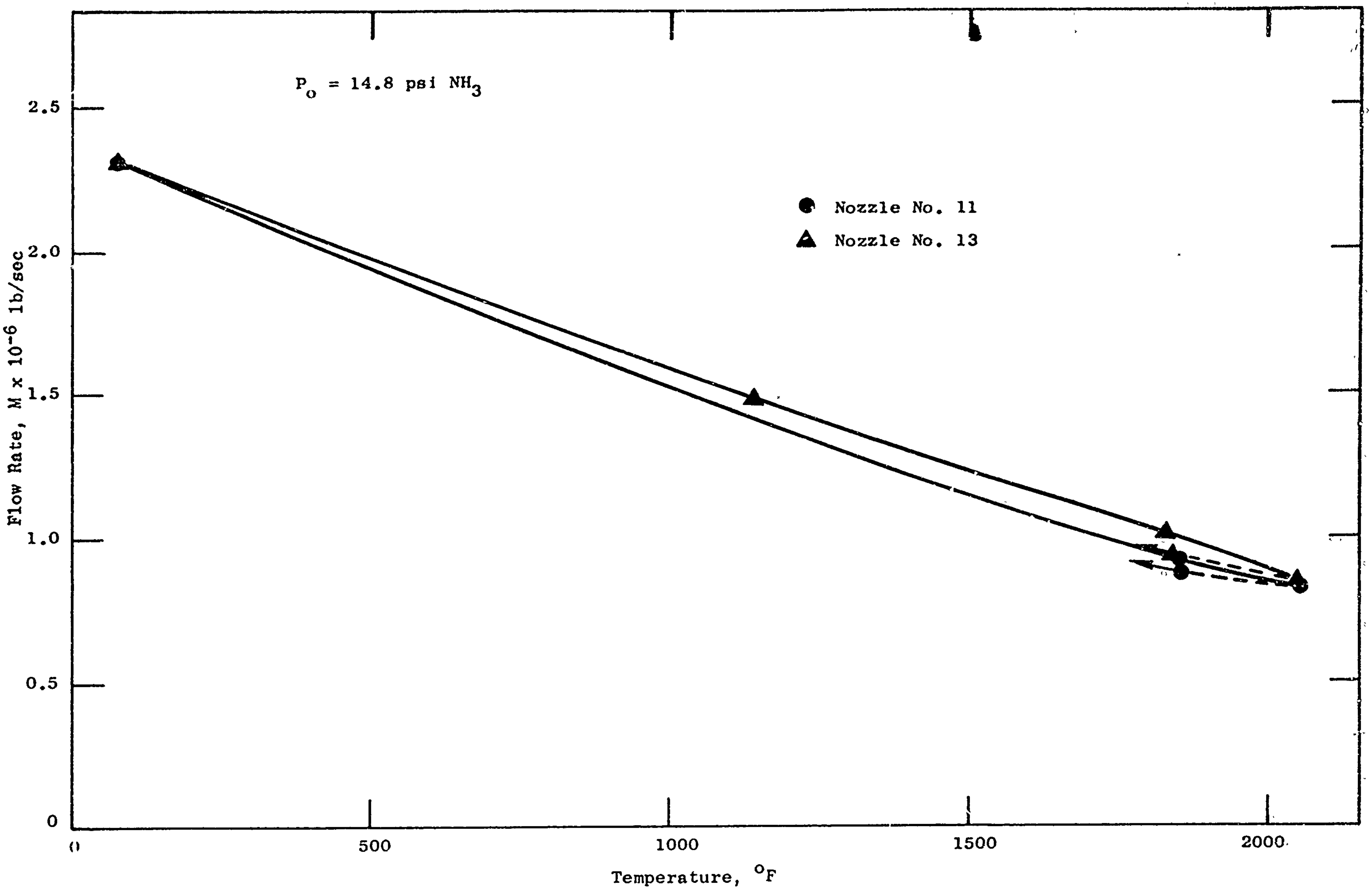


Figure 32. Mass Flow rate for Convergent-Divergent Nozzles, Number 11 & 13, Preliminary Evaluation Thrusters.

curve of specific impulse vs. Temperature (Figure 33) for nozzles Number 11 and 13 represents data procured with the leak-tight system.

All the performance tests on nozzles thus far have been performed without a catalyst bed in place. Therefore an evaluation of the convergent-divergent 200 micropound thrust nozzle No. 14 (physically identical to nozzle No. 13 with a 3.90 mil throat diameter and a 7.9 mil exit diameter) with the decomposition catalyst in place was performed. A curve of Mass Flow Rate vs. Temperature appears in Figure 34. Thrust vs. Temperature is shown in Figure 35. A plot of Specific Impulse vs. Temperature for both the catalyzed and uncatalyzed thruster appears in Figure 36. Examination of these data indicate that the nozzle was not yet properly sized for 200 micropound thrust. Accordingly, arrangements were made to procure three additional nozzles for testing with the intention of more closely bracketing the 200 micropound thrust level:

Additional Nozzles	Throat Diameter Mils ± 0.000025 Mil	Exit Diameter Mils ± 0.000025 Mil
First	0.0044	0.0082
Second	0.0046	0.0084
Third	0.0048	0.0086

Further examination of the data shows that at temperatures other than 2000°F the catalyst did improve the specific impulse to a degree that warranted its inclusion in the fabrication of the 200 micropound portion of the flight thruster. Specifically, at 1500°F, the specific impulse was increased from 110 seconds to 128 seconds, a 16 percent improvement.

At this point in time, discussions with the Applied Physics Laboratory personnel of the Johns Hopkins University and the Wright-Patterson Air Force Base personnel resulted in a decision to operate the thrusters at 1500°F rather than 2000°F to reduce the electrical power consumption in the thrusters. Also, a re-evaluation in the thrust requirements by APL resulted in a suggestion by them that the 200 micropound thrust value be reduced to 100 micropounds. It appeared agreeable to all concerned that the values of 3, 6 and 100 micropounds be nominal values. The 3 micropound thrust value could be anywhere between 2 and 5 micropounds; the 6 micropound between 5 and 10 micropounds; the 100 micropound value between 75 and 120 micropounds. It was expected that the thrust of each thruster would be known and each microthruster system would contain thrust-matched thrusters. It was expected that the variation in thrust from system to system would be nominal.

Because the required operating temperature of the thruster was reduced from 2000°F to 1500°F, subsequent tests on nozzle performance were conducted at temperatures only up to 1500°F. At the same time a thrust value of 100 micropounds at a specific impulse of 100 seconds, minimum was imposed at this 1500°F operating temperature. Thus tests on the 4.6 mil and 4.8 mil diameter nozzles were conducted over a temperature range of 70°F to 1500°F and thrust was examined up to 100

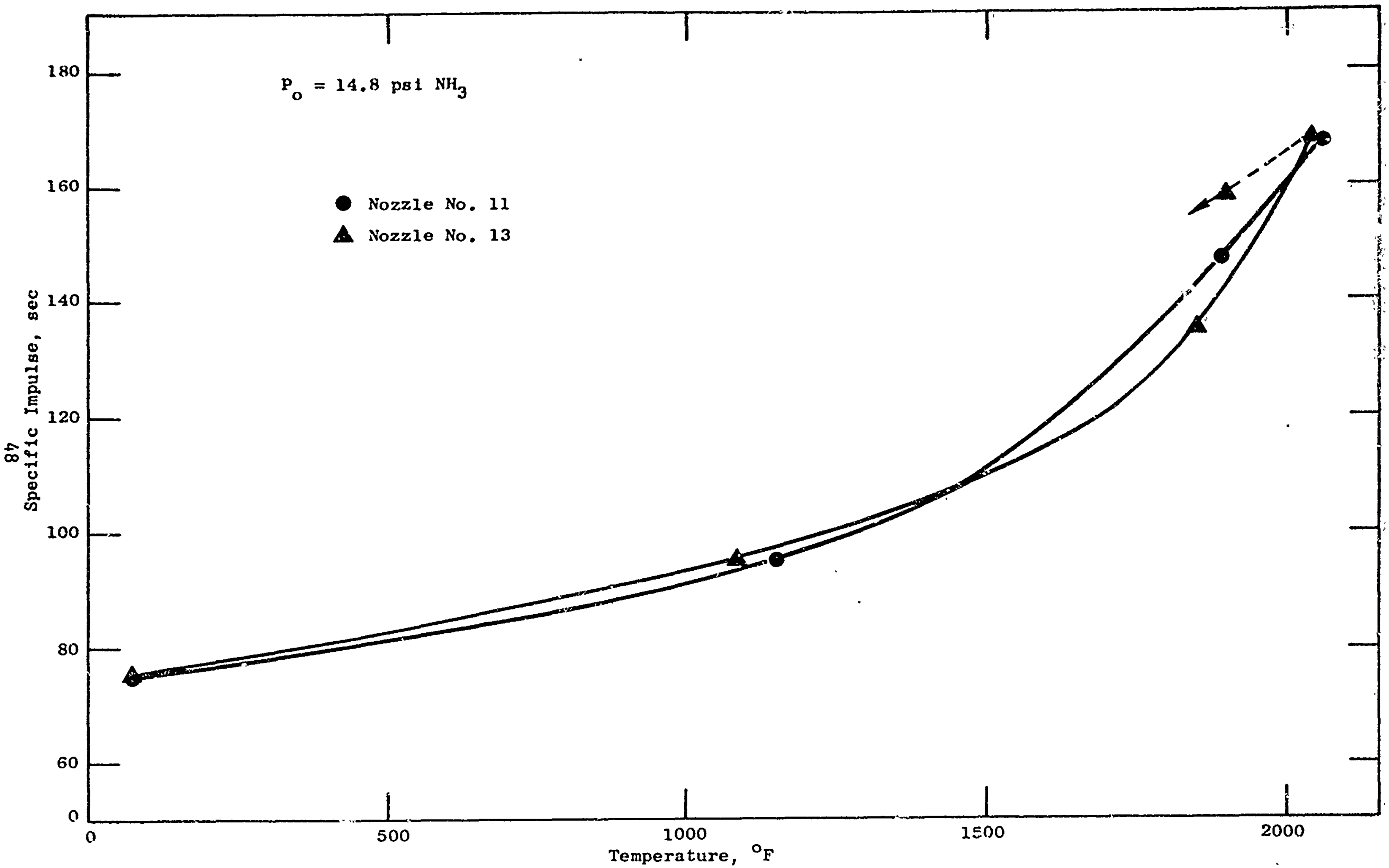


Figure 33. Specific Impulse vs Temperature Characteristics, Nozzles Number 11 & 13, Preliminary Evaluation Thrusters.

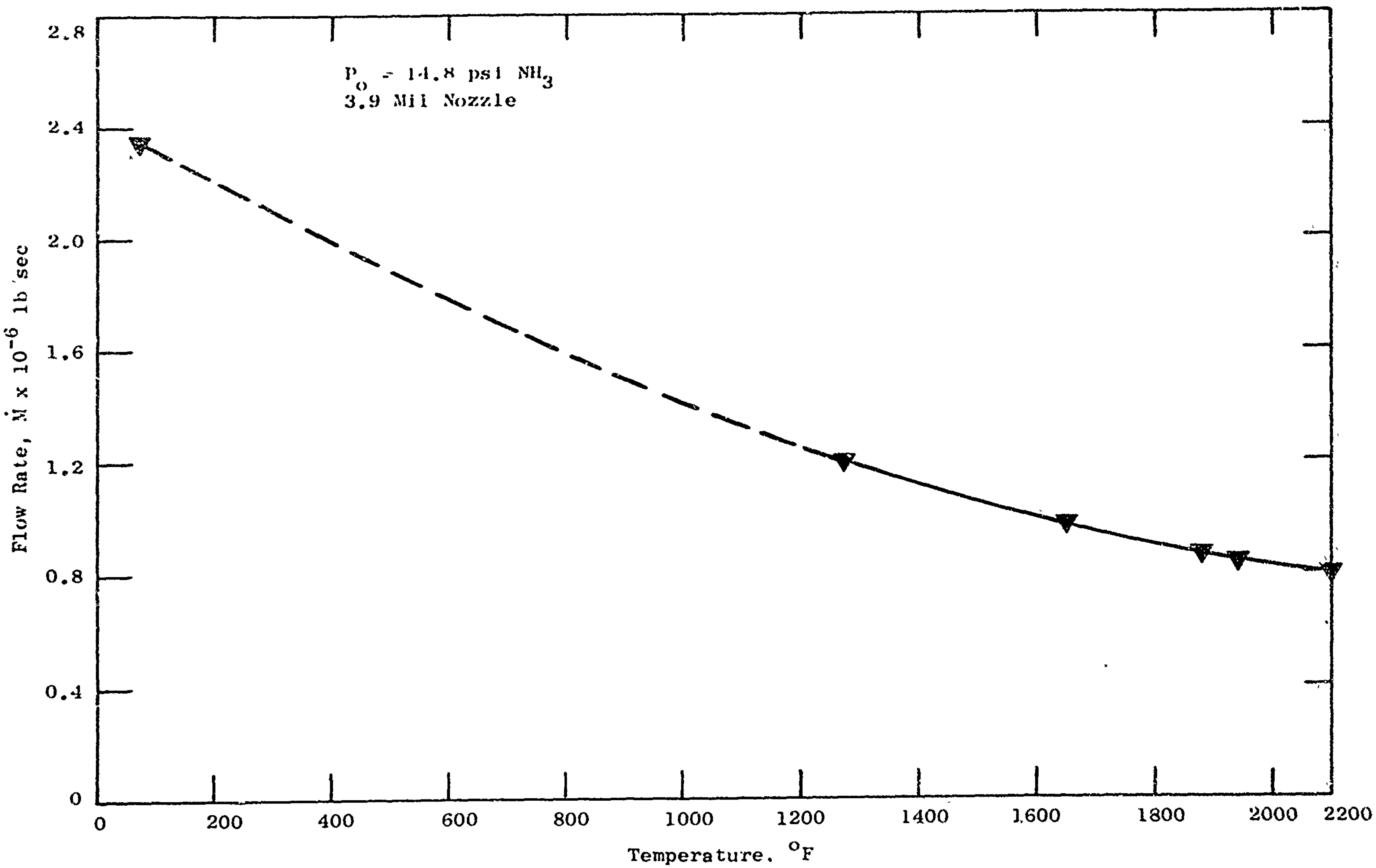


Figure 34. Mass Flow Rate vs Temperature Characteristic; Nozzle No. 14 with Catalyst; Preliminary Evaluation Thrusters.

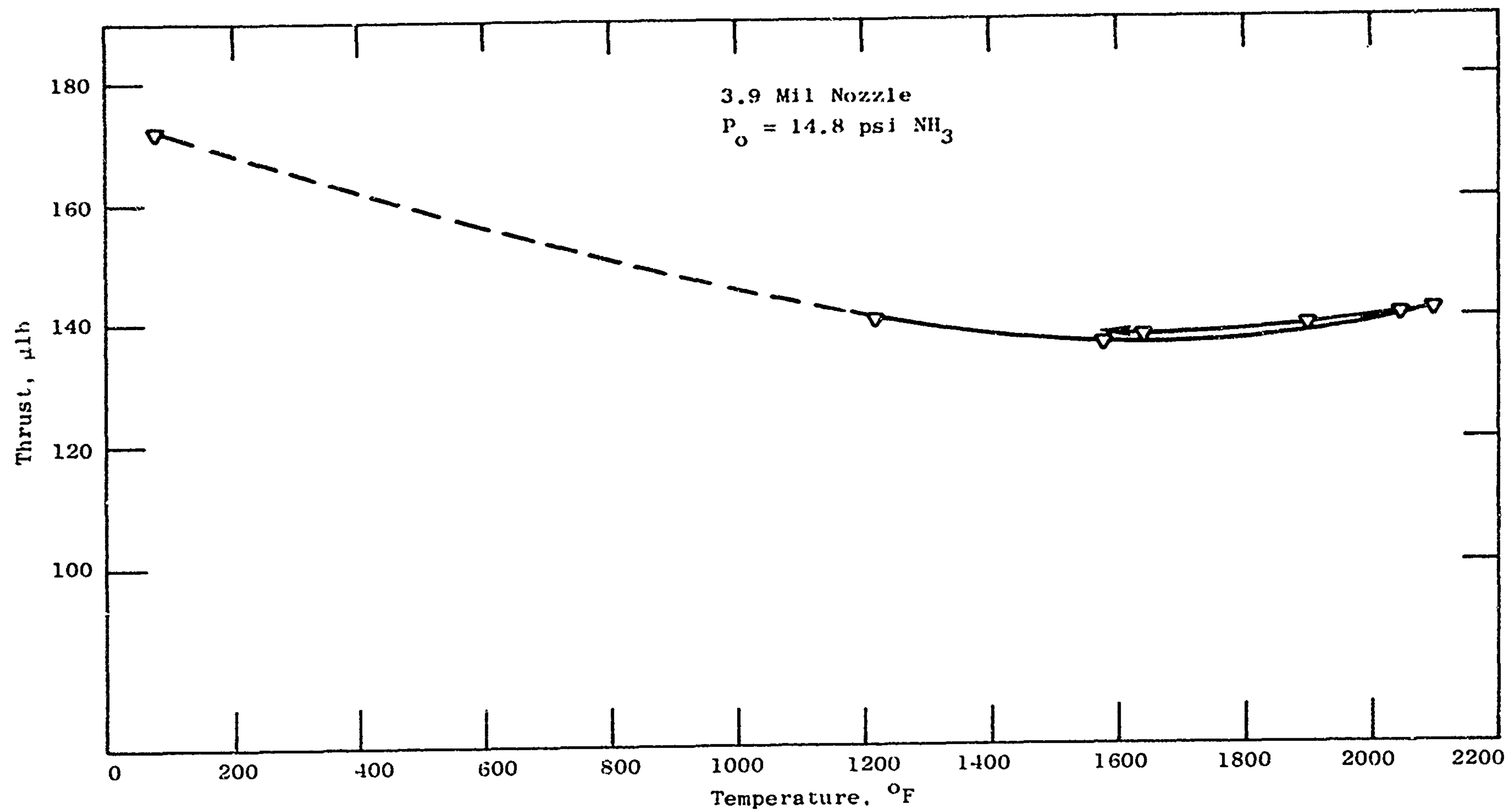


Figure 35. Thrust Vs. Temperature Characteristic; Nozzle No. 14 with Catalyst; Preliminary Evaluation Thrusters.

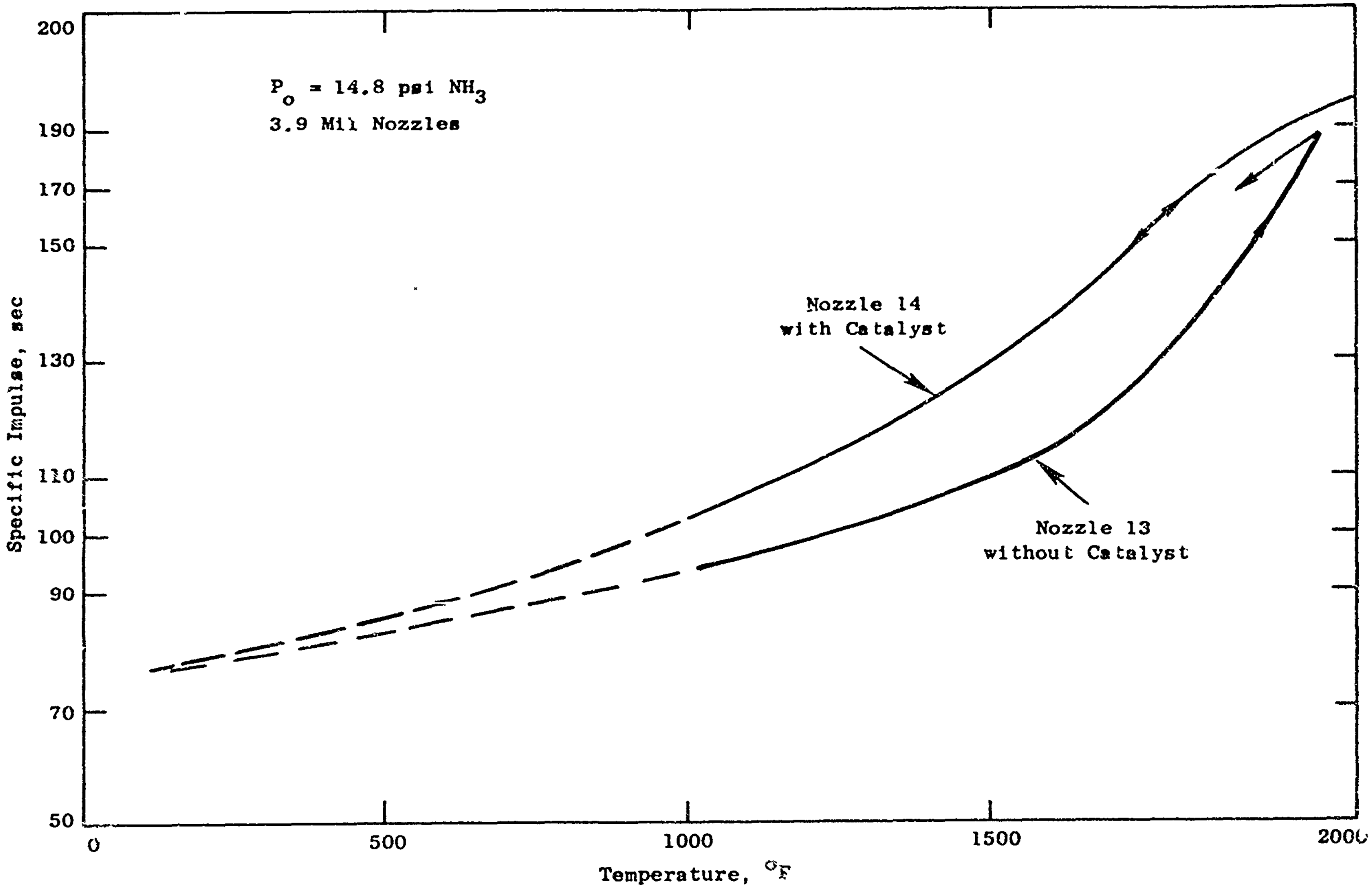


Figure 36. Specific Impulse vs Temperature Characteristic; Nozzles No. 13 (without Catalyst) and No. 14 (with Catalyst); Preliminary Evaluation Thrusters.

micropounds. Thrust data from these tests, along with data previously obtained on nozzle Number 14, were sufficient for determining the size of the nozzle that would produce 100 micropounds at 1500°F and 14.7 psia chamber pressure. This is shown in Figure 37 when the squares of the nozzle throat diameters are plotted vs. the thrust obtained at 1500°F with 14.7 psia NH_3. Since the thrust is directly proportional to the area or diameter squared, a straight line can be drawn through the three experimentally determined points and extrapolated to the 100 micropound level. The intersection at D^2 = 10.8 indicates that nozzle throat diameter should be 3.3 mils.

3. SUMMARY OF RESULTS

Thus the dimensions of the orifices and nozzles were determined to match the performance requirements of the flight thrusters. The results are summarized in Table III. Finalized dimensions are as follows: the 3 micropound orifice should have a diameter of 0.00177 ± 0.00025 inches; the 6 micropound orifice should have a diameter of 0.00177 ± 0.000025 inches; the throat diameter of the 100 micropound nozzle should be 0.0035 ± 0.000025 inches and the exit diameter should be 0.00711 ± 0.000025 inches. It was intended that Figures 21 and 22 would be revised to show these dimensions.

4. DISCUSSION OF ACCURACY OF RESULTS

Although at one time some uncertainity existed concerning the accuracy of mass flow rate measurements as well as the thrust values of the orifice work presented in curve form on Figures 29 and 30, subsequent experience with the use of the Microthrust Measuring System and improved knowledge of mass flow rate measurements indicated that the orifice performance data presented in Table III are realistic and the orifice dimensions described above in "Summary of Results" are acceptable.

The dimensions of the 100 micropound nozzle was determined by extrapolating performance data from other nozzles operating at the 195, 180, and 135 micropound level. Because such an extrapolation (curve "A", Figure 38) showed a substantial thrust value at nozzle diameter zero, some doubt existed as to its accuracy at the 100 micropound point.

Referring again to Figure 38 and assuming that the Thrust vs. Diameter^2 is a linear relationship through zero, at least two of the data points must be somewhat in error. A straight line (curve "B") from the 195 micropound data point to zero would represent the extreme error and it could probabably be assumed that the true value lies somewhere between ("A") and ("B"). Therefore, the 3.3 mil diameter (D^2 = 10.89) nozzle could be expected to produce a thrust of between 93 and 102 micropounds. However, one would expect, due to increasing effect of the boundary layer, that the line would decrease in slope as the diameter approaches zero as indicated by some previously-obtained low thrust date (Figure 39). The extent of this effect is uncertain, but is probably small. Taking all the above assumptions into consideration, it appears that the nozzle size could readily by selected for 100 micropound ± 10%. Such has been done, and it is expected that the thrust obtained with the nozzles as

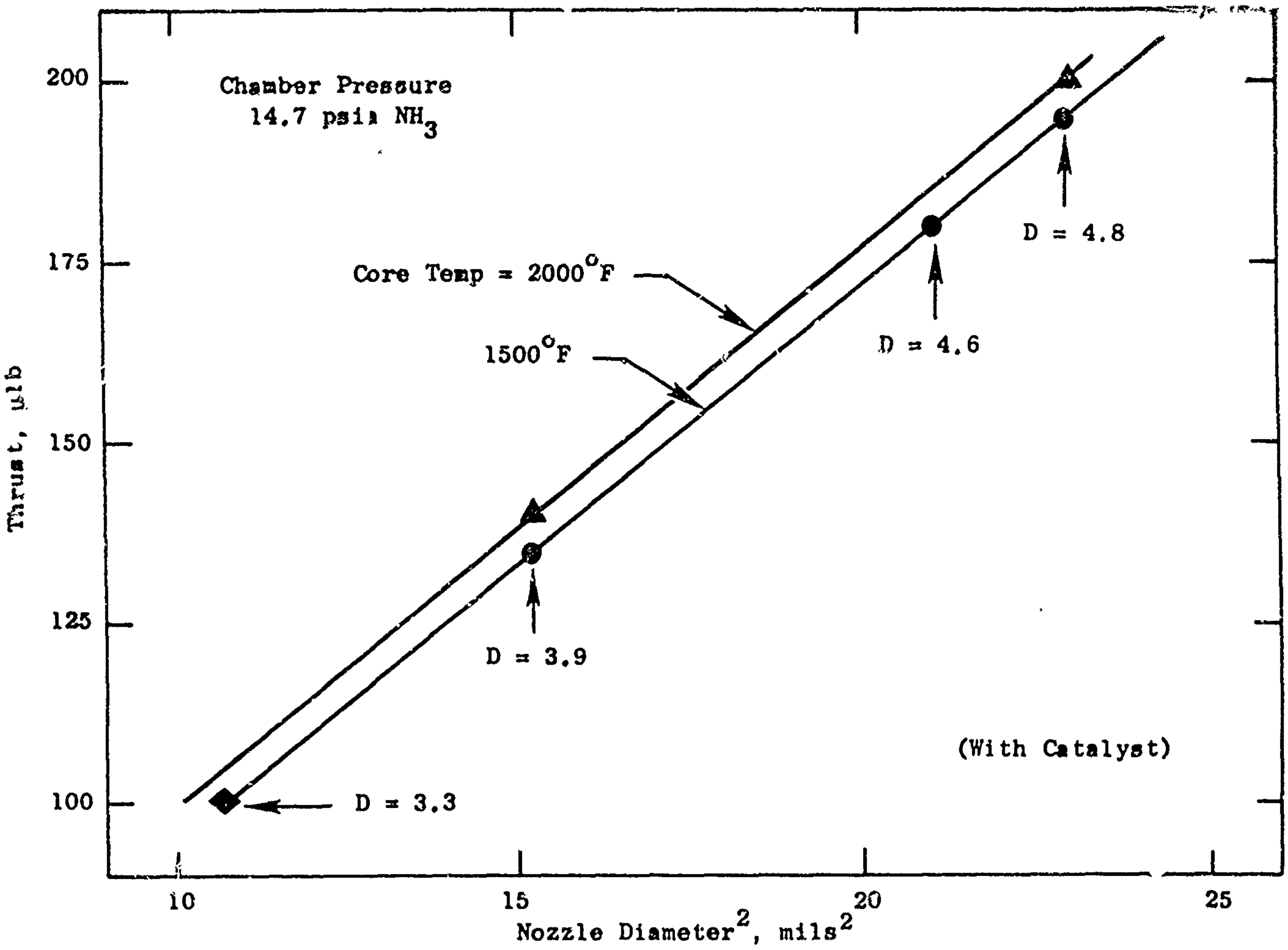


Figure 37. Nozzle Size Determination for Micropounds Thrust at 1500°F and 2000°F.

TABLE III

DATA SUMMARY FOR ORIFICES AND NOZZLE

Throat Machined Diameter (Mils)	1500°F NH_3			70°F NH_3		
	Thrust (Micro-pounds)	Mass Flow Rate (lb/sec)	Specific Impulse (sec)	Thrust (Micro-pounds)	Mass Flow Rate (lb/sec)	Specific Impulse (sec)
1.77	3.0	3.0×10^{-8}	100	4.8	8.0×10^{-8}	60
2.38	6.0	6.0×10^{-8}	100	9.4	16.0×10^{-8}	60
3.30	100	0.87×10^{-6}*	115*	125	2.1×10^{-6}	60

* With Catalyst (Estimated)

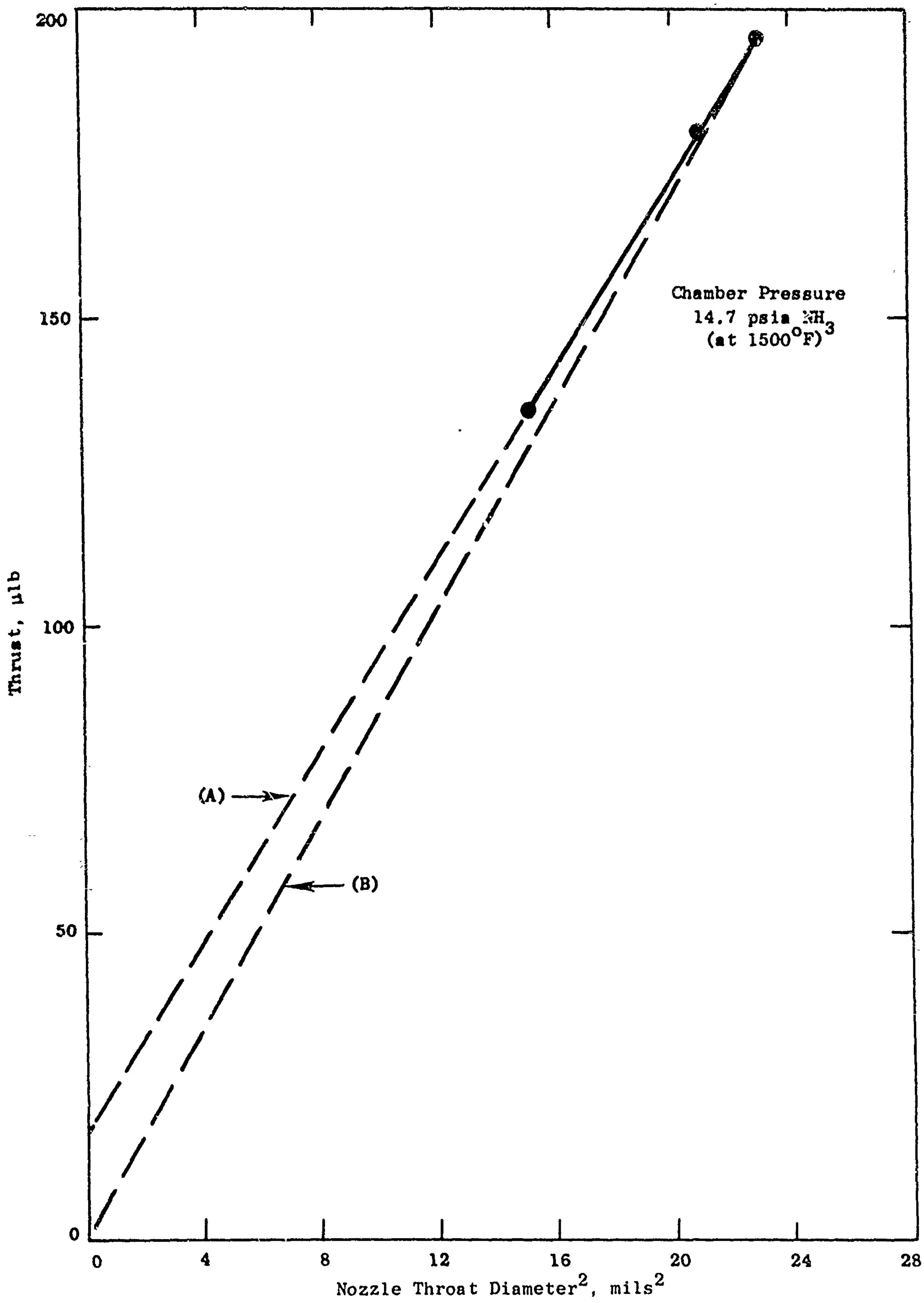


Figure 38. Nozzle Throat Dimensions, 100 to 200 Micropounds Thrust.

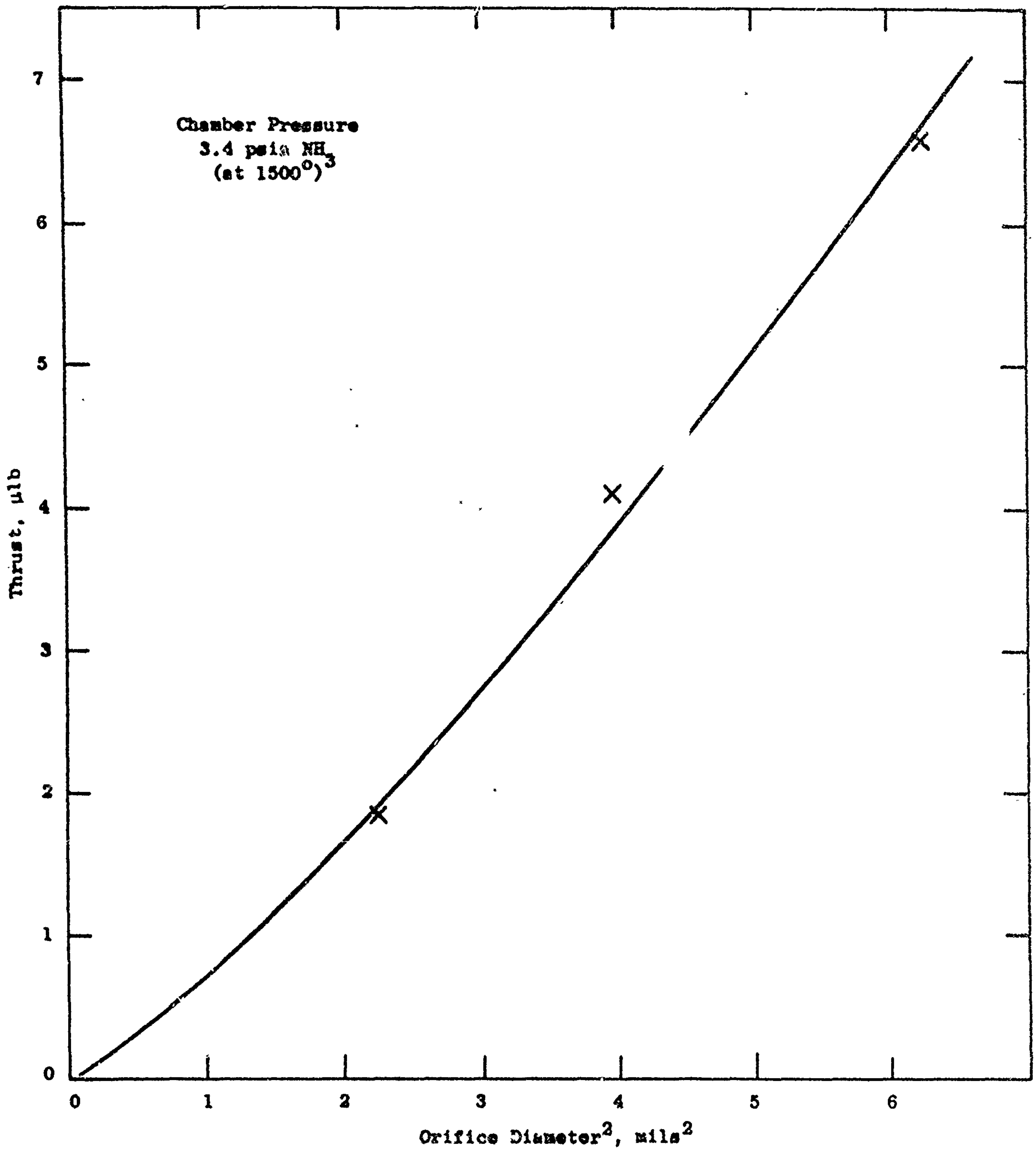


Figure 39. Orifice Dimensions at Low Thrust Values.

dimensioned will be well within the plus 20%, minus 25% (120 to 75 micropounds) tolerances on the 100 micropound value. Therefore, the dimension of the nozzle appearing above in "Summary of Results" are acceptable.

5. DISCUSSION OF TESTING PROCEDURES

Throughout the testing of the Preliminary Evaluation Thrusters, use was made of the Microthrust Measuring System. It was during this portion of the program that experience with the System resulted in significant improvements to the System. Periodically, it was considered necessary to review and check the methods for determining flow rate of gas through the orifices and nozzles and the resultant thrust, particularly if values of specific impulse were of values other than anticipated. Since the thrust measurements were taken with a calibrated force transducer mounted directly behind the thruster, there is little doubt of their accuracy. Therefore, the object of the test described below, was to verify the accuracy of the mass flow rate measurements.

- Mass Flow Rate Measurements

The method by which mass flow rate was determined with the microthrust system was to release the pressure of a known volume of ammonia gas through the nozzle under test at a constant flow rate by means of a pressure regulator upstream of the nozzle (Figure 15). By observing the change in pressure (ΔP), the time elapsed (Δt) and the known volume temperature (T), the mass flow rate ($\dot{M}$) was calculated for ammonia by the relationship:

$$\dot{M} = V \frac{\Delta P}{\Delta t} \left(\frac{2.41 \times 10^{-5}}{14.7}\right)^{*} \frac{560}{T(°R)} \quad \frac{lb}{sec}$$

For the volume used of 48.25 in^3 at room temperature this reduced to:

$$\dot{M} = \frac{\Delta P}{\Delta t} (8.29 \times 10^{-5}) \quad \frac{lb}{sec}$$

As a quick check of this method, the thruster was replaced with a precalibrated needle valve. This valve had been previously calibrated for effective orifice diameter with ammonia gas at 22.2 psia inlet pressure. Figure 40 presents this calibration curve. Assuming the effective area to be the same at 15.2 psia, the valve was opened 0.7 turns to the equivalent of a 4.8 mil effective diameter orifice. This was done for a ΔP of 10 psi as was done with the thrusters. For the 4.8 mil nozzle the time required was 198 seconds. For the valve the time required was 195 seconds. The expected mass flow rate through a 4.8 mil effective diameter orifice at 15.2 psi would be:

* Density of NH_3 at room temperature and atmospheric pressure.

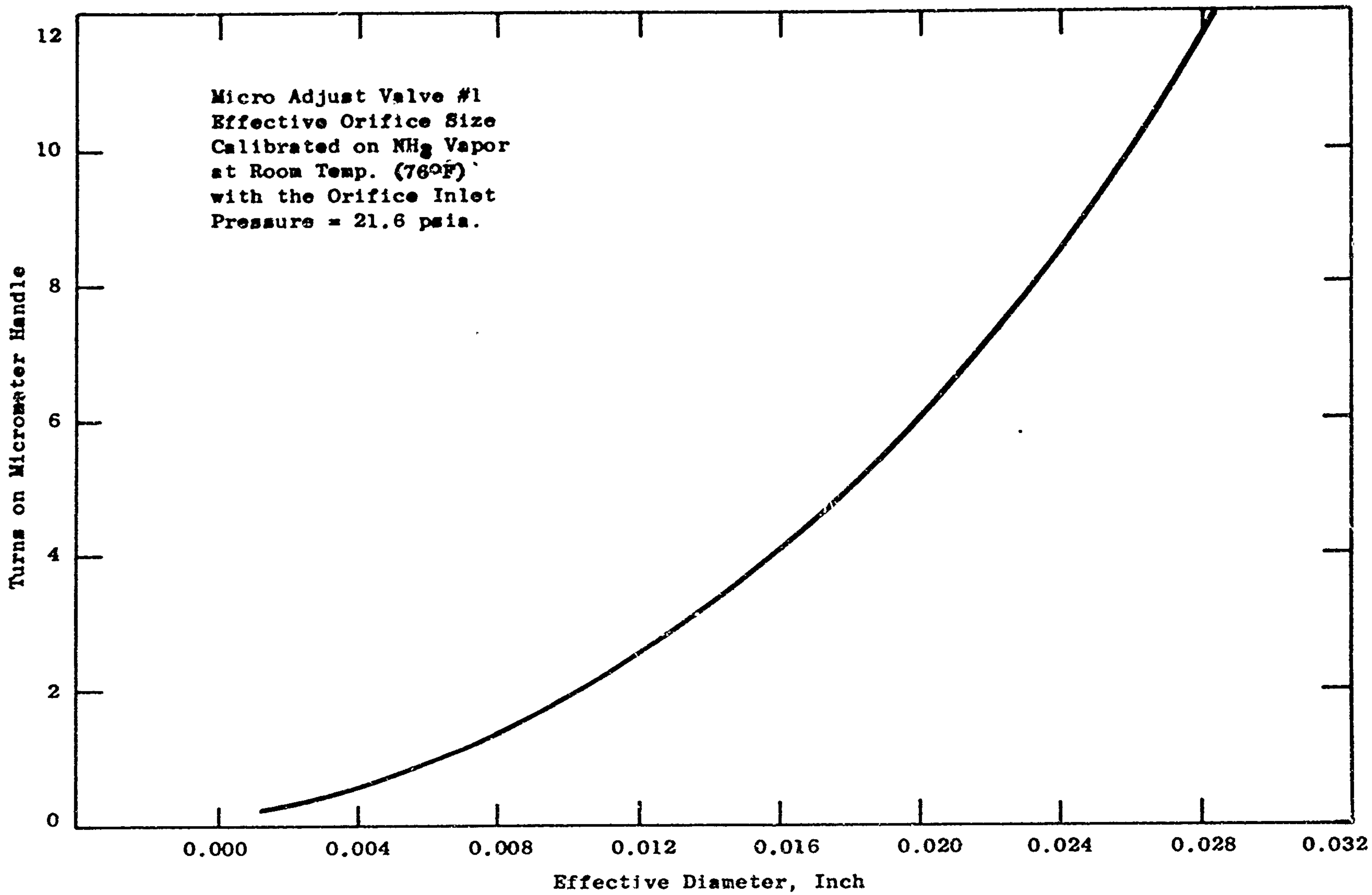


Figure 40. Needle Valve Calibration Curve, Preliminary Evaluation Thruster Mass Flow Rate Measurements.

$$\dot{M} = \frac{0.248\ \text{lb}}{\text{sec. in}^2} \times \frac{P_o}{14.7} \times \frac{560}{T_o} \times A$$

$P_o = 15.2$ psia

$T_o = 535°R$

$A = 18.09 \times 10^{-6}$ in^2 area

$$\dot{M} = 4.778 \times 10^{-6}\ \text{lb/sec}$$

The mass flow rate as determined by the P/t method is:

$$\dot{M} = \frac{\Delta P}{\Delta t}\ (8.29 \times 10^{-5})\ \text{lb/sec}$$

P = 10 psi
t = 195 sec

$$\dot{M} = 4.25 \times 10^{-6}$$

which is 11% less than expected for a 4.8 mil diameter orifice.

Since this was a discrepancy in the wrong direction it was assumed that the needle valve calibration was incorrect or that the calibration did not hold at the lower pressure. In any event, since this rapid check failed to produce satisfactory results it became apparent that a truly positive and accurate method must be used from which the results could be compared with the original data. To attain this end, a completely independent means of mass flow measurement was devised whereby the flow could be directly measured with a rotometer type flowmeter. A variable area flowmeter was chosen which had a 1/16" diameter stainless steel ball float. The meter was first checked with the set-up shown in Figure 41 to determine where the thruster nozzle flow rates read on the meter scale. With the thruster in vacuum, the flow rate was increased until the pressure gauge on the nozzle inlet read 15.2 psia and the meter reading recorded. The 4.8, 4.6, and 4.4 mil nozzles were checked and the calibrated needle valve was checked over the entire meter scale (0-17 units). The data from these tests are presented in Table IV.

The next step was to accurately calibrate the meter with ammonia gas. To accomplish this, the arrangement shown in Figures 42 and 43 was built up whereby the total volume of ammonia passing through the meter could be collected and weighed.

The 4" diameter stainless steel sphere was first evacuated and weighed to within 0.0001 gram and returned to the setup. With V_1 still closed and V_2 opened to vacuum V_3 and V_4 were adjusted for a particular meter reading with 15.2 psia on the meter outlet. Then, quickly, V_2 was closed, V_1 opened, and the sphere immersed in liquid nitrogen to keep the pressure low. The meter outlet pressure was maintained at 15.2 psia at a particular flow rate for approximately 10 minutes. The sphere was then removed, brought to room temperature, and weighed to determine

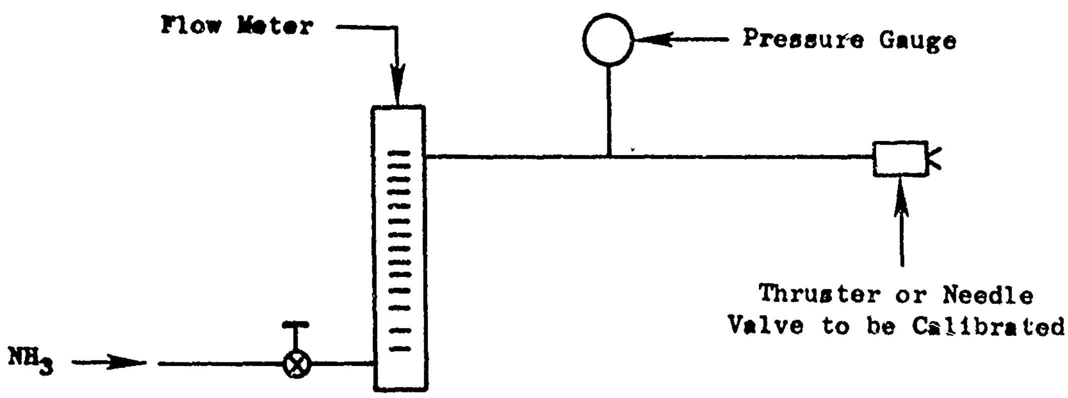


Figure 41. Thruster Nozzle Calibration with Flow Meter, Schematic Diagram.

TABLE IV

NOZZLE AND NEEDLE VALVE FLOW CALIBRATION DATA

Test Piece	Flow Meter Reading
4.8 mil dia. nozzle	5.38
4.6 mil dia. nozzle	4.90
4.4 mil dia. nozzle	4.45
Needle Valve, No. turns open	
0.7	5.40
0.3	1.20
0.4	2.45
0.5	3.55
0.6	4.25
0.7	5.40
0.8	7.15
0.9	8.40
1.0	10.15
1.1	11.50
1.2	13.10
1.4	16.30
0.7	5.30

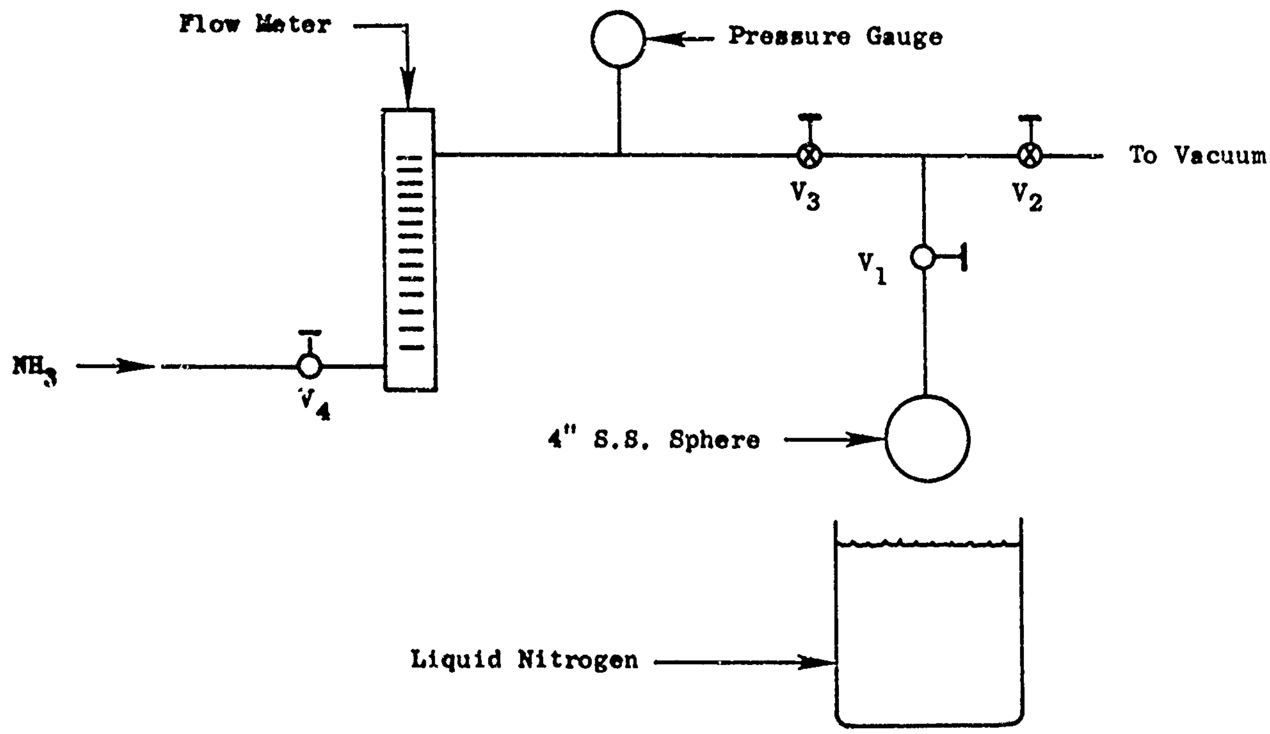


Figure 42. Flowmeter Calibration Arrangement, Schematic Diagram.

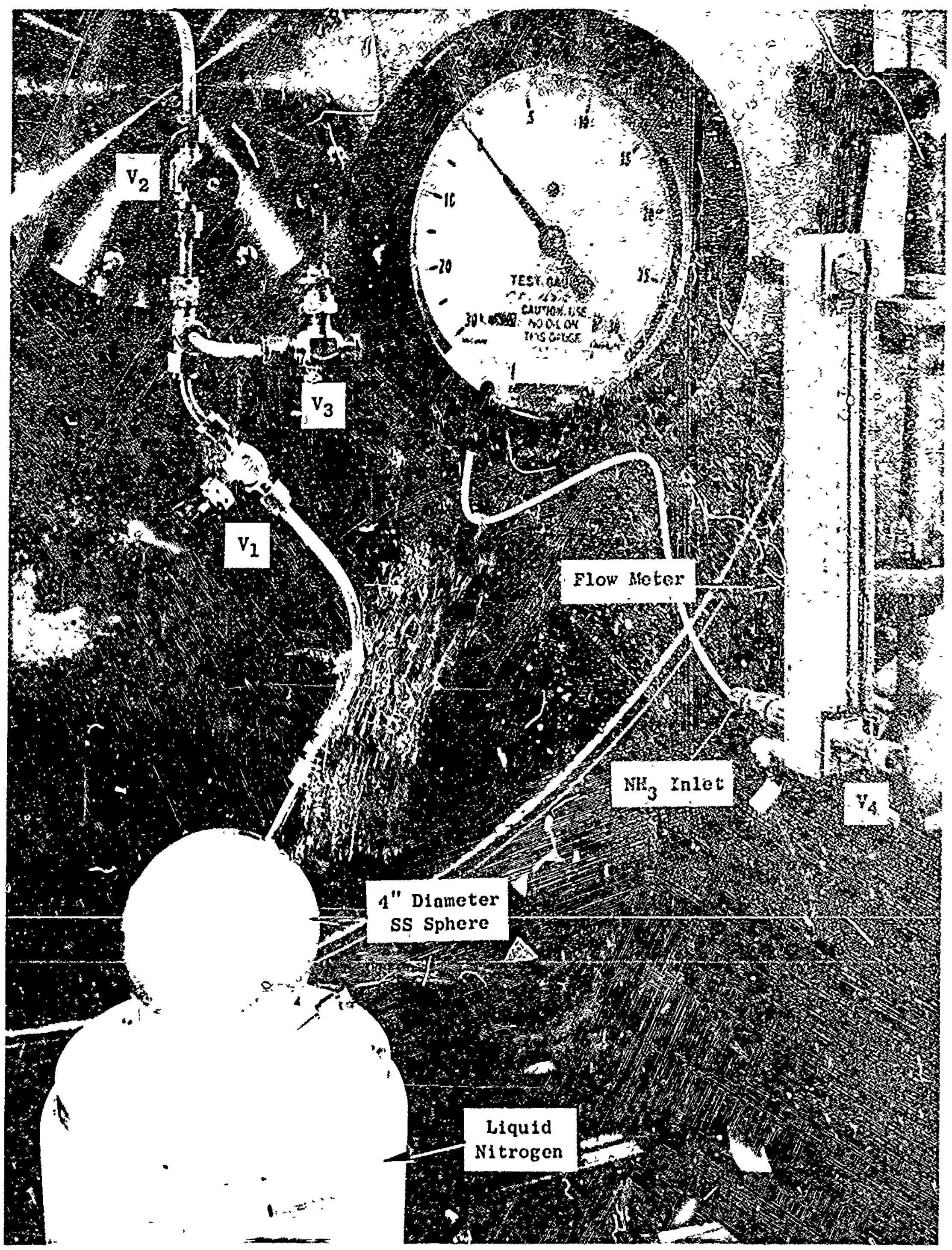


Figure 43. Flow Meter Calibration Arrangement.

the total ammonia mass flow and, of course, with the flow time known, the mass flow rate could be readily calculated. A series of seven runs was made with this setup at various meter readings to cover the range of the nozzle flow rates. One point, 5.5, was run a second time to check for repeatability and data was within 0.7% of the previous run. The data from this series of tests is presented in Table V and the calibration curve of Scale Reading vs. Mass Flow Rate is shown in Figure 44. Using this curve and the data from Table IV, the mass flow rate of the nozzles could be determined.

The 4.8 mil diameter nozzle for example, produced a flow meter reading of 5.38 which corresponds to 4.35 X 10^{-6} lb/sec. The flow rate observed by the $\frac{\Delta P}{\Delta t}$ method as mentioned earlier in this discussion was 4.25 X 10^{-6} lb/sec, or 2.5% lower. This indicated that the $\frac{\Delta P}{\Delta t}$ method of mass flow measurement did not introduce an error anywhere near the 20% that was suspected. The conclusion was that the mass flow rates used to calculate I_{sp} for the 4.6 and 4.8 mil diameter nozzles as reported were correct to within a few percent.

It is interesting to note here that, after the above tests were completed, an even simpler and more direct method of mass flow measurement was conceived. The 4" sphere would be charged with liquid ammonia and weighed. It would then be connected to the thruster through the 15.2 psia regulator in vacuum and allowed to run for a given time and re-weighed. This method could be used to accurately size the flight nozzles and orifices when received, and then the flowmeter could be compared at room temperature and used for further measurements at elevated temperatures. These measurements would be available for comparison with those made by the $\frac{\Delta P}{\Delta t}$ method while making thrust measurements. This would make it possible to detect quickly any malfunction of the thruster, such as a partially plugged nozzle or even a leak in the system, should one develop.

- Typical Test Procedure, 100 Micropound Nozzle

Had it been decided to procure a 100 micropound thrust nozzle of the dimensions described under "Summary of Results" and in a configuration to be used with the Preliminary Evaluation Thrusters, performance tests would have been conducted as described below:

Mass flow rate measurements would first be made. An arrangement as shown in Figure 45 would be utilized. With liquid ammonia in the sphere and V_1 closed, V_2 and V_3 are opened to evacuate the regulator (R), pressure transducer (P), and thruster (T). V_3 is then closed. V_1 is opened for a certain amount of time (t) allowing gas to flow through the nozzle and then closed. By accurately weighing the sphere before and after the test the total mass flow from the sphere can be determined, and, of course, the mass flow rate by dividing by (t). The amount of ammonia lost in the volume between the nozzle and V_1 can be determined by first running a test with t as short as possible. Any error remaining with this method can be minimized by making t relatively long in subsequent

TABLE V

FLOW METER CALIBRATION DATA

Flow Meter Reading	t (sec)	Weight, final Grams	W $\times 10^{-3}$ lb	$\dot{M}$ $\times 10^{-6}$ lb/sec
5.50	639	337.6385	2.860	4.476
2.50	1249	337.2040	1.904	1.524
3.95	550	337.0626	1.593	2.896
7.15	464	337.6240	2.828	6.095
11.95	430	338.3735	4.477	10.412
5.50	899	338.1552	3.997	4.446
9.15	488	338.1058	3.888	7.967

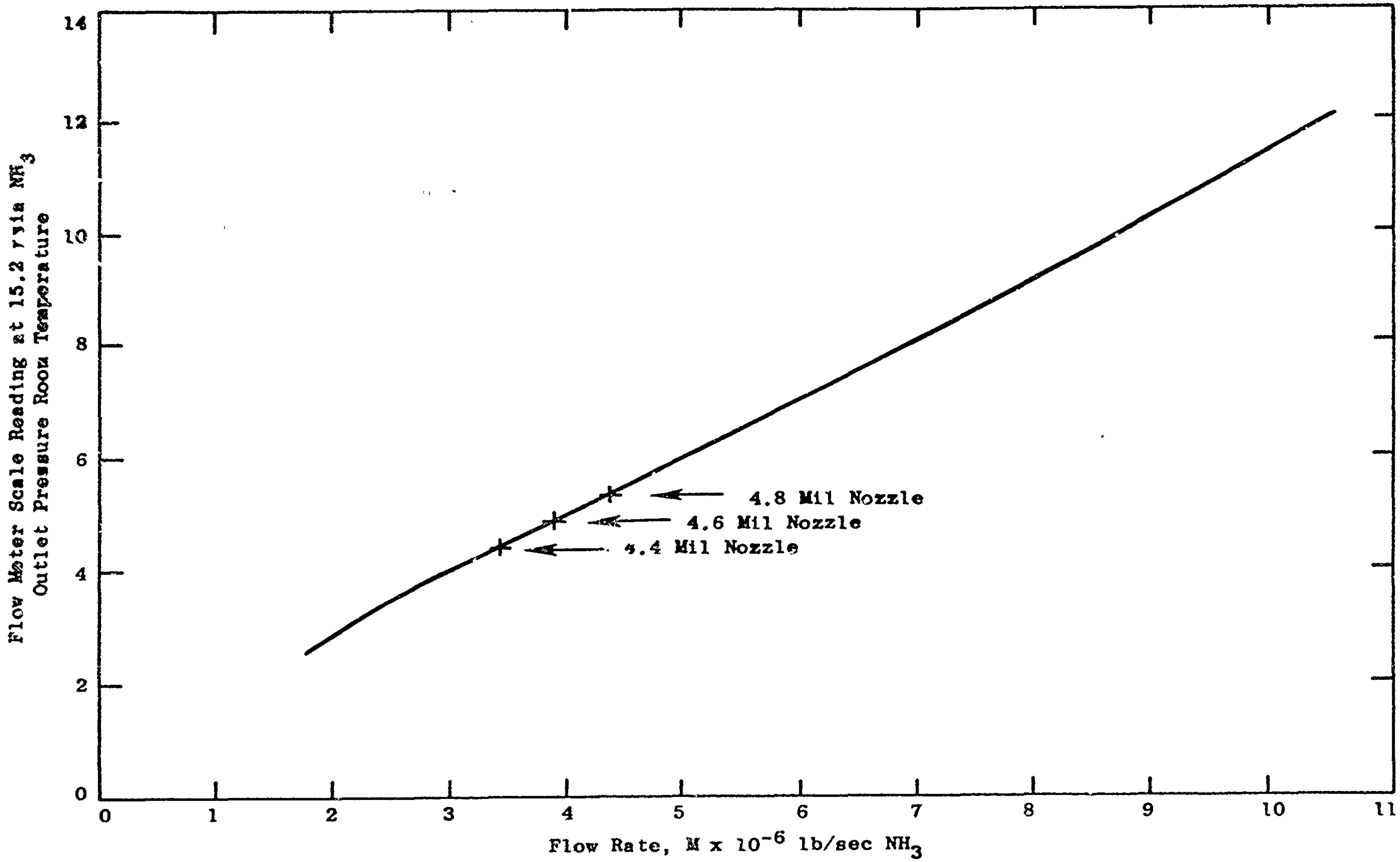


Figure 44. Calibration Curve, Flow Meter.

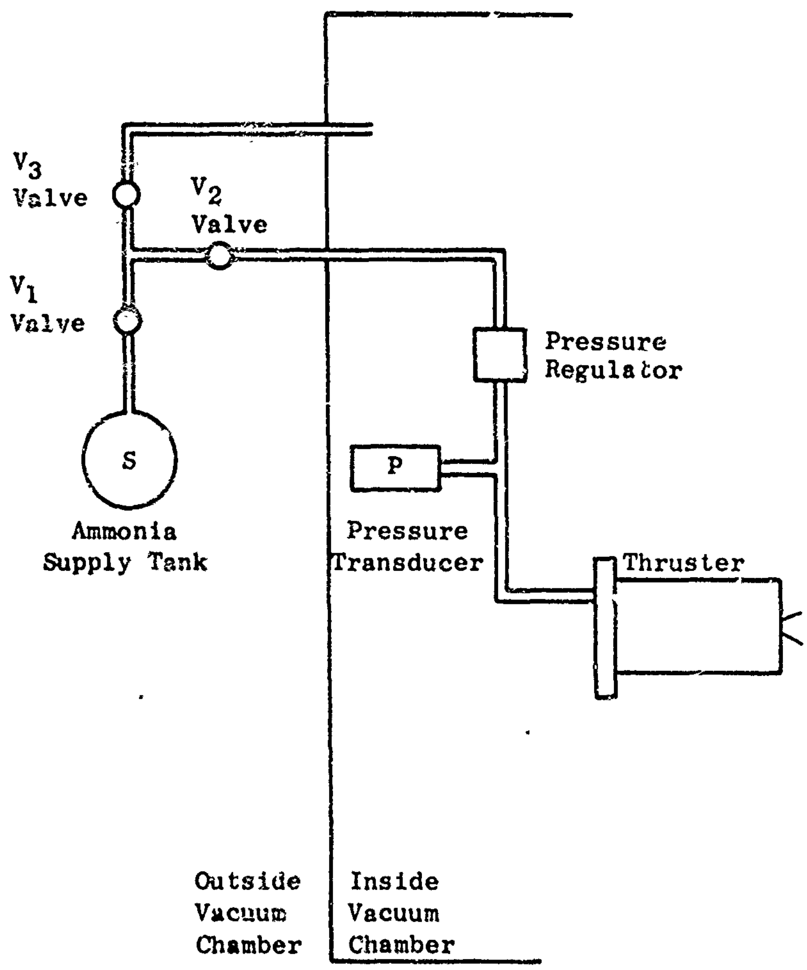


Figure 45. Mass Flow Rate Measurements, Schematic Diagram.

tests.

A new method of calibrating the thrust measuring system as a whole has been devised but not as yet tested. By suspending a weight (w) (see Figure 46) by a thread of length (L) and moving the thruster against it for a distance (x) with the force transducer in a micrometer drive mount, the actual force on the thruster can be calculated and compared to the transducer output. The restoring force on the weight would be for small values of x, $\frac{Wx}{L}$. All three of these quantities can be readily and accurately determined.

- Performance Tests on the 100 Micropound Nozzle, Detailed Procedure

A preliminary evaluation nozzle procured would have a 3.3 mil throat diameter. Upon receipt, the nozzle would be sized (mass flow rate checked) at room temperature using the new procedure described above. It would then be welded with catalyst into a preliminary evaluation thruster body presently on hand and re-sized. Assuming no change, the nozzle would be mass flow rate checked at elevated temperatures of at least 1500°F and 2000°F and then back to 1500°F. Assuming these data to be acceptable, the thruster would then be mounted in the Microthrust Measuring System, the system calibrated for thrust, and the thruster then evaluated (thrust and mass flow rate) at selected thruster temperatures. Should the values of $\dot{M}$ vs. temperature differ from the initial ones, the initial readings would be repeated and the results, if consistent with the first, would be used to modify the pressure drop method normally used with the Microthrust Measuring System. Once this nozzle was thoroughly evaluated, it should then be possible to select a nozzle size within a few micropounds of 100 and very accurately predict its performance.

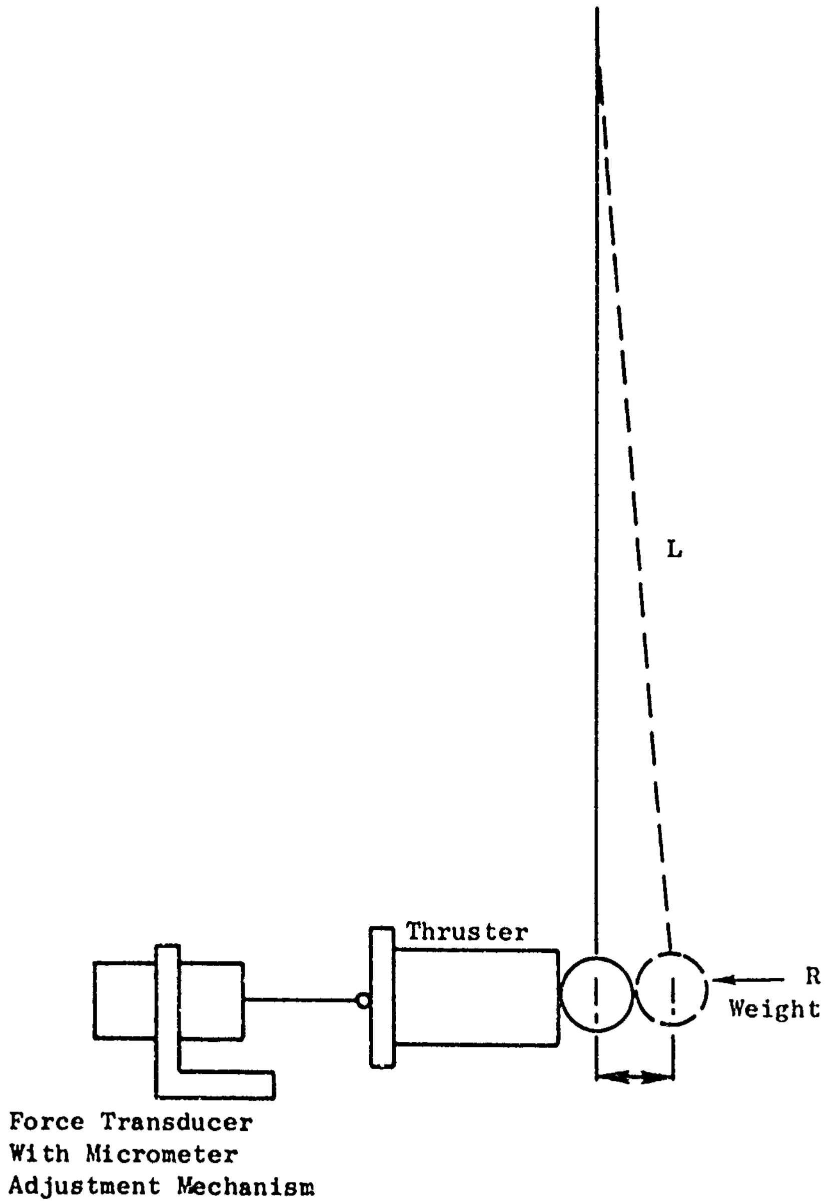


Figure 46. Equipment Arrangement Calibration of Thrust Rig.

SECTION IV

FLIGHT THRUSTERS

1. PERFORMANCE REQUIREMENTS

For its station keeping application each thruster was required to produce 3, 6, or 9 micropounds of thrust with a minimum impulse of 140 seconds. For station changing, the thrust level required was 200 micropounds minimum with a specific impulse of not less than 200 seconds. The allowable electrical power for the thruster was 20 watts. To achieve this performance, it was planned to operate the thruster at 2000°F and the original design provided for this. However, later, the specific impulse for the 3, 6, or 9 micropounds was reduced to 60 seconds, the 200 micropound thrust level was reduced to 100 micropounds (with specific impulse at 100 seconds minimum), and the power imput reduced to 15 watts maximum.

2. CONCEPTUAL DESIGN

The conceptual design of the thruster required the consideration of a number of interdependent factors and spanned a considerable period of time. It consisted of generation and assessment of ideas leading to a final design. These included the selection of the heater material, size and configuration of the heater capsule, and materials for the thruster body with allowance for the insertion of three nozzles (3, 6, and 200 micropounds thrust). In addition, consideration had to be given to the compatibility between selected material and the ammonia propellant, to the requirement of the gas flow passage, the nozzle contours, containment of a catalyst, configuration of the radiation shielding, and cold gas operation.

One of the most important aspects considered in the conceptual design sub-task lay in the fact that a common heater could be used in producing the required thrust levels of 3, 6, and 200 (later changed to 100) micropounds. It was expected that this plan would result in a reduction of weight and power requirements and would permit more effective packaging of the thruster in relation to the satellite's center of mass. The sketch shown in Figure 47 illustrates how the three nozzles were to be packaged around a common heater within the thruster body.

A mechanical means for the compensation for the thrust vector effects upon the perturbation of the satellite was also an initial conceptual design consideration. The thrust vectors were to be sensed and measured using the Microthrust Measuring System as described in Section II, above. When the thrust vectors were established, mechanical means of offsetting them were to be devised so that they could be incorporated into the thruster mount on the satellite.

3. GENERAL DESIGN CONSIDERATIONS

Definition of the machining and fabrication methods were planned as well as the configuration of the plates in which were to be located the

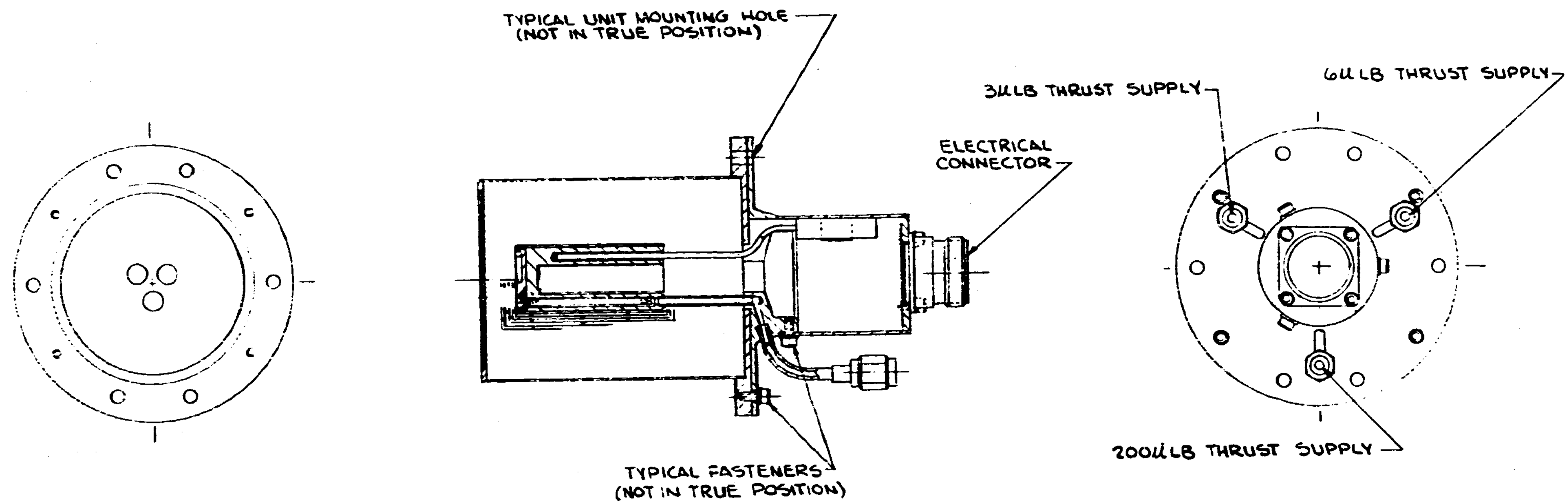


Figure 47. Sketch, Conceptual Design of Flight Thruster.

3 and the 6 micropound thrust orifices and the 200 micropound nozzle. A foil was selected for the heat shielding package. It was planned that the thrusters were to be made of a nickel alloy throughout. All joints were to be welded with the exception of the propellant supply tubes which were to be brazed to the rear mounting flange.

The thruster design required the attachment of small, thin-wall propellant tubes to relatively massive thruster supporting structures. Up until the present time, such attachment had been achieved with tungsten inert gas welding and electron beam welding. Such processes have not always resulted in leak tight assemblies, particularly where the joint area is inaccessible or necessarily miniature in size. Therefore, only such joints in the thruster that are readily accessible were designated as welded joints. A second method of joining parts (wherein the joints are more difficult to reach with these conventional welding methods) is furnace brazing. Such a method of joining has been designed into the fabrication of certain other portions of the flight thruster. To prove the feasibility of such design techniques, a brazing assembly study was initiated, partially supported by the program funding. A description of the study and the results achieved thus far are described in Appendix I entitled "Thruster Brazed Assembly Studies". Ultimate withdrawal of support of this study resulted in the work described in the appendix being an interim report. Results up to that point, however, did indicate the feasibility of the flight thruster brazed joints.

4. FINAL DESIGN

The final design of the completely-assembled thruster is shown in Figures 48 and 49. It consists of an electrically-heated body enclosed with a metallic foil heat shield. The body contains three separate, segmental thrust chambers completely sealed from each other. The orifice plates are made up in three segments and are designed to be welded onto shallow ribs on one end of the thruster body. Before this shallow-rib design could be firmed up, some samples were machined and dummy segments were actually welded into position. Because it was expected that the dimensions of the orifices and the nozzle would be dependent upon the completion of the Preliminary Evaluation Thruster task described above in Section III, the drawing describing the shape and size of the segments was completed very early in the program, the orifice and nozzle dimensions being added at a later date. Within the 100 micropound thrust cavity, the catalyst is contained in a screen capsule. Because the intent of the capsule is to avoid any migration of small particles towards the nozzle opening, the screen is of 25 micron size. This feature required that the fit of the capsule into the cavity be very close, requiring very stringent mechanical tolerances. The capsule also provided for easy handling of the catalyst.

Careful consideration was also given to the design of the heater. Its fit into the thruster body had to be close enough to result in an acceptable temperature gradient, yet generous enough to permit its withdrawal if necessary. The electrical design of such a heater is described in Appendix II. Temperature of the thruster body was originally planned to be sensed by locating two separate thermocouples within the body. It was later decided to imbed a single thermocouple within the heater itself. Electrical connections to the heater and thermocouples

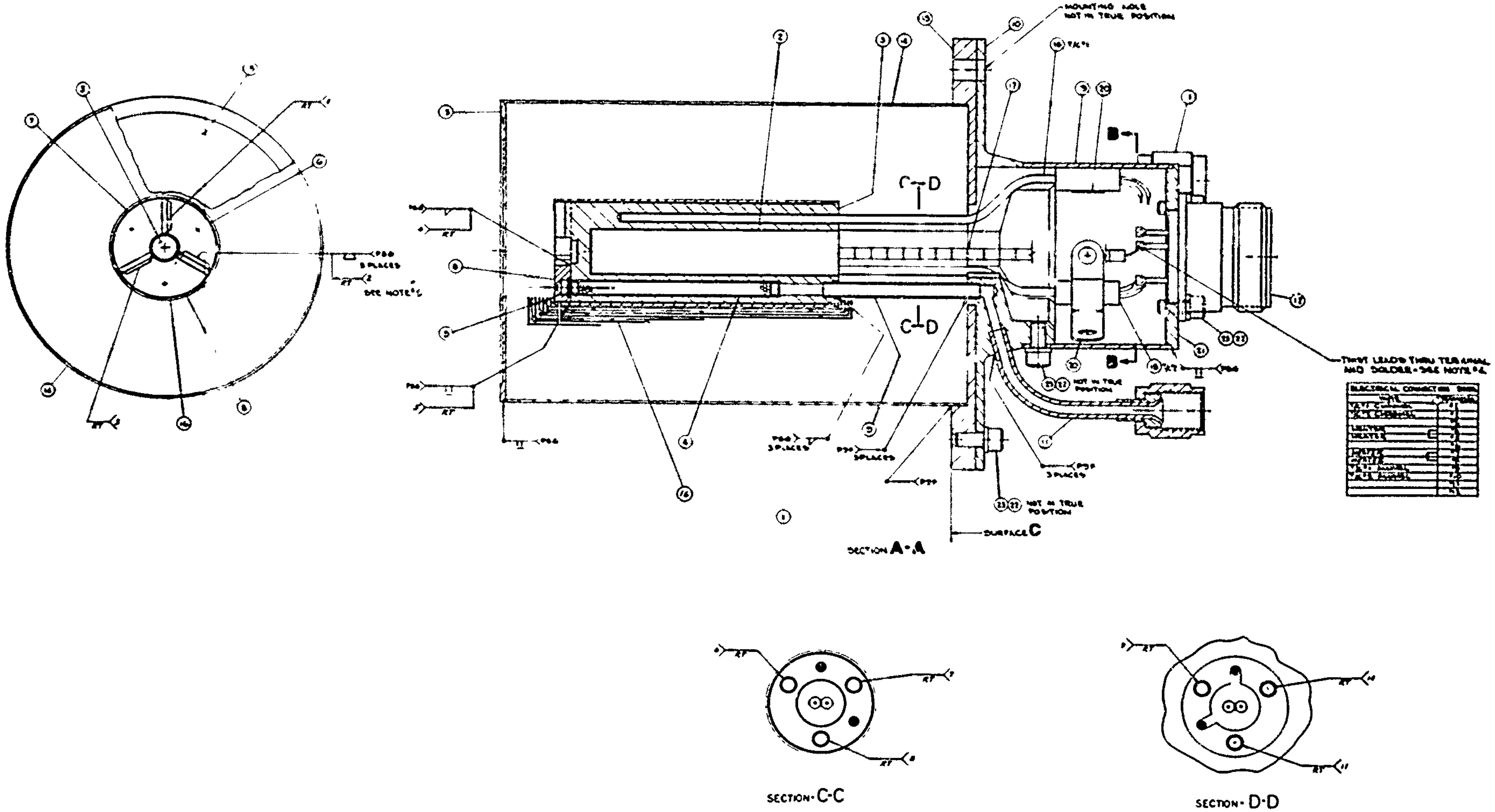
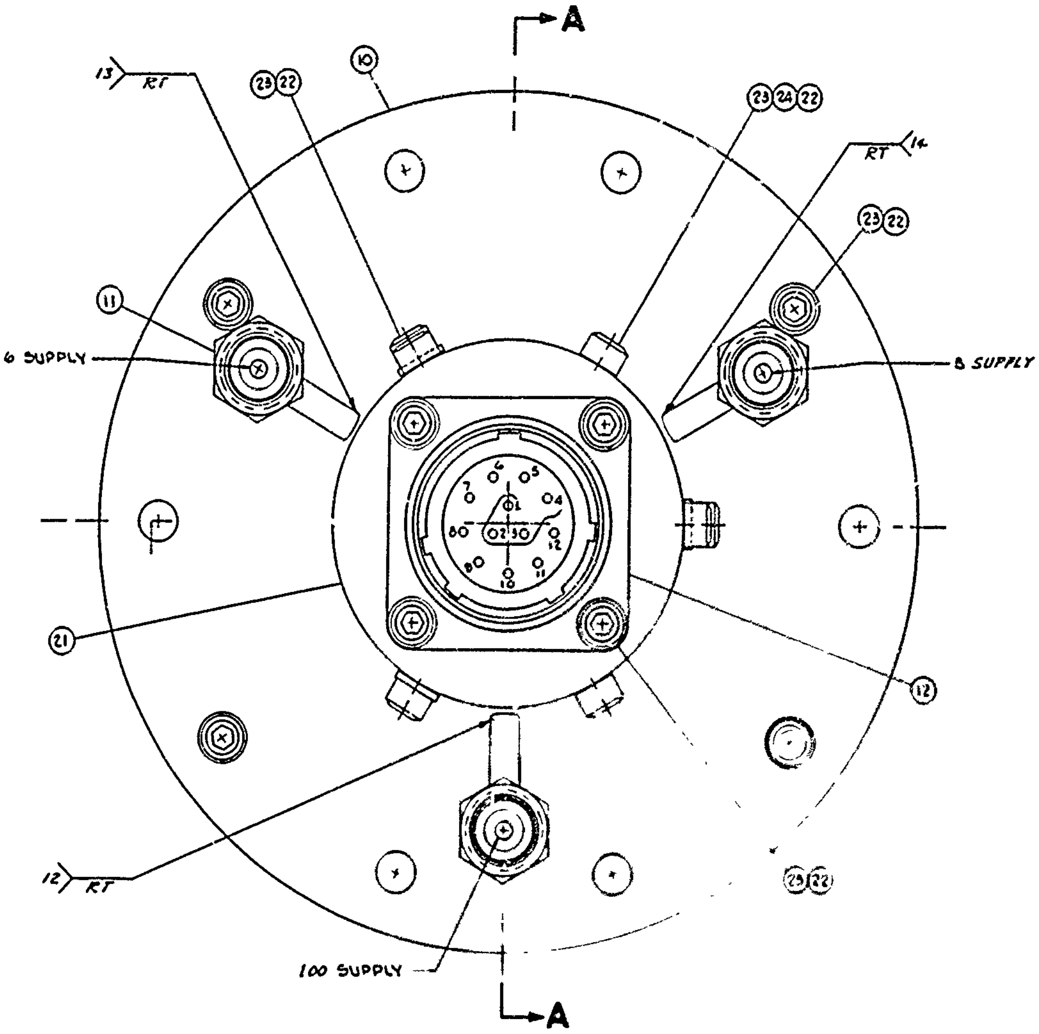
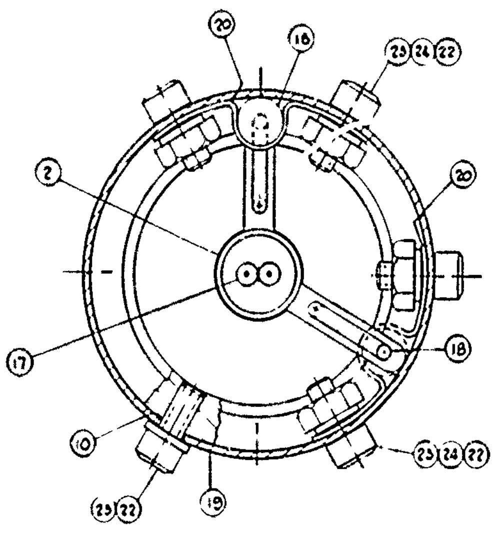


Figure 48. Final Design, Side View, Flight Thruster.





SECTION B-B

Figure 49. Final Design, End View, Flight Thruster.

were provided for by locating an electrical connector receptacle on the thruster assembly. Because the reliable connection of these leads to the connector pins required definition, a short "joining" investigation was conducted, as reported in Appendix III. The results obtained were incorporated into the design.

Periodically, design reviews were held and both the manufacturing and quality control group representatives proffered pertinent recommendations.

With the completion of the design, a model of the thruster was prepared and is shown in Figure 50. This cutaway view shows how ammonia gas may be introduced into each of the separate chambers of the thruster body through separate tube connections, as well as the details of the heat shielding.

5. THRUSTER BUILDUP

It was planned to build up 9 thrusters to meet contractual requirements. Accordingly, requisitions were placed for the procurement of all the necessary components. Those with the largest delivery times were, of course, ordered first. Each incoming shipment of machined parts were subjected to inspection by Quality Control personnel. Because it was expected that the definition of dimensions of the thruster orifices and nozzle would appear towards the end of the scheduled program, machining work on the segmental plates was initiated, but a hold placed on the orifice/nozzle shaping.

An examination of the overall schedule showed that it would be possible to build up one pilot model thruster (except for orifice and nozzle plates) in advance. It was deemed prudent to plan to subject this unit to electrical power consumption tests at its rated operating temperature prior to its flight performance evaluation tests.

- A Problem in Welding Propellant Tubing to Thruster Bodies

The thruster body and its three propellant tubes are shown in Figure 51. The tubes were electron beam welded but leakage was indicated at a location which was well-defined. Initially it was believed that the leakage resulted from a minute crack in the thruster body but further checking pin-pointed a small weld-damage opening in one of the tubes. When set up for a repair welding operation, during this rework it was damaged beyond repair by an involuntary malfunction of the electron beam welding equipment. This damaged body is shown in Figure 52. The cause of the fault was determined to be an equipment fault and steps were immediately taken to correct it. The remaining 8 thruster bodies were examined for cracks and then cleaned in preparation for welding and brazing.

An effort to weld the three type 316 stainless steel thruster tubes of a second unit again resulted in failure. One of the tubes was melted away and the body partially damaged. This was caused by an unexpected deflection of the electron beam by a shoulder area located between the tubes. Consequently, it was decided to remove the shoulder

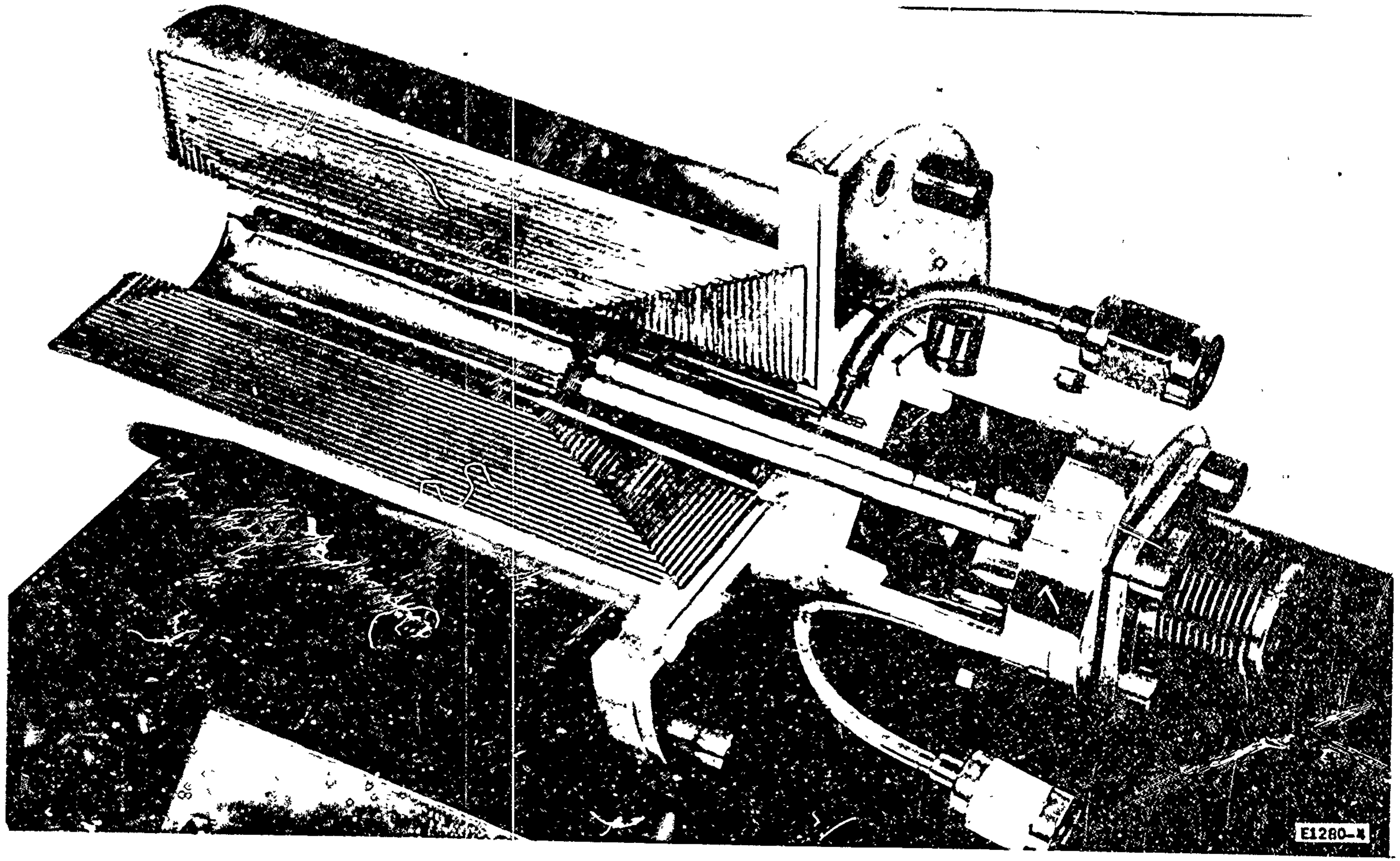


Figure 50. Model, Flight Thruster.

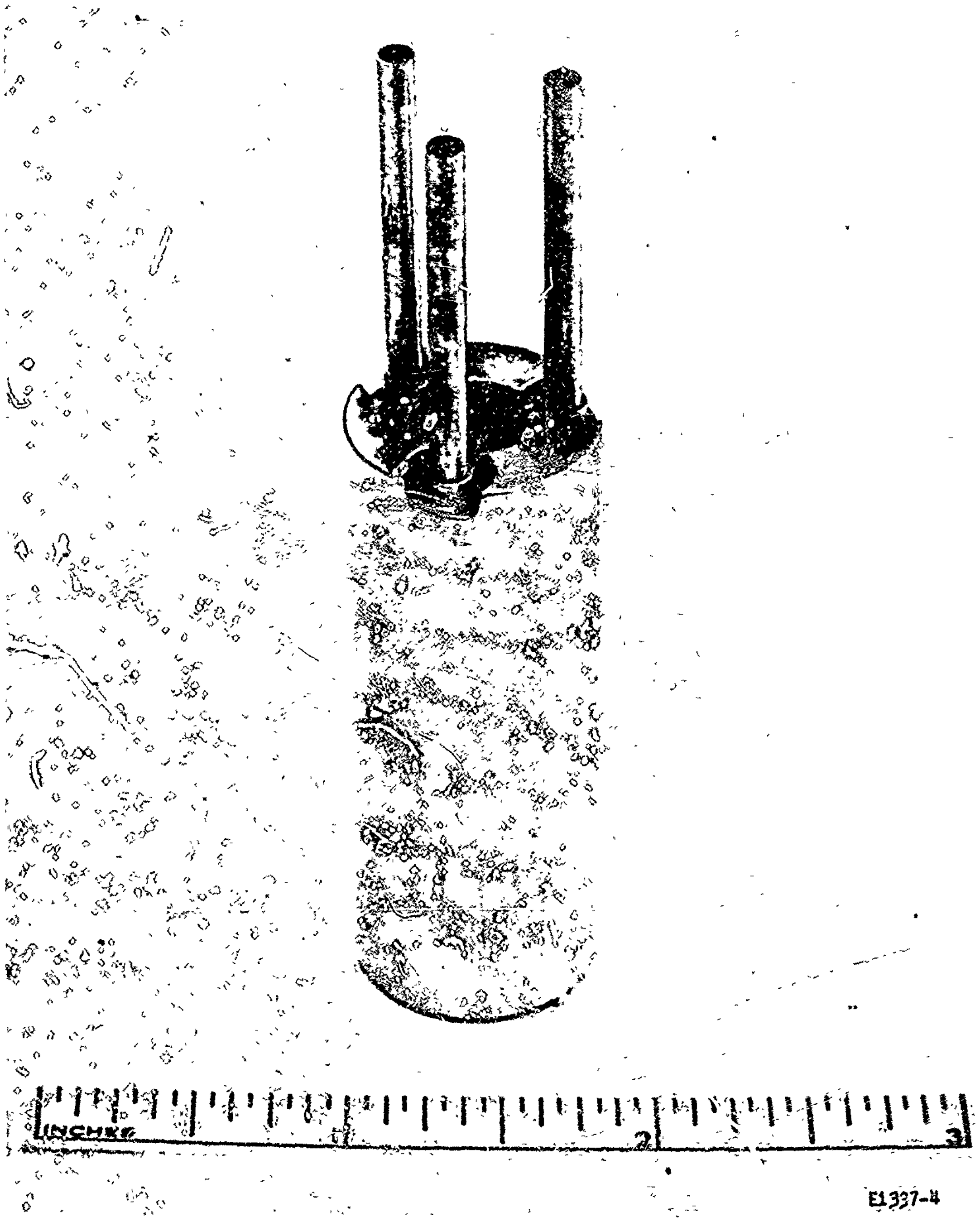


Figure 51. Flight Thruster Body With Propellant Tubes in Position For Electron Beam Welding.

Figure 52. Flight Thruster Body Damaged During Electron Beam Welding of Propellant Flow Tube.

areas from all 8 thruster bodies. The results of welding the first three modified bodies were an unqualified success. In addition, the previously damaged body (the second one) was salvaged and made serviceable. Leak testing proved that all 3 thruster bodies were leak-tight. Welding efforts were continued with the remaining thruster bodies. Those already welded (Figure 53) were scheduled for further assembly by having the thruster tubes brazed to the thruster flange/heat shield housing combination. It did, however, become apparent that welding of the propellant tubes of the subsequent units would be a rather tedious and somewhat unsatisfactory method of joining, since the close proximity of the tubes tended to deflect the electron beam. The removal of the "shoulders" from the thruster body greatly improved the situation but the welding results were not consistently of leak-tight quality. It became apparent the weld zone was so thin that the joints had to go through additional brazing cycles in order to seal leaks. This was the method used with the third unit in which the leak problem was rectified by brazing with gold-nickel alloy. Because the brazing worked out so well, it was decided to use this process, after all, for joining the tubes to the body rather than electron beam welding. This was considered to be a major breakthrough because all the subsequent joints were successfully made.

- Buildup of Heat Shield Packages

The heat shield packages, Figure 54, were built up of die punched parts, an innovation in heat shield processing. It became readily apparent that this resulted in a more uniform quality assembly as well as contributing to a saving in labor cost. A photograph showing how the heat shield fits over the thruster body is shown in Figure 55.

- Continued Buildup of the Thrusters

As the welding and brazing problems were solved, the buildup of the thrusters continued. In record form, the activities were described as below:

Assembly No.	Description of Activities
1	a. Three (3) tubes were EB welded to the thruster body and assembly was leak tight per SPPS Spec. 03-0013-00B.
	b. Six (6) tube joints were brazed to thruster flange with Au-18Ni at 1800°F for 3 minutes, and all were leak tight. However, a leak developed at one of the EB weld joints.
	c. Leak at weld repaired by brazing with Au-18Ni and total assembly was leak tight.
	d. Heat shield housing brazed with Au-18Ni.
2	a. Three (3) tubes EB welded to thruster body and assembly was helium leak tight.

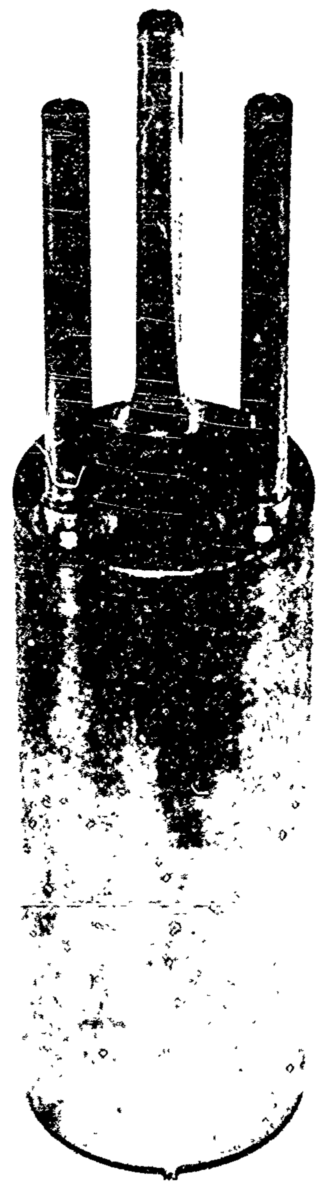


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Figure 53. Flight Thruster Body with all Three Propellant Tubes Welded in Position.

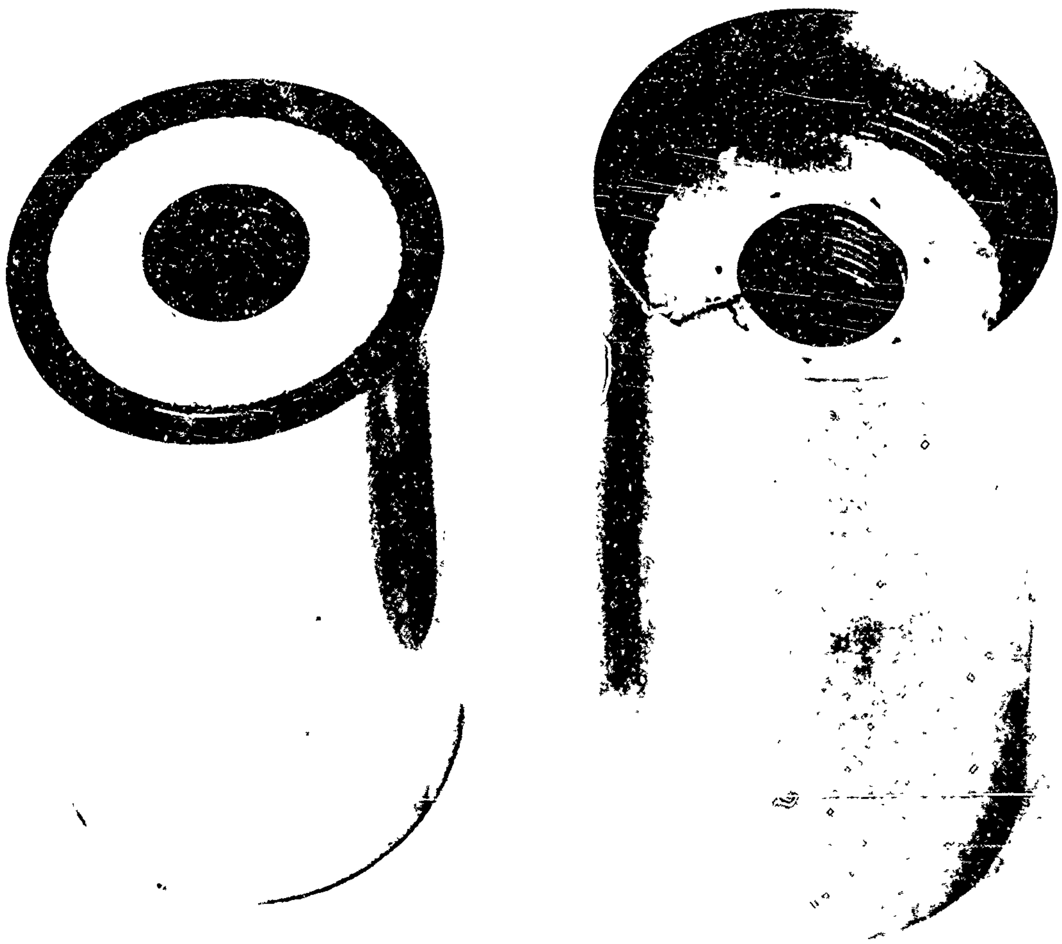

Figure 54. Heat Shield Package for Flight Thruster.

Figure 55. Assembly, Heat Shield and Flight Thruster Body.

b. Assembly exposed to simulated (Au-Ni) braze cycle (1800°F for 3 minutes). All weld joints remained helium leak tight.

c. Six (6) tube joints were brazed to thruster flange with Au-18Ni. Assembly was leak tight.

d. Heat shield housing brazed with Au-18Ni.

3 a. Three (3) tubes EB welded to thruster body and assembly was helium leak tight.

b. Assembly exposed to simulated (Au-18Ni) braze cycle - 1800°F/3 minutes. All three joints leaked.

c. Leaks at joints were repaired by brazing with Au-18Ni. Three braze cycles were necessary because of additional crack formation. Assembly leak tight after third cycle.

d. Six (6) propellant tubes were brazed to thruster flange with Au-18Ni. Temperature at repaired weld joints reached 1590°F during this braze cycle. Assembly was helium leak tight after this braze cycle.

e. Heat shield housing brazed with Au-18Ni.

4 a. One (1) tube to thruster body was EB welded, no leak check performed.

b. Remaining two (2) tubes to thruster body were brazed with Pd-38Co alloy at 2280°F/2 minutes. A ring of braze alloy was also applied at the EB weld joint. Assembly was helium leak tight.

c. Propellant tube brazing (Au-18Ni) to thruster flange will be completed by 3-10-68.

d. Heat shield housing brazing will be completed by 3-15-68.

5, 6, 7, 8, a. Tubes to thruster bodies were brazed with Pd-38Co alloy. All four (4) assemblies were helium leak tight.

b. Propellant tube brazing (Au-18Ni) to thruster flange will be completed by 3-10-68.

c. Heat shield housings for all four (4) assemblies to be completed by 3-15-68.

NOTES

Assembly S/N #3 was considered to be the pilot model because of the excessive cracking at the repaired EB weld joints.

NOTES, cont'd

Assemblies S/N #3 and S/N #4 were radiographically inspected to check flow through of braze material and resultant integrity of assemblies.

All brazing conducted in vacuum per GE-SPPS Spec. 03-0039-00A.

Views of Pilot Model Thruster Body appear on Figure 56 and 57.

Views of Pilot Model Thruster partially assembled, appear on Figures 58 and 59.

- Thermocouple Problems

Twenty thermocouples that were initially ordered were rejected because of poor workmanship. The breakage at the point at which the thermocouple lead wires are joined to the thermocouple junction wires may be seen in Figure 60. It is doubtful that the glass-bead method, used at the junction point would be satisfactory for the eventual assembly into the flight package. The junction point, shown magnified in Figure 61, had a tendency to break during shipping and handling. Consequently, the vendor was requested to devise a fabrication method which will eliminate this problem.

Being uncertain that the vendor would devise a successful fix and deliver them on time, a duplicate order was placed with another vendor. These were found to be completely acceptable. As the buildup of the thrusters progressed, however, it became apparent that the better place for a thermocouple was within the thruster heater itself. Accordingly, these separate thermocouples were laid aside and not used.

- The Thruster Heater

Because the heater was an integral part of the thruster, and because its procurement generally resulted in late delivery, an order was placed in June 1967 for heaters suitable for use in the 2000°F flight thrusters. Because it had not been determined at that time whether a heater element of recrystallization-resistant platinum wire or of an alloy consisting of 50% molybdenum - 50% rhenium was better for long continuous use as well as intermittent use, some of each were ordered. Also because an analysis of Power and Signal Conditioner losses was not complete, it was not known at that time whether lower power losses would result from use of a 12.5 to 13 volt heater, or a 24 volt heater. Accordingly, a minimum quantity of 8 each of these heaters were requisitioned and were as described below:

13 volt, DC 20 watt, recrystallization-resistant platinum heater element wire, per GE Drawing 47C143254P1.

24 volt, DC 20 watt, Mo-50Re heater element wire, per GE Drawing 47C143255.

12.5 volt, DC 20 watt, Mo-50Re heater element wire, per GE Drawing 47C143256

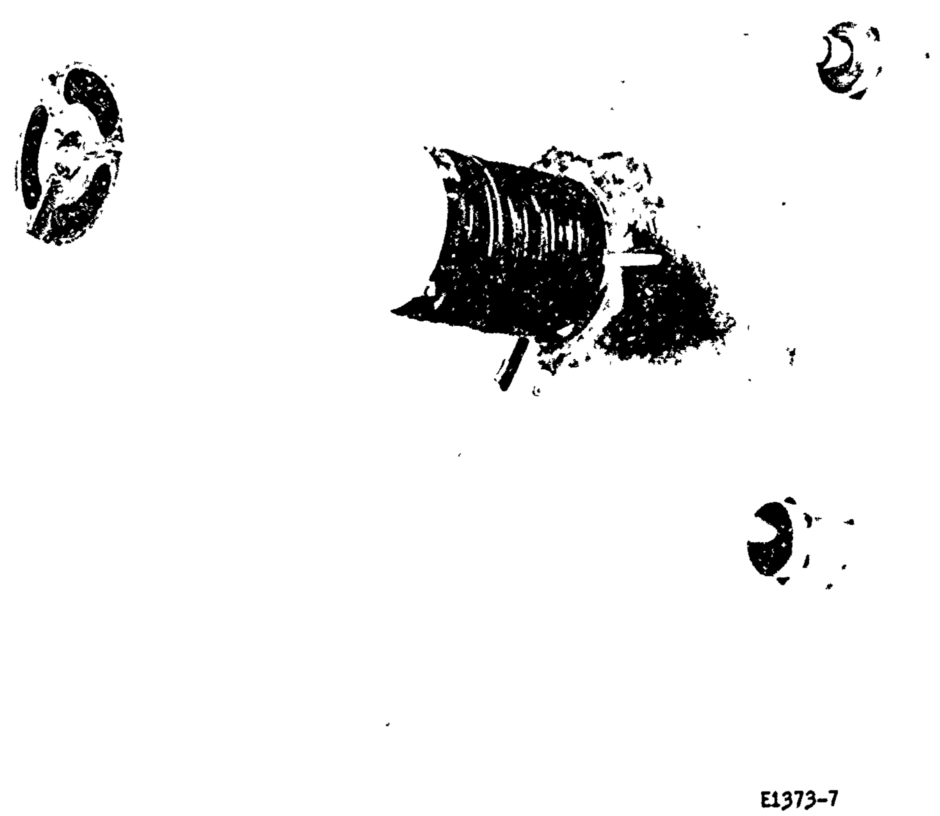

Figure 56. Pilot Model Thruster Body, Showing Catalyst Cartridge Chambers.

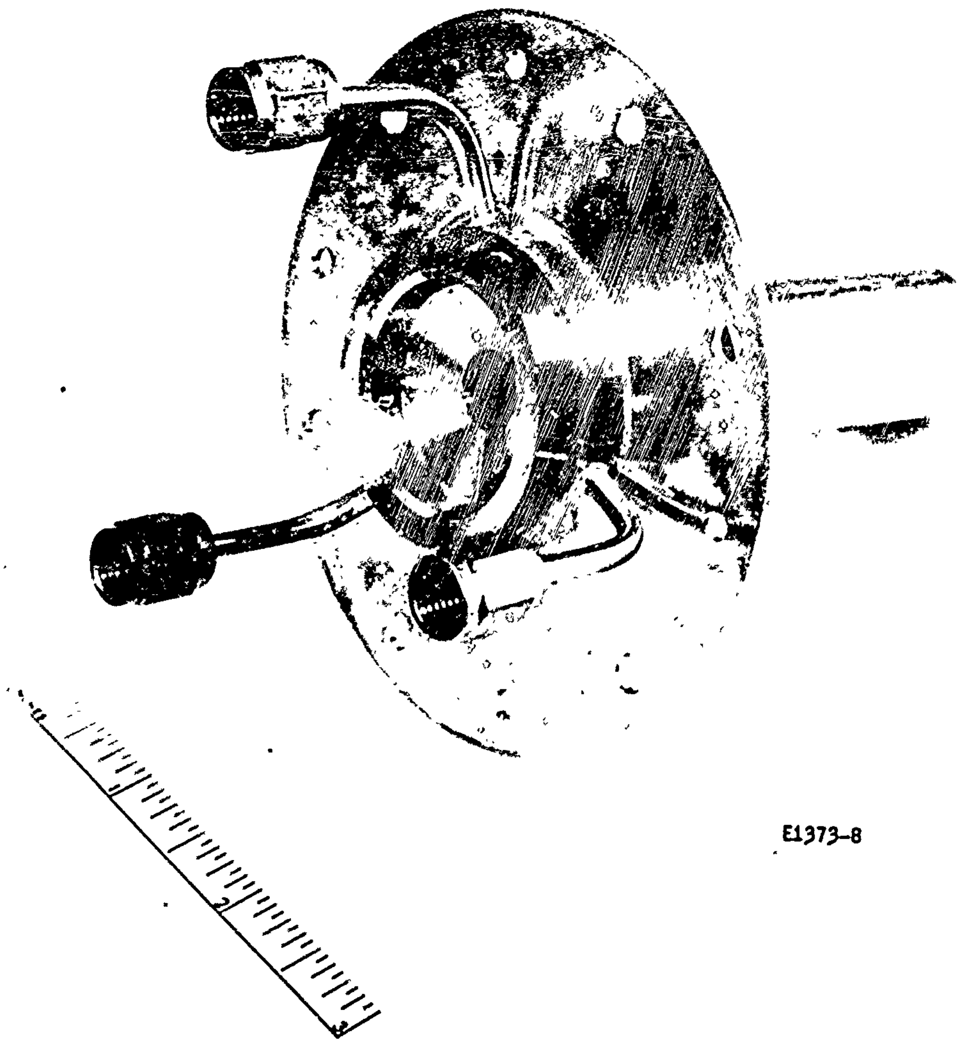

Figure 57. Pilot Model Thruster, Showing Brazed Fuel Tubes.

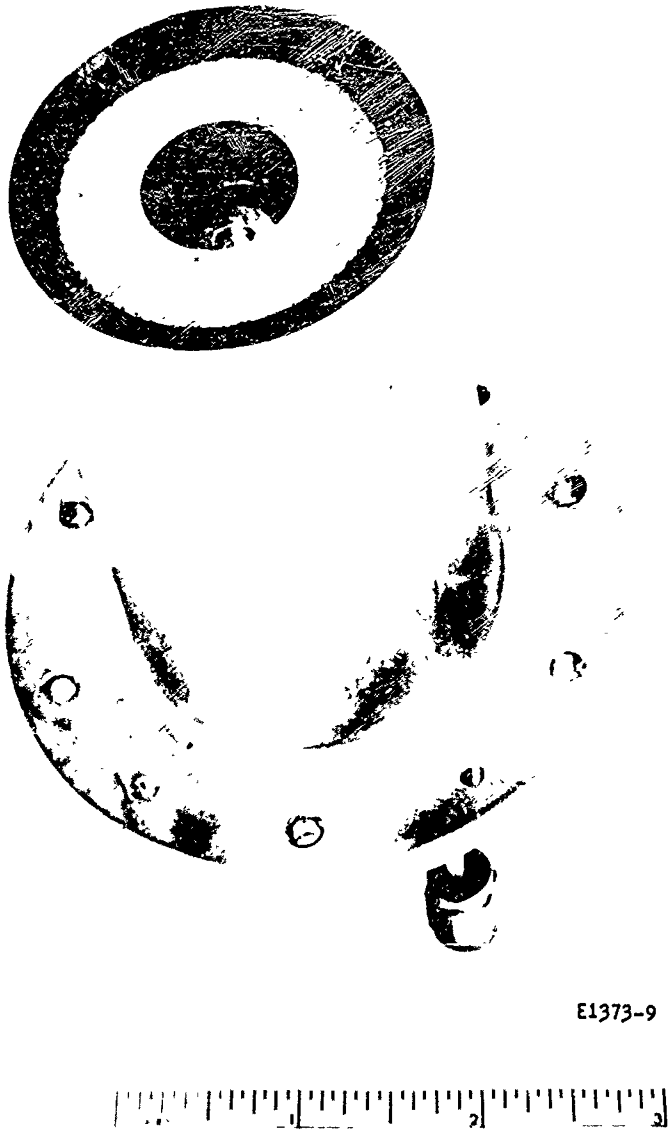

Figure 58. Partial Assembly, Thruster, Pilot Model.

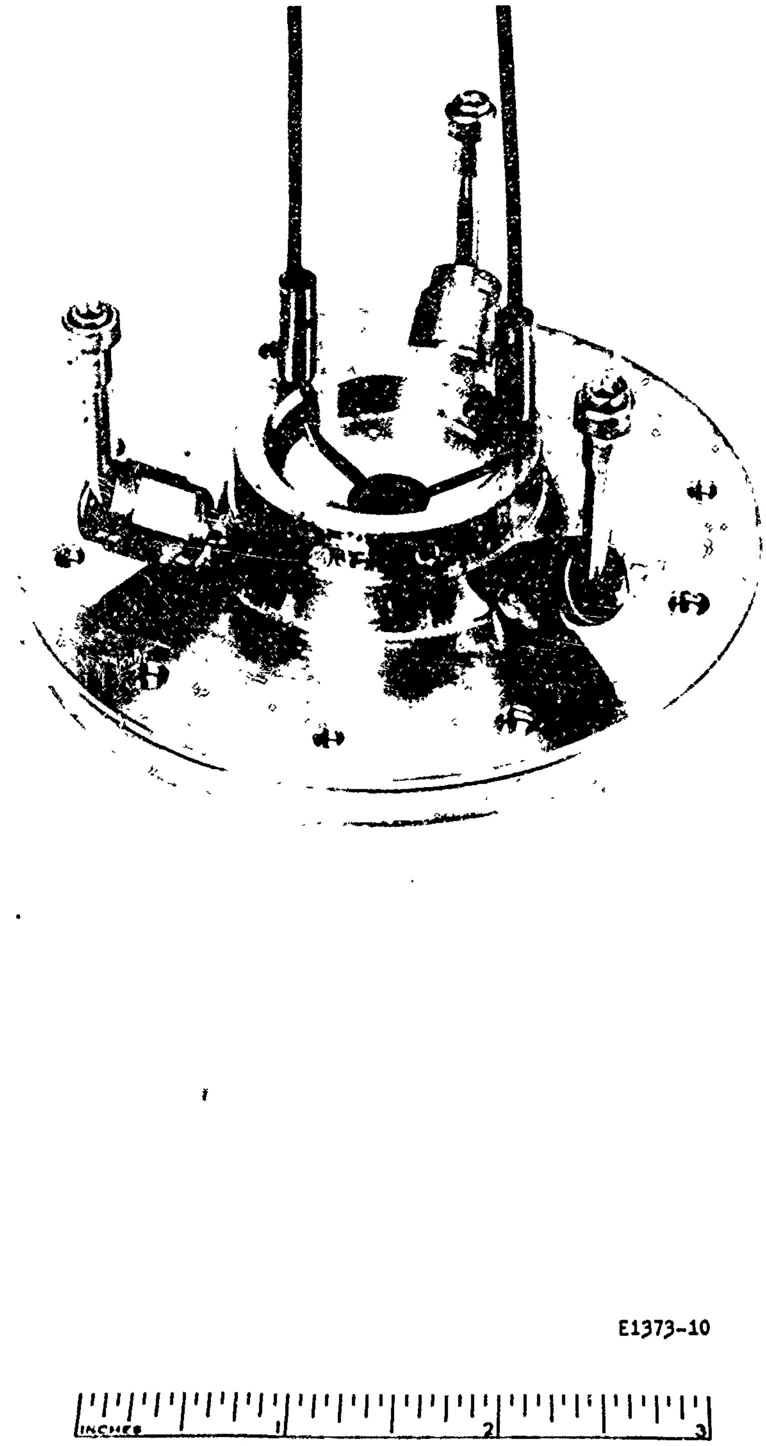

Figure 59. Partial Assembly, Thruster, Pilot Model, Showing Thermocouple Insertion.

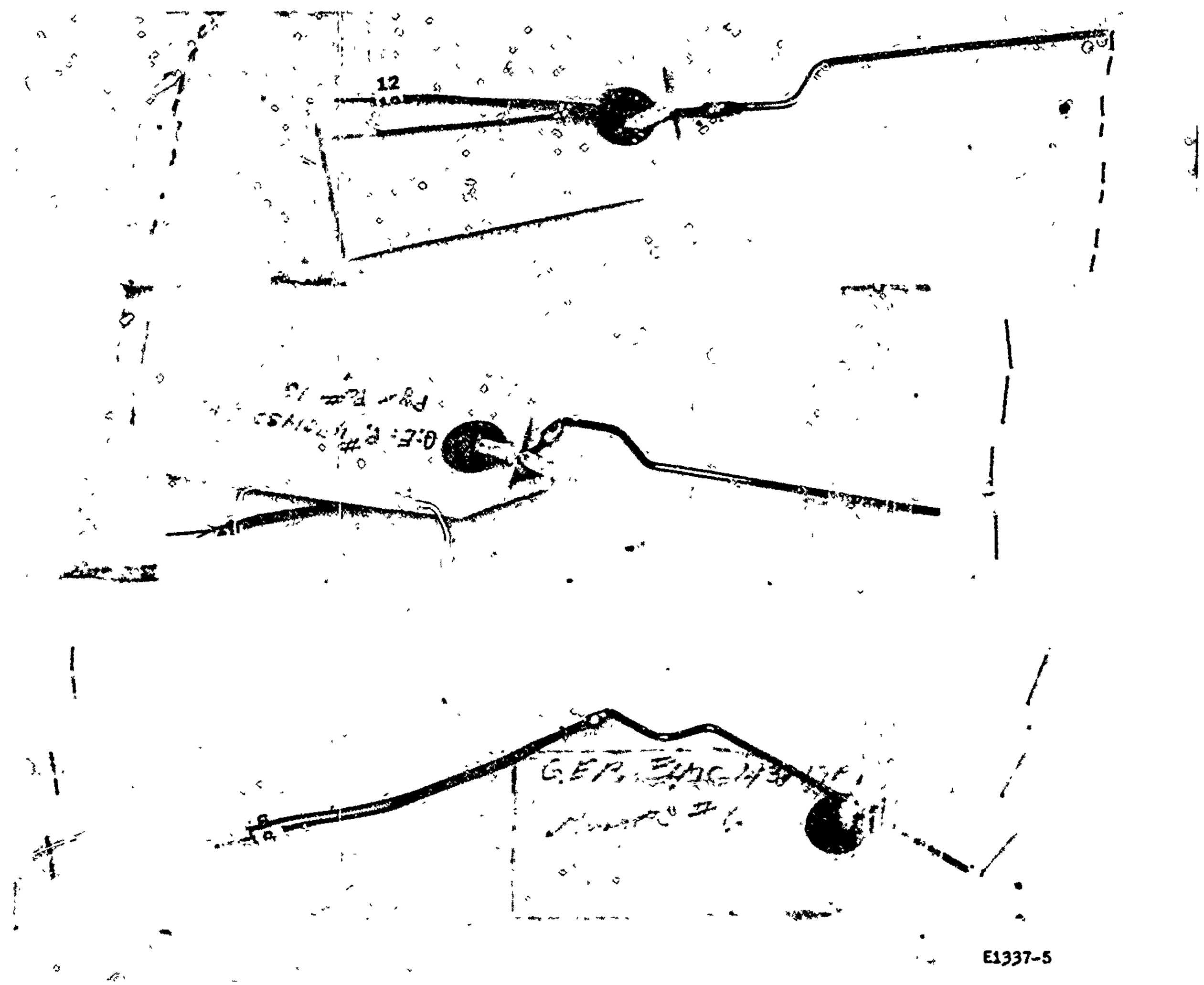

Figure 60. Flight Thruster Thermocouples As Received, Showing Broken Junction.

Figure 61. Flight Thruster Thermocouple Showing Broken Junction.

The supposition that delivery would be long was confirmed by these delivery dates:

Partial delivery	8/1/67
Partial delivery	8/15/67
Remainder of heaters delivered	10/20/67

Expecting that the thrusters would be built on schedule and that their testing could be delayed by a lack of heaters, a quantity of six, 13 volt DC, 20 watt heaters built up of platinum - 10% rhenium element wire and with an internal chromel-alumel thermocouple junction imbedded within its core were borrowed in June 1967 from another program. These were machined down to the dimensions of the DODGE-M heaters, inspected, tested, and placed in storage for possible use. Figure 62 shows their final form.

It is important to note the amount of effort that was applied to the heater task before heaters could be actually ordered. The following specifications were prepared to control quality of the heater:

Specification Number	Title
01-0076-00-A	Heater wire, Recrystallization - Resistant Platinum
01-0077-00-A	Boron Nitride Powder
01-0078-00-A	Swageable Magnesia Electrical Heater Cores
01-0079-00-A	Heater Wire, Mo-50Re
01-0202-00-A	Special Processing Requirements, Swaged Electrical Heaters

While these specifications were being prepared, designs were being made of each of the three heaters to be ordered. As an example, actual calculations for the 24 volt DC Moly-50Re heater appears in Appendix II. A procurement drawing finally had to be made up. The heater as received had yet to be machined to its final dimensions. However, before machining, it was subjected to an extensive testing schedule. The procedure is reproduced as Appendix IV. Typical inspection results are shown on Appendix V.

Before some doubt was expressed by key personnel concerning the true characteristics of the satellite electrical power that would be made available to the Microthruster System and whether or not the thruster would operate at 2000°F, a heater design had already been selected. Studies showed that the lowest power losses in the Power & Signal Conditioner would result if the higher voltage heater was used. Also, laboratory testing proved that the Mo-50Re heater was superior to the others. Therefore the heater planned for flight use was the 24 volt DC Mo-50Re heater.

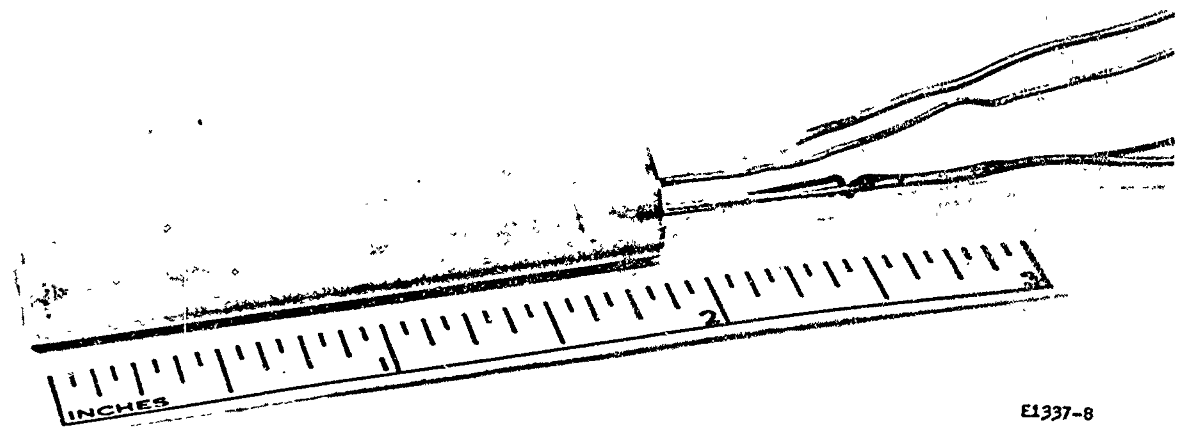

Figure 62. Interim Flight Thruster Heater (Equipped With Internal Chromel-Alumel Thermocouple Junction)

However, it was planned that as soon as the characteristics of the final electrical power available to the Microthruster System could be made known, a new heater would be designed so that a minimum power loss would occur in the heater circuit of the power and signal conditioner. Also, because the thruster was to be operated at 1500°F rather than 2000°F, readily-available Nichrome wire could be used as its element, thereby improving delivery.

6. FINAL STATUS OF THE THRUSTERS

The thruster buildup was carried through the point described above in "Continued Buildup of the Thrusters". Tests were conducted in a vacuum facility on the pilot model thruster with a 3 mil thick disc of Nichrome foil in place of the orifice/nozzle plate. The results showed that to maintain the thruster body at 1500°F, the electrical power required was 12.3 watts. This was well within the 15 watts allowable. It was planned to rerun the test to secure a complete thermal profile of the assembly. This was never realized. The weight of this assembly was 1 pound, 10 ounces.

SECTION V

TANKAGE

1. GENERAL SCOPE OF THE PROBLEM

The problem consisted of (a) designing a fuel tank for ammonia which would separate liquid and vapor in such a manner that vapor only is presented to the exit orifice under zero or very low gravity conditions, and (b) confirming the effectiveness of the design by motion pictures under low gravity conditions, simulating the fuel tanks with clear plastic models.

The design problem was simplified by several factors. First of all, the proposed application was very close to true zero gravity, so that the forces involved were almost wholly adhesion-cohesion forces. This meant that only the direction and not the magnitude of the forces had to be considered. Second, the maximum propellant flow rate was so low that cooling effects at the liquid surface were negligible. Finally, there existed a considerable body of theoretical and experimental information on the subject from previous studies.

The general approach to the problem was to design the tank geometry by applying existing theoretical relationships which have been well confirmed experimentally. The initial interface and the interface prevailing after partial depletion of the fuel had to be considered. The method of experimental confirmation was photography of the liquid behavior in simulated clear plastic fuel tanks during low gravity KC-135 flights. Only those aspects of behavior which occured in a few seconds could be confirmed experimentally. Behavior requiring longer periods of zero-g than the KC-135 flights can supply, such as evaporation-condensation phenomena, heat-transfer, slow tank draining, and slow translation between configurations of nearly equal energy, can only be treated theoretically.

2. THEORETICAL BACKGROUND

Extensive treatments of the theory of liquid-vapor interface at low gravity have been made elsewhere, and will not be repeated here. A few relationships will suffice to allow the selection of a simple fuel tank geometry for a wetting liquid under conditions of very low acceleration and heat transfer.

- Stability of Bubbles vs. Size

In a general way it can be shown that the relative stability of bubbles and drops of various sizes favor the coalescence into larger sizes. For the simple case of spherical bubbles of uniform size, the number N and diameter D are related to the fixed volume V by

$$V = \frac{\pi}{6} N D^3 \qquad (1)$$

The interfacial area is given by

$$A = \frac{\pi}{4} N D^2 \qquad (2)$$

which is, of course, proportional to the surface energy. Substitution of D from (1) into (2) gives

$$A = \frac{(9\pi V^2)}{16}^{1/3} N^{1/3} \qquad (3)$$

so that the surface area and energy are minimized as N approaches 1 for any non zero volume. Although coalescence may be hindered or delayed by surface charge, surface film tension, or kinetics, converse statements of the above relationship are generally true. That is, larger spherical bubbles do not spontaneously break up into smaller ones, nor does a larger spherical bubble collapse while a smaller one grows under isobaric and isothermal conditions.

As regards the present tank design problem, the general principle may be established that large vapor bubbles, wherever formed, do not spontaneously collapse, but must be collapsed when desired by adverse pressure due to an unbalanced temperature or surface energy effect. A second general principle can be established that a spherical vapor bubble of any size in any real liquid will stick to the wall in preference to floating free. This can be shown by comparing the liquid-vapor interface area (hence surface energy) for any given volume of vapor. The wall bound bubble will always have a smaller liquid-vapor interface.

- Interface Equilibria for Concentric Tubes

Consider two connected, axisymmetric thin-walled tubes of radii r_1 and r_2, closed at the bottom end at one gravity. If the tube diameters are of the order of several centimeters, the liquid level will be approximately a plane surface as shown in the sketch on the left in Figure 63, since the Bond number (ratio of acceleration forces to surface tension) is so high. Transition to zero gravity of such a system results in a new balance of forces depending on surface tension only. The upward pressures in the inner and outer tubes are given by

$$P_1 = \frac{2\pi r_1 \sigma}{\pi r_1^2} = \frac{2\sigma}{r_1} \qquad (4)$$

$$P_2 = \frac{2\pi \sigma (r_2+r_1)}{\pi (r_2^2 - r_1^2)} = \frac{2\sigma}{(r_2 - r_1)} \qquad (5)$$

and the pressures are equal only if r_2 is exactly twice r_1. If r_2 is greater than twice r_1, then the upward pressure in the inner tube P_1 will exceed P_2, and the liquid will rise in the inner tube and fall in the

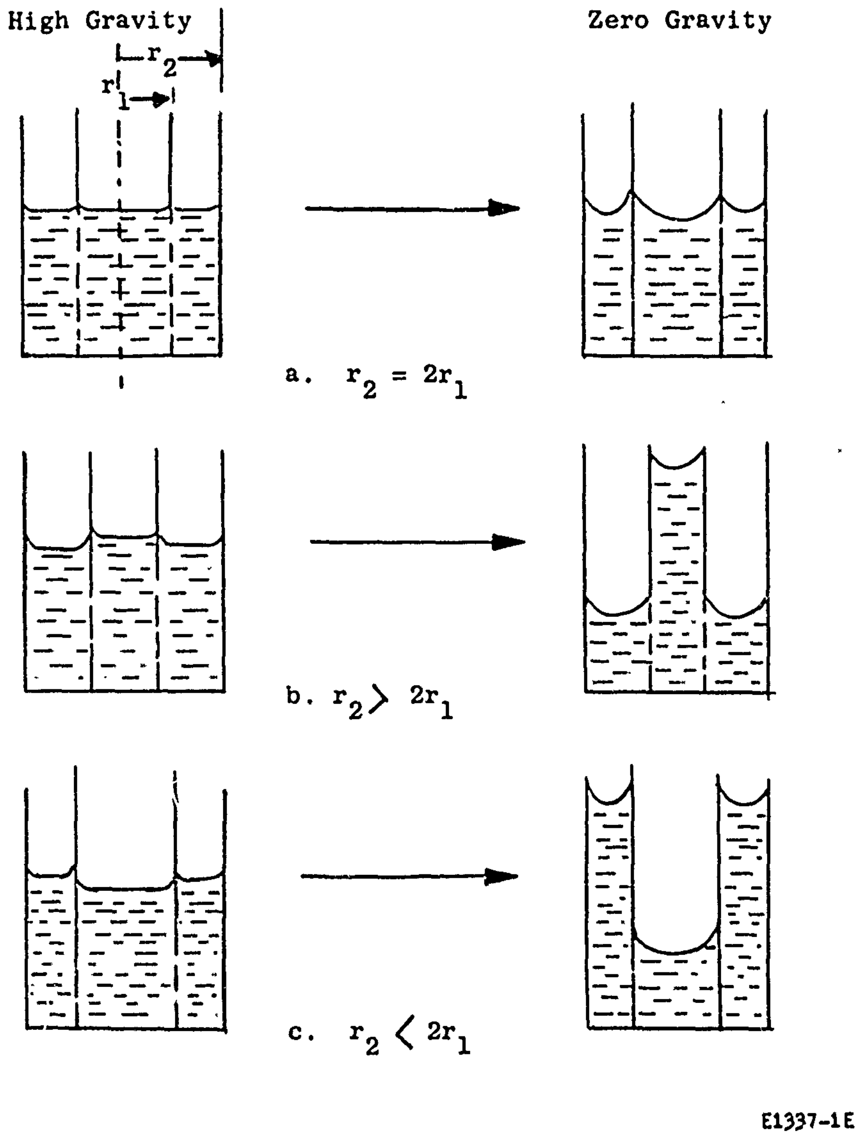

Figure 63. Zero Gravity Interface In Concentric Tube Geometry.

outer tube. If r_2 is less than twice r_1, the reverse will occur. Also, since there is no hydrostatic head at zero g, and since the radii are constant, this process will continue until one of the tubes is either completely empty or completely full. The interface geometry expected at zero-g for the three cases is shown on the right in Figure 63.

Now consider a similar case, where the tubes (cylinders) are replaced by axisymmetric cones, as shown in Figure 64. Equations (4) and (5) can now be modified to include the change in radius with height as follows: The upward pressures in the inner and outer cones are given by:

$$P_\theta = \frac{2\pi h_\theta \tan \frac{1}{2}\theta\, \sigma}{\pi h_\theta^2 \tan^2 \frac{1}{2}\theta} = \frac{2\sigma}{h_\theta \tan \frac{1}{2}\theta} \quad (6)$$

$$P_\phi = \frac{2\pi (r_2 + r_3)\sigma}{2\pi \frac{(r_2 + r_3)}{2} 2h_\phi \tan \frac{1}{2}\phi} = \frac{\sigma}{h_\phi \tan \frac{1}{2}\phi} \quad (7)$$

now, however, since the radii change with height, the flow between cones will not necessarily continue until one of the cones is either full or empty, as in the case of cylinders, but only until the changing pressures are equal. This is the case of interest, where $P_\theta = P_\phi$, and

$$\frac{h_\phi}{h_\theta} = \frac{\tan \frac{1}{2}\theta}{2 \tan \frac{1}{2}\phi} \quad (8)$$

Figure 65 illustrates the three different types of interface geometry possible for partially filled concentric cones, according to (8). In case a., $\tan \frac{1}{2}\theta$ is exactly twice $\tan \frac{1}{2}\phi$, and h_θ and h_ϕ as defined in Figure 64 are equal. In case b., the height of the liquid in the central cone will rise, and in case c., it will fall. Similar reasoning can be applied to a series of concentric cones.

One important distinction is that the central segment is a cone, whereas all others are like circular crevasses converging to a point. As long as the cone bases remain open, the behavior is as predicted by equations 6, 7, and 8. But when the cones are enclosed in, say, a spherical tank, the behavior becomes more complex.

- Equilibrium Interface for Axisymmetric Cones in Spherical Tanks

Consider the design shown in Figure 66, of two concentric cones in a spherical tank. The geometry is complicated over that of the concentric cones shown in Figure 64 by the introduction of angles made at the juncture of the cone bases with the spherical walls, $\alpha, \beta, \gamma, \delta$. The equilibrium position will, of course, include a portion of the liquid hanging up in these crevasses, since they are similar to the angle ϕ, except for size. If we neglect the curvature of the outer sphere for the short distance involved, the equilibrium distribution of liquid in the

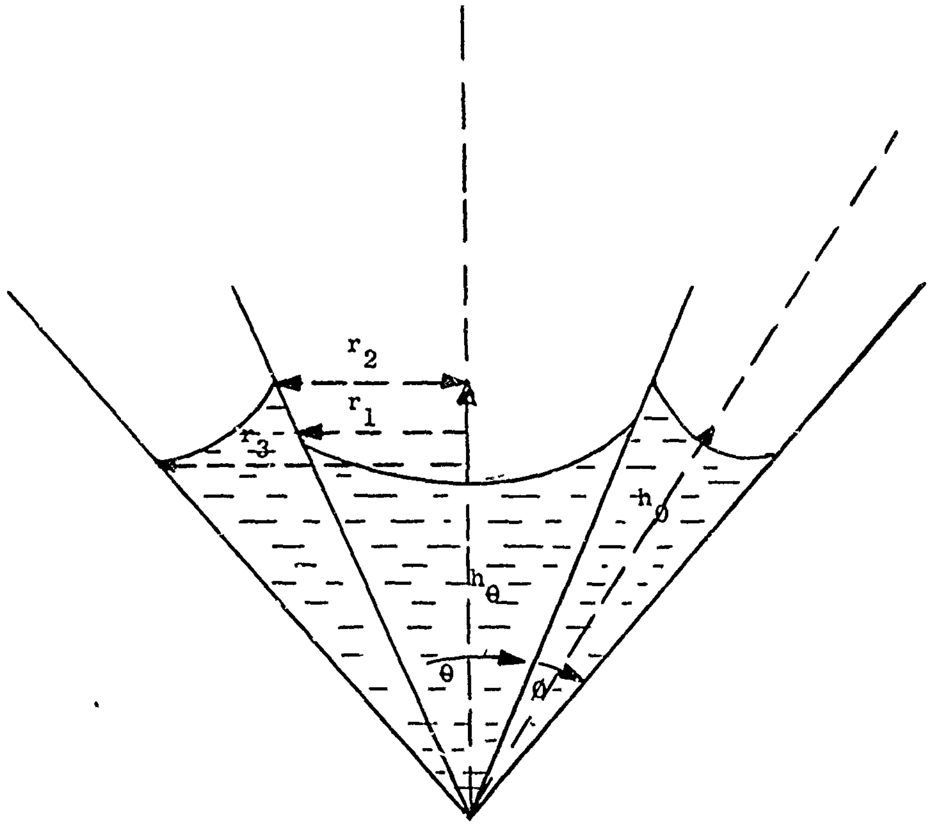

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Figure 64. Definition of Symbols in Equations 6, 7, 8.

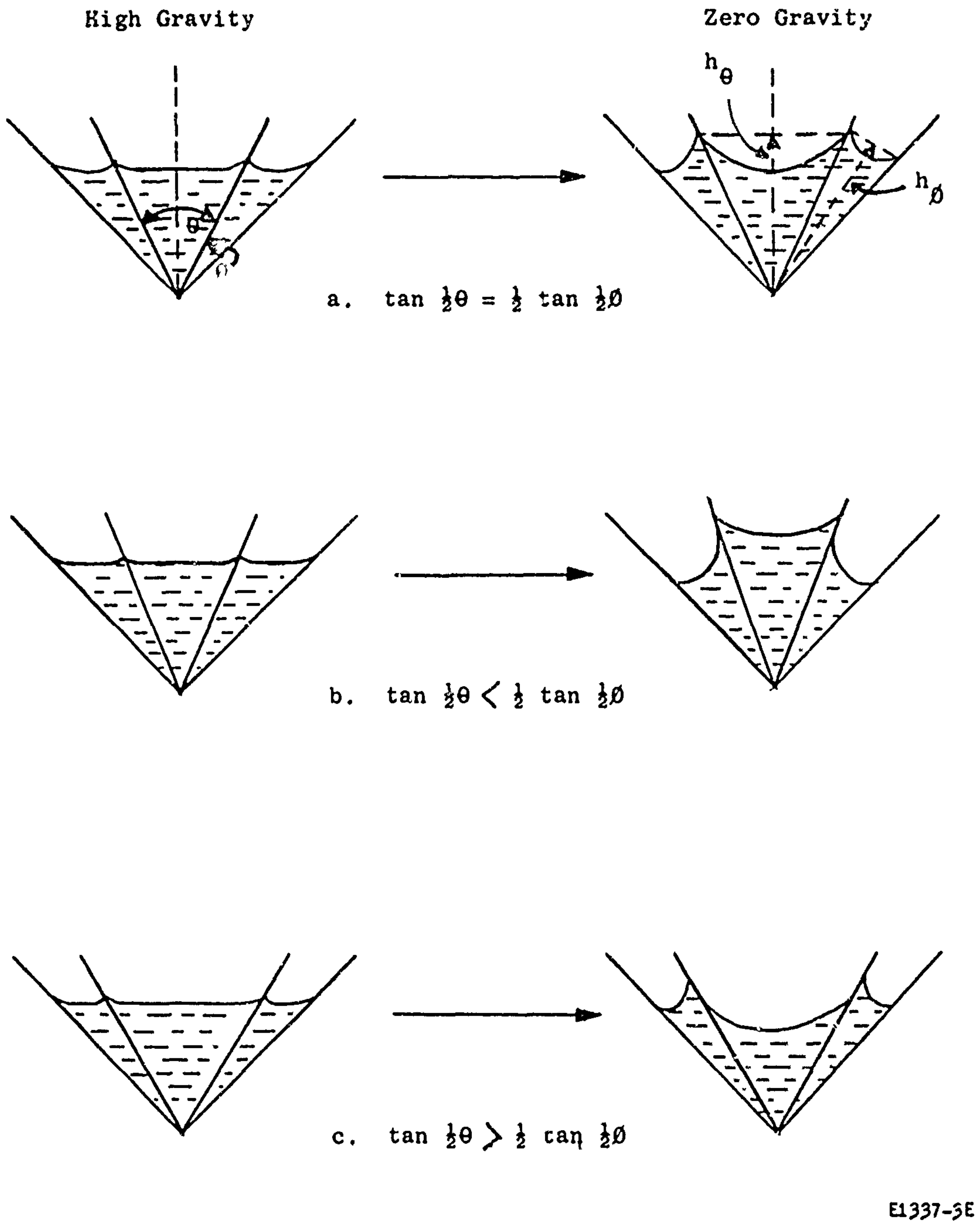

Figure 65. Zero Gravity Interface in Concentric Cone Geometry

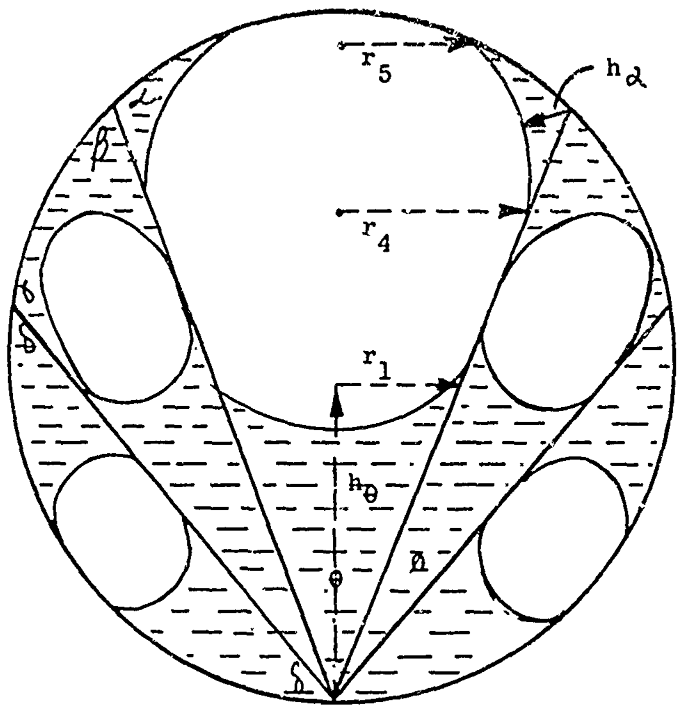

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Figure 66. Equilibrium Interface Predicted for Case of Appreciable Voids in Each Segment

various crevasses can be simply derived. At the apex of the inner cone the pressure was given by (6), and around the angle formed by the base with the sphere

$$P_\alpha = \frac{2\pi\sigma\,(r_4 + r_5)}{2\pi\,\frac{r_4 + r_5}{2}\;2\,h_\alpha\,\tan\frac{1}{2}\alpha} = \frac{\sigma}{h_\alpha \tan\frac{1}{2}} \qquad (9)$$

and at equilibrium

$$\frac{h_\alpha}{h_\theta} = \frac{\tan\frac{1}{2}\theta}{2\tan\frac{1}{2}\alpha} \qquad (10)$$

so that if α is kept slightly larger than θ, θ being of the order of 40-60°, h_α will be of the order of 1/5 the magnitude of h_θ.

Equations (7) and (9) are identical in form, and hold for all crevasses similar to α and ϕ, that is β, Γ, and γ. Equation (6) determines the pressure of the liquid in the apex of the cone only. Thus, at equilibirum the pressure P_θ, P_ϕ, P_α, etc. must be equal, so that

$$\frac{2}{h_\theta \tan\frac{1}{2}\theta} = \frac{1}{h_\phi \tan\frac{1}{2}\phi} \qquad \frac{1}{h_\alpha \tan\frac{1}{2}\alpha} = \frac{1}{h_\gamma \tan\frac{1}{2}\gamma}, \text{ etc.} \qquad (11)$$

Similar equations would result if the curvature of the sphere were not neglected, except that $h_{\alpha}, \beta, \gamma,$ would be slightly larger due to the channel being slightly narrower than expected from the assumption of plane surfaces. According to this equation, if all surfaces in the tank design shown in Figure 66 are assumed to be coated with a film of the liquid, the lowest energy condition, when appreciable voids are present in all sections, is one in which the heights of liquid in each crevasse are inversely proportional to the tangent of the half angle of the crevasse. In the case of the central cone, this will be true whenever the vapor volume is large enough to form a spherical bubble tangent to the sphere and the cone walls. In the outer segments, a spherical vapor bubble is, of course, a lower energy configuration than a toroid. At high fill levels, any voids in the outer segments will be spherical bubbles up to the size where they are tangent to both the spherical shell and the conical walls. At lower fill levels, a surface energy balance will be attained between an oblate partial toroid and multiple spherical bubbles, until the void is so large as to make a toroid necessary. At this point, a symmetrical oblate torus will be formed as shown in Figure 66.

In summary, we have three simple factors governing the liquid-vapor interface equilibrium at zero gravity; first, the tendency for large vapor voids to grow at the expense of smaller ones; second, the tendency for bubbles to attach to the wall; and third, the behavior governed by equation (11) and the qualitative discussion succeeding it.

3. TRANSITION FROM HIGH GRAVITY TO ZERO GRAVITY FOR VARIOUS CONDITIONS

Consider the design shown in Figure 66, but filled at one gravity nearly to the base of the inner cone as shown at left in Figure 67a. On transition to zero gravity, the toroidal bubble at β is unstable, since its collapse to a spherical bubble and the growth of the large bubble in the central cone gives a smaller surface energy in each segment. For this case the predicted equilibrium interface is as follows. If the total vapor volume is small enough to be contained in a single spherical bubble in the central cone, the result in Figure 67a will occur. If this volume is exceeded somewhat, result 67b will occur, a balance between elongation of the central bubble and maintenance of a small spherical bubble in the middle segment. If the fill level is approximately as shown in 67c then bubbles in both segments will be oblate.

When the fill level is at or slightly below the base of the second cone at one gravity, then equation (11) should hold for both inner segments at zero gravity while the outer segment remains filled as shown in Figure 67a. When the fill level at one gravity is somewhat lower, as in Figure 67b, then oblate bubbles in the central and middle segments will be in equilibrium with a spherical bubble centered in the outer segment. At very low fill levels, as in Figure 67c, equation (11) will hold for all segments.

- The Effect of Previous Gravity

The predicted equilibrium shown in Figures 67 and 68 are based on the assumption that the tanks are mounted so that the apices of the cones point down at one gravity. If the tanks are otherwise mounted, or if an appreciable off-axis acceleration is experienced just prior to transition to zero gravity, entirely different equilibria are predicted.

If we define the apices of the cones as the south gravity direction, then the effect of north gravity (mounting the tanks with apices up or reverse thrust) is as shown in Figure 69. In this configuration the inner two segments are narrow with respect to the outer segment at levels a and b, and they grow narrower faster as the liquid height is increased. Hence the predicted interfaces are those in which the inner segments are entirely filled. At level c the outer segment is the narrow channel, and will fill up somewhat while the levels in the inner segments drop. The low surface energy for large spherical, wall-bound bubbles will drive the equilibrium toward the illustrated geometry, not too much different from Figure 68.

The effect of east or west gravity is shown in Figure 70. At high fill levels, such that the central cone is filled at one gravity, there will be no tendency toward formation and growth of a small bubble in the central cone, at the expense of the already existing voids in the outer segments. At lower fill levels, the common equilibrium position of Figures 68 and 69 will be attained.

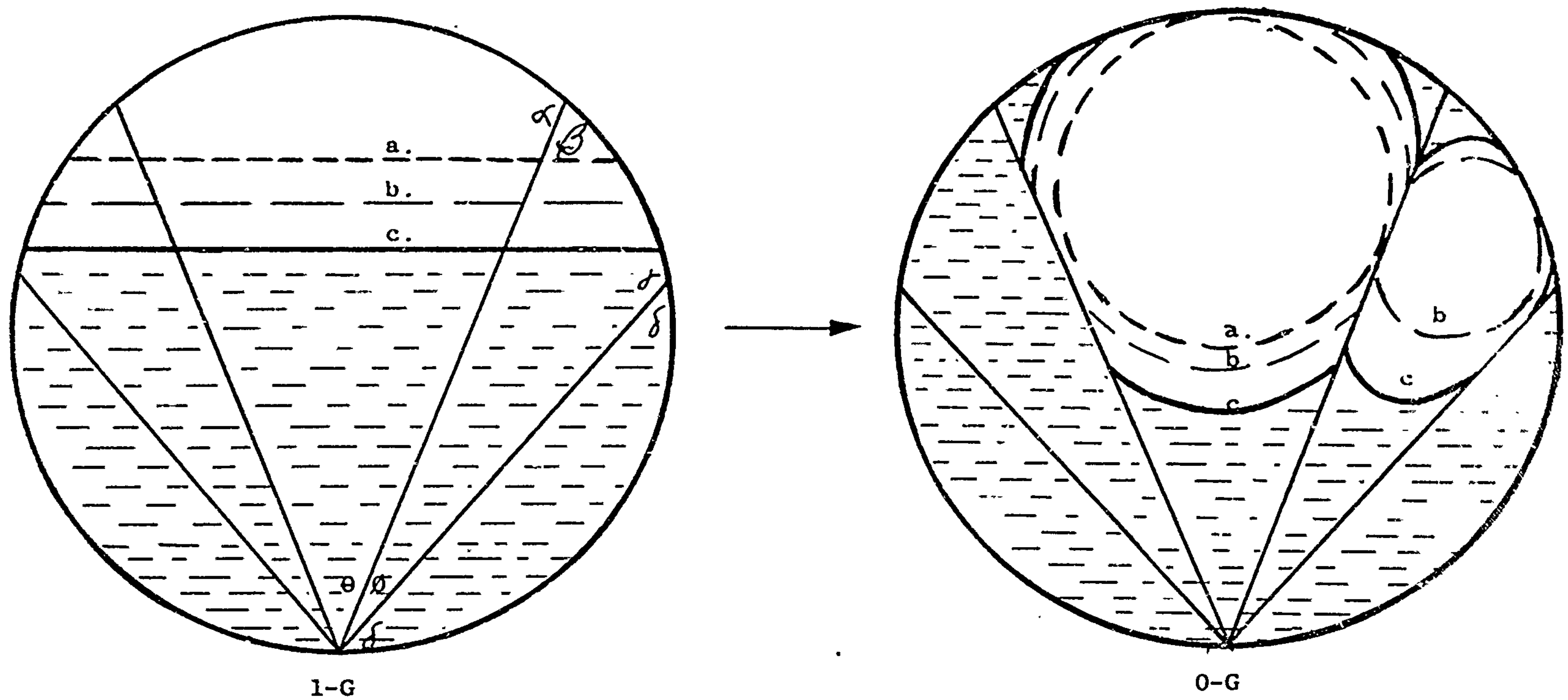

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Figure 67. Transition from High Gravity to Zero Gravity
for High Fill Levels

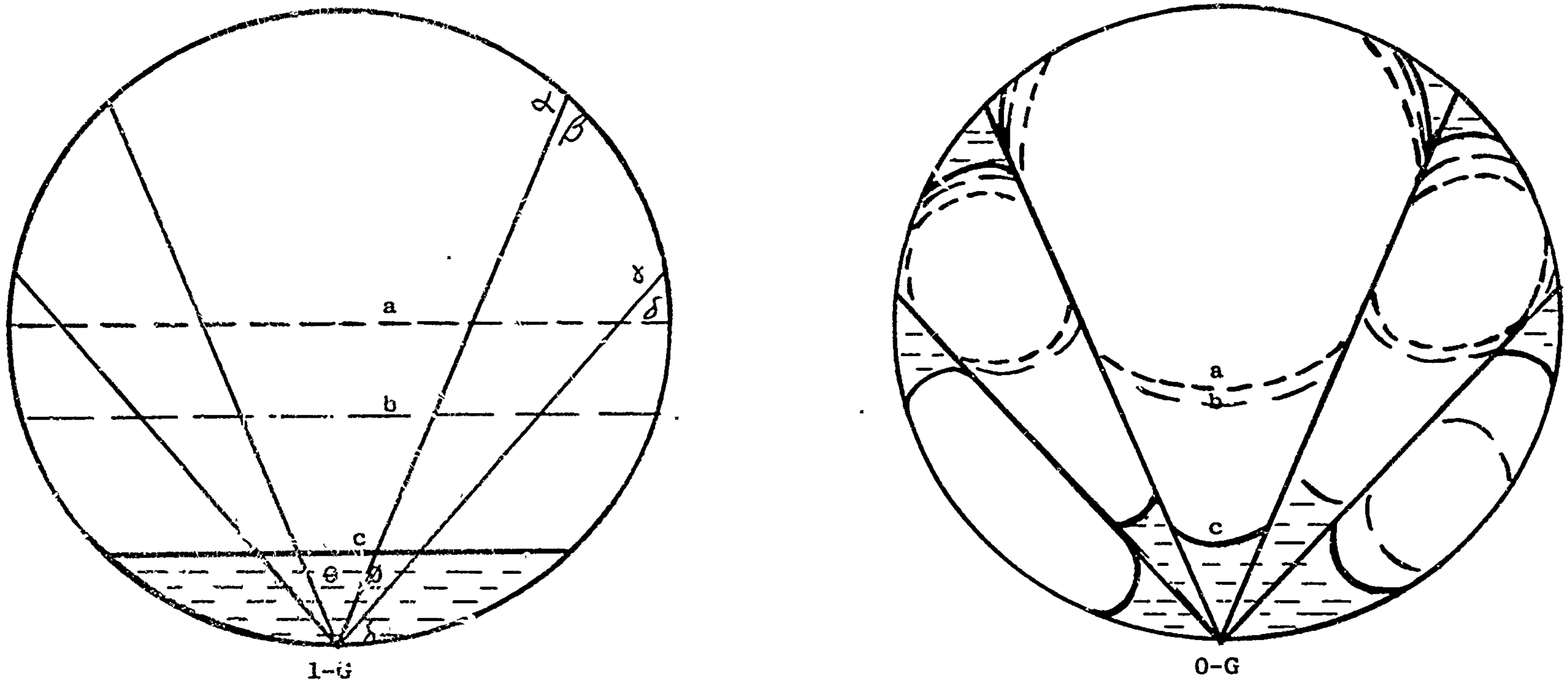

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Figure 68. Transition from High Gravity to Zero Gravity for Low Fill Levels

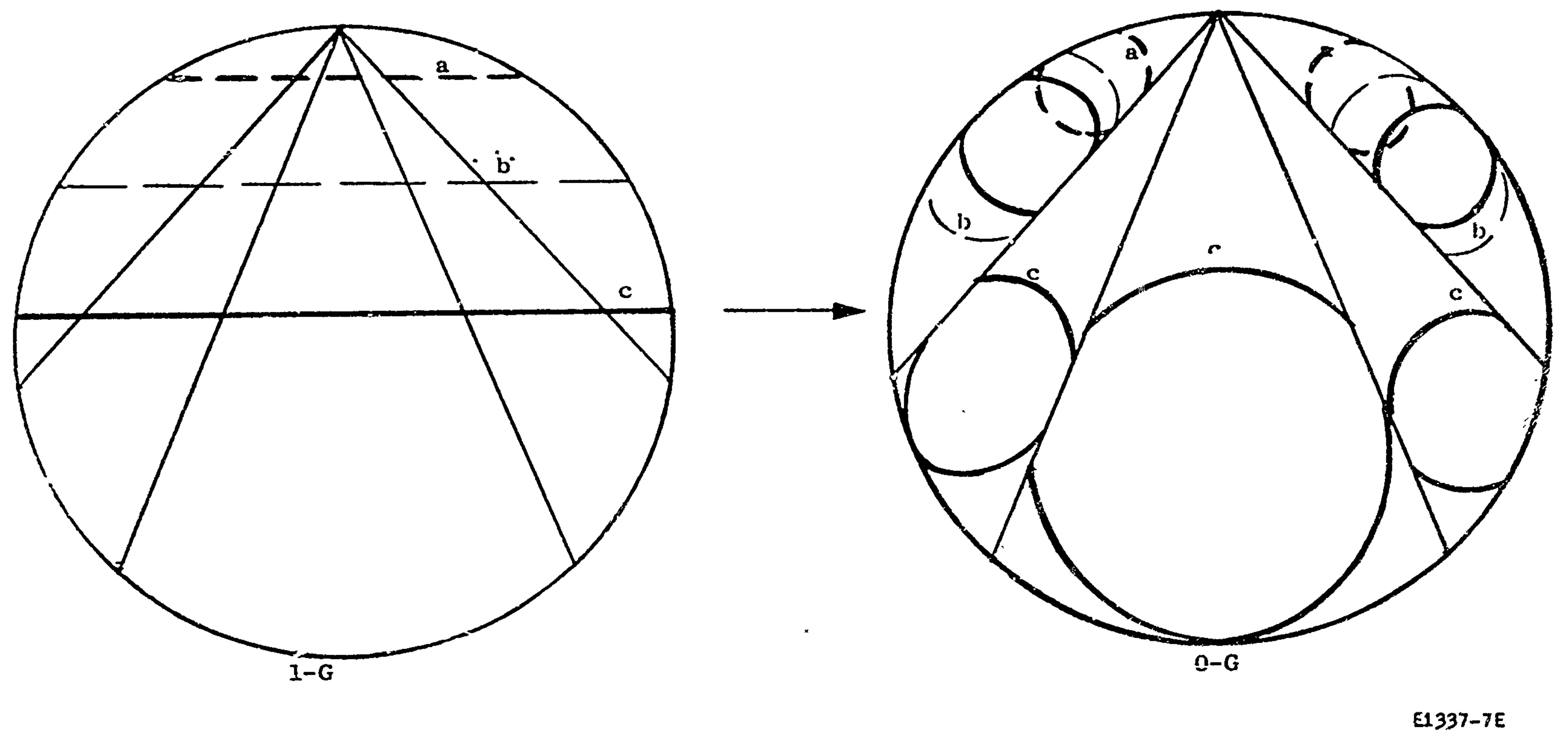

Figure 69. The Effect of North Gravity on Equilibrium Interface.

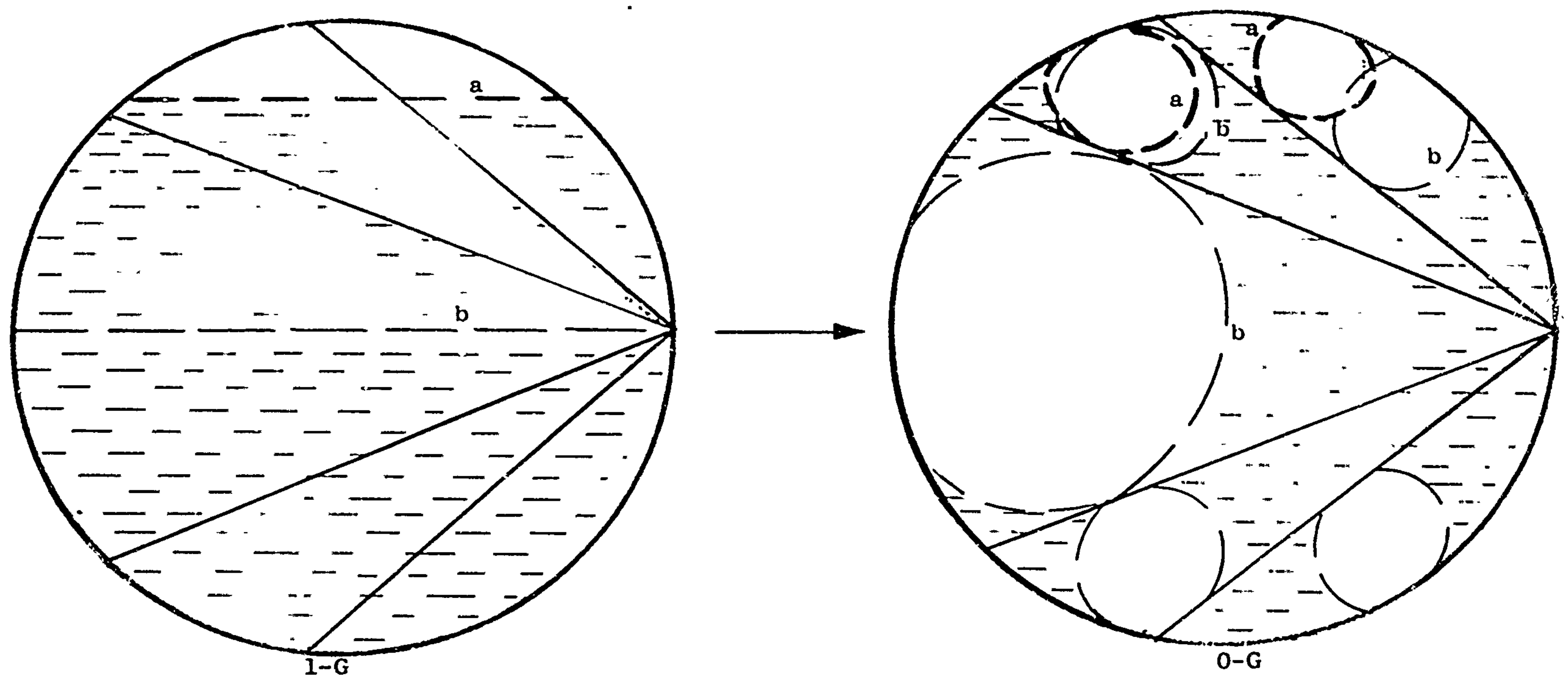

Figure 70. The Effect of East or West Gravity on Equilibrium Interface

- The Effect of Drainage History

Consider the configuration shown in Figure 67a filled to a reasonably high level during boost into orbit. Upon transition to zero gravity, a spherical vapor bubble in the central cone is the predicted interface. If the fuel is evaporated and used isothermally, there will never be enough driving force to initiate small, high surface energy bubbles in the outer segments until the entire central cone is empty. When the cone is empty, a small bubble will next be formed, probably at the most acute angle available, Ø, and this bubble will grow as the fuel evaporates further. As soon as this bubble formation occurs, a completely empty inner cone is no longer stable, and the central cone will be refilled until an equilibrium is established between the void spaces in the inner and middle segments. Then both segments will be emptied completely before bubble formation in the outer segment, followed by partial refilling of the two inner segments, in equilibrium with the growing bubble in the outer segment. In other words, the energy barrier to nucleation is large enough to keep filled segments completely filled while partially empty segments are completely emptied.

If this drainage sequence is interrupted by a period of acceleration, two entirely different configurations can result at the same fill level. For instance, if south gravity is experienced momentarily when the central cone is almost empty, the sequency shown in Figure 71 is predicted. Thus the equilibrium configuration during tank drainage cannot be shown during brief periods of zero-gravity merely by starting with a lower fill level.

- The Effect of Fuel Sloshing

More or less violent sloshing is expected during boost, spin removal, and major orbit correction. The effect of sloshing will be to produce many small vapor bubbles. Should the transition to zero-g occur immediately after violent mixing, the equilibrium condition will be metastable, with many vapor bubbles trapped almost in random fashion in local energy minima. Transition of such states shown in Figures 67 and 68 may occur during a long period of true zero-g, but this cannot be demonstrated during KC-135 flights.

4. GENERAL TANK DESIGN CRITERIA

From the above treatment it appeared that a design consisting of a series of concentric cones in a spherical tank would serve the purpose. It is desired that a vapor bubble be positioned at the center of the cone base at zero gravity. To accomplish this, the tangent of the cone half angle $\frac{1}{2}\theta$ should be greater than twice that of the other apex angles between adjacent cones, in order to insure that the height in the central cone will be less than the height in the outer segments. In addition, the angle γ should be kept large to avoid excessive liquid hangup in this area. The number of conical baffles should be chosen so as to minimize weight while providing enough surface forces to hold the liquid-vapor interface in the desired position against minor disturbing forces.

Off axis thrust and sloshing should be avoided as much as possible.

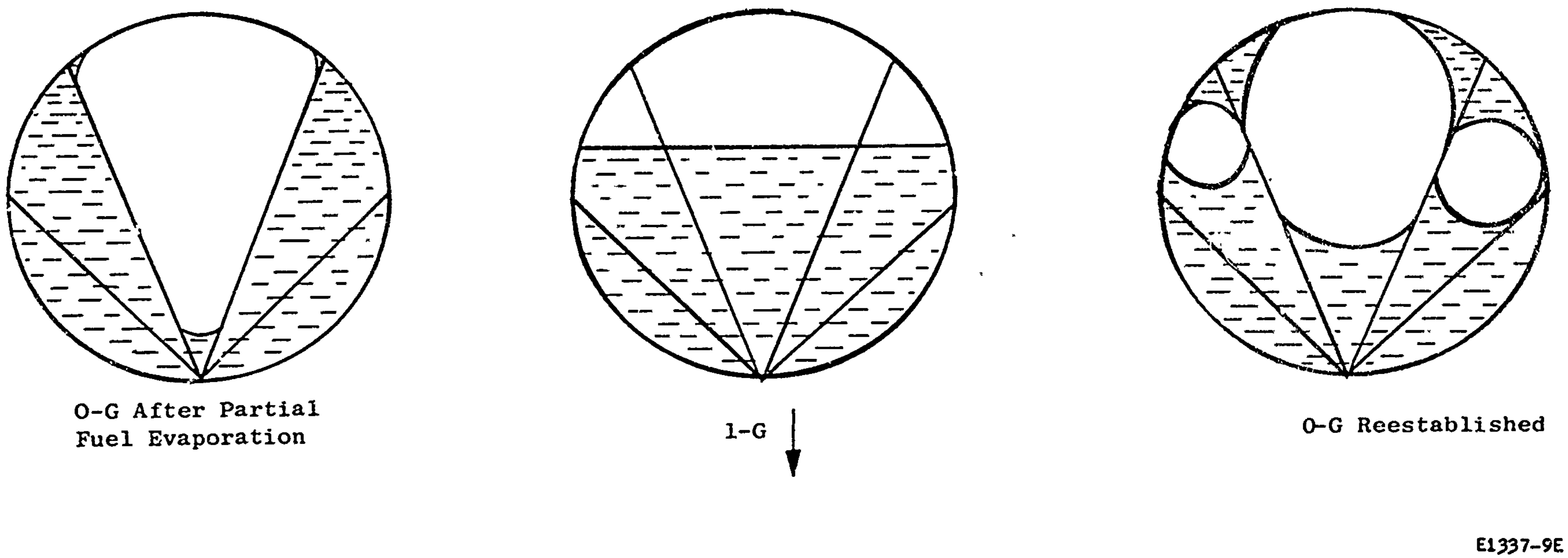

Figure 71. The Effect of Momentary Acceleration During Tank Drainage

In order to counteract any tendency toward undesired metastable configurations which result from necessary off-axis forces, the propellant outlet and the upper end of the central cone should be coated with a non-wetting material, such as teflon. Finally, some source of heat should be considered, such as the use of high emissivity coatings on the surface near the outlet, or even a small heater on the outlet.

5. FINAL TANK DESIGN SELECTION

The final mechanical design of the tank was based upon information procured from 3 sources:

The General Tank Design Criteria described in the preceeding paragraph.

An investigation to determine the most suitable material, as described in Appendix VI.

An analysis of strength, volume capacity, weight, and heat transfer, as described in Appendix VII.

The tank consisted of two 0.046 inch thick aluminum hemispheres which, when put together, formed a sphere 11 inches in diameter. A one-quarter inch aluminum fill tube was then welded into position on each hemisphere. Within the hemisphere designated as the "top", an assembly of four concentric cones of thin aluminum screening was fastened by welding. This assembly was oriented so that the base of each cone was directed towards the fill tube of that "top" hemisphere. The final weld was that to the girth, joining the two hemispheres. This is as shown on Figure 72.

As the tanks were being fabricated, the results of the KC-135 flights (as described in a following paragraph of this Section) were observed very closely. If indications had appeared that more surface tension forces were required to improve the liquid-vapor separation, plans were made to either add 3 additional cones to the existing nest of cones or to replace the existing screen cones with ones made of perforated metal. Both of these alternates were carried up to the point of procuring quotations from the vendor for reworking the parts that had been fabricated up to that point in time into one or the other of these alternate configurations. Fortunately, the flights indicated in sufficient time that the original mechanical design was suitable, so no further action was taken on the two alternates.

A total of five tanks were ordered from a selected vendor, the first being a qualification model (Figure 73). Production of the remaining four tanks (only two were actually delivered prior to stoppage of this work) was delayed until this model successfully passed vibration and g loading tests, which it did. These tests were conducted with 18.7 pounds of water occupying 85% of the tank's volume and pressurized with 150 psig of nitrogen gas. It was supported in its test fixture by a clamp and tension strap assembly that were a very close approach to what was visualized as flight design hardware (Figures 74 and 75). X-rays (Figure 76) made at the conclusion of

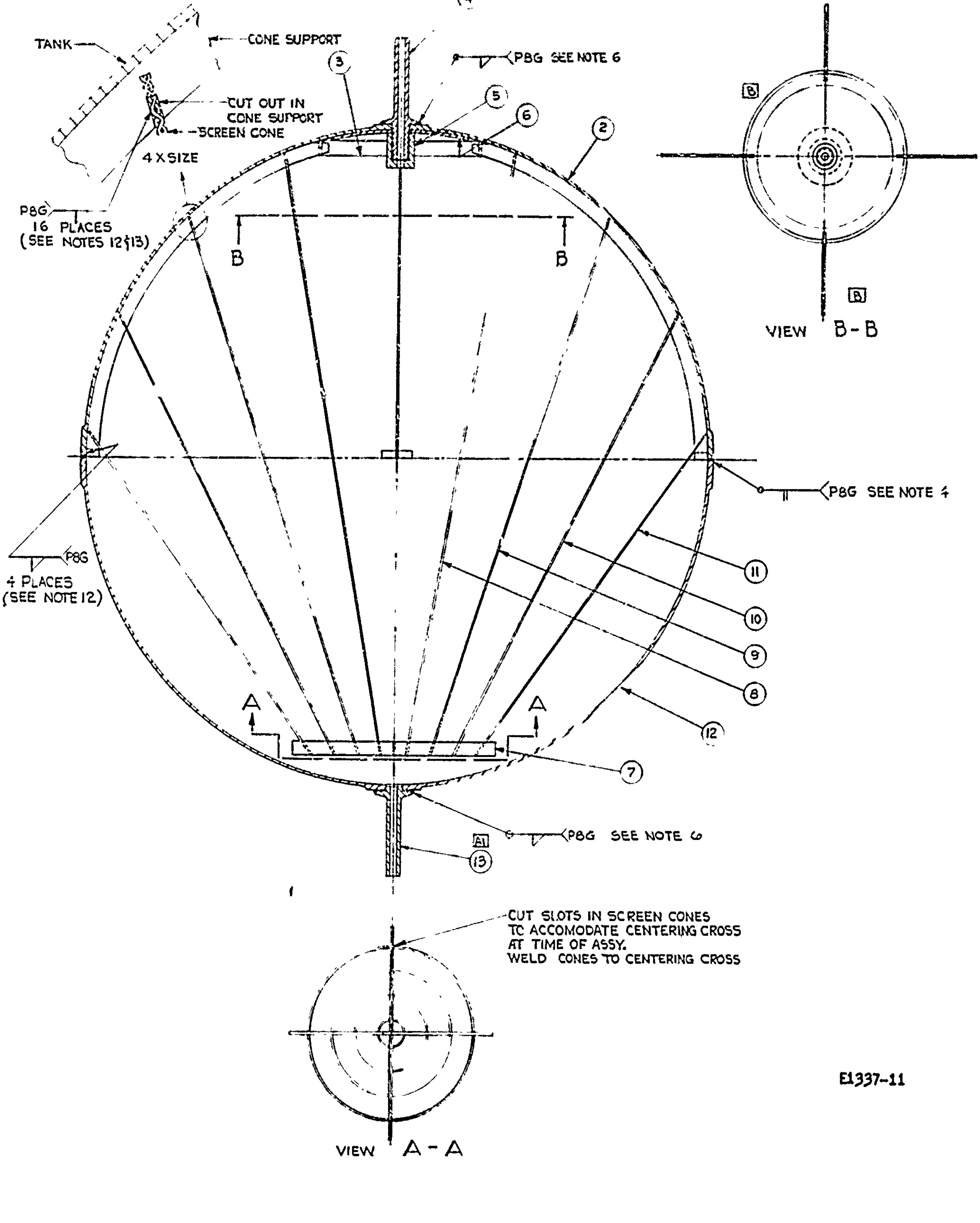

Figure 72. Ammonia Fuel Tank, Mechanical Assembly.

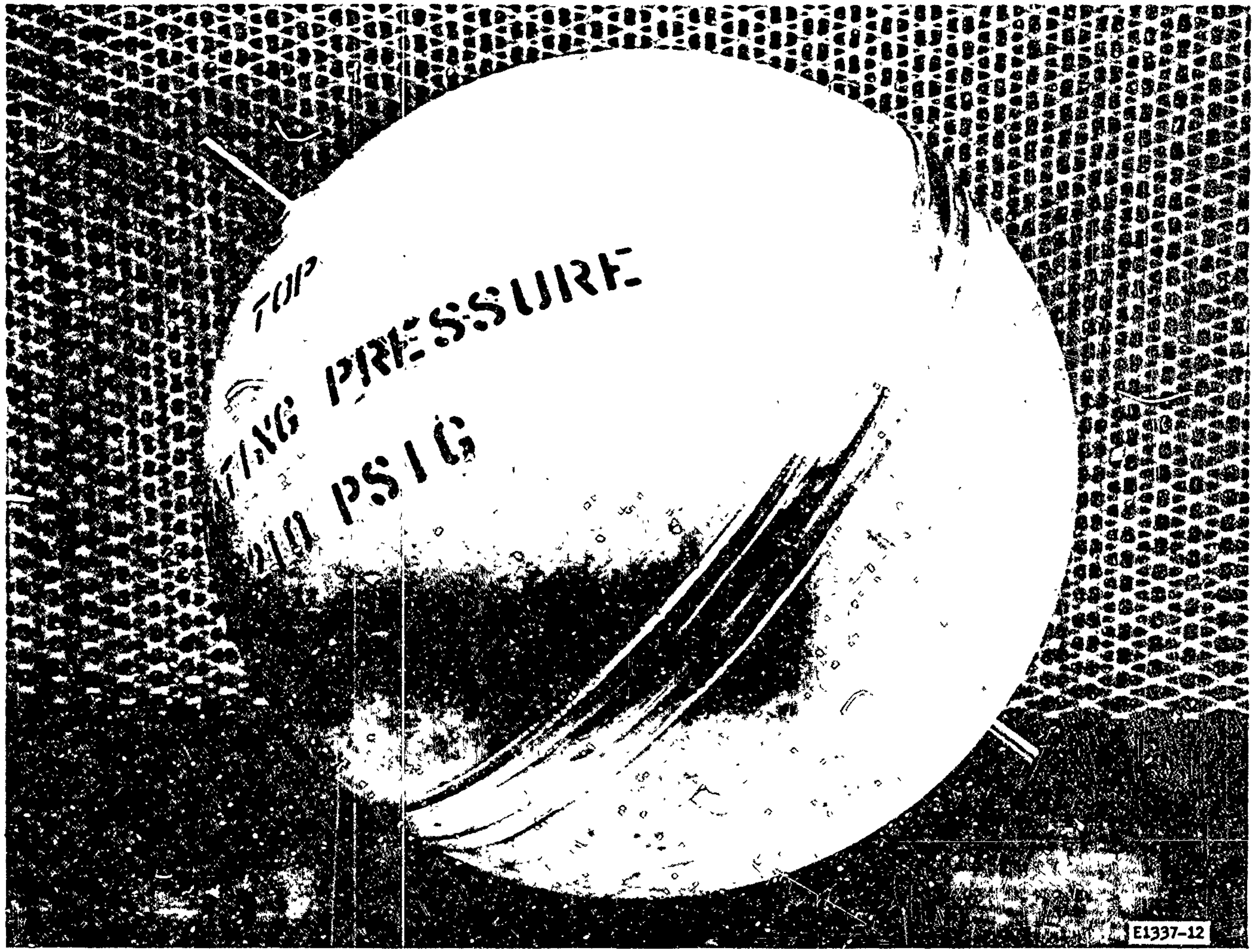

Figure 73. Ammonia Fuel Tank.

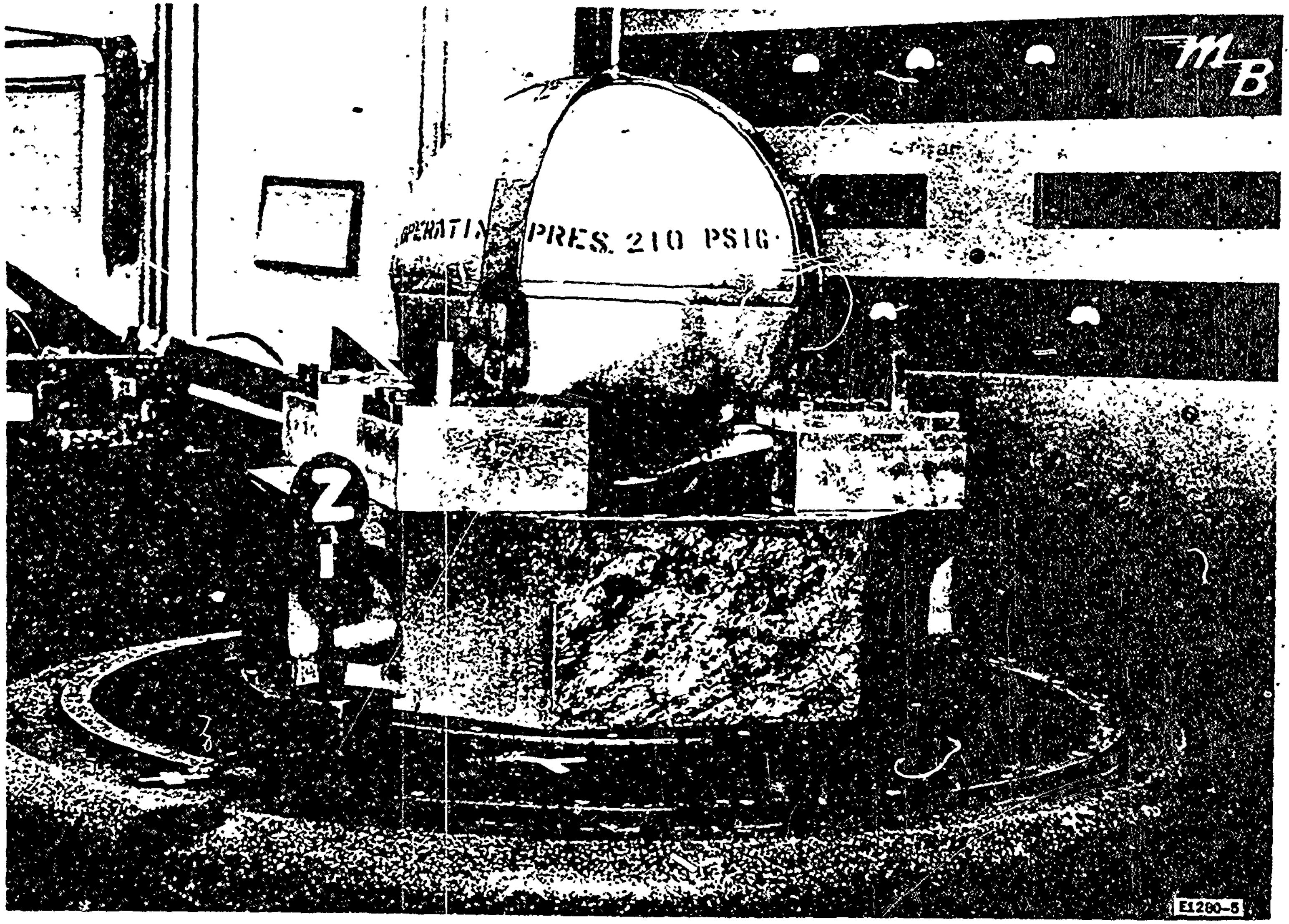

Figure 74. Vibration Test Portion of Propellant Storage Tank Qualification Tests.

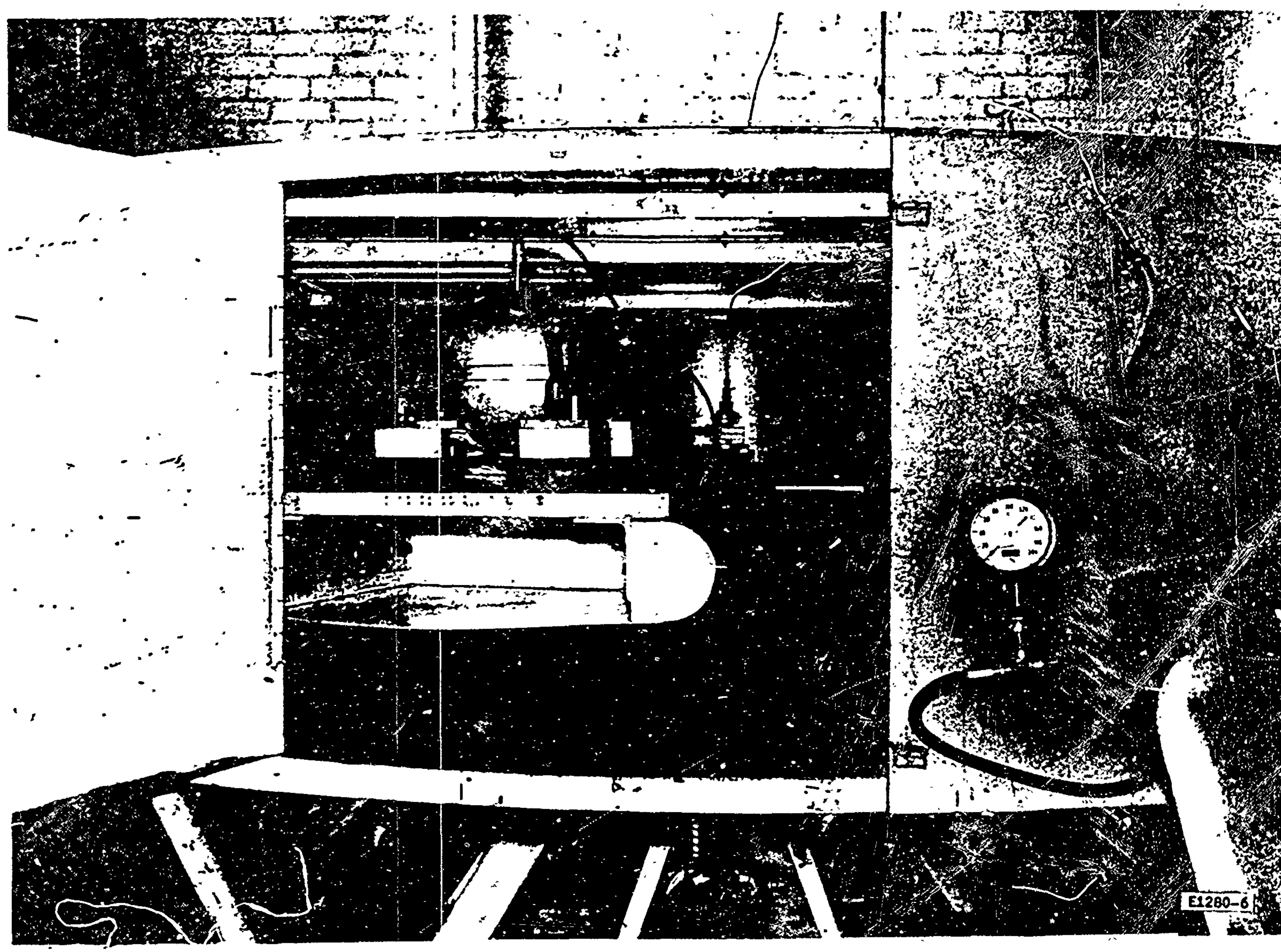

Figure 75. G-Loading Test Portion of Propellant Storage Tank Qualification Tests.

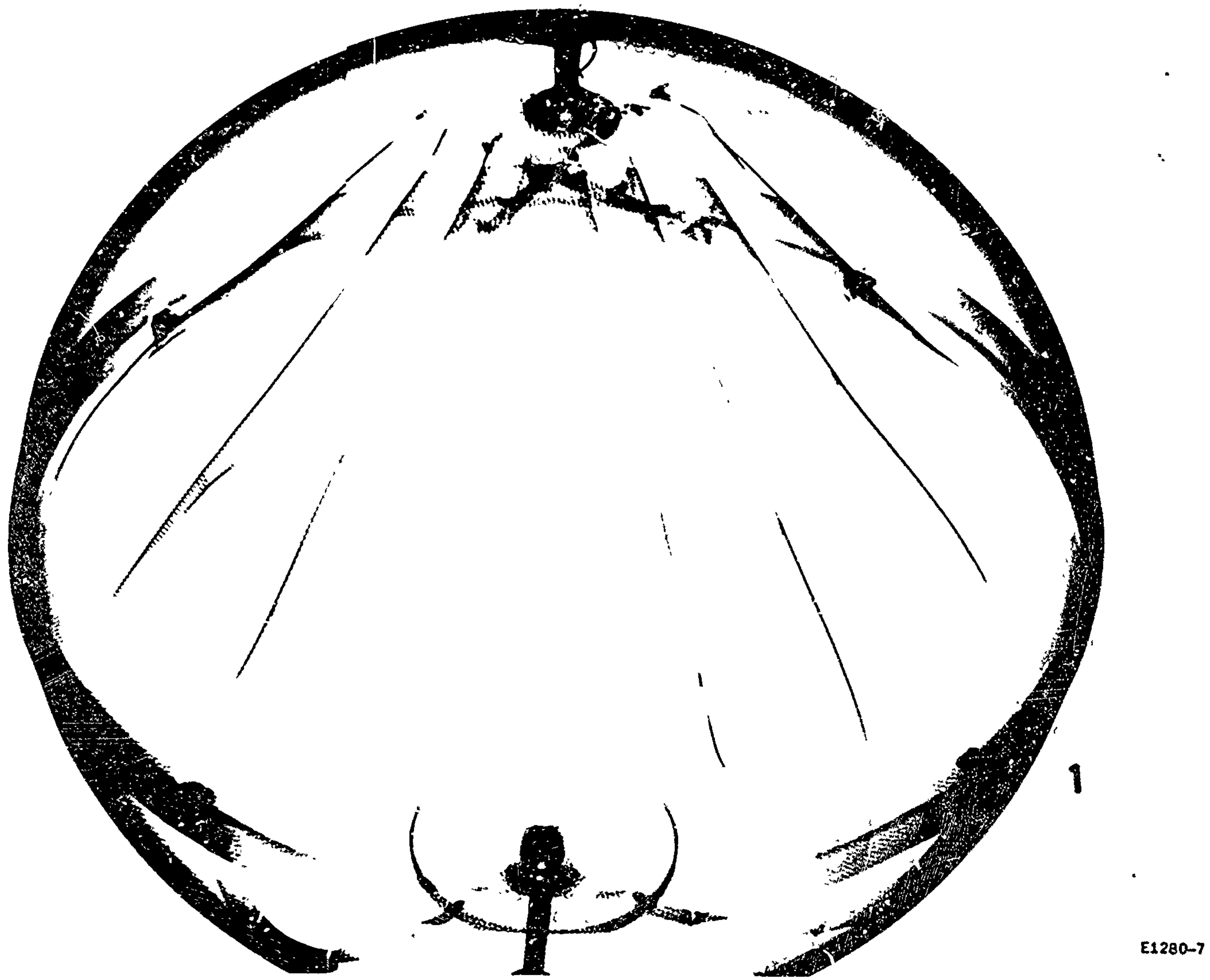

Figure 76. X-Radiograph of Propellant Storage Qualification Tank at Conclusion of Vibration and G-Loading Tests.

these tests showed no change in shape of the interior tank structure and no breakage at any weld. A detailed description of this test appears in the tank manufacturer's report. (Reference 1)

1. A final test on this qualification model tank consisted of subjecting it to an internal pressure of 440 psig (twice its maximum internal pressure rating) for a period of 10 minutes. Measurements of the outside diameter at several points before and after the pressure test showed no significant change in shape.

6. KC-135 TESTS

In order to test the capability of the propellant tank to perform as desired at zero gravity, the plastic models shown in Figure 77 were built and flown many times on KC-135 low gravity parabolic flights, while color motion pictures were taken of the action of the contained liquid. The spheres were mounted in the flight capsule shown in Figure 78. Figure 79 is a schematic of the arrangement within the capsule.

Many of the flights were turbulent, and the films taken during these flights are of no value. The combined footage of all the good flights was collected and edited into a single reel of about 900 feet of 16 mm film. Three copies of this film are kept on file, two at the Aeropropulsion Laboratory at Wright-Patterson Air Force Base, Ohio, and one at Nuclear Systems Programs, General Electric Company, Evendale, Ohio.

In general, the photography confirms the theoretical treatment given in an earlier section of this report, and the configurations shown in Figures 67 through 71 were attained.

The effect of sloshing and off axis accelerations can only be fully appreciated by viewing the actual motion of the liquid. Once a stable interface is established, whether it be the most stable or only a metastable one, it is not changed by mild disturbances over the short periods observed.

Violent sloshing or relatively high acceleration ($> 10^{-2}$g) completely overpowers any of the interfaces observed, and the new equ librium when zero gravity is reattained can be completely dissimilar.

7. CONCLUSION

In the absence of off axis thrust and sloshing, transition to zero gravity results in the predicted and desired liquid-vapor interface, and these configurations are stable against small disturbances.

Larger off-axis forces at any time can result in undesirable metastable equilibria, and the transition from the metastable configuration to that of lowest energy may be very slow, depending on the height of the energy barrier.

The geometry of the tank design was verified, but some driving force in addition to the surface energy must be used to establish the desired interface whenever significant off axis forces are encountered.

Figure 77. Plastic Models Of Fuel Tank For KC-135 Zero-G Tests.

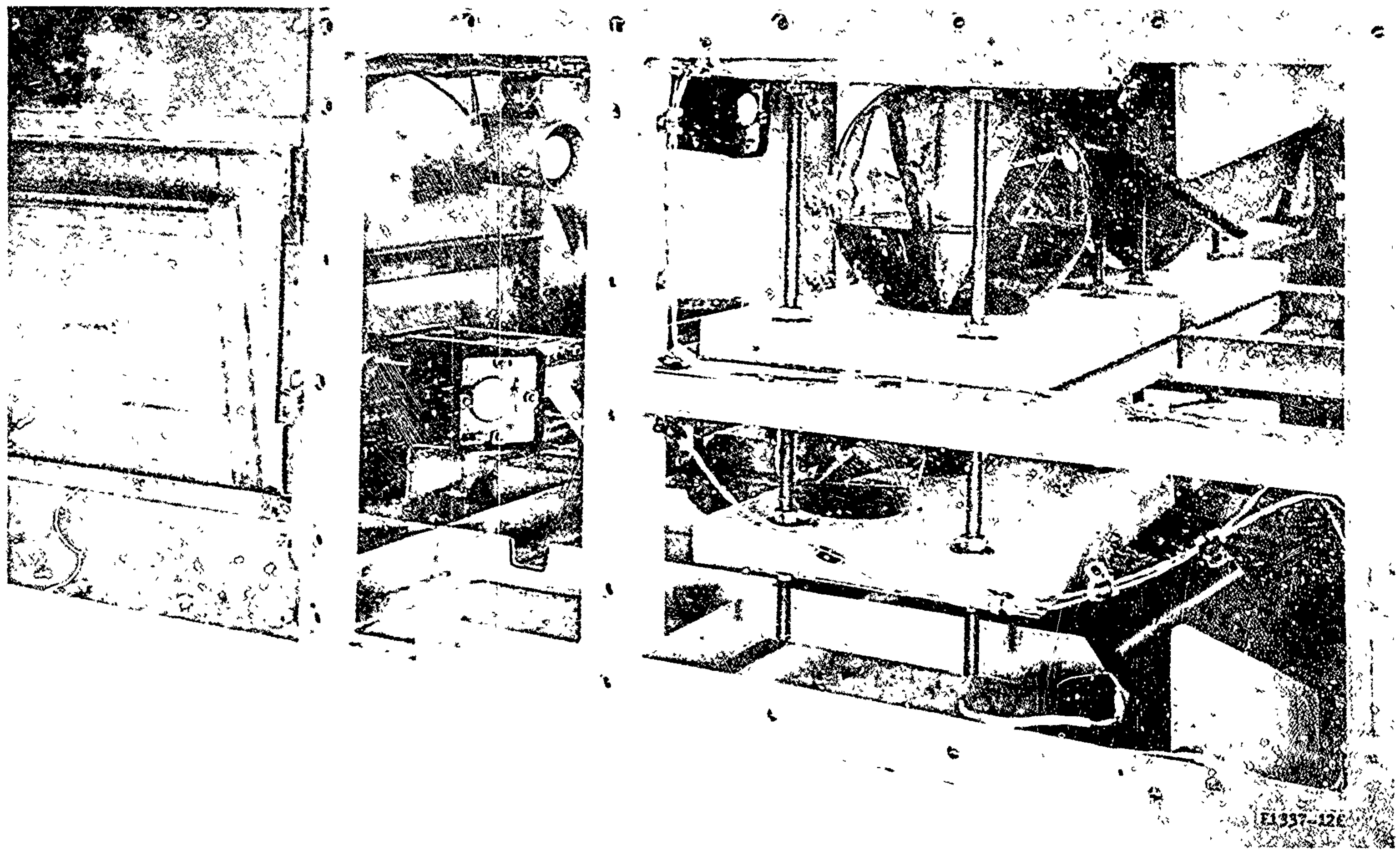

Figure 78. The Capsule Prepared For The KC-135 Flights Containing Initial Models Of The Fuel Tank.

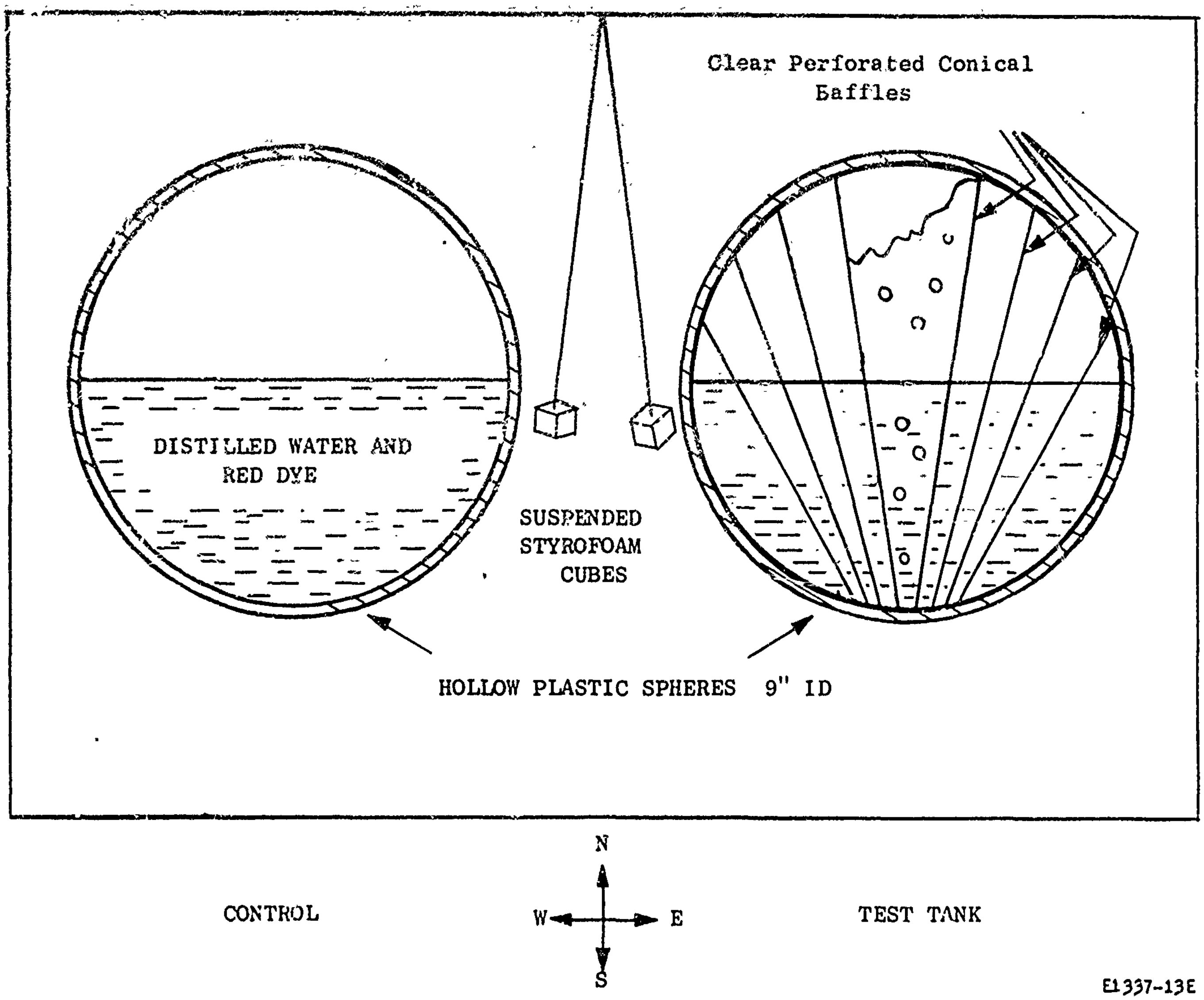

Figure 79. Inside the Flight Test Capsule

SECTION VI

AMMONIA GAS FLOW CONTROL

1. SCOPE

A schematic diagram showing the flow control and instrumentation components required for one propulsion system appears in Figure 1.

Upon definition of the system, the components were identified by written specifications. These documents described the design requirements, the performance requirements, the cleanliness requirements and the quality assurance requirements for the explosive valves, filters, pressure regulators, pressure transducers, and solenoid valves. The quality assurance requirement portion of the specification for each was actually a group of requirements, consisting of acceptance tests, review of the vendor's in-process quality control plan, and formulation of an acceptance test procedure. Suppliers of these components were selected on the basis of having produced similar units that were qualified for aerospace use as well as their manufacturing capability, delivery reputation, and total costs. Finally, each selected item was defined by a G.E. drawing showing its size and shape as well as other pertinent physical characteristics of the component. Photographs of certain of these components appear in Figures 80 through 83. The quantities required for each propulsion system and the quantities procured are shown in Table VI.

As each component was received, it was subjected to a strict incoming inspection process. If acceptance test results were a part of the shipment, the data would be checked for conformance to the procurement requirements.

2. SOLENOID VALVE AND PRESSURE REGULATOR QUALIFICATION TESTING

The particularly demanding requirements of performance for the solenoid valve and the pressure regulator posed a challenging procurement problem. Based upon the best information available at the time, designs and a supplier were selected. For performance reassurance, a representative solenoid valve (as well as a miniature version of that valve) and a representative pressure regulator were procured well in advance of delivery of the flight units and subjected to stringent qualification tests. The test procedures are described in Appendices VIII and IX. The conduct of the tests and the results are described in Appendices X and XI and Reference 2.

The results of the valve testing showed that both valves successfully passed the qualification tests. However, in the final selection, the standard size valve was chosen for flight hardware because it was more manufacturable.

The only difficulty experienced with the advance model of the pressure regulator had to do with a dampening mechanism, one designed to reduce "flutter" at high flow rates of gas. When this dampening

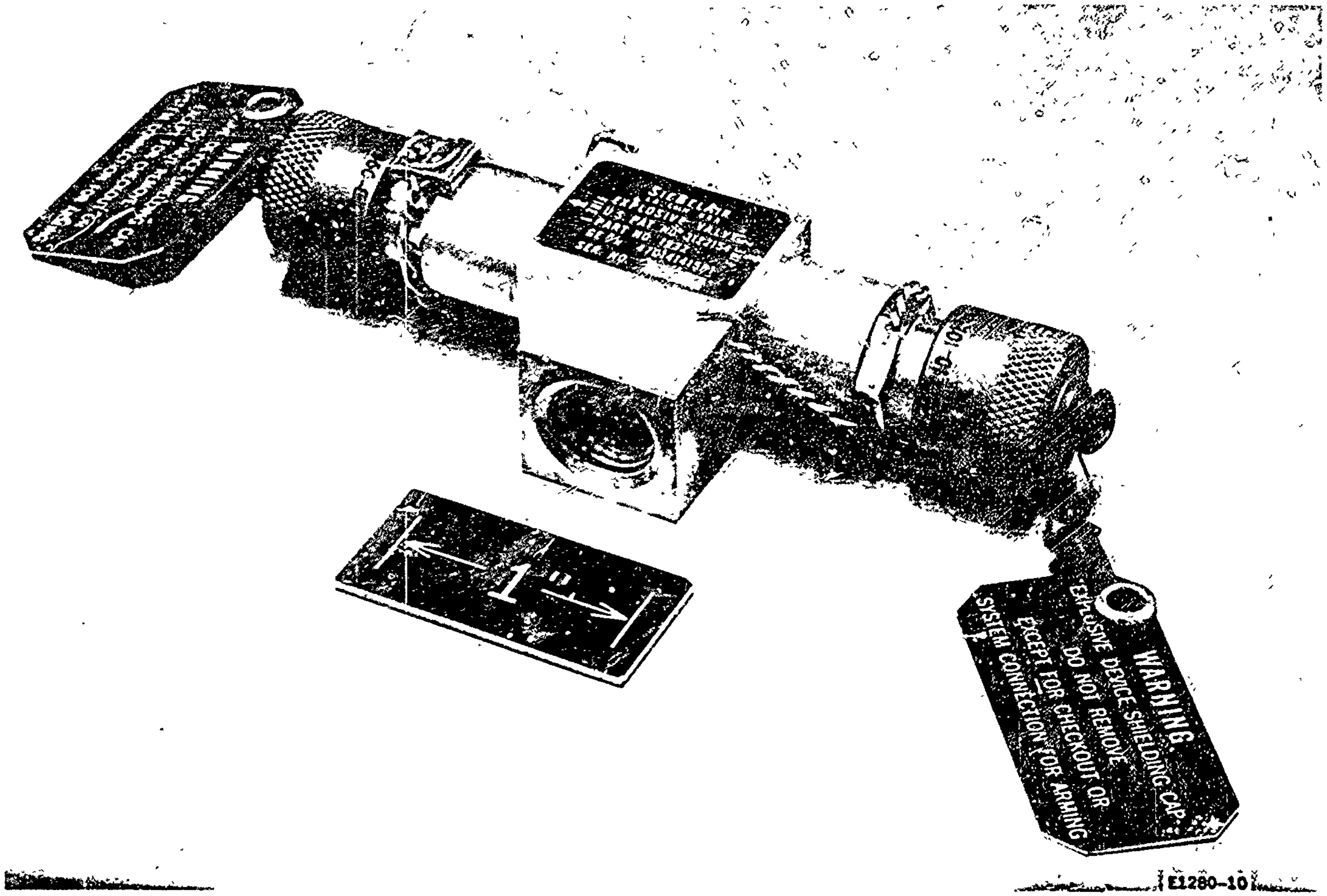

Figure 80. Normally Open/Normally Closed Explosive Valve.

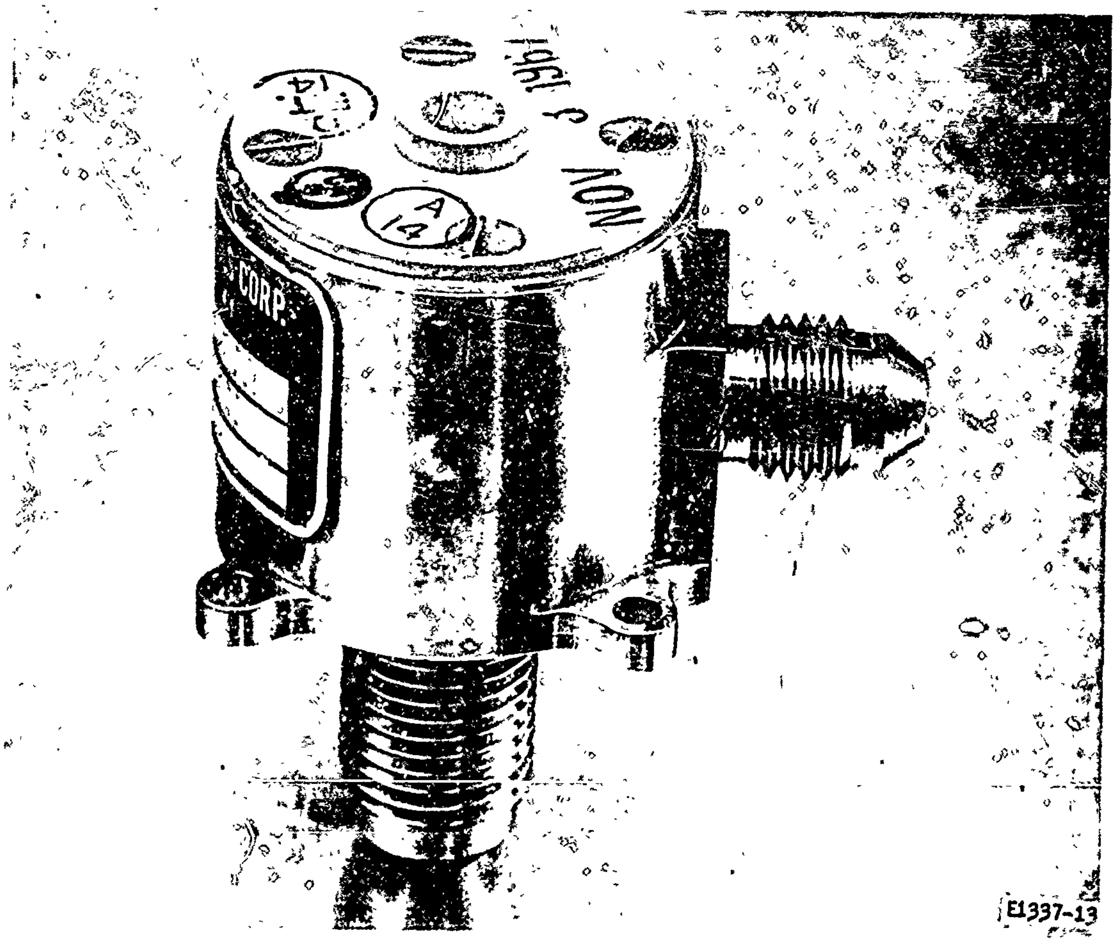

Figure 81. Gas Pressure Regulator.

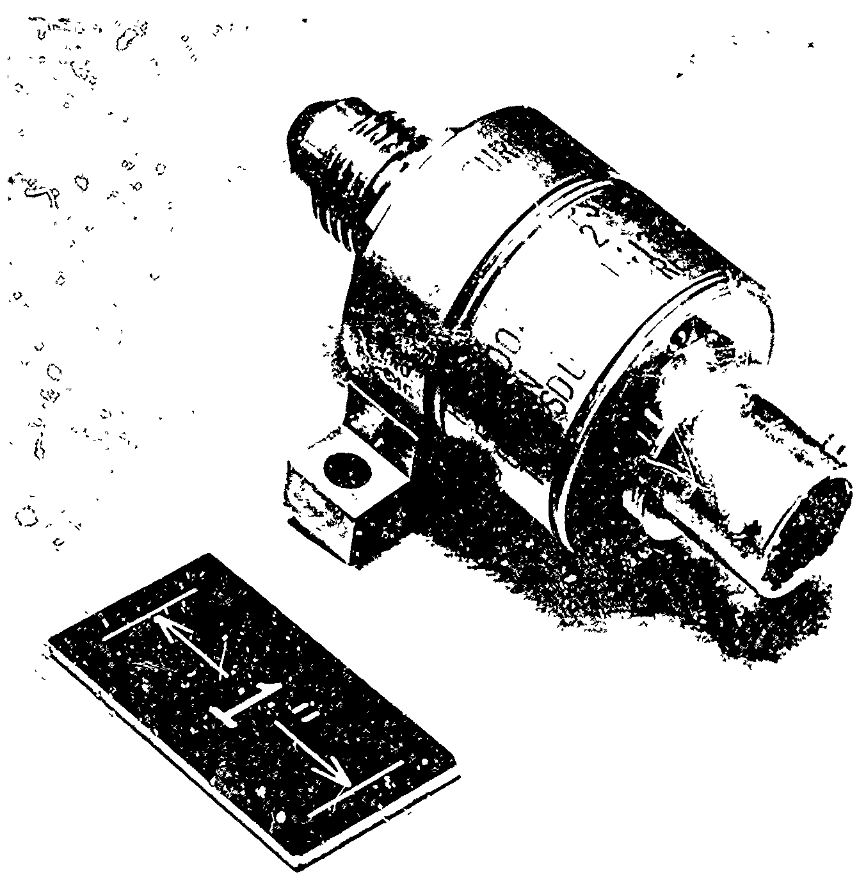

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Figure 82. Pressure Transducer.

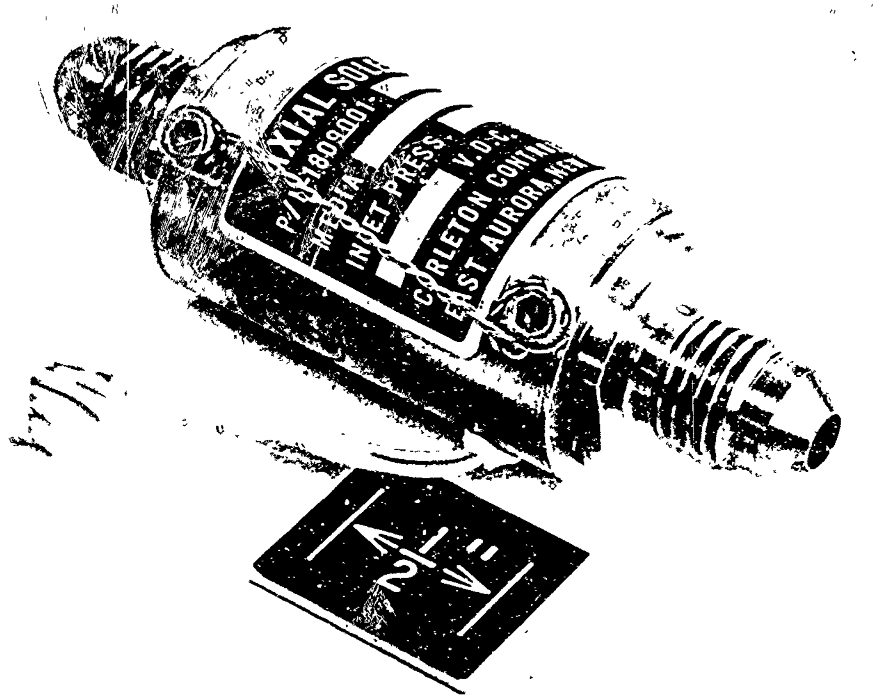

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Figure 83. Solenoid Valve.

TABLE VI

QUANTITIES REQUIRED AND PROCURED
FLOW CONTROL AND FLOW INSTRUMENTATION COMPONENTS

Item	G.E. Part No.	Quantity: Required Per Flight System	Quantity: Procured
Tank Pressure Transducer	47C141413P2	1	3
Explosive Valve	47C141415P1	3	14
Station Keeping System Pressure Regulator	47C141387P1	2	7
Station Changing System Pressure Regulator	47C141387P2	1	4
Station Keeping System Pressure Transducer	47C141413P1	3	7
Station Changing System Pressure Transducer	47C141413P1		
Pressure Switch	47B115388P1	6	15
Solenoid Valve	47B115407P1	12	42
Filter	47C141416P1	6	22

mechanism was removed from the regulator, its performance - particularly the regulated pressure and internal leakage during no-flow (lockup) conditions - became acceptable. The dampening mechanism was not built into the flight regulators delivered on this Program.

3. SPECIAL TESTS ON SOLENOID VALVES

Special tests were conducted on two of the flight solenoid valves, subjecting them to certain "severe-case" conditions. These consisted of exposure to liquid ammonia and to cycling with ammonia gas. The criterion for performance was determined to be a check for internal leakage. A final test was an investigation of the capability of the valves to close tightly after having been open for an extended period of time.

Although it was not intended that the valves were to be exposed to liquid ammonia (and the procurement specification defined the medium which the valve was intended to pass an "anhydrous gaseous ammonia") when built into the Microthruster System, they were set up in such a way that liquid ammonia was applied to their inlet end. This was checked by opening the valves and allowing the liquid ammonia to pass through and escape from the outlet end. Internal leak tests on the seat were made before the test and again after being subjected to the liquid ammonia for 40 hours. The results are shown on Table VII. At the pressure the valves were most likely to see (15 psia) there was no indication of any change in leak rate. However, when subjected to the very severe test pressure of 200 psia, one valve did start to leak, but not at a severe rate.

The same two valves were then arranged as indicated in Figure 84 to subject them to a cycling test with gaseous ammonia. This test configuration was unique in that it provided a check on whether or not the valves actually opened when their coils were energized. The electrical schematic diagram, Figure 85, shows that an electrically-operated counter was energized when the valve coils were energized. A pressure switch in the downstream line of each valve was actuated by the gas being released through that valve. Each pressure switch operated its own independent electrical counter. Thus a check - by comparing the counter readings - could be made whether or not the valves really opened when energized. It was worthy of noting that at the end of 51,933 cycles of operation, all the counters read the same. Leakage through the valves was checked after approximately each 10,000 cycles of operation for a total of approximately 50,000 cycles. The results are shown in Table VIII. It is interesting to note that at the end of the test and at the pressure

TABLE VII

INTERNAL LEAKAGE TEST ON SOLENOID VALVES
(LIQUID AMMONIA, AT ROOM AMBIENT)

VALVE CONDITION	VALVE S/N 1		VALVE S/N 40	
	Inlet Pressure PSIA	Leak Rate, Std CC/Sec Air at Indicated Inlet Pres.	Inlet Pressure PSIA	Leak Rate, Std CC/Sec Air at Indicated Inlet Pres.
Vendor guarantees that, when shipped to GE, leak rate will not exceed:	215	0.37×10^{-6}	215	0.37×10^{-6}
Actual leak rate when valve was shipped by manufacturer:	5 15 200	1.79×10^{-8} 2.21×10^{-8} 1.2×10^{-7}	5 15 200	1.76×10^{-9} 1.98×10^{-9} 0.59×10^{-7}
Leak rate when put on test at GE:	30 215	No Indication* 3.4×10^{-7}	30 215	No Indication* 5.0×10^{-7}
Leak rate after 40 hours' exposure to room ambient liquid ammonia:	30 75 215	No Indication* - 3.4×10^{-7}	30 75 215	No Indication* 1.0×10^{-6} -

1. Leak rates measured by manufacturer with Veeco Model MS-9 (actual max. sensitivity 1.87×10^{-9} Std. cc/sec air)

2. Leak rates measured by G. E. with GE Mass Spectrometer Model M60 (max. sensitivity 5.0×10^{-10} std. cc/sec air)

3. All tests conducted with valves exposed to atmospheric pressure.

4. All leak rates measured with one milliamp DC passing through valve coil.

5. To convert leak rates to Std. cc/sec helium, divide Std. cc/sec air value by 0.374.

6. Valves tested were: DODGE-2 Flight Quality

* Less than 5×10^{-10} std. cc/sec air

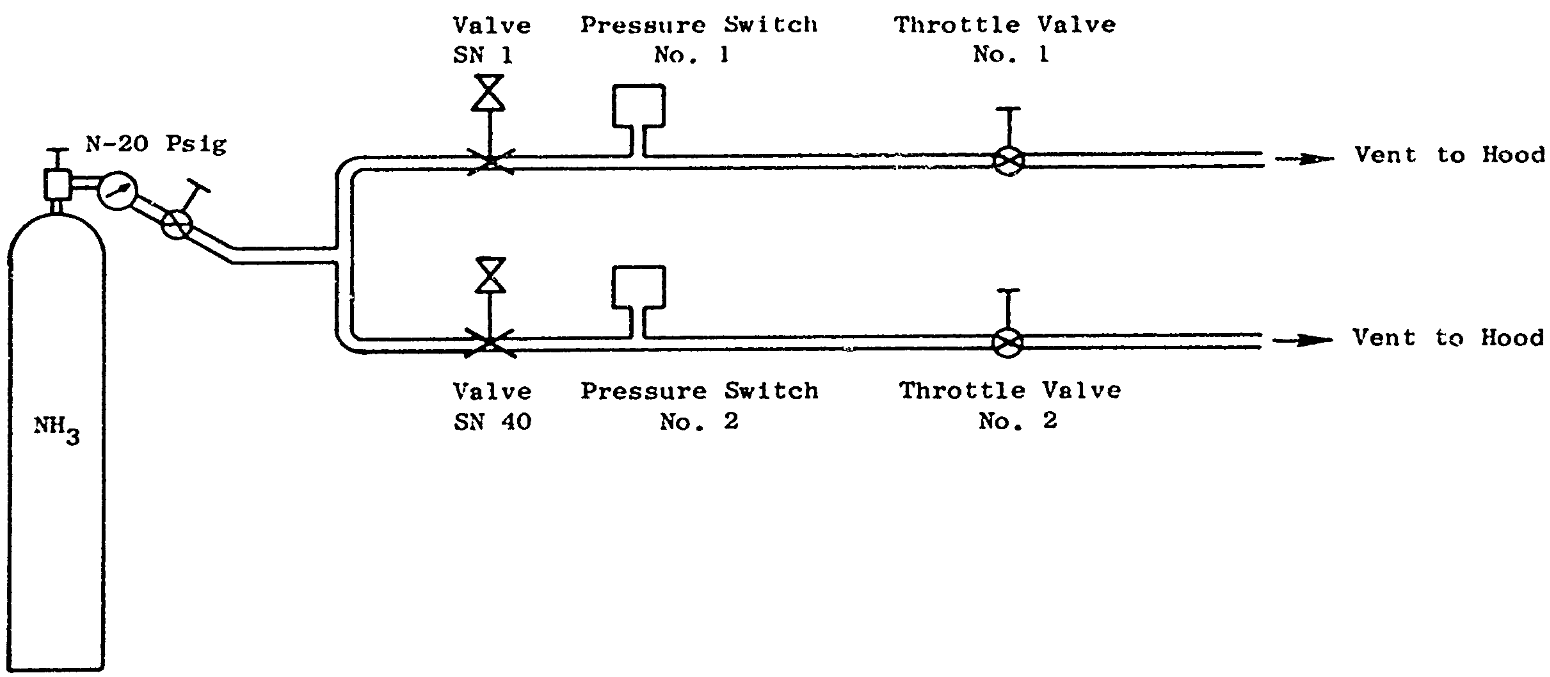
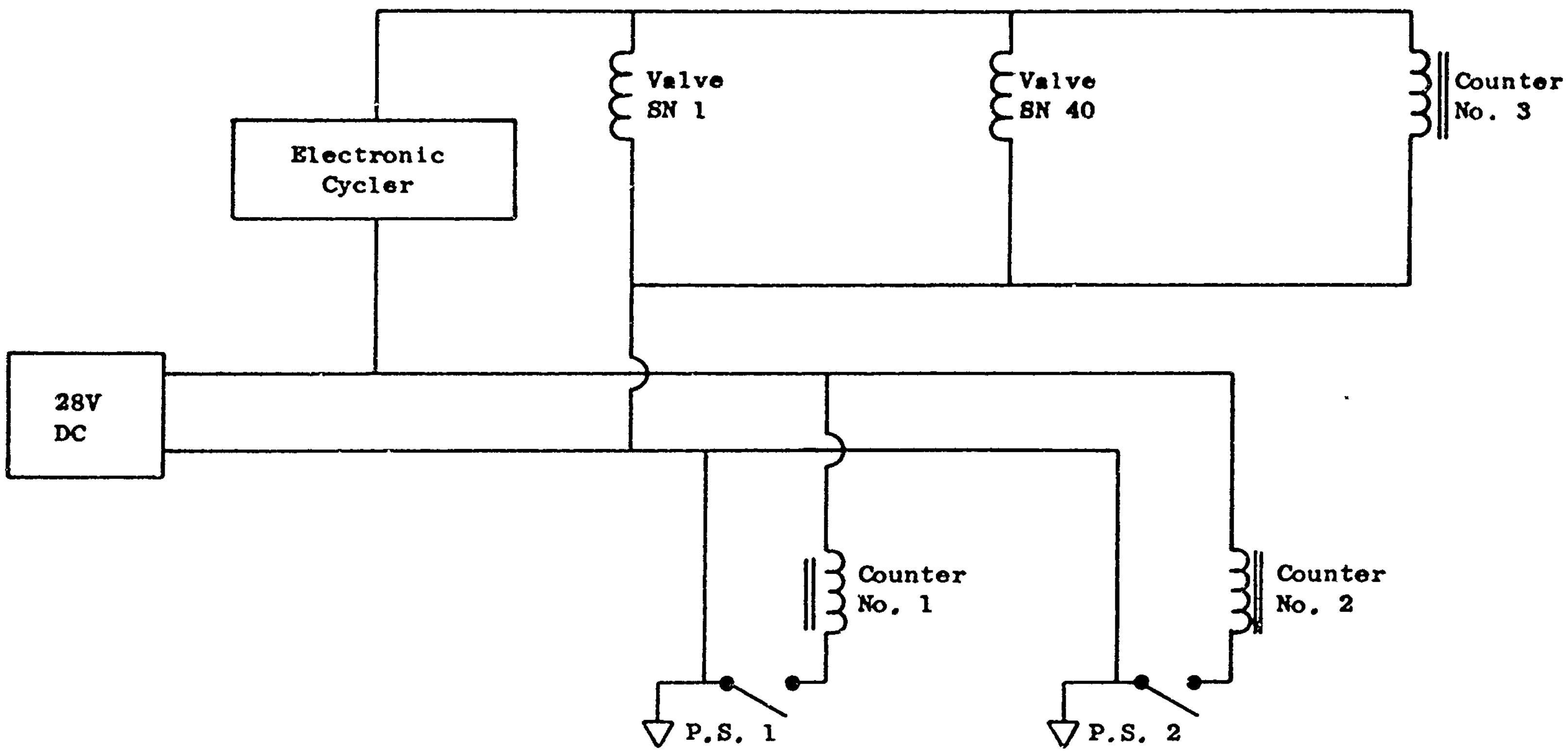


Figure 84. Pneumatic Diagram, Cycling Test for Solenoid Valves.



Figures 85. Electrical Schematic Diagram, Cycling Test for Solenoid Valves.

TABLE VIII

INTERNAL LEAKAGE TEST, SOLENOID VALVES

(CYCLED WITH AMMONIA GAS)

VALVE CONDITION	VALVE S/N 1		VALVE S/N 40	
	Inlet Pressure PSIA	Leak Rate, Std CC/Sec Air at Indicated Inlet Pres.	Inlet Pressure PSIA	Leak Rate, Std CC/Sec Air at Indicated Inlet Pres.
Leak rate after 40 hours exposure to room ambient liquid ammonia (just prior to cycling test)	30 75 215	No Indication* - 3.4×10^{-7}	30 75 215	No Indication* 1.0×10^{-6} -
After 11,000 cycles of operation	15 215	No Indication* 1.1×10^{-7}	15 215	No Indication* 1.5×10^{-7}
After 23,800 cycles of operation	15 200 215	2.0×10^{-8} - 1.8×10^{-7}	15 200 215	1.7×10^{-7} 1.0×10^{-6} -
After 32,700 cycles of operation	15 115 165 200	5.2×10^{-8} - - 5.6×10^{-7}	15 115 165 200	6.0×10^{-9} 1.1×10^{-7} 2.5×10^{-7} 3.4×10^{-7}
After 43,400 cycles of operation	15 160 200	6.1×10^{-9} 1.0×10^{-6} -	15 160 200	8.4×10^{-9} - 3.3×10^{-7}
After 51,900 cycles of operation	15 65 95 115 195 200	7.8×10^{-9} 6.2×10^{-7} 1.8×10^{-6} 3.9×10^{-6} 6.5×10^{-6} 8.4×10^{-6}	15 65 95 115 195 200	4.4×10^{-9} - - - - 1.7×10^{-7}

1. Leak rates measured by G. E. with GE Mass Spectrometer Model M60 (Max. sensitivity 5.0×10^{-10} std cc/Sec Air)
2. All tests conducted with valves exposed to atmospheric pressure.
3. All leak rates measured with one Milliamp DC passing through Valve Coil.
4. To convert Leak Rates to Std. cc/Sec Helium, divide Std. cc/Sec air value by 0.374.
5. Valves tested were: DODGE-2 Flight Quality,
6. Valve Cycling Rate: One second open, six seconds closed)

* Less than 5×10^{-10} Std cc/Sec Air

(15 psia) that the valves were most likely to see, the leak rate of each was much less than the leak rate of the valves as received from the vendor (Table VII). However, when subjected to a very severe test pressure of 200 PSIA, one valve indicated a leak rate of approximately one decade worse than the design value of 1.0×10^{-6} Std cc/sec helium and the other showed an improved leak rate.

It was apparent from these limited tests of a very small group of samples that it is unlikely that liquid ammonia when applied to the Neoprene seat of the valve for limited intervals of time will cause catastrophic leaks through the valve. It was also apparent that when assembled into the Microthruster System, operating the valve with gaseous ammonia would not materially increase its leak rate.

The last of this group of special tests consisted of energizing them for one week while installed in a chamber evacuated to 5×10^{-6} mm hg and passing ammonia gas. Leak rates were measured before the valves were opened and then again after they closed. The results are shown on Table IX. The conclusions drawn in the above paragraph still hold true.

TABLE IX

INTERNAL LEAKAGE TEST, SOLENOID VALVES

Long-Time Open Test

In Vacuum Chamber at 5×10^{-6} mmhg

Valve Condition	Valve S/N 1 Inlet Pressure PSIA	Valve S/N 1 Leak Rate, Std cc/Sec Air at Indicated Inlet Pres.	Valve S/N 40 Inlet Pressure PSIA	Valve S/N 40 Leak Rate, Std cc/Sec Air at Indicated Inlet Pres.
Prior to being energized	15	7.8×10^{-9}	15	4.4×10^{-9}
	65	6.2×10^{-7}	65	-
	95	1.8×10^{-6}	95	-
	115	3.9×10^{-6}	115	-
	195	6.5×10^{-6}	195	-
	200	8.4×10^{-6}	200	1.7×10^{-7}
After 1 week open	15	3.4×10^{-9}	15	3.9×10^{-9}
		6.3×10^{-8}	40	1.4×10^{-7}
		2.2×10^{-7}	80	1.4×10^{-7}
		6.4×10^{-7}	120	2.5×10^{-7}
		9.5×10^{-7}	160	2.2×10^{-6}
		2.4×10^{-6}	200	8.4×10^{-6}*

1. Leak rates measured by G. E. with GE Mass Spectrometer Model M-60 (Max. sensitivity 5.0×10^{-10} Std cc/Sec Air).

2. All leak rates measured with one milliamp DC passing through Valve Coil.

3. To convert Leak Rates to Std. cc/Sec Helium, divide Std. cc/Sec air value by 0.374.

4. Valves tested were: DODGE-2 Flight Quality.

* Maximum measurable leak with M-60.

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SECTION VII

POWER AND SIGNAL CONDITIONER

1. GENERAL DESCRIPTION

The power and signal conditioner was to convert the electrical power available on the satellite into the forms required by the resistance jet microthruster system. In addition, electronic circuitry was necessary to condition the various instrumentation signals into a form acceptable by the vehicle telemetry system for ground transmission. All circuits were to be constructed from components suitable for space operation. Verification of the capability of the power and signal conditioning package to sustain the launch into orbit was to be made by subjecting the prototype unit to flight qualification tests equivalent to those conducted on the thruster systems. Both of the final units were to be acceptance tested before delivery was made. The design and development of this electronics package was based upon the following nominal constraints:

a. Total Input Power - $<$ 32 watts at nominal input voltage

b. Total Weight - $<$ 5 lbs.

c. Total Volume - $<$ 150 in^3

d. Power Conversion Efficiency - $>$ 75% at nominal input voltage while supplying power to one thruster heater, 2 pairs of solenoid valves, and associated instrumentation.

These requirements were based upon an input power of the form and frequency specified in Figure 86. Input voltage was to be either 28 volts, or 40 volts peak to peak, with a variation of +20% to -10%. The Johns Hopkins University "Propulsion System Specification" (Ref. 3) was used as a design guide for power and signal characteristics.

- Power System Restraints

The power and signal conditioner was required to protect the power system from all overload failures which could develop within this device.

The satellite power system was to have the following characteristics:

a-c voltage:	40V $^{+20\%}_{-10\%}$ peak-to-peak
frequency:	976.41 c.p.s.
frequency stability:	$1:10^{10}$/day
a-c wave shape of all inverters:	Per Figure 86

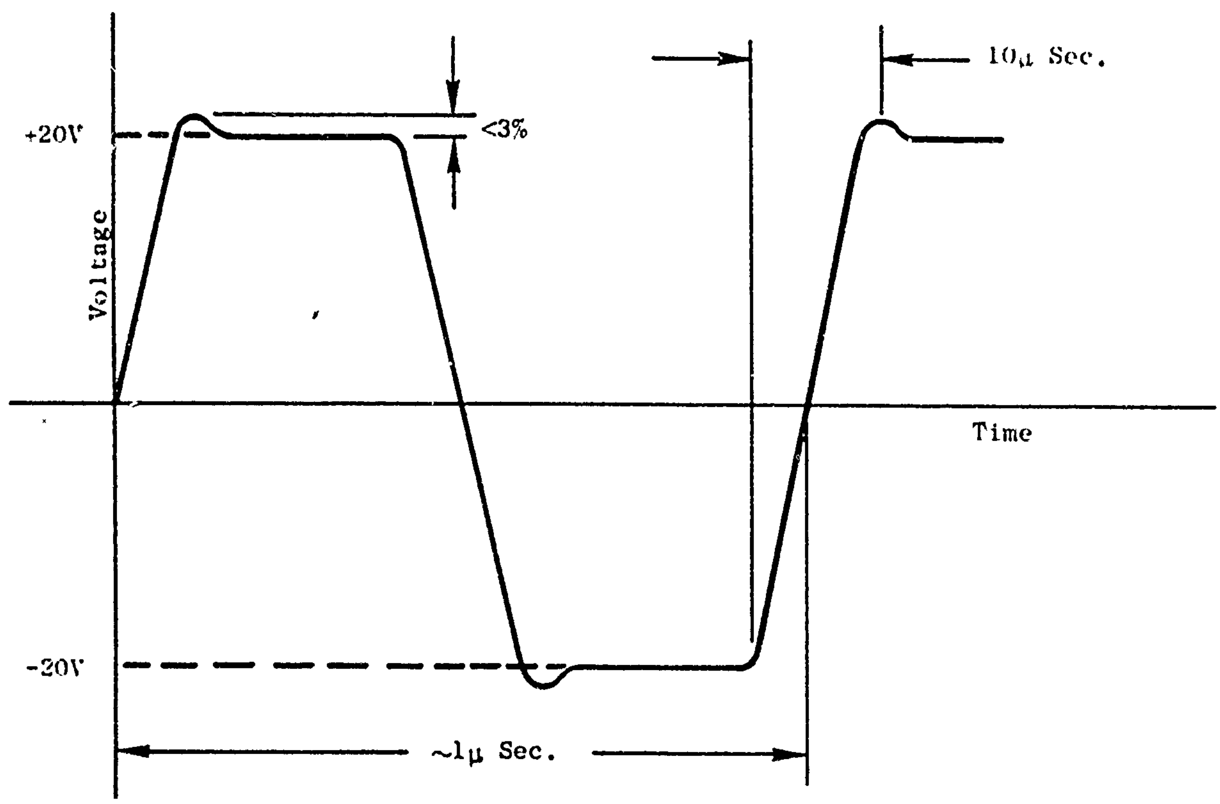

Figure 86. Input Voltage Waveform

In addition to the above, the power and signal conditioner was required to transform, rectify, and filter the a-c power where necessary to operate solenoids, relays, and other required components.

It should also be noted that both sides of the a-c input line to the thrusters had to be isolated from ground potential.

- Command and Telemetry System Requirements

Command System

The satellite command system normally provided DPDT, magnetic latching relays which were rated at 2 amperes per contact.

Telemetry

The telemetry analog-to-digital converter operated over an input voltage range of $\pm$ 0.250 volts with a resolution of 1 part in 256. Signal voltages greater than this were to be attenuated by the satellite circuitry. Voltages less than this would result in less than full scale readings. In no case was the system to provide TM input voltage the magnitude of which exceeded 0.30 volts after attenuation. The source impedance was not to exceed 5,000 ohms. Tell-tale (flag) signals were to be less than 0.5 volts for a "1" state and 0.4 milliamps at +5 volts or more for a "0" state.

2. DESCRIPTION OF THE OPERATION OF EACH CIRCUIT

- Thruster Heater Circuit

The thruster heaters, rated 20 watts, are supplied by a current-limited control which provides approximately 11 to 14 volts at the heater terminals. The circuit in Figure 87 consists of a step-down transformer, power switching relay (located in the command module) a full-wave rectifier, and adjustable current reference. Several voltage taps are available in the secondary winding to match the heater voltage best. This secondary AC voltage is rectified by a full wave rectifier circuit. Each diode used has about 0.7 volt drop (Unitrode UT 8110). The rectified voltage is then applied to a current regulator.

The mathematical model of constant current regulator is shown on Figure 88. The current through the heater is measured by the voltage drop across the series resistor R. This voltage is compared to the reference voltage obtained from the Zener diode. The error signal causes a current through the base of the transistor, which regulates the voltage drop across the transistor and hence the voltage applied to the heater.

The operation of the basic circuit is described by the equations below:

$$V_R = V_Z - V_{BE}$$

$$I_R = I_B + I_C \qquad I_B = I_R - I_C$$

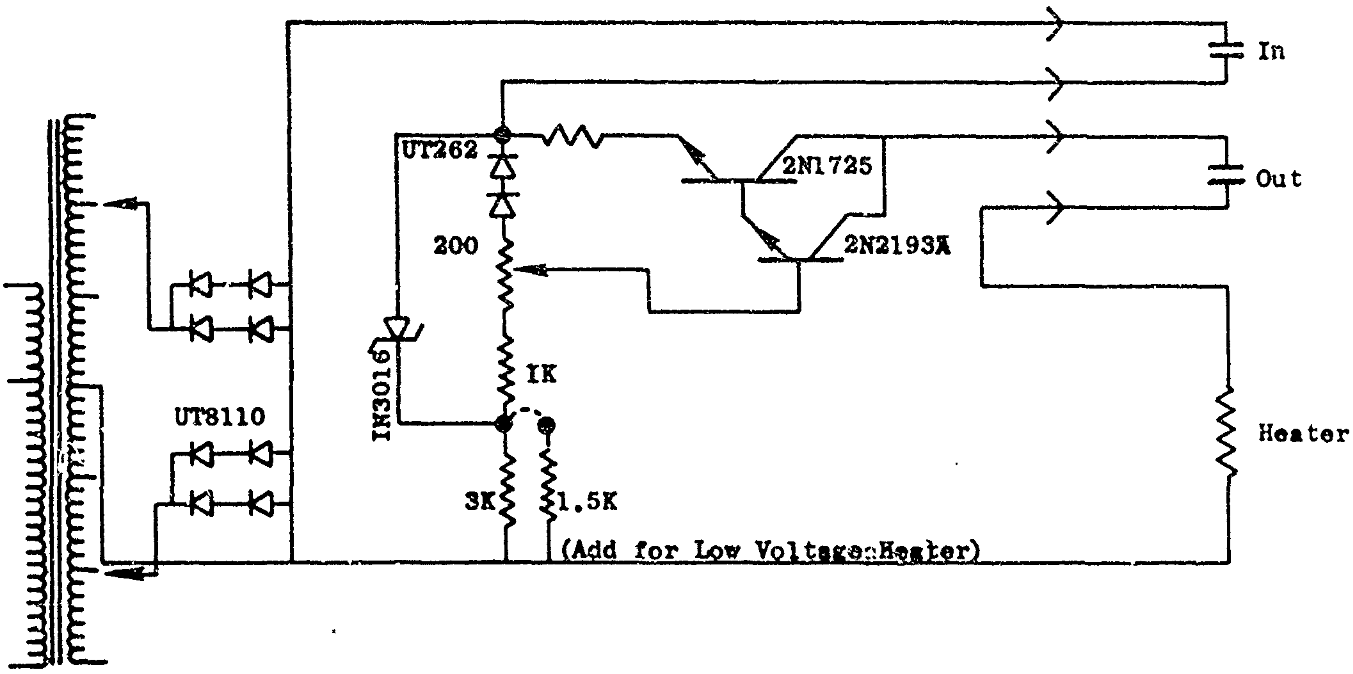

Figure 87. Heater Circuit.

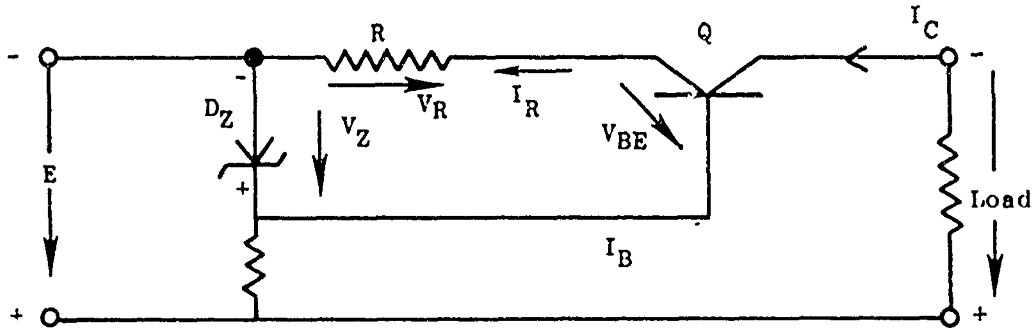

Figure 88. Constant Current Regulator.

$$I_C = I_B h_{FE}$$

$$I_C = I_R h_{FE} - I_C h_{FE}$$

$$I_C = \frac{I_R h_{FE}}{1 + h_{FE}} \qquad \text{but } I_R = \frac{V_Z - V_{BE}}{R}$$

$$I_C = \frac{(V_Z - V_{BE}) h_{FE}}{(1 + h_{FE})\ R} \qquad \text{since } h_{FE} \gg 1$$

$$I_C = \frac{V_Z - V_{BE}}{R}$$

The actual heater operating circuit is somewhat different than the mathematical model, and it was shown in Figure 87. Instead of the single transistor (in mathematical model) a super-alpha connection is used for higher gain, and the reference voltage is established in a manner which provides some adjustment. The two diodes used in the reference voltage circuit provide temperature stabilization.

- Solenoid Valve

Valve characteristics:	Pull in voltage	18 VDC maximum
	Dropout voltage	6 VDC maximum
	Power at 28 volts	1 watt

The circuit which is used to operate the solenoid valves is a voltage-doubler, as shown in Figure 89.

C_2 is selected such that the time constant of the valve resistance and C_2 is long enough, and that the voltage remains sufficiently high until the valve is actuated. The valve of C_1 determines the steady state current for the valve.

While the switch SW is open the capacitor C_2 charges up to about twice the value of the input voltage Vac_{in}. At the time of closing SW the capacitor C_2 is supplying most of the energy for opening the valve. As the voltage drops across C_2 due to the valve resistance, C_2 is trying to supply energy for both C_2 and the solenoid valve. However, its value is such that it limits the current to a value which corresponds to a valve voltage somewhat higher than the dropout voltage.

The actual circuit for operating two solenoid valves is given in Figure 90.

To increase reliability it will be noted that the rectifier diodes are used in "quads". Also, there are two series capacitors to improve reliability against shorts in the capacitors. The two series diodes across each solenoid valve are "flywheel" diodes.

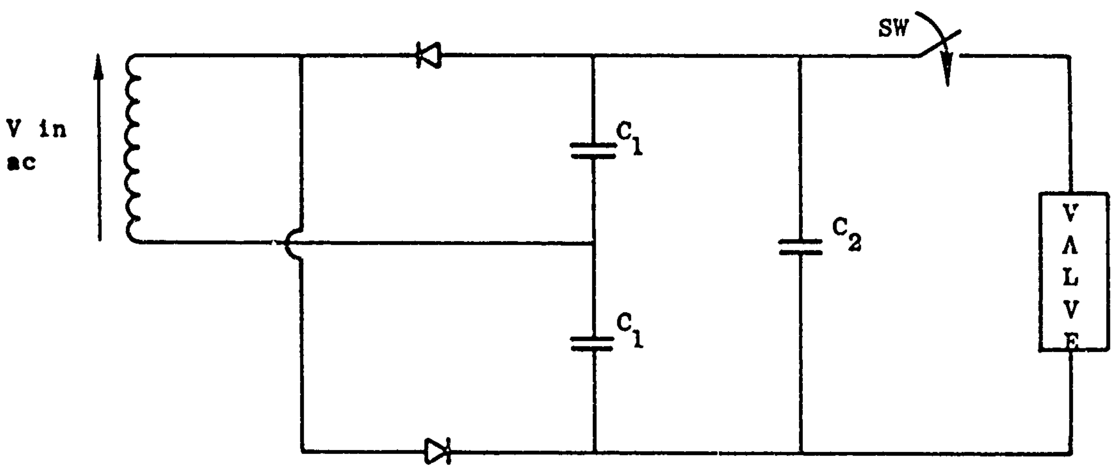

Figure 89. Voltage Doubler Circuit.

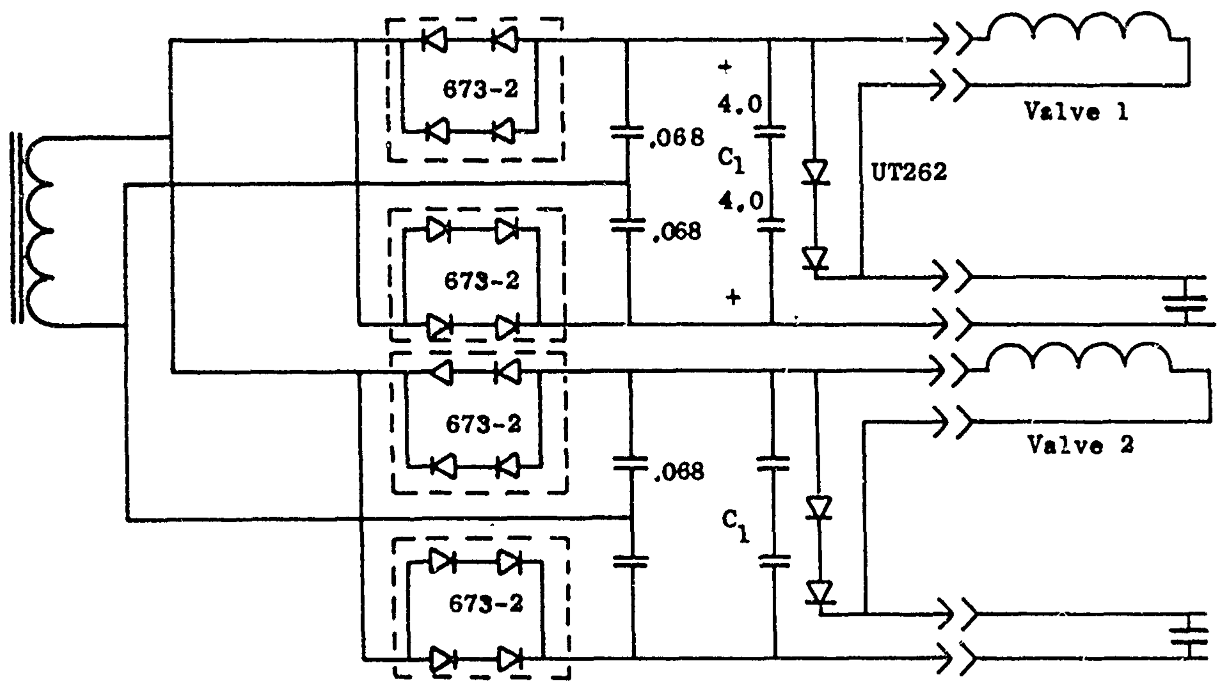

Figure 90. Solenoid Valve Circuit.

• Telltale Signal Indicator

The function of this circuit is to indicate whether the contacts of the pressure switch are open or closed. A zero volt output indicates a closed contact and a five volt output indicates an open contact. The circuit for three pressure switches is shown in Figure 91. The value of R is selected such that the telemetry system can draw 0.4 ma at 5 volts. A conservative choice of resistor valve is 10 K ohms with an output DC voltage from the rectifiers of 10 volts.

• Temperature Amplifiers

The purpose of this circuit is to convert the low level thermocouple signal into a signal of $\pm$ 0.25 volt for telemetry.

Operation: Input AC is rectified and then fed through voltage regulators. One 5-volt regulator powers the thermocouple reference junctions, the other regulator powers the operational amplifiers. Each thruster heater is equipped with a sheathed chromel-alumel thermocouple junction for measuring thruster temperature. This thermocouple signal is fed into a T/C reference junction which provides a compensated signal referred to a known temperature and also makes the transition to copper leads. The output of the compensated chromel-alumel thermocouple, at the thruster temperature level, is 0 to 50 millivolts. The compensated thermocouple signal then is fed into an amplifier whose output is -2.5 to 2.5 volts at a power level which is suitable for telemetry input (Figure 92).

• Short Circuit Protection

The details of this circuit have not been fully investigated. However, two proposals were considered for short circuit protection:

a. Since all of the circuit provides some type of short circuit protection, therefore for the whole power conditioner it is not necessary to have a very sophisticated protective circuit and a small resistor could be used, acting as a fuse. Heater circuit, solenoid valve circuit, pressure switch circuit, and the temperature amplifier circuit all have built-in short circuit protections against shorts occurring in the load which they supply. In the heater circuit if the heater shorts then all of the voltage would appear across the transistor In the solenoid valve circuit and the telltale circuit resistors protect against short circuits in the load. Temperature amplifier circuit uses an operational amplifier which has built-in short circuit protection.

b. A small resistor in series with the power conditioner can be used to provide a signal for actuating a relay which would work as a circuit breaker. Reset of the relay would be done by a command signal.

Of course, more sophisticated protective circuits such as magnetic amplifier current sensing, or Hall effect sensor circuits have also been briefly considered.

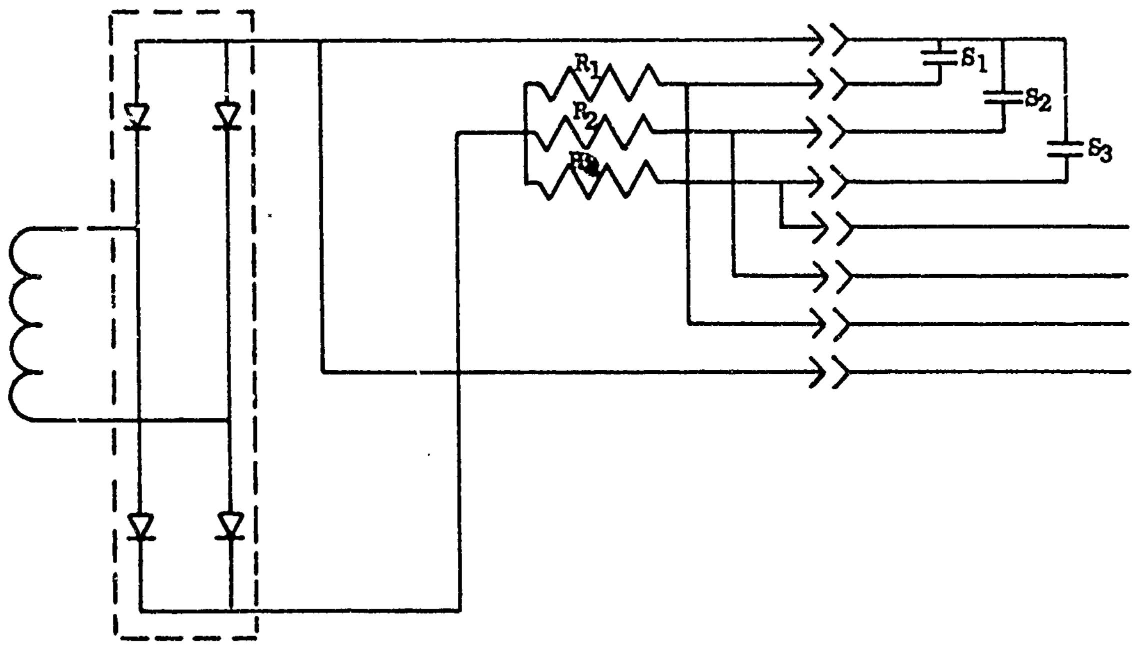

Figure 91. Telltale Circuit.

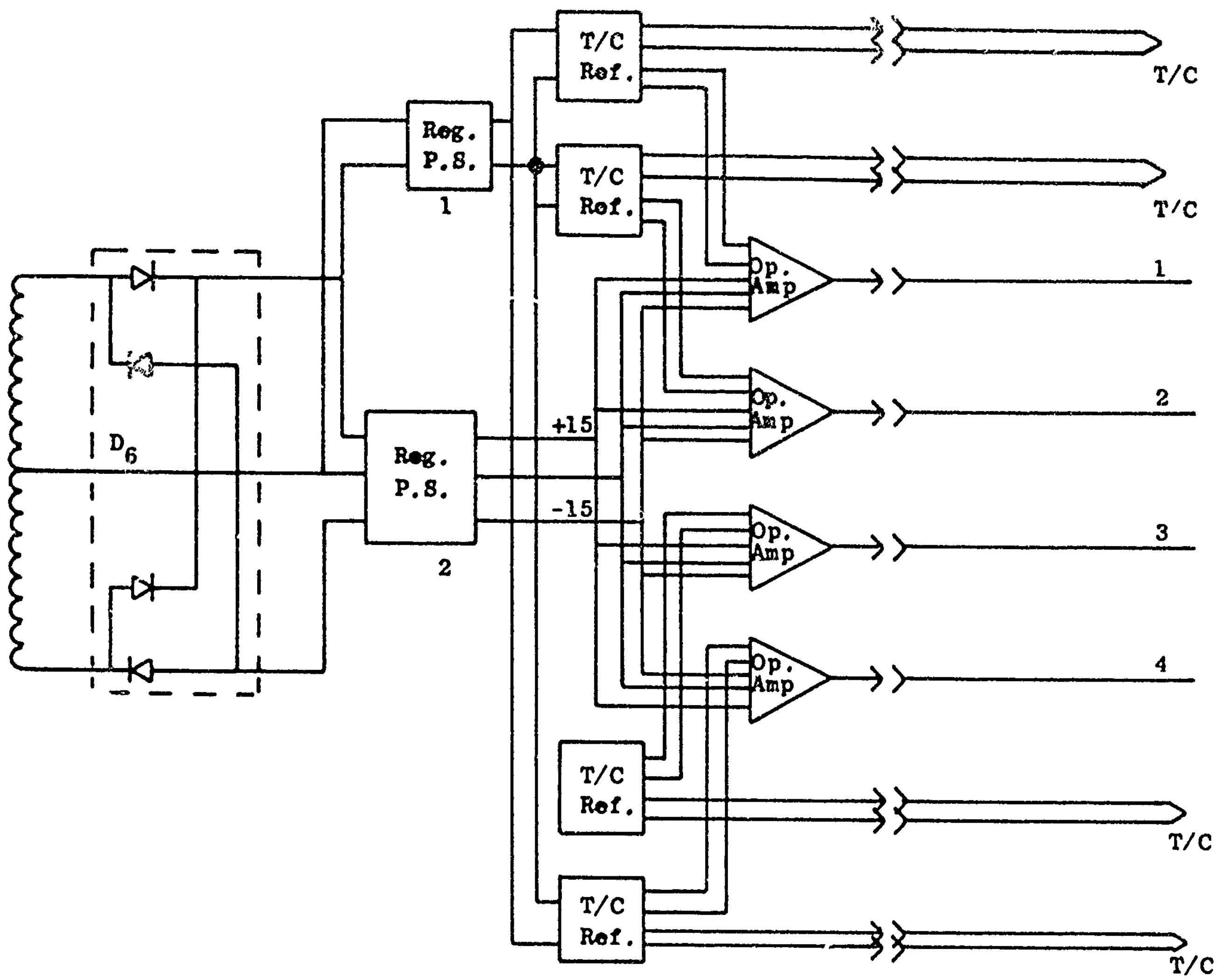

Figure 92. Temperature Amplifiers.

• Proposed Component List

POWER & SIGNAL CONDITIONER
PARTS LIST FOR FLIGHT TYPE

	Diagram Figure 93 Reference	Quantity	Part Description
GENERAL	T_1	1	Power Supply Transformer 1000Hz (SP 80950 SPECIAL) Polyphase Instrument Co.
HEATER CIRCUIT	D_1	1	Unitrode Diode UT 8110 (100V, 12A)
	D_2	8	Unitrode Diode UT 262 (200V, 2A)
	Z_{D1}	2	Zener Diode IN 3016 (6.8V)
	Q_1	2	Transistor 2N1725
	Q_2	2	Transistor 2N2351A
	R_1	2	Resistor .25, .5 or .75 ohm (5 watts)
	R_2	2	Resistor 1 K ohm (1/4 watt)
	R_3	2	Resistor 1.5 or 3 K ohms (1/4 watt)
	R_4	2	Resistor and ohm (1/4 watt)
SOLENOID VALVE	D_3	8	Unitrode Bridge Rectifier 673-2
	D_4	12	Unitrode Diode 262 (200V, 2A)
	C_1	8	Capacitor Non-polarized .068 μf, 75V (Aerovox Hi Q MC 80V 683 AK)
	C_2	8	Capacitor, Tantalum 40 μf, 75V (General Electric 6K105 AA2) std Part #R2023
PRESSURE SW SIGNAL	D_5	1	Unitrode Bridge Rectifier 673-2
	R_5	6	Resistor K ohms (1/4 watt)
T/C SIGNAL CONDITIONER	D_6	1	Unitrode Bridge Rectifier 673-2
	P.S. 1	1	5V Regulated Power Supply (Bourns) Model 3965
	P.S. 2	1	$\pm$ 15V Regulated Power Supply (Burr-Brown) Model [illegible]5/25
	T/C REF	4	T/C/ Reference Junction (Consolidated Ohmic Devices) Model JR 361
	OP AMP	4	Chopper Stabilized Operational Amplifier (Analog Devices) Model 203

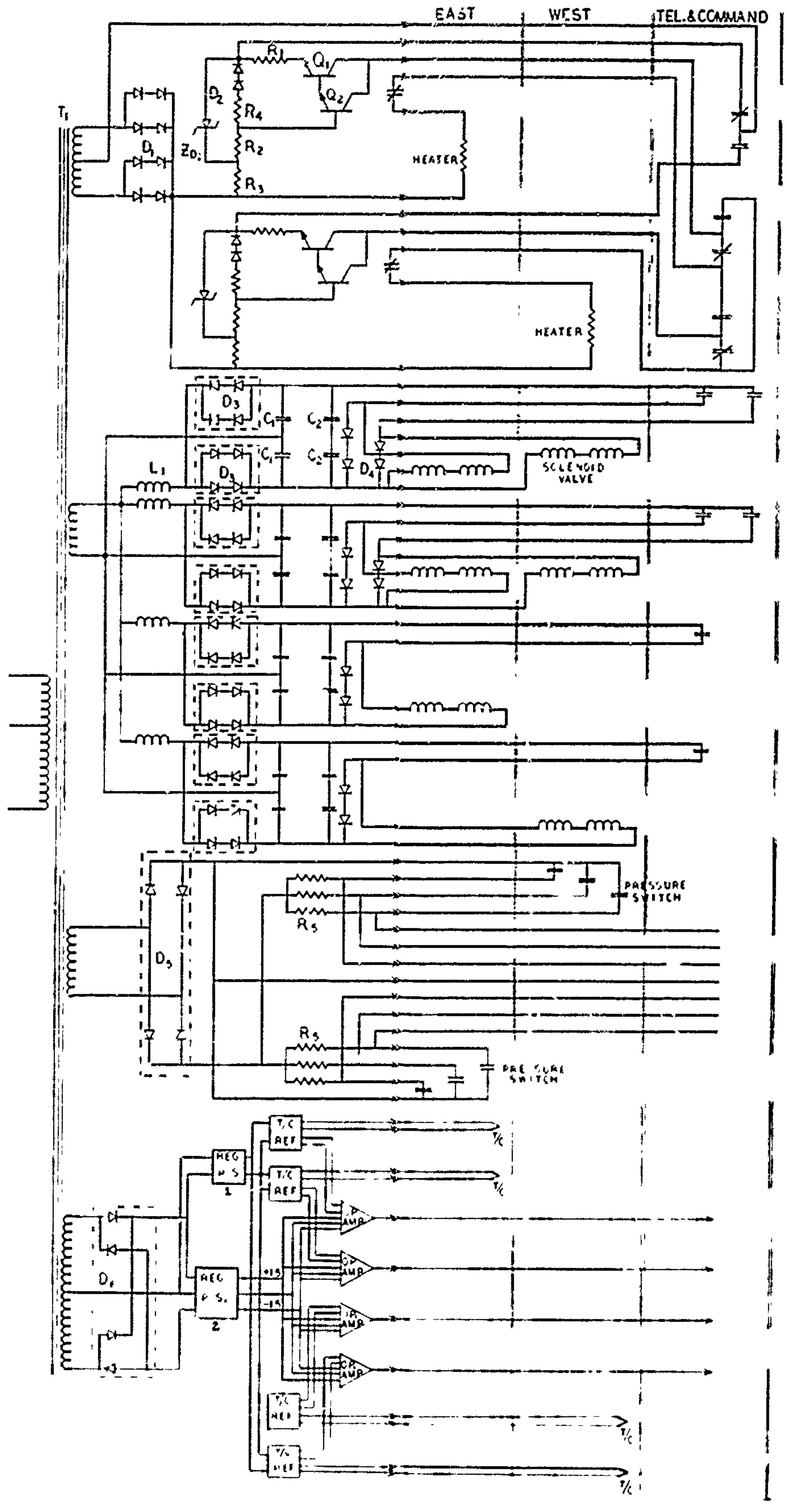

Figure 93. Electrical Schematic Diagram, Power and Signal Conditioner.

	Quantity	Part Description
ALTERNATE FOR T/C CONDITIONER	4	Telemetry Amplifier Model 5201, Bourns
	1	5V Regulated Power Supply (Bourns) Model 3965

3. BREADBOARD BUILDUP AND TEST RESULTS

A breadboard was built and was used to test the operation of each of the circuits. It is shown in Figure 94 and 95. The power supply was simulated by a square wave generator and a 60 watt high fidelity audio amplifier. The output of this power supply was connected to the primary of a transformer which accommodates the 28 or 40 volt levels. Separate secondary windings provide the heater and other outputs for the valves, pressure telltale, and temperature indicating circuit.

Two toroidal transformers were purchased for the power conditioner. The difference between them is that they have different secondary voltages for the heater circuit. Drawings for the transformers, including specifications, are shown in Figures 96 and 97.

- Heater Circuit

The curves in Figures 98 and 99 show the performance characteristics of the heater circuit for two types (resistances) of heaters. Table X indicates the performance drift as a function of component temperature.

The basic operating point for the curves of Figure 98 (high resistance) was established at 20 volts and 1 ampere. At this point the heater power is 20 watts and the heater circuit operates at about 79% efficiency. The current corresponding to the cold heater resistance is 1.08 amperes or only 8% higher than the normal operating current. When the transformer input voltage was raised by 20%, the output voltage increased by 10%, thus increasing the heater power by 20% from 20 to 24 watts. On the other hand, when the input voltage was lowered by 10%, the heater voltage decreased by 10%, and the corresponding heater power decreased by 20%.

The operating point on the curves of Figure 99 (low resistance) was set at 10 volts and 2 amperes. At this point the heater power is 20 watts but the heater circuit operates at 57% efficiency. The current corresponding to the cold heater resistance is 2.12 amperes i.e. 6% higher than the normal operating current. For a 20% increase in the input voltage, the heater voltage increased by 6%, thus increasing the heater power by 12% from 20 to 22.4 watts. When the input voltage was lowered by 10%, the heater voltage decreased by 7% corresponding to a 14% decrease in heater power.

A second operating point (refer to Figure 99) was established at 11.4 volts and 1.9 amperes corresponding to a heater power of 21.6 watts. At this operating point, the heater circuit operates at 64% efficiency, seven percent better than the first operational point. However, the 20% input voltage rise increases the heater power by

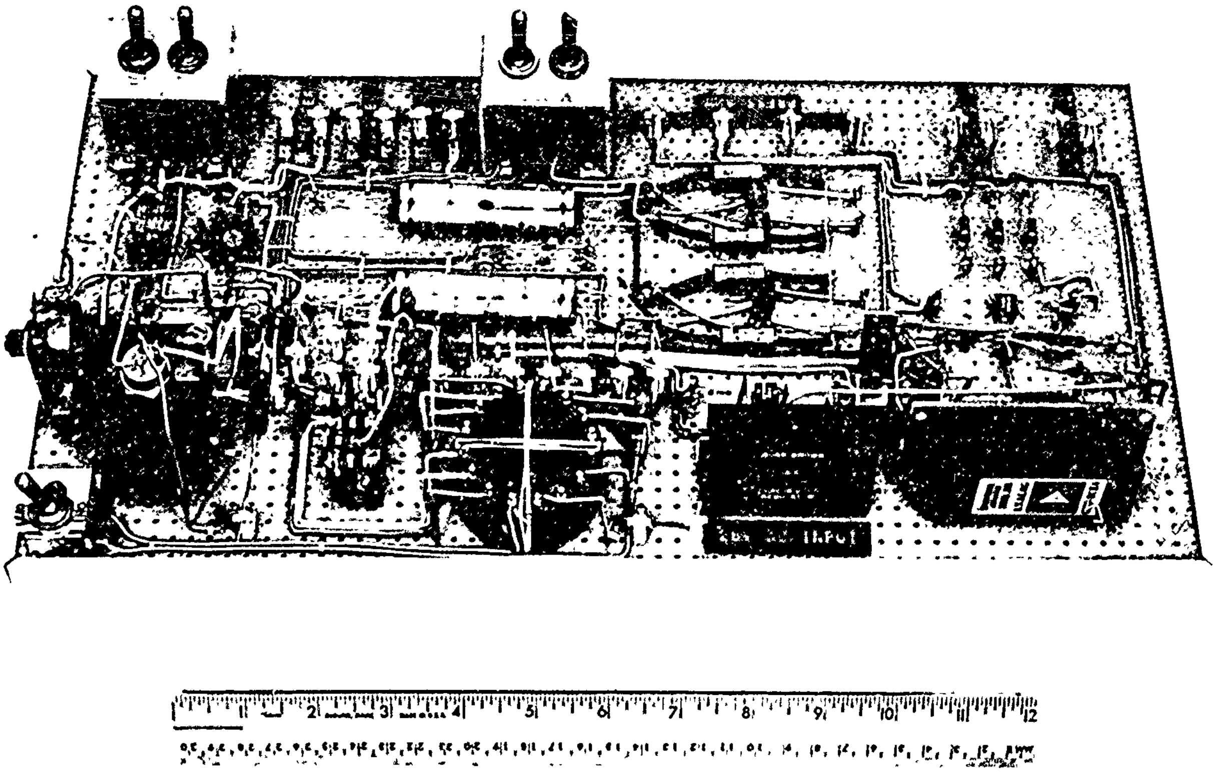

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Figure 94. Power and Signal Conditioner in "Breadboard" Configuration for Use in Testing and Analyzing all Circuits, with Transformer Lower Center of Assembly.

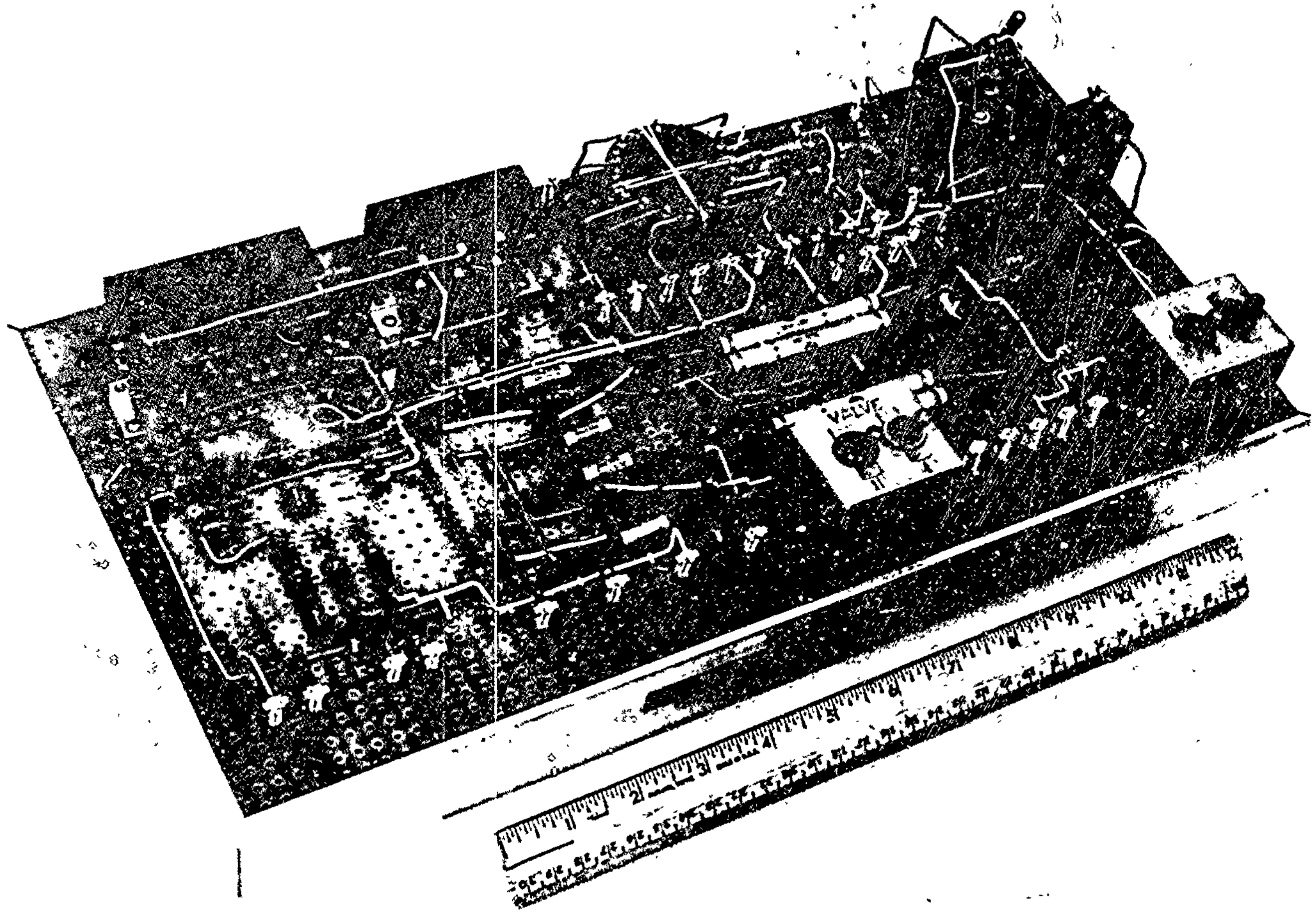
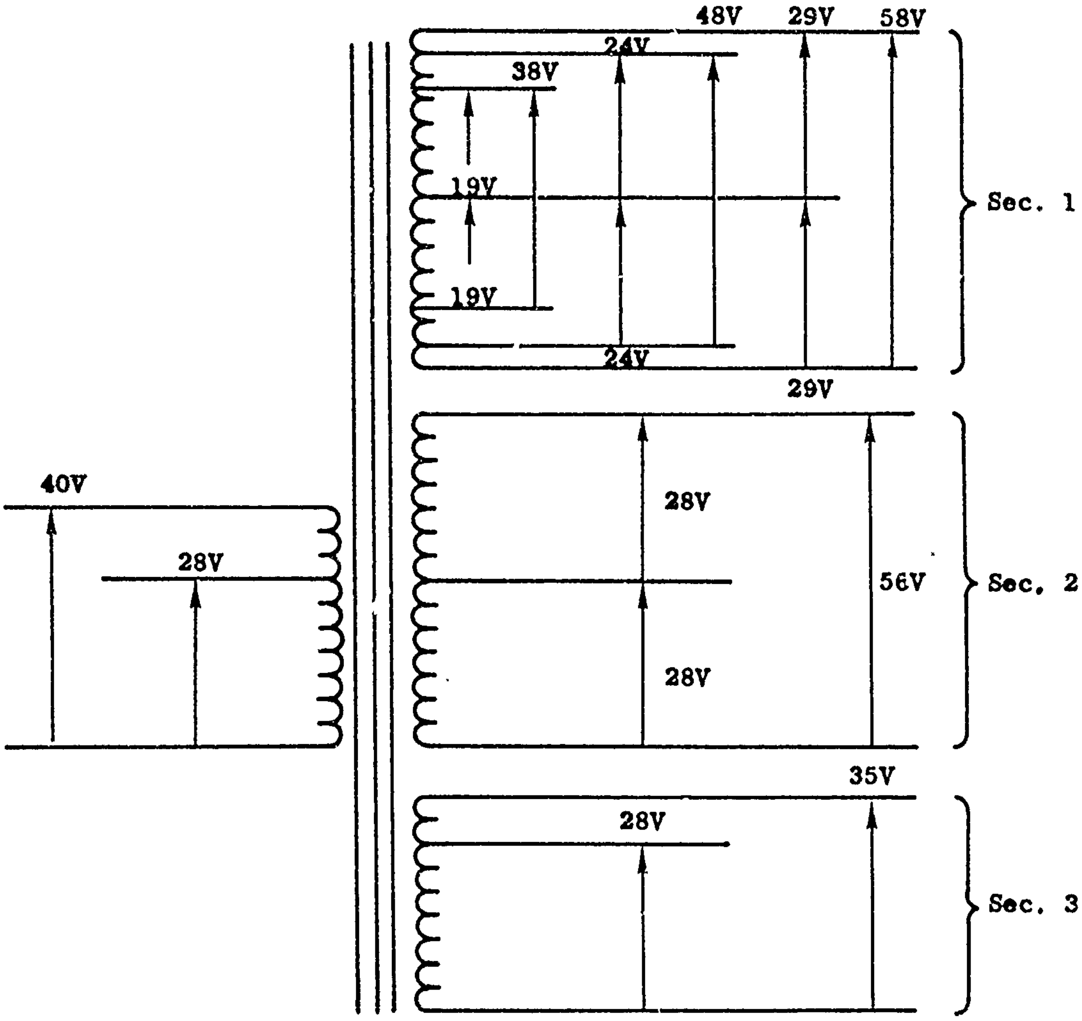


Figure 95. Breadboard (Auxiliary View).



Input: Square Wave
976 Hz

Sec. 1 - Power drain through full wave rectification into resistive load 27 watts. Transformer should be able to deliver 27 watts at any one of the voltage ratings (38V, 48V, or 58V)

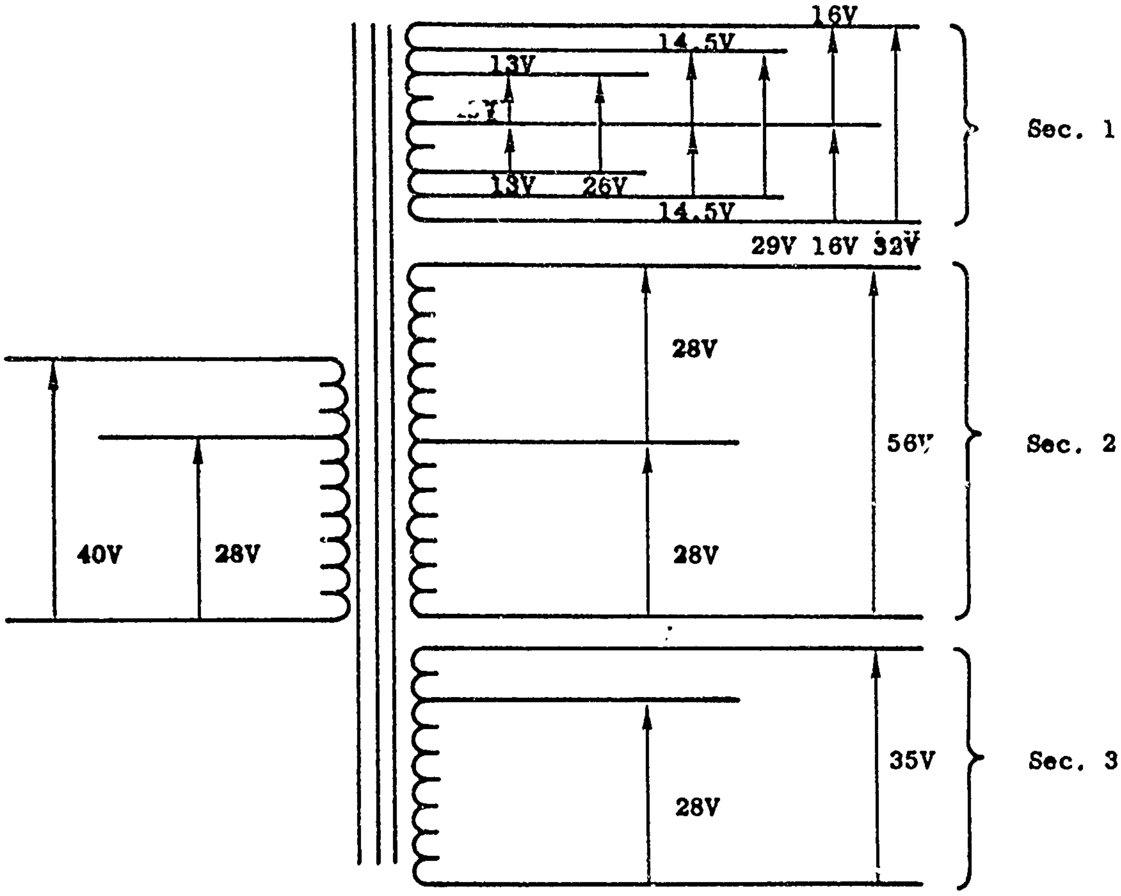
Sec. 2 - Power drain through full wave rectification 4 watts.

Sec. 3 - Power drain through full wave rectification (Large capacitor is being charged, large leakage reactance is desirable) Normal power drain .7 watt.

Construction: Open frame, fully impregnated, capable of passing MIL-T-27B Grade 5, Class R, altitude unlimited. Minimum weight required.

Efficiency: >95%

Figure 96. Power Supply Transformer. Schematic Diagram and Performance Specifications.



Input: Square Wave
976 Hz

Sec. 1 - Power drain through full wave rectification into resistive load 27 watts. Transformer should be able to deliver 27 watts at any one of the voltage ratings (26V, 29V, or 32V)

Sec. 2 - Power drain through full wave rectification 4 watts.

Sec. 3 - Power drain through full wave rectification (Large capacitor is being charged, large leakage reactance is desirable) Normal power drain .7 watt.

Construction: Open frame, fully impregnated, capable of passing MIL-T-27B Grade 5, Class R, altitude unlimited. Minimum weight required.

Efficiency: >95%

Figure 97. Power Supply Transformer II. With Different Secondary 1 Voltage.

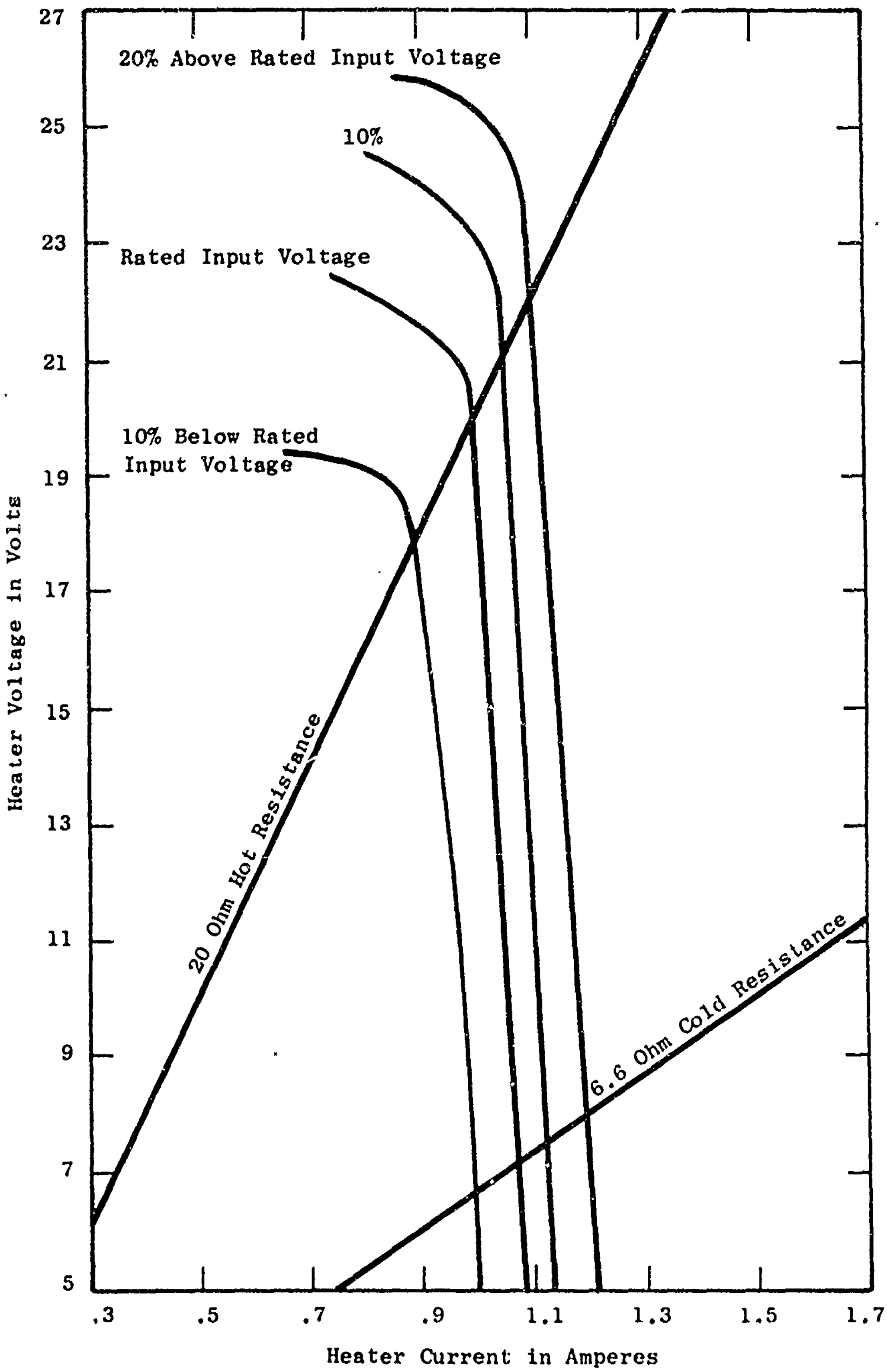


Figure 98. Characteristics of High Resistance Heater.

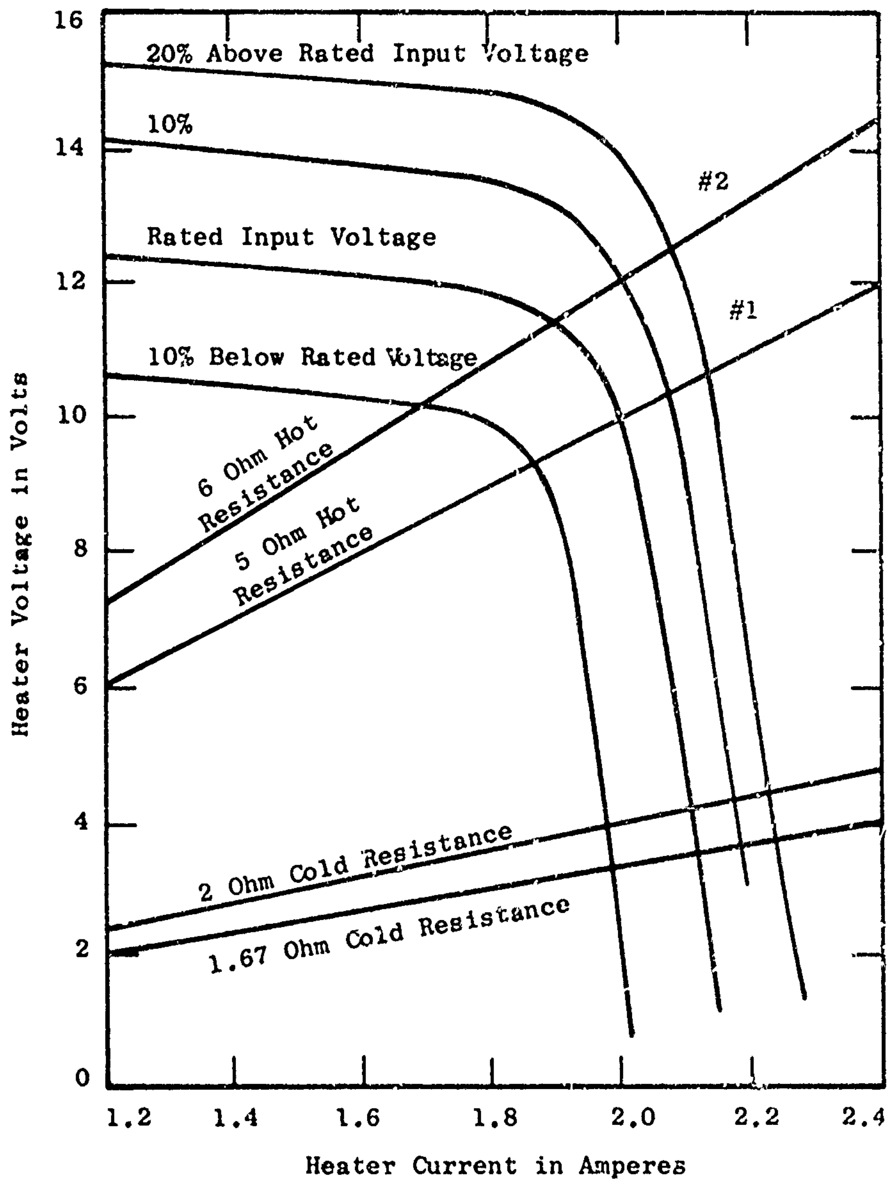


Figure 99. Characteristics of Low Resistance Heater.

TABLE X

PERFORMANCE DRIFT AS A FUNCTION OF TEMPERATURE

		98°F	150°F	175°F	200°F
Input to the transformer	Voltage (Volts)	23.6	23.5	23.5	23.5
	Current (Amperes)	1.10	1.10	1.11	1.12
Output to the heater	Voltage (Volts)	20.2	20.2	20.4	20.5
	Current (Amperes)	1.02	1.01	1.02	1.03

20%, and the 10% input voltage drop increases the heater power by 20%. The first operating point provides better line regulation whereas the second operating point setting results in better efficiency.

While adjustable resistances were used to simulate the heaters, it should be noted that the high resistance range of values corresponds to the moly-50 rhenium heater. The low resistance range corresponds to the recrystallization resistant platinum heater.

- Temperature Amplifiers

The circuit shown in Figure 100 was set up to check out the temperature amplifier. The temperature reference junction is not shown since it had not been purchased. Instead of a thermocouple, a known thermocouple signal (a certain millivolt signal) was fed into the amplifier. A room temperature test showed that the output was a linear function of the input. The gain was adjusted such that the output signal was $\pm$ 2.5 volts for the operating range of the heater. Telemetry circuits would be used to produce a ten to one attenuation. Temperature versus output voltage is shown in Figure 101.

At this stage of development two circuits were considered for the temperature amplifiers. The circuit shown in Figure 100 had an analog operational amplifier which is physically large (1.3 in. x 1.0 in. x 2.8 in.) and does require a well-regulated power supply. Another circuit which was considered used a Burr-Brown amplifier and does not require a regulated voltage. This amplifier is smaller (.5 in. x 1 in. x 1 in.) but it costs at least twice as much as the previous one and does not require a small regulated power supply at the output to shift the level of the voltage for the telemetry.

- Solenoid Valve Circuit

The solenoid valve circuit was breadboarded as it is shown in Figure 90.

Tests on two flight valves showed that the valves had been designed very conservatively. One operating condition was established by supplying a voltage above the maximum pull in voltage (28 volts) and letting it drop to a voltage which is somewhat higher than the maximum dropout voltage (about 7 volts). A test series was run under these conditions. The steady state voltage across two valves in series was 14 volts and the corresponding current was 9 ma. This corresponds to a valve power of less than .2 watt. To protect the C_1 capacitors, a resistance of 3 ohms/volt will be put in series with these capacitors.

This solenoid valve circuit draws from the transformer current in the form of spikes. The 9 ma rms current, from oscilloscope pictures shown in Figure 102 has peak currents of about .4 ampere. When this current is added together with the heater current, the spikes are hardly noticeable. In figure 102 the no load picture shows the 100 Hz square wave input to the power conditioner. Second picture shows the input current to the power conditioner when the heater draws 1.65 amperes. The third oscilloscope picture shows the input current to the

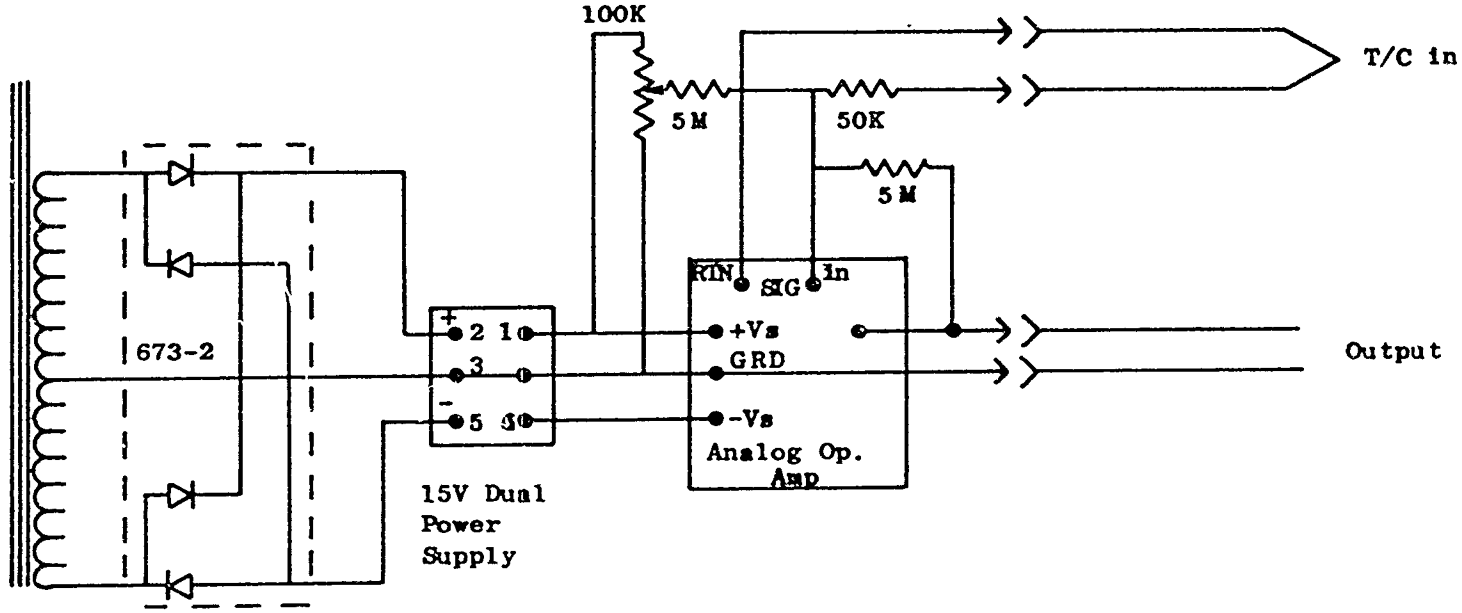


Figure 100. Temperature Amplifier Circuit for Breadboard.

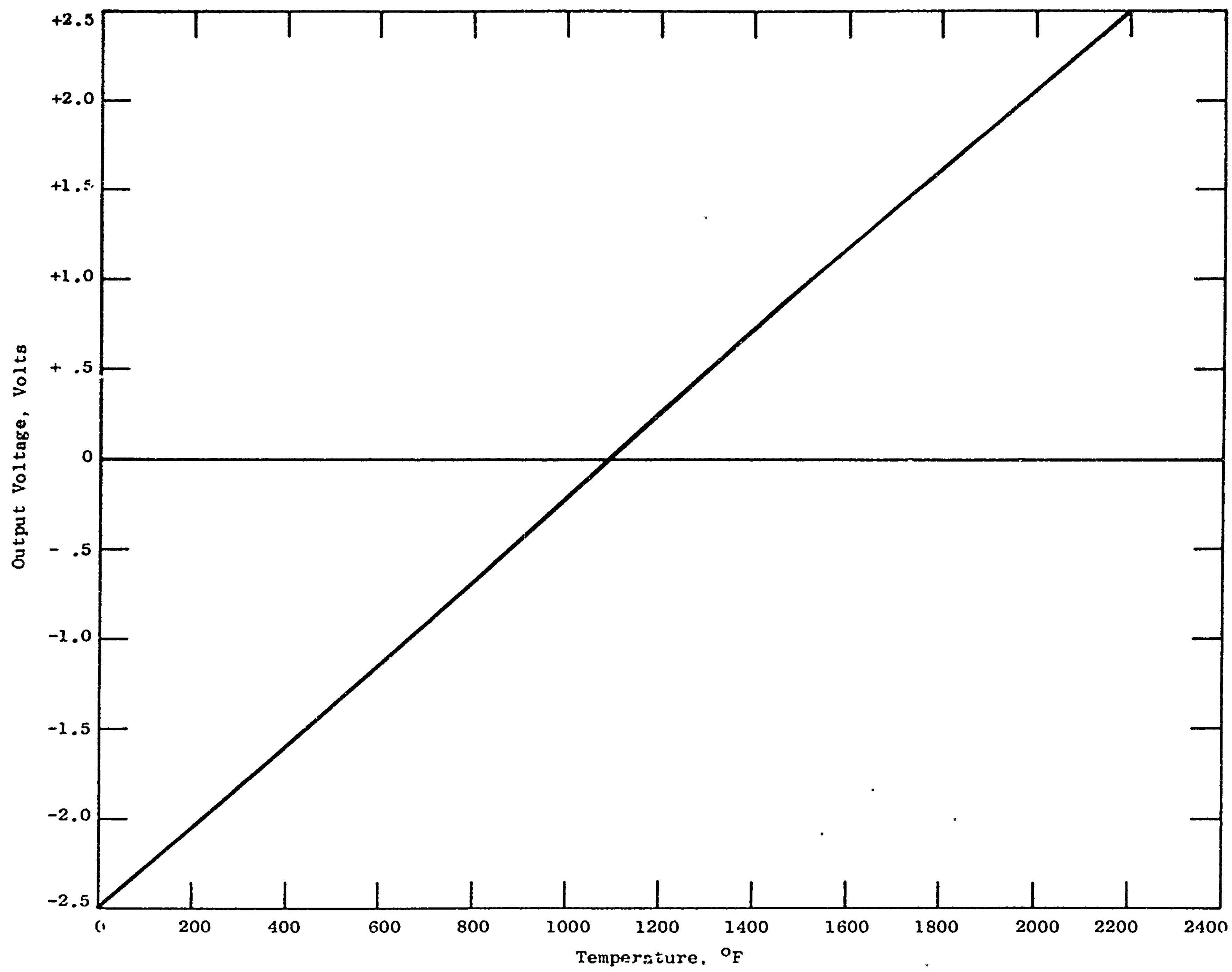
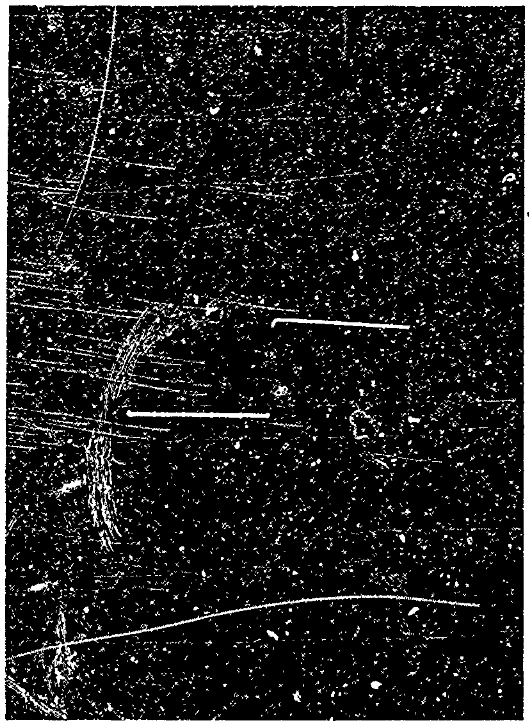
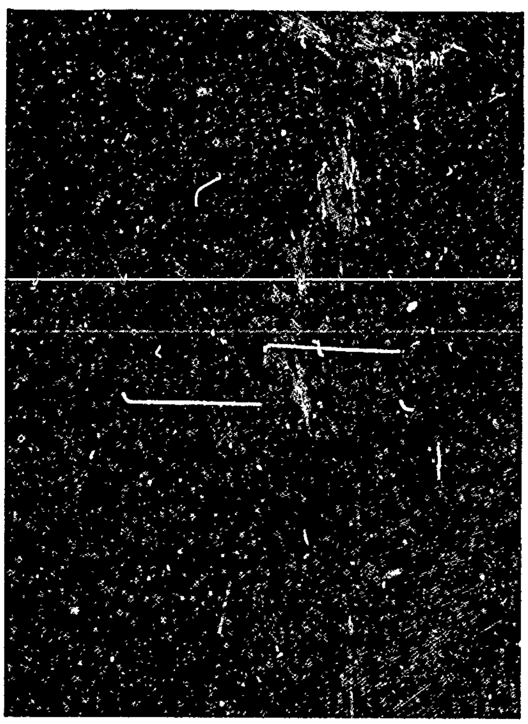


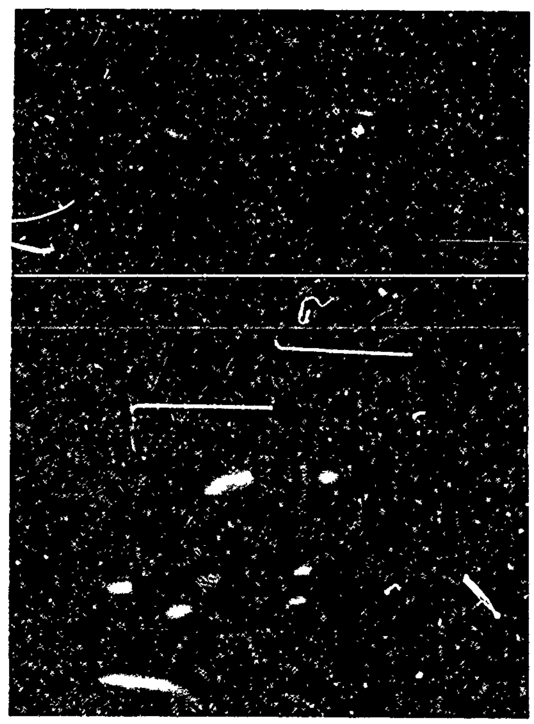
Figure 101. Temperature Versus Output Voltage for Temperature Amplifier.



No Load



Heater on 1.65 amp



Heater Plus Solenoid Valve

Figure 102. Waveforms.

power conditioner when the heater draws 1.65 amperes and also draws current for operating the solenoid valve. The oscilliscope pictures were taken across a small resistor (.1 ohm) which was placed in series with the power conditioner.

4. PACKAGING

An enclosure was made for the engineering prototype power conditioner. The dimensions of the box are shown in Figure 103. A photograph of the actual box is shown in Figure 104. The box was made of aluminum and the joints were TIG welded together. The overall shape was determined by requirements and space in the satellite, where the power conditioner will be mounted in a "book case" arrangement. Maximum volume for the prototype box was below the specified 150 cu. in. Mounting to the supports (which are also the "heat sinks") would be through the projections, front and back. Connections to the wiring harness would be through connectors. Those on front are for connections that are removed during check-out. Those in back (not easily accessible after mounting) are for connections that are "permanently" made at installation.

5. SUITCASE TESTER

It was a requirement of the project that means be developed for conveniently checking the Power and Signal Conditioner, in the factory, at the satellite assembly location, and at the launch pad. A piece of test equipment was therefore proposed which would have the necessary wide range of test capability and be built into a "suitcase" (approximately 15 in. x 18 in. x 6 in.) for easy transportation.

Table XI indicates the test capability which was proposed for the tester. It lists the components which will be included in the system, and then the connecting procedures and testing sequence for bench (at manufacturer) assembled system and launch pad. At the launch pad the test was planned as a "go, no-go" type of sequence for simplicity and minimum disturbance to the final connections in the vehicle. An auxiliary cable connector would be mounted for umbilical attachment to the tester for speed in making the test.

Figure 105 is a sketch for the preliminary layout of the suitcase tester. Connectors for input and output cables are in the lower right hand corner. Switches are used to select the circuits under test and simulate the loads and certain power conditioner circuits. Miniature instruments, rather than lights, are used to indicate the results of each test.

6. SUMMARY

All work on the Power and Signal Conditioner and the "suitcase" tester was stopped due to lack of definition of the characteristics of the electrical power to be made available to the Microthruster System. The foregoing tasks were completed up to the points indicated. The Power and Signal Conditioner was carried through to breadboard form. No components were procured for the flight units. The "suitcase" tester was just carried to the planning stage.

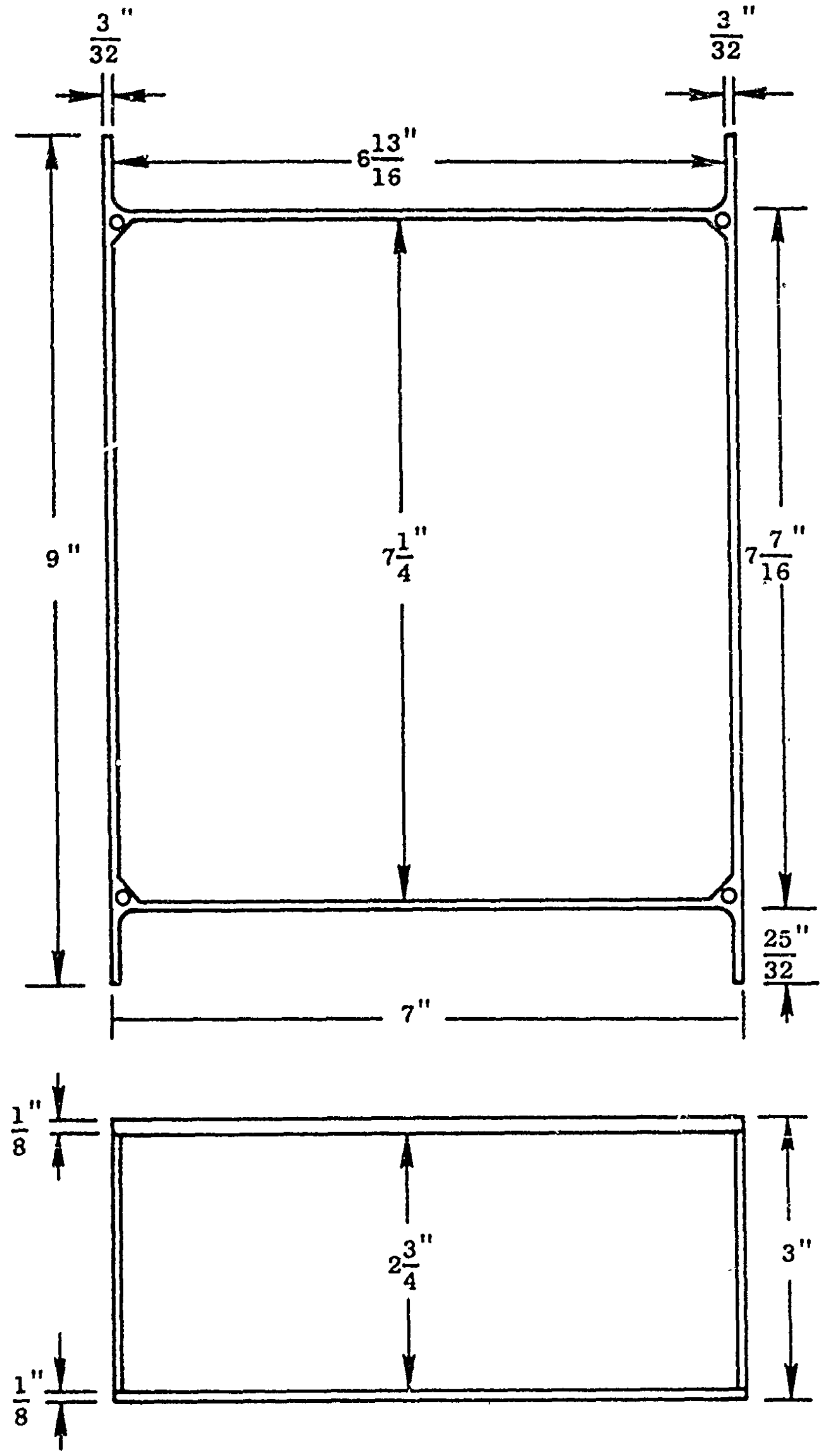


Figure 103. Enclosure for Engineering Prototype Power Conditioner.

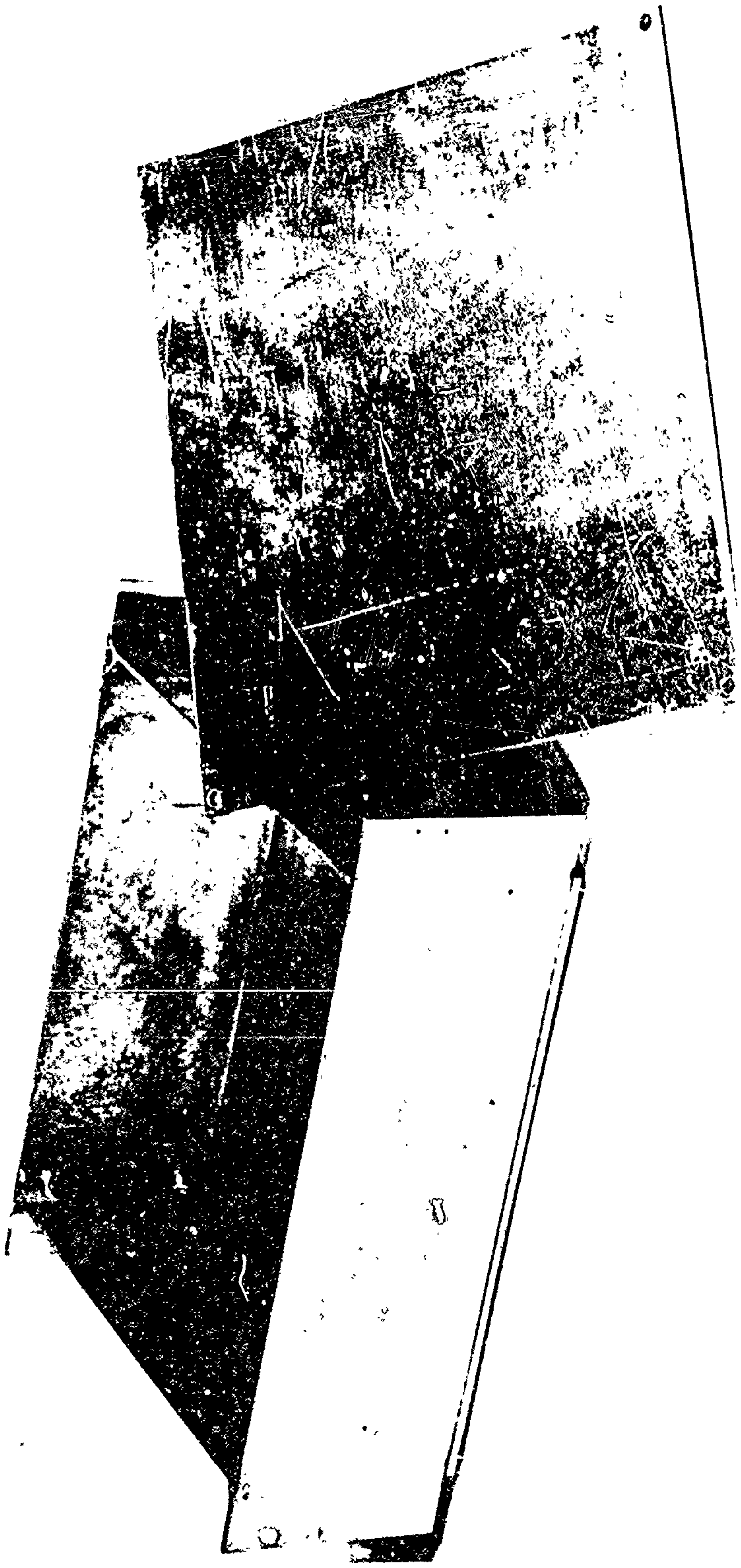


Figure 104. Enclosure for Engineering Prototype Power Conditioner.

TABLE XI

TEST CAPABILITY OF "SUITCASE" TESTER

1. Tester Will Include

   A. Input power (1000 cps) connections and control.

   B. Resistances to simulate heater at minimum and maximum temperature.

   C. 3 relays comparable to valve loads.

   D. Main selector switch.

   E. Second selector switches for valves and pressure switches.

   F. Indicating lights.

   G. DC ammeter (for heater current).

   H. DC voltmeter (multi-range).

   I. Connectors and cables.

2. For Launch Pad or System Testing

   A. Tester cable output to special connector on P&S Conditioner (or to flight plug and connector on vehicle which is connected to P&SC). Note: Probably use different "test cable" for lab and pad testing.

   B. Tester connection to 1000 cps power supply.

   C. Testing sequence is as follows:

      1. Simulate command module contacts. Check East heater and West heater power circuits briefly, or use external resistor to simulate heater.

      2. Measure resistance of each heater circuit ... 3 ohms approx.

      3. Check output of T/C operation amplifiers to telemetry - should be - .25 volts at room temperature. Note: This will be measured by system telemetry.

      4. Apply gas to valve/piping system (dry N_2 gas applied to lines at flight check-out). Operate each valve and check for pressure switch tell-tale signal. Also operate any combination of 2 valves on East or West and check for tell-tale signals. Signals to be checked by telemetry system.

      5. Consider any other "passive" monitoring desirable.

Table XI (Cont'd)

3. For Complete Bench Check-Out of Power and Signal Conditioner

   A. Cable connections made to all inputs and outputs of P&SC (except pad or "system check-out connection).

   B. Connect to 1000 cps power supply (not built into tester).

   C. Testing Sequence is as follows:

      1. Check input voltage

      2. Select "Heater East" - Read current and voltage with minimum and maximum resistance.

      3. Select "Heater West" - Read current and voltage with minimum and maximum resistance.

      4. Select "Valves East" - Use 2nd switch to select 3, 6, or 200 micropound and any 2 combinations. (Light on relays simulating valves, indicate operation.)

      5. Select "Valves West" - Repeat sequency shown in 4.

      6. Select "Pressure Switches" - A 2nd switch will simulate tell-tales of the 6 switches and output shown as indicating lights.

      7. Select "T/C Response" - A 2nd switch will apply voltage comparable to T/C outputs for room temperature and for 200°F. Output values read on meter (range of -.25 to +.25 volts).

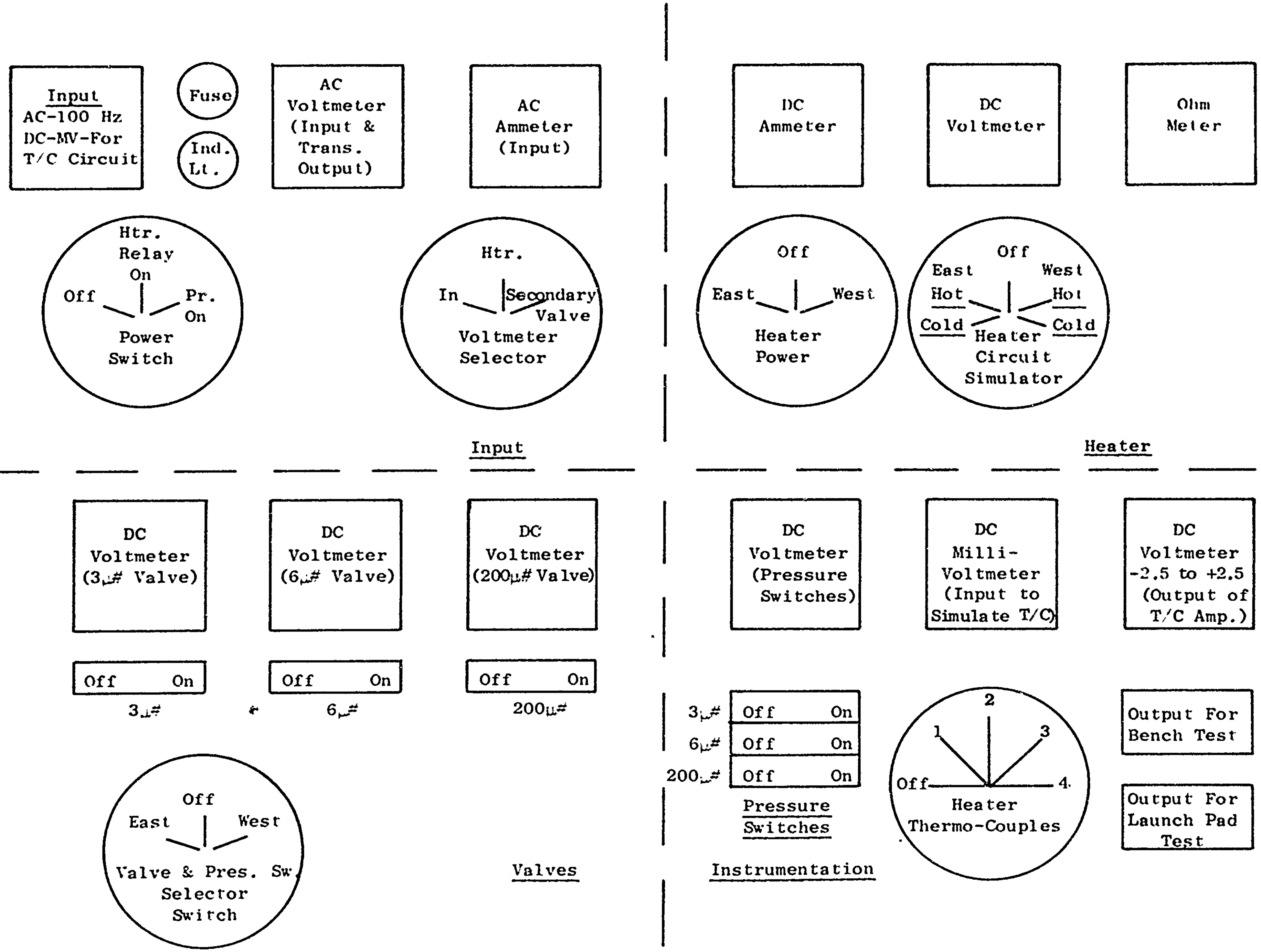


Figure 105. Layout of Suitcase Tester.

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SECTION VIII

SYSTEM INTEGRATION

1. LIAISON

Continuous contact with personnel at the Applied Physics Laboratory (APL) of The Johns Hopkins University was a contractual requirement. The Work Statement "Exhibit A" dated September 12, 1966 (Appendix XII) required that the General Electric Company provide interface information to APL in the areas of weight, volume, power, thermal conditions, and the like.

Initial results of these efforts appeared in the "Resistance Jet System Interface Document", dated November 25, 1966 (Appendix XIII). This document described the technical requirements and agreements between General Electric Company and APL regarding the resistance jet microthruster system mechanical and electrical interface with the Dodge-M satellite. Some portions were made obsolete by the issuance by the Air Force of the revised Work Statement "Exhibit A-1", dated April 21, 1967 (Appendix XIV). Had the proposed additionally revised Work Statement "Exhibit A-2", dated February 7, 1968 (Appendix XV) been made effective, the document would have been subject to still further changes.

Design reviews were held periodically with APL and WPAFB personnel so that the microthruster system would be completely acceptable for installation and use on the satellite. Progress was reported monthly (Reference 4) to WPAFB.

As a guide for the preparation of flight hardware, APL personnel made available their 9 page "Space Division Standard Manufacturing Practices", drawing number 7205-9915 (Reference 5). The pertinent requirements of this document were considered when General Electric Company prepared the "Quality Control Manual, Resistance Jet Thruster Systems, Dodge-M", (Reference 6).

2. PROPELLANT TANK SUPPORT AND FLOW CONTROL COMPONENT SUBASSEMBLY

Initially, it was planned to mount the flow control and flow instrumentation components in "viewband" packages (as shown in Figure 106) remote from the fuel tank. Discussions with APL personnel, however, indicated that installation of the microthruster sy em on the satellite would be simpler if all the viewband packages w .e built onto one "platform", along with the fuel tank. This concept is shown in Figure 107. Further discussions, however, with both APL and WPAFB personnel ultimately resulted in the packaging concept shown in Figure 108. This design was finalized using 1/4 inch tubing to interconnect the components. In an effort to conserve weight, the design was subsequently altered to use 1/8 inch tubing. Completed designs are shown in Figures 109, 110, and 111. Partially finished assemblies appear in Figures 112 and 113. A considerable amount of effort was put into this design so that it would be a flight worthy assembly. For example, as far as practicable, tubing connections were brazed rather than made with flared tube fittings, reducing the possibility of leaks.

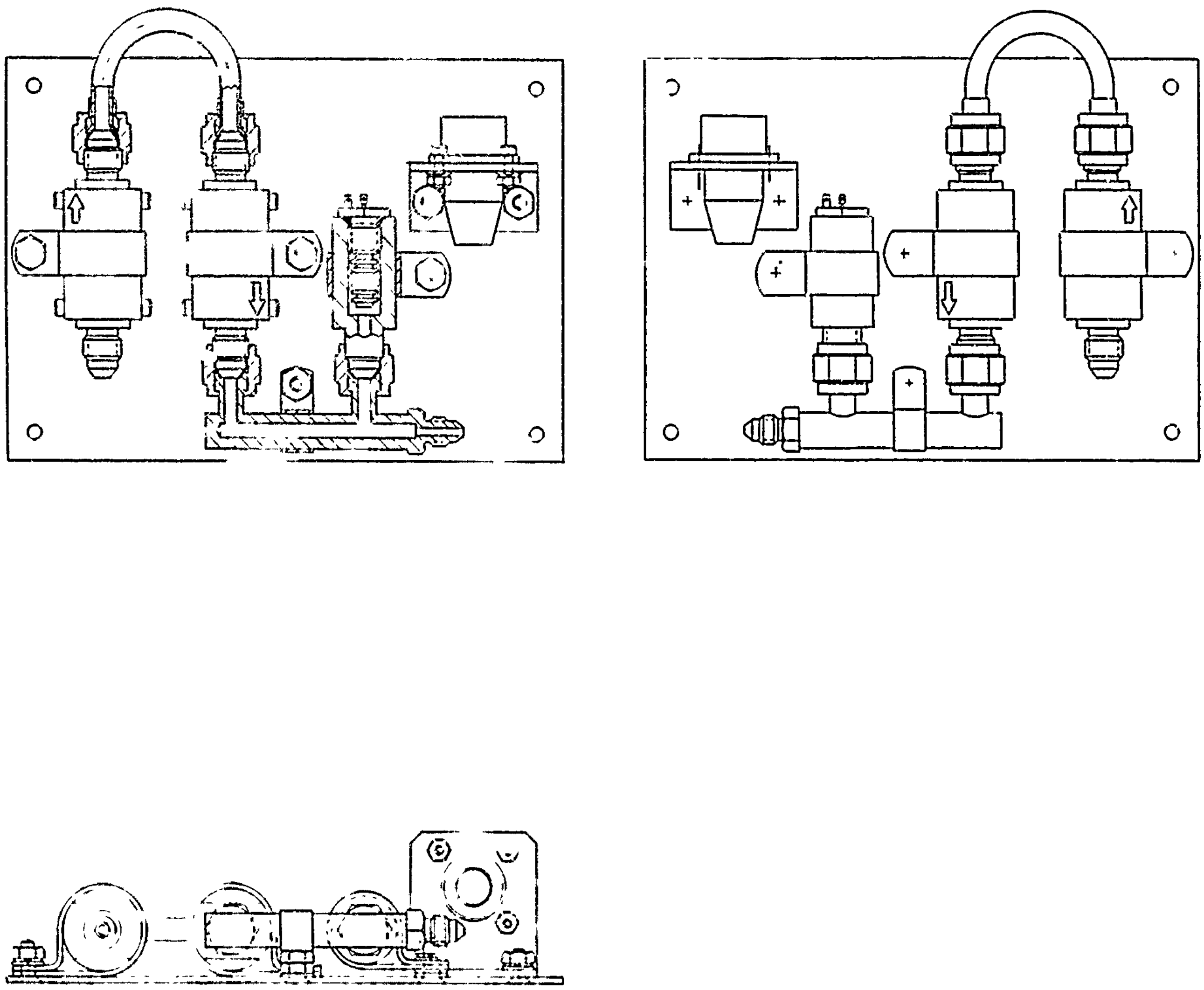


Figure 106. "View Band" Packaging of Flow Control Components.

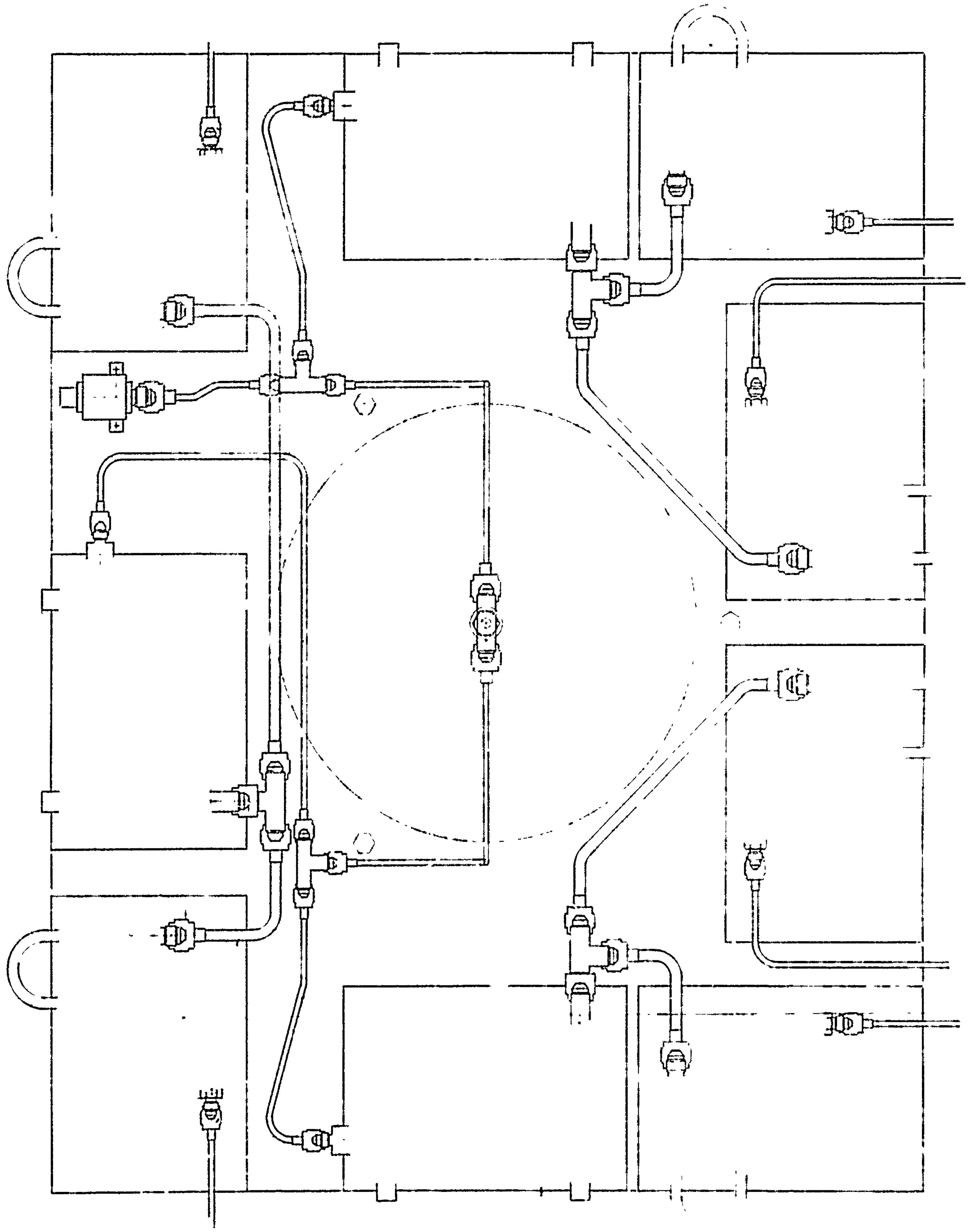


Figure 107. "Platform" Packaging of Flow Control Components with Propellant Tank.

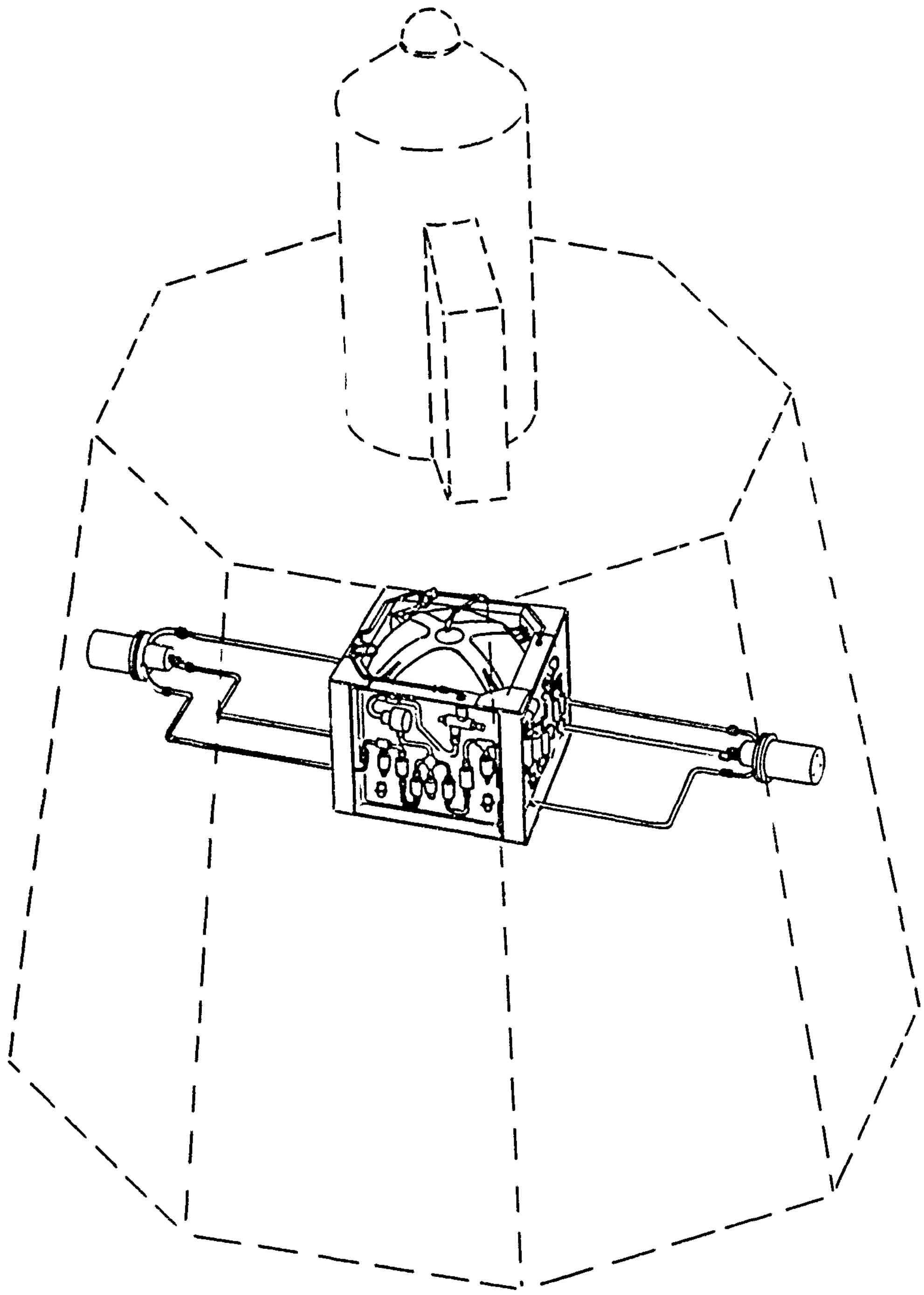


Figure 108. Arrangement of Resistance Jet Microthruster System on Satei ite.

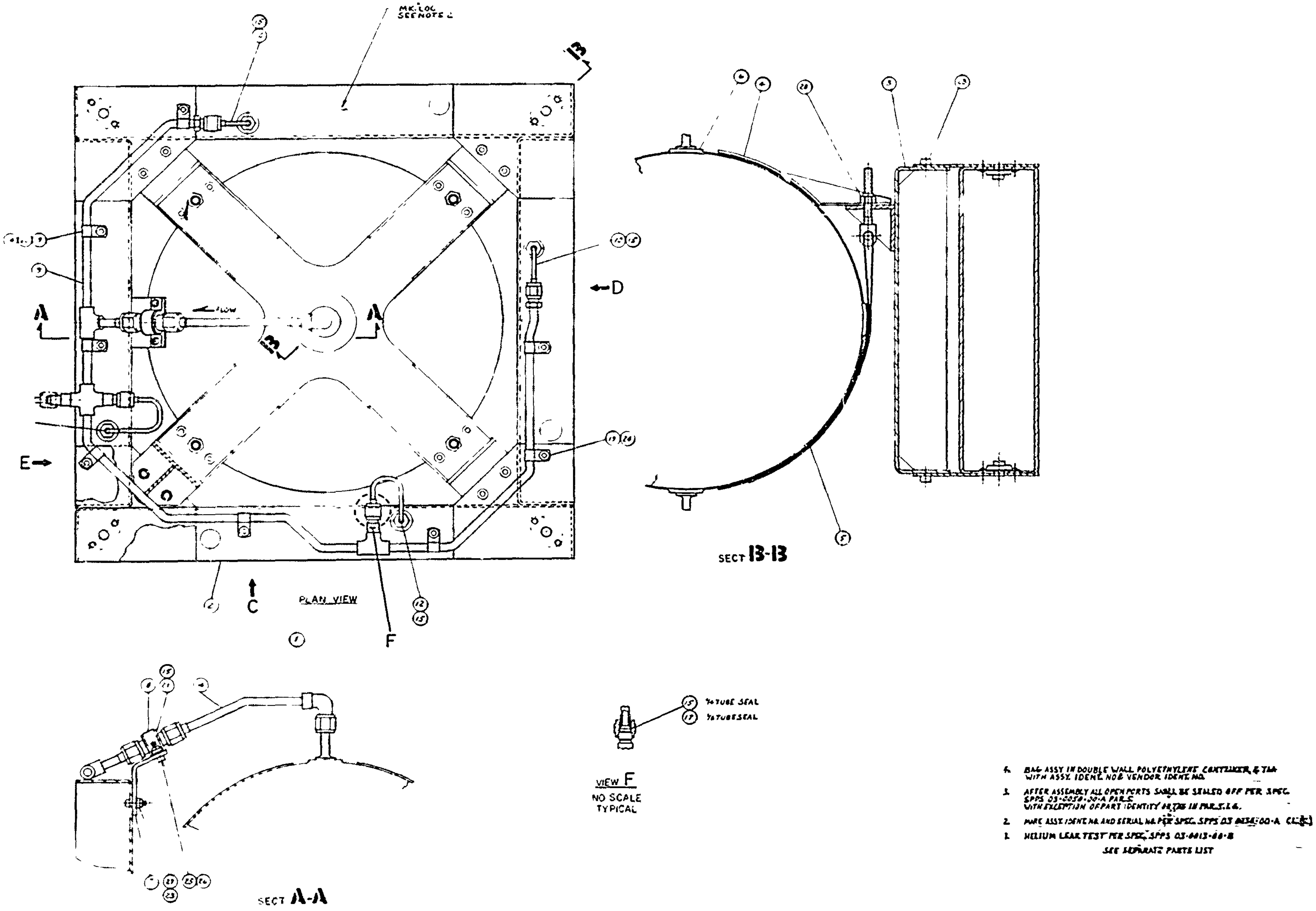


Figure 109. Frame Structure for Propellant Tank and Flow Control Components, Top View.

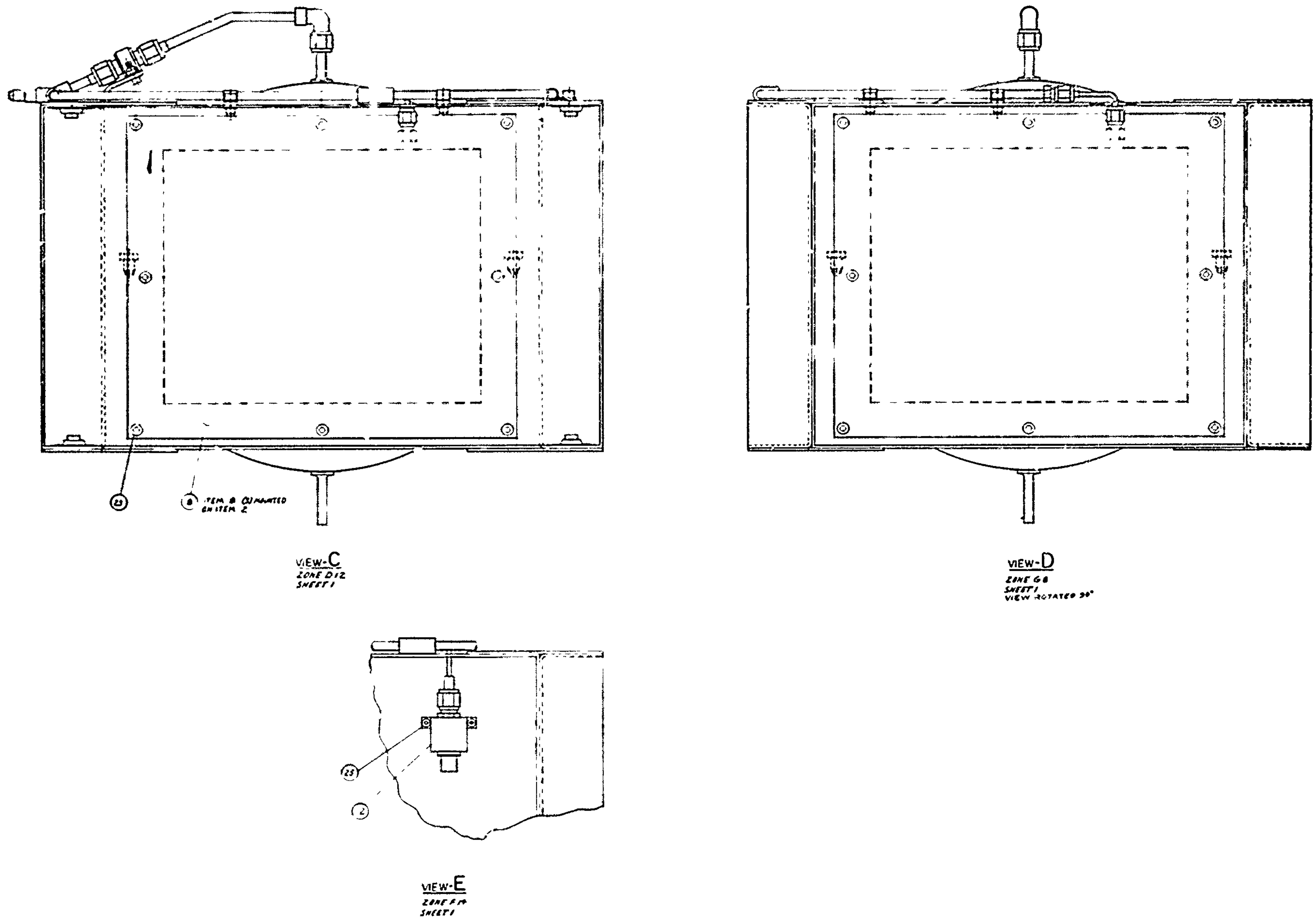


Figure 110. Frame Structure for Propellant Tank and Flow Control Components, Side View.

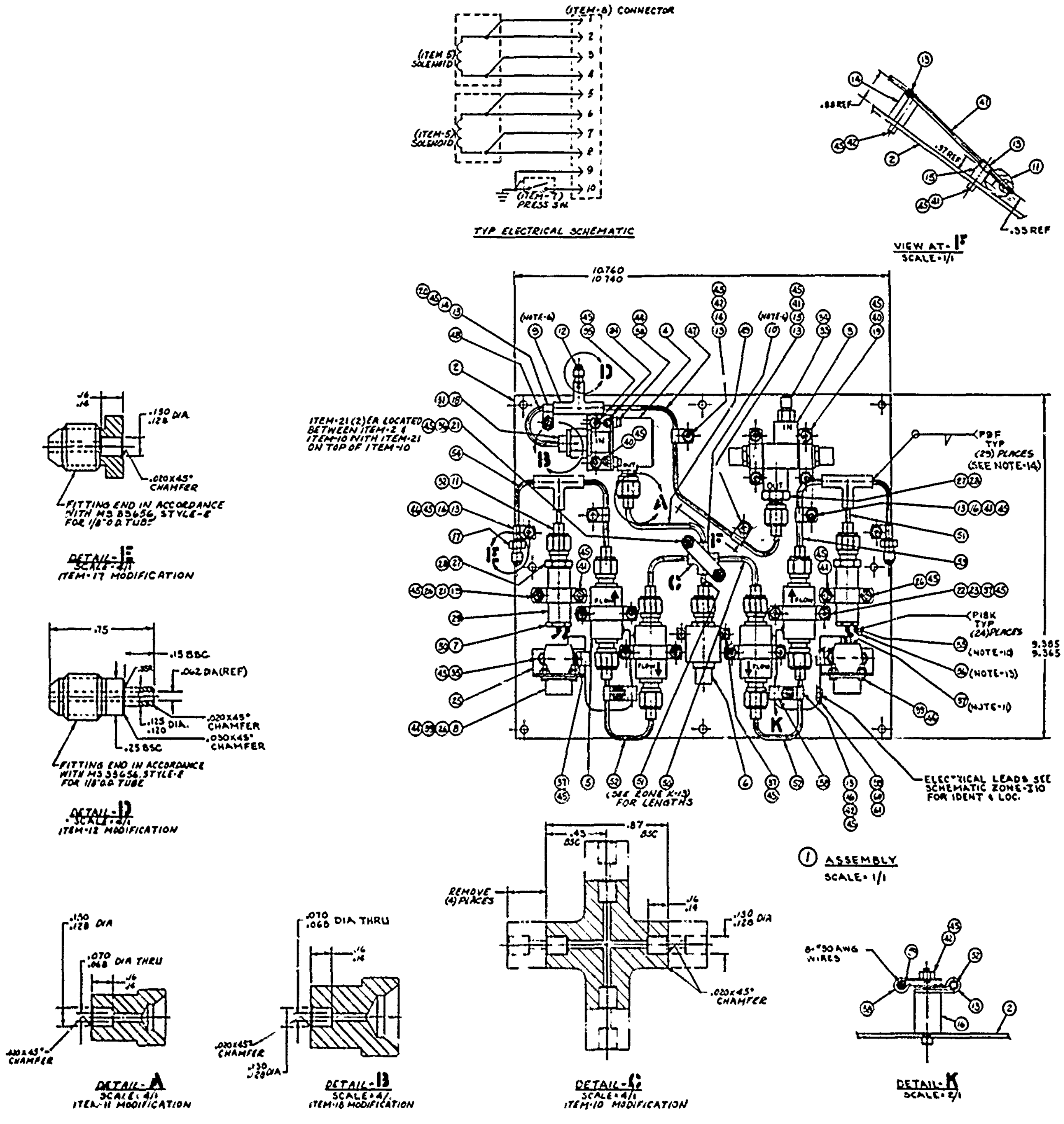


Figure 111. Assembly of Flow Control Components on Panel.

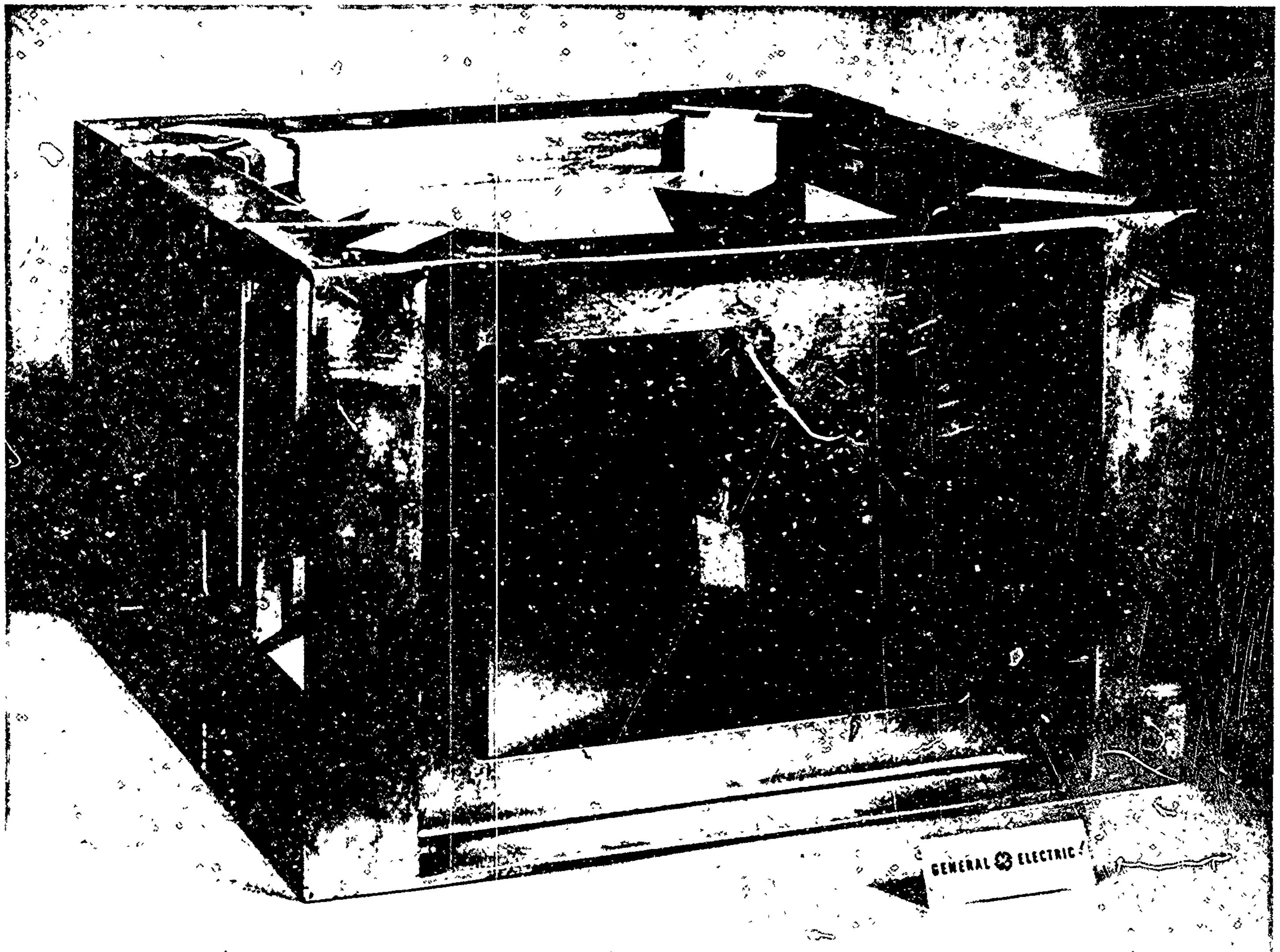


Figure 112. Frame Structure for Propellant Tank and Flow Control Components, Unfinished Assembly.

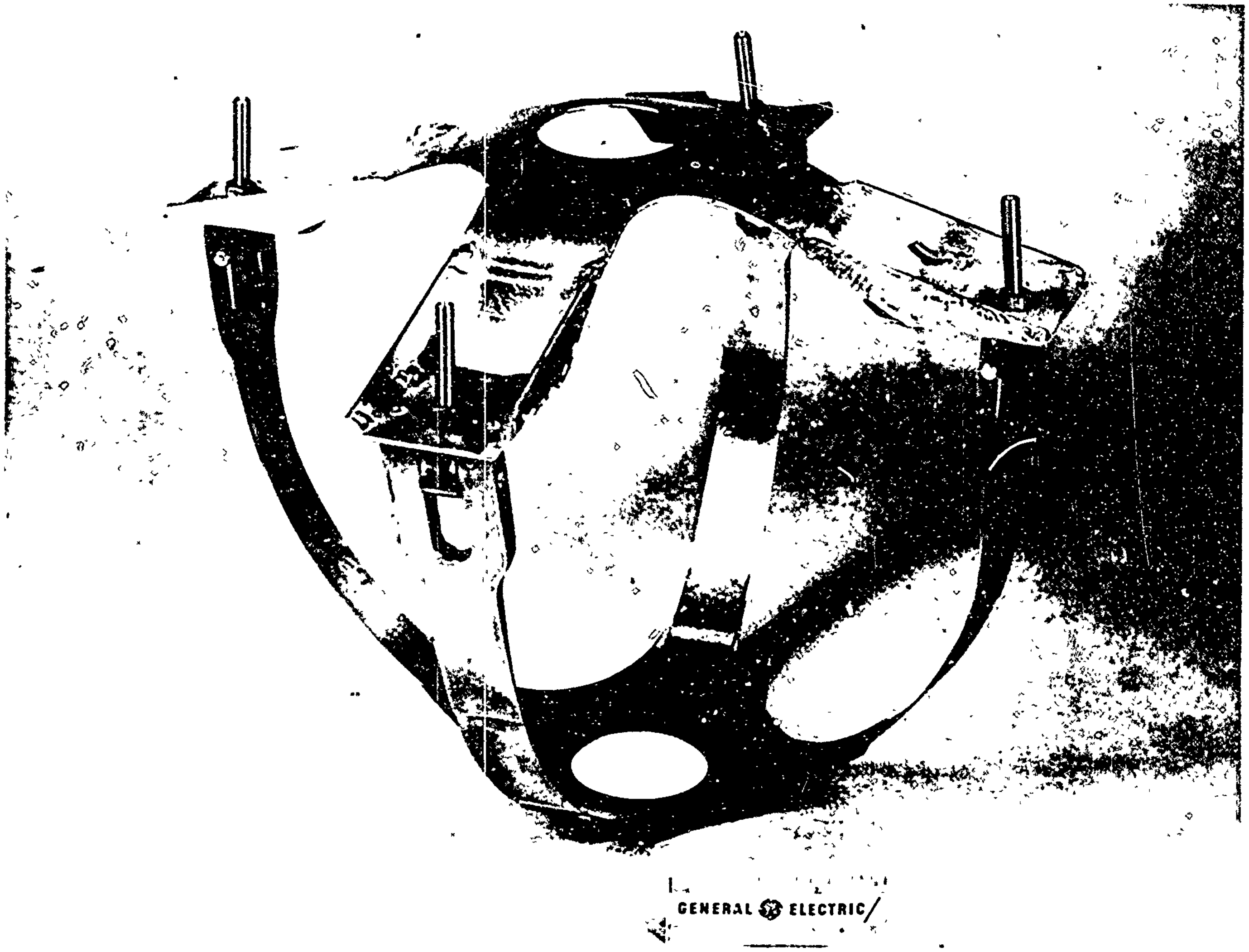


Figure 113. Tank Support Straps for use in Frame Structure, Unfinished Assembly.

Welding was used for joining the basic framed aluminum structure which supported the propellant tank and the sub-assembly plates for the flow control components. It was planned to subject the initial unit to vibration and g-loading tests, using an ammonia tank containing a suitable amount of water.

3. ENERGY MANAGEMENT

Because of deep interest expressed by personnel at APL and WPAFB concerning the performance of the Resistance Jet Microthruster System when the thruster was operated at temperatures lower than its nominal 2000°F, the data appearing on Tables XII and XIII were prepared. They appear in curve form on Figures 114 and 115. Recognizing that although two thrusters are built into one microthruster system, only one is heated at any one time. Thus the total system power includes that to operate only one heated thruster. These data were presented both to APL and WPAFB personnel in separate meetings. The predicted power consumption for a single thruster at various heater operating temperatures is shown on Figure 116.

4. WEIGHTS AND VOLUMES

As the designs progressed, weight and volume became primary considerations. A final design aim (Work Statement Appendix XV) became 3600 cubic inches for the assembly of the tank and flow control components. A single thruster was to be limited to a cylindrical volume of 60 cubic inches. The power and signal conditioner was to be limited to a rectangular volume of 150 cubic inches. For the complete fueled flight package, the weight limit was to be 39 pounds (including 5 pounds of ammonia). A number of design reviews were held and weights and volumes studied. Some changes were made and confidence was expressed that the goals could be met.

5. AMMONIA TANK FILL SYSTEM

A system was devised for transfer ing suitable quality ammonia from a specially prepared and cleaned storage tank, through cleaned transfer lines of suitable diameter and material and into the cleaned ammonia fuel tank. This was to be accomplished after the fuel tank had been evacuated and then back flushed several times with suitable quality ammonia gas. Filters were to be in all lines to assure that the original fuel tank cleanliness level was maintained.

Transfer was to be accomplished simply by a gravity feed process, the storage tank being maintained at a higher elevation than the fuel tank as shown in Figure 117. For transfer of a specific weight of ammonia, the fuel tank could be supported on a scale with flexible lever attached.

Such a system would be used for filling the ammonia tank while the vehicle was in position on its launch pad. A key component was the fill valves located on the skin of the vehicle. This valve, Figure 118 is a leaktight, positive shut-off type. Prior to being selected for incorporation into the system, samples were placed on test, being

TABLE XII

3 AND 6 MICROPOUND

THRUSTER SYSTEM CHARACTERISTICS

290 lb-sec

Temp. °F	System Power Watts	Specific Impulse Sec	Propellant Weight Lbs.
2000	32	125	2.3
1500	20	100	2.9
ROOM	0	60 - 70	4.0

TABLE XIII

200 MICROPOUND

THRUSTER SYSTEM CHARACTERISTICS

800 lb-sec

Temp °F	System Power Watts	Specific Impulse Sec	Propellant Weight Lbs
2000	32	170	4.7
1500	20	130	6.2
1000	6	105	8.1
ROOM	0	75	10.6

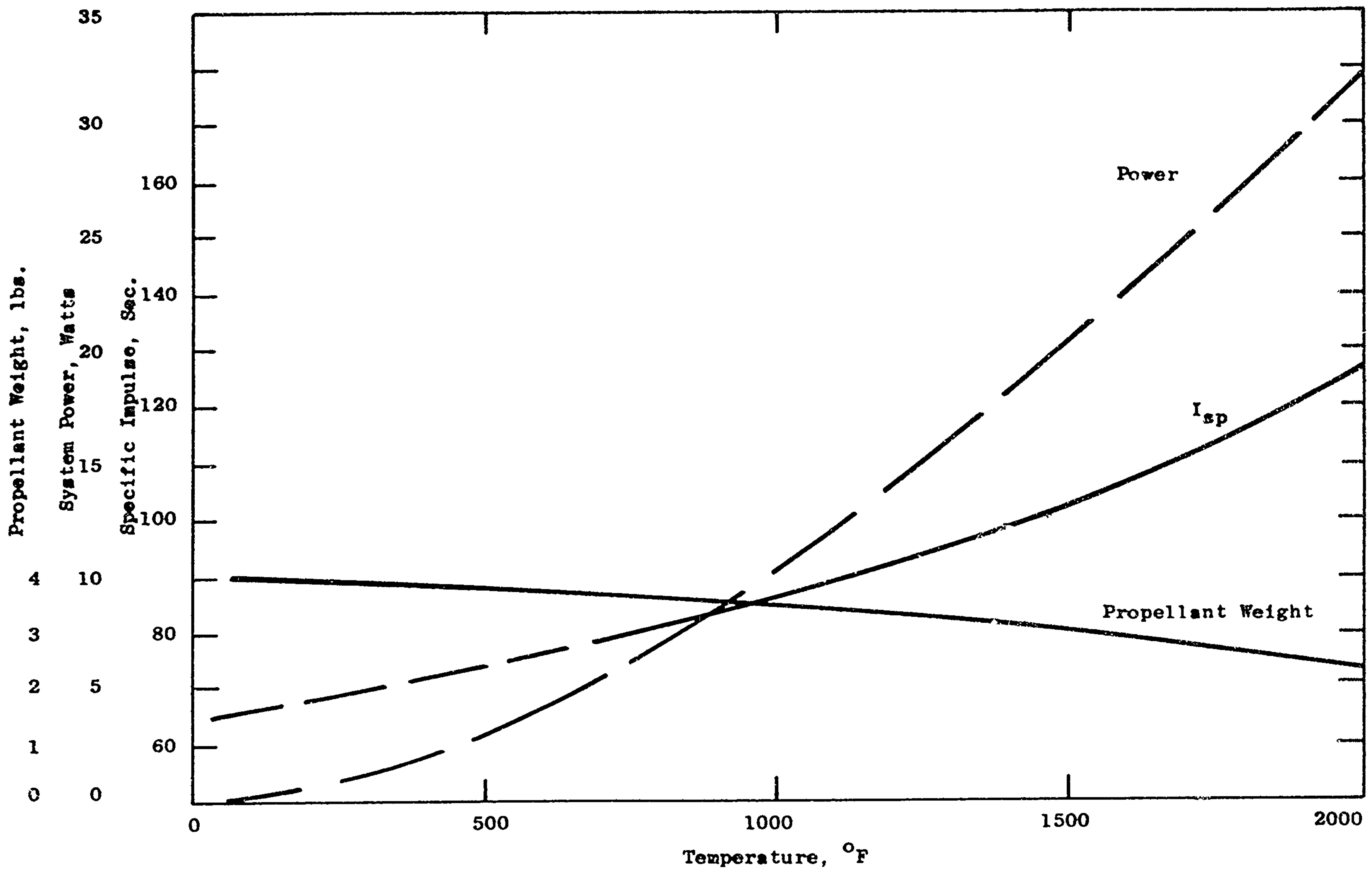


Figure 114. Thruster System Characteristics - 3 and 6 Micropound (Non-Catalyzed) - 290 lb/sec.

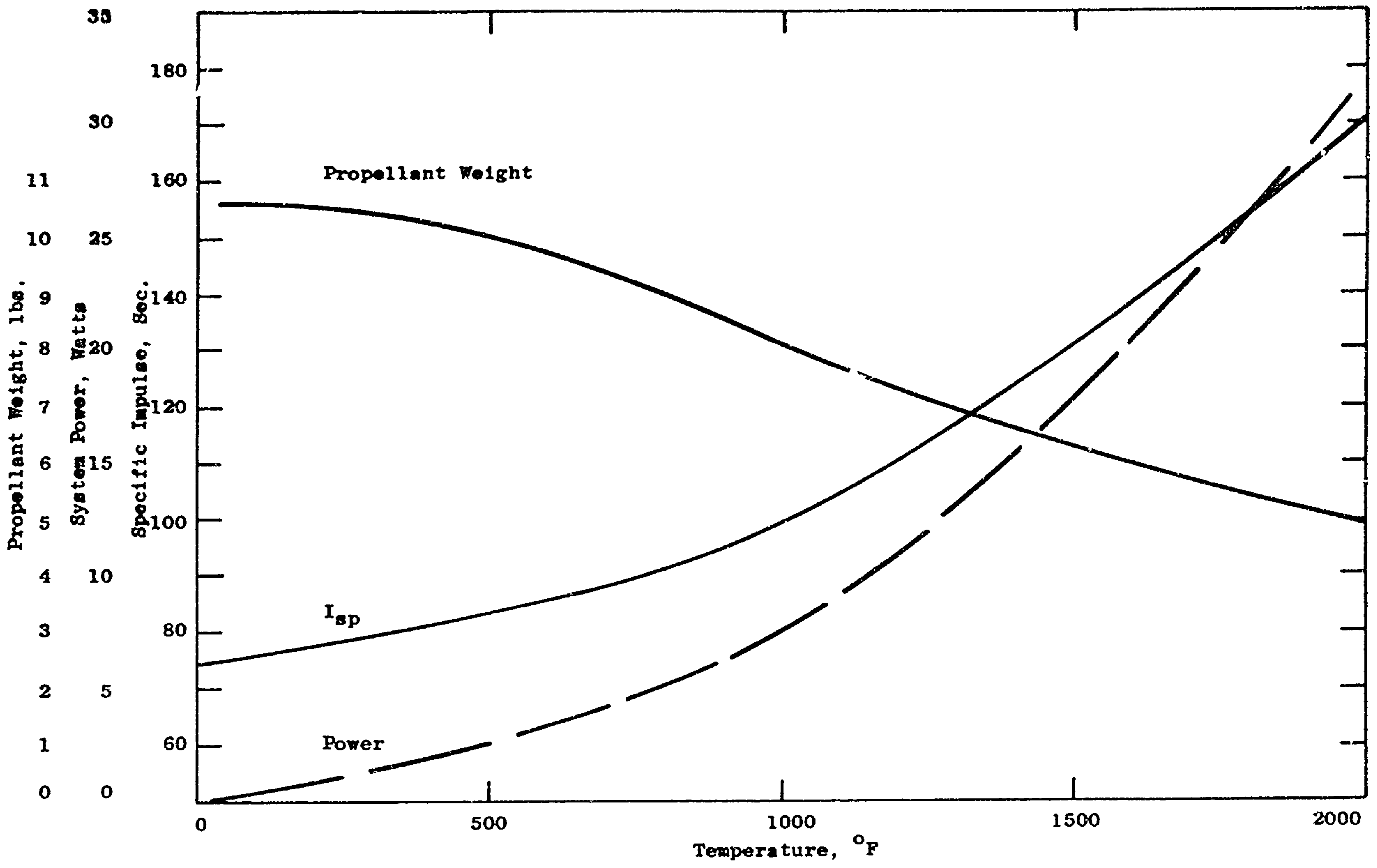


Figure 115. Thruster Systems Characteristics - 200 Micropound (Catalyzed) - 800 lb/sec.

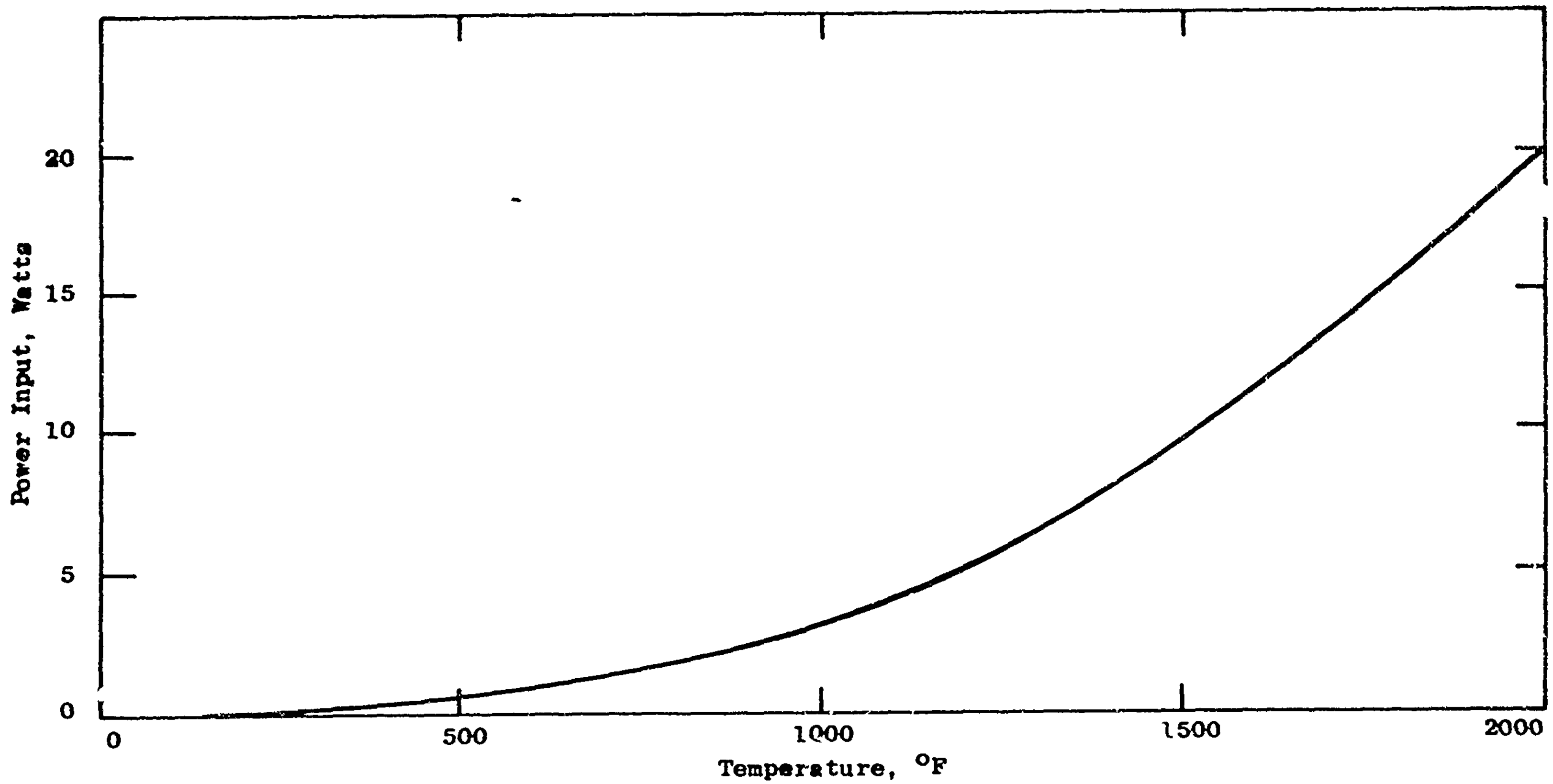


Figure 116. Anticipated Heater Power Requirements, Flight Thruster.

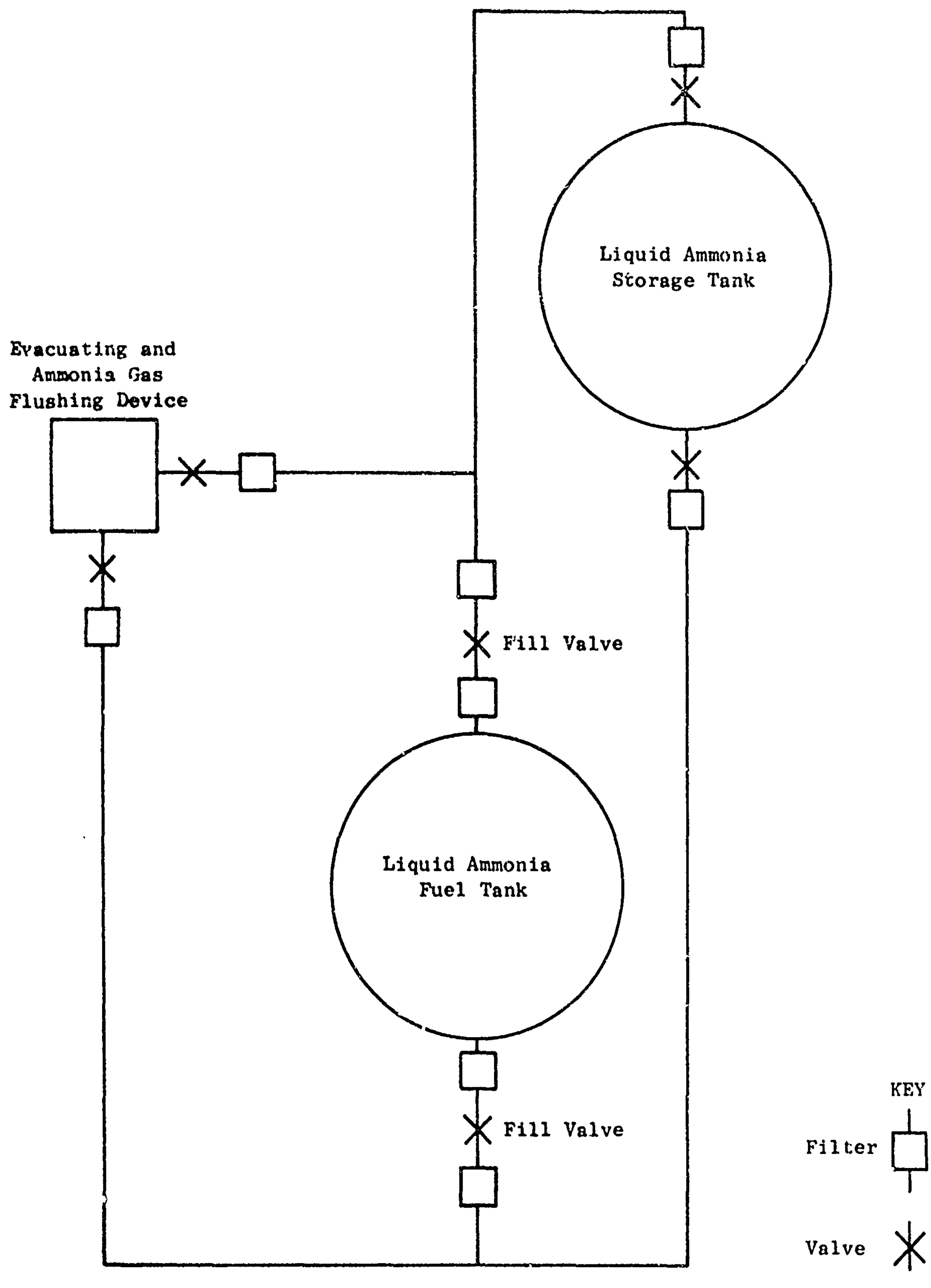
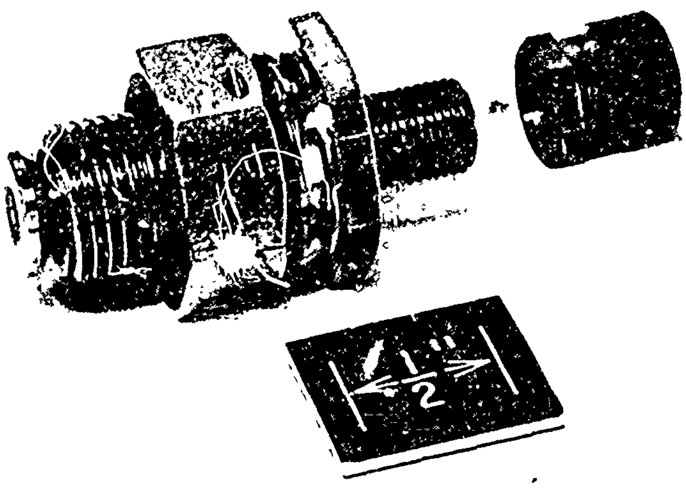


Figure 117. Ammonia Fill Method.



E1280-12

Figure 118. Propellant Tank Ammonia Fill Valve, Positive Shut-off Type.

constantly exposed to liquid ammonia. The valve was periodically opened and closed (closing with a predetermined torque value) and periodically checking them with a mass spectrometer leak detector. After a period of four months, no leak could be detected.

6. MICROTHRUSTER SYSTEM TEST SCHEDULE

A description of the tests to which the Microthruster System would be submitted was prepared and issued February 17, 1967. This was later (September 25, 1968) rewritten to reflect more recent concepts of component and system testing and appears as Appendix XVI of this report. It was planned that this Schedule would be revised as required.

SECTION IX

ISSUED DRAWINGS

The design scope of the Program may be indicated to some extent by a listing of issued drawings (Appendix XVII). It must be appreciated that periodically during the design effort that resulted in such drawings, reviews were held to make certain that the finished products would be suitable for their intended use. Flight quality hardware designs were always examined for minimum weight and suitability for their use.

SECTION X

SUMMARY

Participation in the Dodge-M/Dodge-II program has resulted in the build up of a system that is usable for measuring thrust of suitably-mounted thrusters in the low micropound ranges. Results are repeatable and calibration is straightforward. The Preliminary Evaluation Thrusters served their purpose by providing suitable data on orifices and nozzles, properly sizing them for use in the flight thrusters. The flight thrusters, themselves, were fabricated up to the point of affixing the orifice and nozzle plates. Tests on one unit showed that the electrical power consumption level would be within specified limits. The ammonia fuel tank design was completed and three units were fabricated. Qualification tests on the first of these proved its structural integrity. Plastic models of this tank flown on KC-135 aircraft proved that, in the absence of severe off-axis forces, ammonia gas would be made available to the thruster system. The flow control and flow instrumentation components were selected for flight quality. Tests were performed on certain of these to confirm their complete suitability. The designed assembly of the fuel tank and flow control components resulted in a package well adapted for installation in the satellite. The power and signal conditioner design was carried through to a breadboard stage. It was well directed towards successfully powering and providing for control of the microthruster system. An ammonia tank fill system was devised that provided for easy fueling and refueling of the ammonia tank. A test plan was prepared for qualifying one system for flight use, including a life test of 2000 hours. Plans were also prepared for acceptance testing the flight systems. Weight, volume, power, and operational requirements were well on their way to being realized at the conclusion of the program.

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2. "Determination of Structural Stability of Components When Subjected to Vibration and Acceleration Tests", Report No. 8277-SPPS-119, Missile and Space Division, General Electric Co., Evendale, Ohio, February 16, 1967

3. "Propulsion System Specification, S2P-S-003" The Johns Hopkins University, Applied Physics Laboratory, Scaggsville, Md., October 17, 1966

4. Monthly Progress Reports #1 through #14; January, 1967 to February, 1968, Contract F33615-67-C-1163, General Electric Company, Evendale, Ohio

5. "Space Division Standard Manufacturing Practices", Dwg. No. 7205-9915, The Johns Hopkins University, Applied Physics Laboratory, Silver Spring, Md. September 22, 1965

6. "Quality Control Manual, Resistance Jet Thruster System, Dodge-M", Report No. 08.100.06, General Electric Company, Evendale, Ohio, January, 1967

7. "Installation and Operating Manual, Microthrust Measuring System", November, 1968. Air Force Contract F33615-67-C-1163, General Electric Company, Evendale, Ohio

APPENDIX I

THRUSTER BRAZED ASSEMBLY STUDIES

ILLUSTRATIONS

TABLES

Thruster Braze Assembly Studies

Attachments of propellant feed tubes and orifice plates to resistance jet assemblies are presently achieved through the use of tungsten inert gas (TIG) and electron beam welding techniques. The effective miniaturization of these thruster assemblies makes fabrication of the indicated joints by welding difficult because of poor accessibility and the necessary size of the weld joint. An alternate method for achieving the necessary propellant supply connections is furnace brazing, using a braze filler material which is metallurgically compatible with the basic materials of construction such as Hastelloy-X and Inconel 600 at operating temperatures of 2000°F throughout the anticipated 10,000 hour life of the assemblies; i.e., the braze should not corrode, embrittle, or otherwise adversely effect the reliability of the joint during its operating life.

A study was therefore initiated to determine the feasibility of fabricating appropriate sections of the thruster assemblies by brazing techniques, realizing that the brazed joints must have long-time metallurgical stability and an ability to contain the propellant at 2000°F. Palladium-base brazing alloys were selected for this investigation because of (1) their appropriately high melting points, (2) their previously established ability to produce ductile as-brazed assemblies with nickel-base alloy components, and (3) their potential catalyzing effects on the decomposition of the ammonia propellant. The particular alloys selected were Pd-38Co and Pd-40Ni.

Initial brazing trials and postbraze thermal exposures were conducted in vacuum at 2000°F on samples prepared with the Pd-Co alloy; subsequent sample preparation and testing with the nickel-bearing alloy will be conducted depending on the results obtained from these initial tests. This order of approach in the investigation of brazes was chosen since potentially adverse diffusion effects would be more readily apparent in a braze of a more dissimilar composition from the nickel-base parent metal. Based upon the Hume-Rothery rules for substitutional alloying, more rapid interdiffusion rates were anticipated between the cobalt-containing braze alloy and the nickel-base thruster material than between a high-nickel-content braze and the nickel-base thruster components. Rates of palladium diffusion into the Inconel 600 and Hastelloy-X base metals should be equivalent for both of the selected braze alloys since the concentrations of palladium in each are essentially the same. Thus, if no containment problems were encountered with the Pd-Co braze alloy after long-time, high-temperature exposure, it could be assumed that equivalent or superior results would be found with the Pd-Ni alloy. Should previous testing of the Pd-Co samples dictate, this relative diffusion stability will be verified at a later date by exposing appropriately brazed (Pd-Ni) samples to short-duration thermal cycles and by subsequent microstructural comparisons with similarly exposed Pd-Co specimens.

A total of twelve simulated thruster brazed joint samples were prepared using the Pd-Co braze. Six samples were brazed in each of two configurations; the Hastelloy-X and Inconel 600 base metals were alternated within each group to permit selection of more superior alloy for the eventual application. Two configurations were selected to provide different lengths of brazed joints and different braze filleting geometries. The first will affect joint leakage while the second will show the selective corrosive effects of varying amounts of locally accumulated braze alloy.

Since it is possible that corrosive braze-base metal reactions may result in failure of the thin-walled propellant feed tubes either during brazing or as a result of deterioration by diffusion processes, a different fabrication method is also being considered. This approach entails the electron beam welding of thin-walled tube segments to small cylindrical body sections and subsequent brazing of these sections to a larger assembly, thus isolating the brazing alloy from the tubes. Two braze samples of this type were prepared, again using the Pd-38Co braze filler material: Hastelloy-X was used for the main body in one sample while Inconel 600 was used in the second.

Details of Sample Preparation

Representative braze joints between small-diameter, thin-walled Inconel 600 tubes and bodies of both Hastelloy-X and Inconel 600 were prepared using the Pd-38Co braze alloy. Configurations A and B are shown in Figure I-1. The samples prepared by the combination electron beam welding-brazing processes are depicted in Figure I-2 as Configuration C. Table I-I outlines the scope of work being performed and summarizes the materials, joint configurations, current test time at 2000°F, and status of helium leak test results obtained to date on various joints.

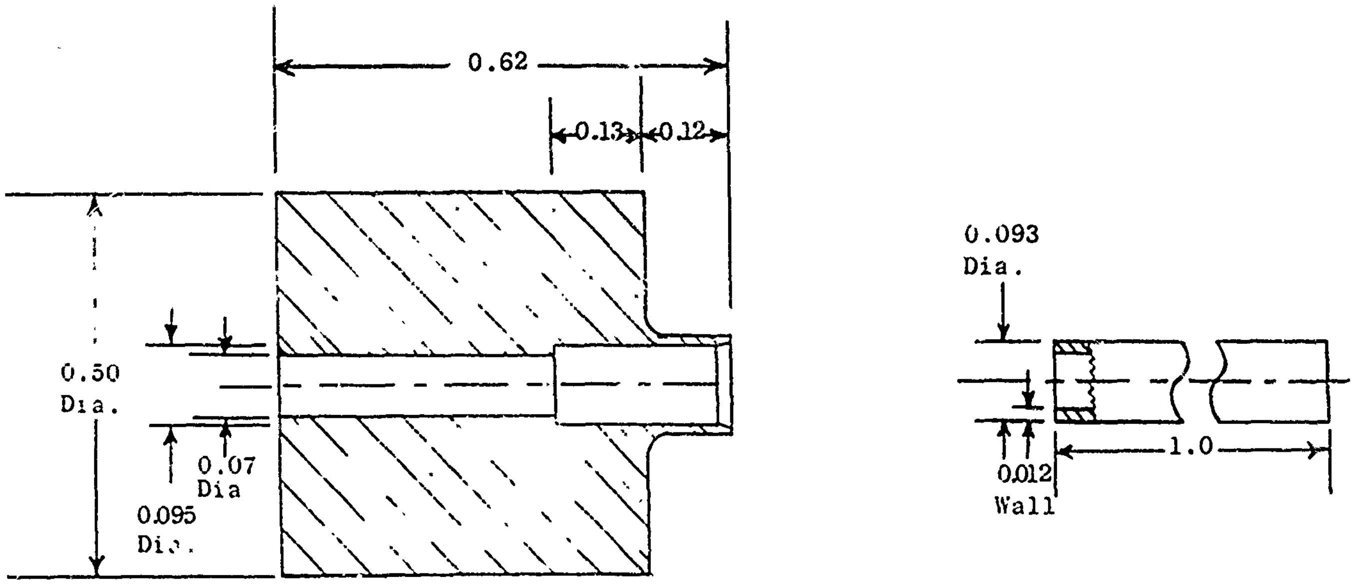
Individual components of the various specimens were cleaned in preparation for brazing or welding by the following sequential processes:

1. Degrease with acetone and ethyl alcohol; dry.
2. Pickle in a solution containing 50% H_2O, 25% HNO_3, 25% HCl (% by volume).
3. Rinse with tap water.
4. Rinse with distilled, deionized water.
5. Flush with ethyl alcohol; dry.

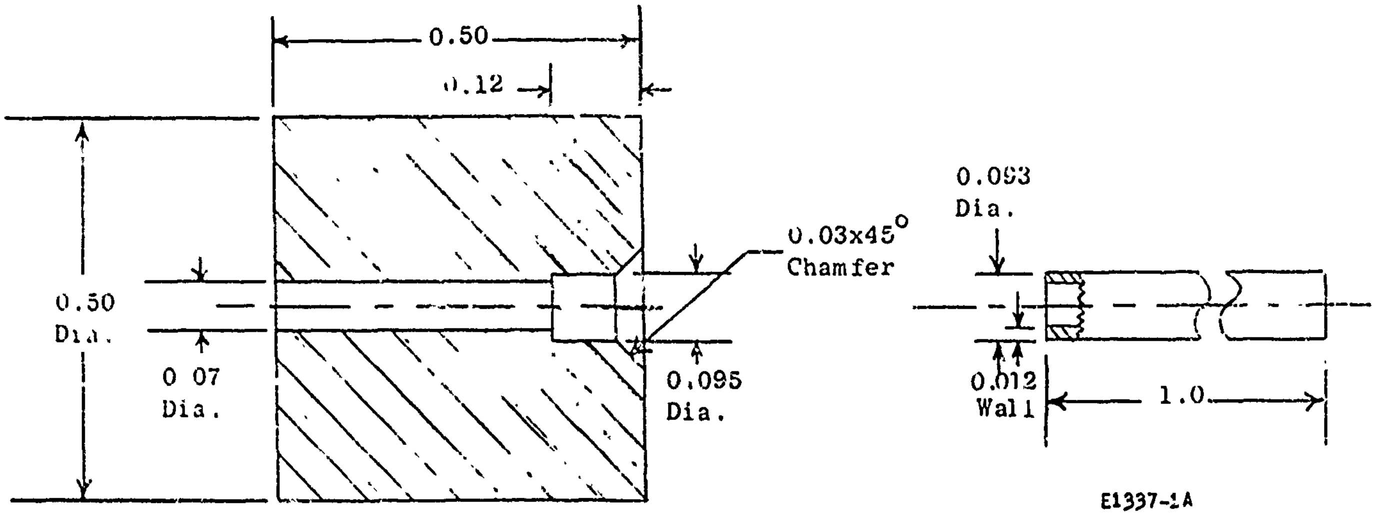
Electron beam welding was performed in the chamber depicted in Figure I-3; the brazing operations were conducted in the vacuum furnace shown in Figure I-4. Brazing alloy was applied to the joints in wire form (0.020-inch diameter) and held in place, prior to brazing, with Nicrobraz Cement. All samples were subsequently brazed at 2275-2285°F for three minutes in the above vacuum furnace; the furnace pressure, during the heating and cooling cycles, was maintained at less than 5 x 10^{-5} torr. Following the individual brazing and/or welding operations, each joint was helium mass spectrometer leak tested and found to be leak tight.

Results

As stated previously, the maximum anticipated operating life of the thruster assemblies at 2000°F, is 10,000 hours. Therefore, the testing schedule for the sample joints was set up to attain that ultimate goal, and thus determine a measure of the reliability as well as thermal stability of the brazing process for fabrication of such assemblies. The actual testing consisted of exposing the sample joints to a temperature of 2000°F in a vacuum environment (5 x 10^{-5} torr) for times ranging from 200 hours to 10,000 hours. At appropriate times, the testing will be, or already has been, interrupted for removal of representative samples, these will be studied metallographically to determine microstructural changes. After



Configuration B



Configuration A

Figure I-1 Braze Sample Joints - Configurations "A" and "B".

TABLE I-I

SUMMARY OF BRAZE[1] SAMPLE CONFIGURATIONS, MATERIALS COMBINATIONS,

POSTBRAZE THERMAL EXPOSURES AND HELIUM LEAK TEST RESULTS

Sample Number	Materials[2] Tube	Main Body	Joint Configuration[3]	Total Time at 2000°F (hours)	Leak Test Results[4]
1	Inco 600	Hast. X	A	200	Leak Tight
2	Inco 600	Hast. X	B	200	Leak Tight
3	Inco 600	Inco 600	A	200	Leak Tight
4	Inco 600	Inco 600	B	200	Leak Tight
5	Inco 600	Hast. X	A	1000	Leak Tight after 200 hrs. Not Leak Tight after 1000 hours.
6	Inco 600	Hast. X	B	1000	Leak Tight
7	Inco 600	Inco 600	A	1000	Leak Tight
8	Inco 600	Inco 600	B	1000	Leak Tight
9	Inco 600	Hast. X	A	1700+	Leak Tight after 1000 Hrs.
10	Inco 600	Hast. X	B	1700+	Leak Tight after 1000 Hrs.
11	Inco 600	Inco 600	A	1700+	Leak Tight after 1000 Hrs
12	Inco 600	Inco 600	B	1700+	Leak Tight after 1000 Hrs.
13	Inco 600	Hast. X (Inco 600 insert) (EB Weld)	C	1700+	Leak Tight after 1000 Hrs.
14	Inco 600	Inco 600 (Inco 600 insert) (EB Weld)	C	1700+	Leak Tight after 1000 Hrs.

---

[1] All samples brazed in vacuum at 2275-2285°F/3 minutes.

[2] Compositions of basic alloys:

Inco 600: Ni - 76.0; Fe - 7.2; Cr- 15.8.
Hast. X : Ni - 44.5; Fe - 20.0; Cr - 23.0; Mo - 9.0; Co - 2.5; N - 1.0.

[3] Configurations A, B, and C are depicted in Figures 1 and 2; braze alloy is Pd-38Co.

[4] Helium leak tests performed with a General Electric Type M-60 Mass Spectrometer, leak tight indicates no leakage greater than 5×10^{-10} std cc's/sec of helium at the various joints.

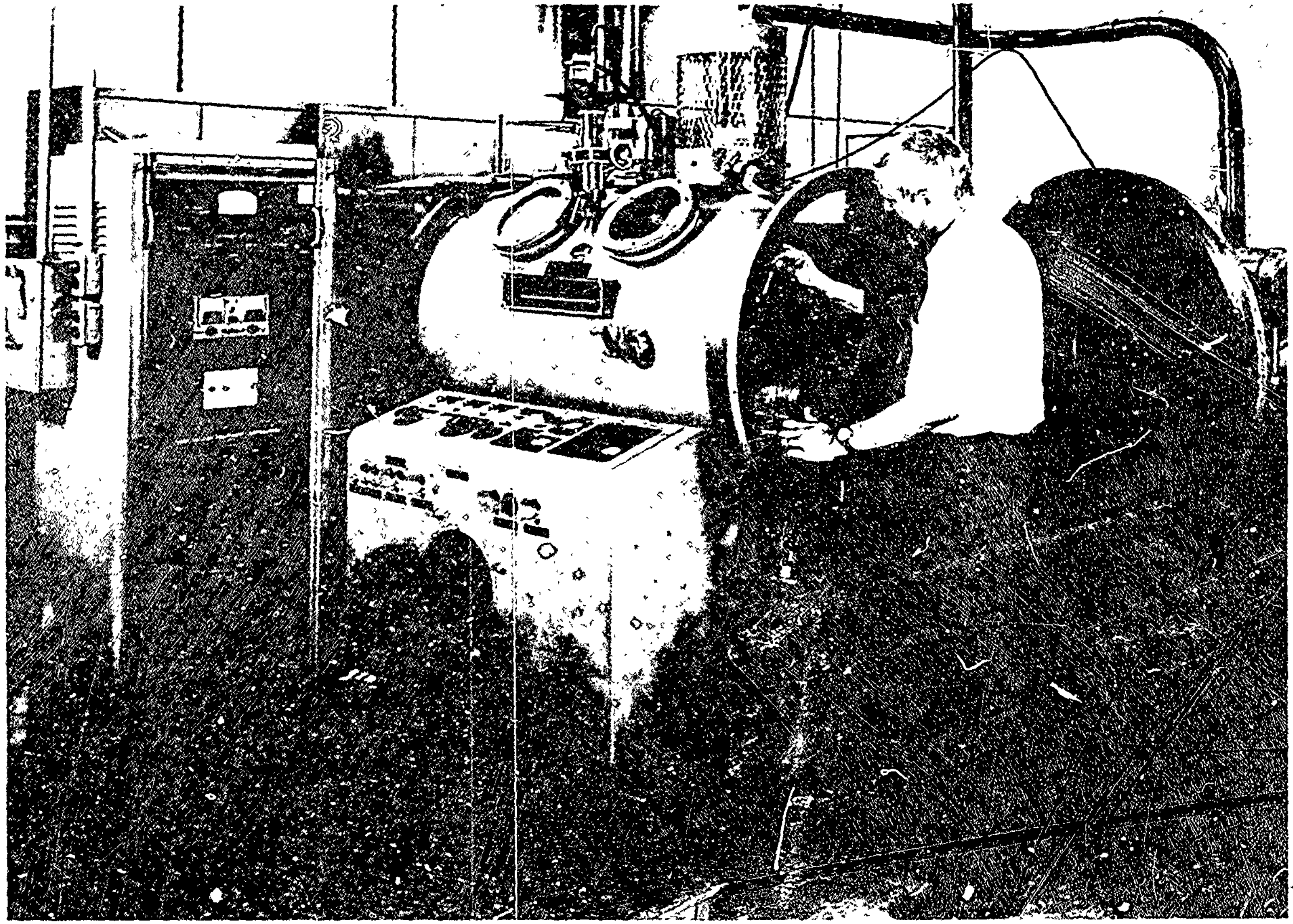


Figure I-3. 30 KV Electron Beam Welding Machine.

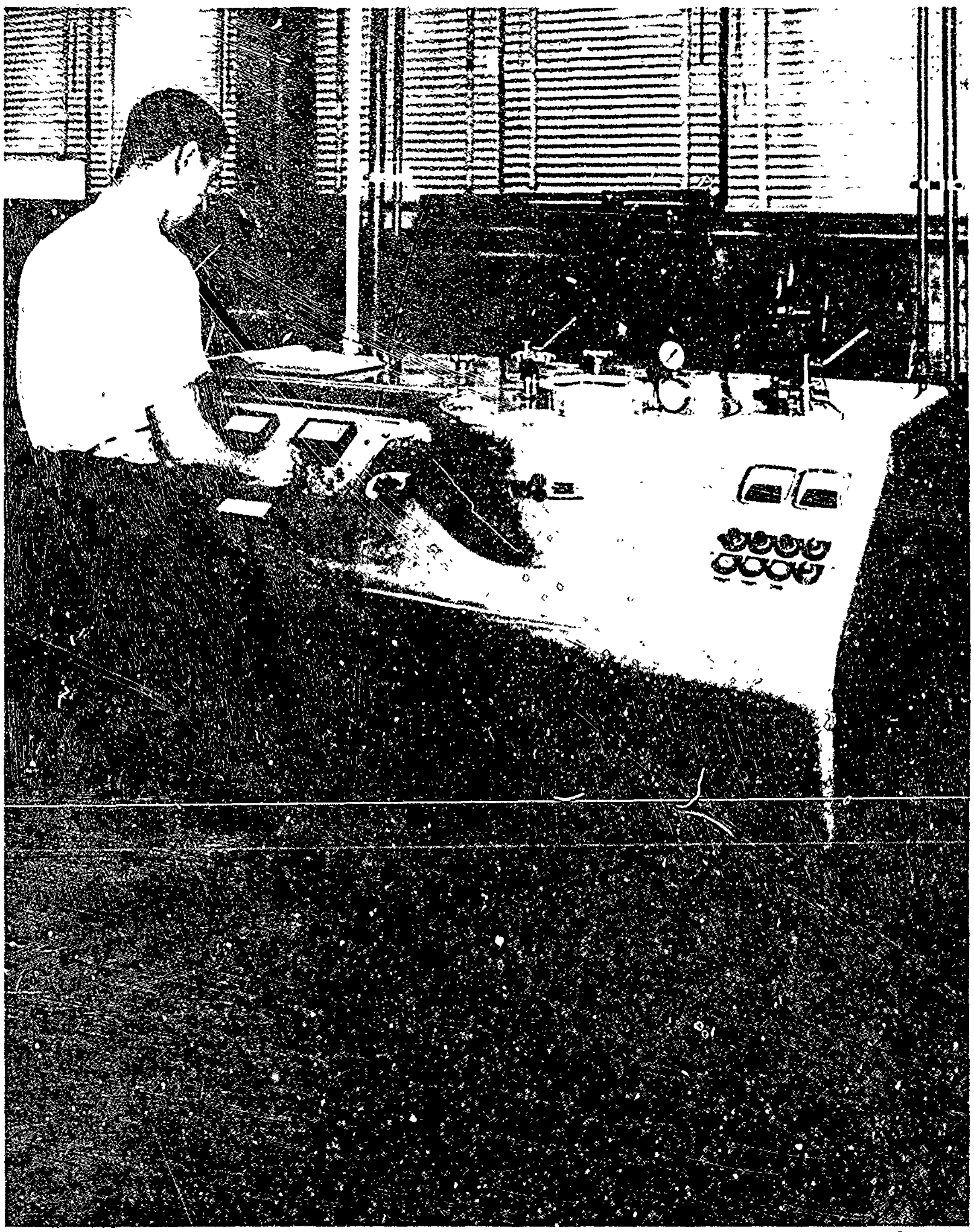


Figure I-4. 4500°F Cold Wall Vacuum Furnace.

200 hours and 1000 hours of exposure all samples were helium leak tested as shown in Table I-?. These results show that all samples were leak tight after 200 hours; after 1000 hours all were still leak tight with the exception of Sample #5 (a Configuration A joint consisting of an Inconel 600 tube brazed to a Hastelloy-X body). Note that Sample #9 is a duplicate of Sample #5 and that no leakage was detected in this duplicate sample at the conclusion of the 1000 hour exposure. The results of Sample #5 suggest that the above materials and configuration combination, such as Samples #1, 5, 9 may not produce reliable joints. Verification of this statement depends on microstructural examination. The preparation and metallographic evaluation of samples exposed for 200 hours and for 1000 hours have not been completed yet; hence, further comments regarding joint leakage, thermal stability and materials selection cannot be made at present.

To date, the joints remaining on test have accumulated 1700 hours of exposure at 2000°F. The next scheduled break in testing will occur after 5000 hours; metallurgical studies on samples tested 200 hours and 1000 hours will be completed by then. If results obtained by that time indicate that the 10,000 hour goal is meaningful, two samples will be returned to test, i.e., electron beam welded and brazed samples #13 and #14. Brazing with the Pd-40Ni alloy will also be initiated at that time if appropriate. The welded and brazed samples were selected for the longest time exposure because it was believed that the combination processing would provide the optimum reliability for the intended applications.

APPENDIX II

ELECTRICAL DESIGN OF A 24V. DC, 2000°F Mo-50Re HEATER

Design Requirements

Voltage:	24 volts
Wattage:	20 watts
Temperature:	2000°F
Wire Core ID:	0.210 inch
Wire Core Length:	1.5 inches
Space Between Turns:	1.5 x wire diameter

Calculation Procedure

1. Determine required wire resistance:

$$\text{Watts} = EI$$

$$I = \frac{20 \text{ watts}}{24 \text{ volts}} = 0.834 \text{ amps}$$

$$\text{Watts} = I^2R$$

$$R = \frac{20 \text{ watts}}{(0.834 \text{ amperes})^2} = 28.8 \text{ ohms}$$

2. Calculate hot resistance of various wire sizes wound on 0.210-inch diameter core for length of 1.5 inches with a spacing between wires equal to 1.5 times wire diameter.

Example for 32 AWG wire (0.00795-inch diameter):

$$\text{No. of Turns} = \frac{1.5 \text{ inches core length}}{2.5 \times 0.008 \text{ inch wire diam.}} = \frac{1.5}{0.020}$$

$$= 75 \text{ turns.}$$

$$\text{Wire Length} = 0.210\text{-inch core diam.} \times \pi \times 75 \text{ turns.}$$

$$= 49.5 \text{ inches length of 32 AWG Mo-50Re wire.}$$

Mo-50Re wire resistivity at 2000°F is 20.9×10^{-6} ohm inches.

Recommended Heater Electrical Design

24 Volts	33 AWG Mo-50Re Wire, (7.08 mils diam.)
0.834 Amperes	
20 Watts	84 turns, 54 in. active heater wire length
2000°F	28.8 ohms total resistance
1.5 inch heater length	
0.210 inch heater coil diam.	

APPENDIX III

INVESTIGATION OF THE FASTENING OF THRUSTER HEATER LEADS AND THRUSTER THERMOCOUPLE LEADS TO THE THRUSTER ELECTRICAL CONNECTOR RECEPTACLE

The Problem

The attachment of the Mo-50Re heater and Chromel-Alumel thermocouple leads from the thruster to the gold-plated pins of an electrical connector must be of such a nature as to minimize local high resistance at the joint. High-resistance joints, if present at these locations, could produce considerable power losses and/or erroneous temperature readings.

Method of Approach to Solution

Connections having low resistances may be reliably achieved by intimate metallurgical bonding of the different materials using either brazing or soldering fabrication techniques. Soldering is generally used to manufacture transition joints of this nature; however, the 500°F maximum anticipated service temperature and operation in a vacuum environment (< 10^{-2} torr) precludes its use since soldering alloys contain elements, such as cadmium and zinc, which have prohibitive vapor pressures at that temperature. Brazing the heater and thermocouple leads to the gold-plated pins would eliminate the problems of metal vaporization and transfer if the proper braze alloys were selected. Such an alloy should have the lowest practical melting point (approximately 1000°F). It should be compatible with the base materials, and it should be available in forms suitable for application to the joints. The alloy selected for evaluation was a BT, Braze, ASTM BAg-8, (72Ag-28Cu) alloy which has a melting point of 1435°F.

Efforts Towards Solution and Solution Obtained

Experiments were conducted with the BT alloy to determined whether sound joints could be fabricated without producing detrimental effects in the body of the electrical connector. The following braze process factors were investigated: (1) braze alloy wettability on the Mo-50Re and Chromel-Alumel wires, (2) plating of the wires should poor wettability to the bare wires be encountered, (3) methods of heating joints to braze flow temperature, and (4) possible fluxless brazing.

Initial efforts were extended toward pretinning of the bare wires by dipping samples into a crucible containing molten braze alloy and a protective layer of NF Silver Brazing Flux. Bath temperatures, ranging from 1450°F to 1750°F, and soak times up to five minutes were used in these trials. Complete lack of wetting was observed regardless of prior treatments such as pickling, grit blasting, and mechanical polishing.

Because of a lack of melting, the method of pretinning to prepare the wires for brazing was abandoned, and the alternate approach of nickel plating the ends of the wires was initiated. Samples of the Mo-50Re, Chromel, and Alumel wires were nickel plated by both the electroless and conventional electroplating processes. These samples were subsequently used in the brazing trials.

The construction materials of the electrical connector prevented uniform furnace heating of the entire electrical connector and its connector pins. Thus, a method for locally heating the pin-wire joints to the brazing temperature was required. It was necessary to bring the pin-wire joint rapidly to brazing temperature and to complete the brazing in a short period of time. A graphite-tipped, tweezer-type resistance heater was selected in preference to the use of an oxyacetylene torch because better temperature control could be maintained. The nickel-plated wires were bent into the shape of an eyeloop and inserted through the holes in the gold-plated pins of a sample electrical connector. For brazing in air, a small quantity of the NF silver brazing flux was applied to the respective joints and to the braze alloy wire used. The individual joints were then heated to the braze temperature by gripping them with the tweezers. Input power to the heater and resulting joint temperatures were varied by adjustment of an external variable electrical power unit until desired conditions were attained. Satisfactory brazed joints between all three wires (Ni plated-electroless and conventional) and the gold-plated pins were thus fabricated. Temperatures at the connector body were maintained below 200°F during all of these brazing cycles.

Since the presence of flux in the braze joint could serve as a possible source of contamination during thruster operation, a method to eliminate the possible entrapment of flux in the joints during brazing was also briefly investigated. Electroplated samples of each of the three wires were brazed to the gold-plated pins of another sample connector in a vacuum-purged inert atmosphere glove box; argon was used as the backfill gas. The tweezer resistance heater was placed inside the chamber along with the wires, braze alloy, and other required equipment. Temperature was again controlled by adjustment of the Powerstat located outside the glove box. Again, satisfactory brazing of all three wires was observed, and, since no flux was used, this method of fabrication appears superior to brazing in air, in which case flux is required.

APPENDIX IV

TEST PROCEDURE
INCOMING INSPECTION OF RESISTANCE JET HEATERS
WITH INCONEL 600 SHEATHS

GENERAL ELECTRIC SPACE POWER & PROPULSION SECTION CINCINNATI, OHIO 45215	PROCEDURE NO. TP- 0104-00-A
# TEST PROCEDURE	ISSUE DATE November 9, 1967

PROGRAM RESISTANCE JET HEATER DEVELOPMENT	ORIGINAL CONTRACT
TITLE OF TEST Incoming Inspection of Resistance Jet Heaters with Inconel 600 Sheaths	

PREPARED BY: H. A. Williams H A Williams DATE 10/31/67

APPROVED BY: W. H. Townsend WH Townsend DATE

APPROVED BY: W. N. Neiman W. N. Neiman DATE 10/25/67

APPROVED BY: M. L. Bromberg M L Bromberg DATE

APPROVED BY: DATE

ISSUED BY: D. A. Pritchett MAIL ZONE N-21 PHONE 5729
DRAFTING

SP 1162 (9/66)

SUPERSEDES SPECIFICATION NO. NEW	DATED	SEE SPPS INSTRUCTIONS SERIES 03.111

SP 1073 A

Incoming Inspection of Resistance Jet Heaters with Inconel 600 Sheaths - CONTINUED	DATE 11-9-67	NO. TP-0104-00-A

1 SCOPE

1.1 This procedure covers the test requirements for incoming inspection and initial processing of resistance jet heaters with Inconel 600 sheaths.

2 OTHER APPLICABLE DOCUMENTS

2.1 None.

3 PROCEDURE

3.1 Identification. Permanently mark serial number on heater in two places per Spec. 03-0034-00-A, Method 2D.

3.1.1 Around the periphery at the top of the sheath.

3.1.2 At the bottom of the end plug.

3.2 X-ray. X-ray, two views, 90° to each other. Identify the heaters by putting the serial numbers on the film.

3.2.1 Count the number of turns of element wire for one heater per lot.

3.2.2 Determine that active heater length meets print requirements.

3.2.3 Determine that heater wire spacing is uniform.

3.2.4 Measure the distance between the closed end plug of the cartridge and last turn of element wire. Check this dimension with that shown on applicable procurement drawing.

3.2.5 Check that MgO core is centrally located in the sheath.

3.3 Mechanical Measurements. Measure mechanical dimensions of heater.

3.3.1 Outside diameter in three (3) places (both ends and mid-point) using 1" micrometer caliper.

3.3.2 Concentricity or bow by placing heater on flat surface and inserting shims at center of heaters.

3.4 Hardness. Four (4) impressions around the periphery, 90° apart, at the end plug. (To be performed on only one heater from each lot.)

3.5 Electrical Measurements. Electrical measurements with heater elements at room temperature.

3.5.1 Record room temperature.

SP 1073 A Incoming Inspection of Resistance Jet Heaters with Inconel 600 Sheaths - CONTINUED	DATE 11-9-67	NO. TP-0104-00-A

3.5.2 Using G.R. 1650A Impedance Bridge, measure resistance from very end of power leads.

A. Make three (3) readings.

B. Adjust the lead holding means per reading.

3.5.3 Repeat 3.5 with thermocouple wire (if so equipped).

3.6 Insulation Resistance. Insulation resistanc, at room temperature and 2000°F.

3.6.1 Record room temperature.

3.6.2 Using Megger, measure (at 50 v and 500 v DC) the resistance between:

A. One power lead and outside sheath;

B. One thermocouple lead and outside sheath;

C. One thermocouple lead and one element lead;

D. Two power leads.

3.6.3 Place heater in vacuum furnace, pump pressure down to 1 x 10^{-5} mm Hg or less. Attach one ground lead to outside sheath. Connect the two power leads, ground leads, and thermocouple leads (if so equipped) to feedthroughs.

A. Increase furnace temperature from room ambient to 2000°F gradually so that pressure does not rise above 4 x 10^{-5} mm Hg. (This generally takes approximately 1/2 hour.)

B. Hold at 2000°F ±25°F.

3.6.4 Repeat operation 3.6.2. (Do not repeat 500 v measurement.)

3.6.5 Allow to cool down in vacuum to room temperature.

3.7 Thermocouple. Heater internal thermocouple checkout (if so equipped) at 2000°F. Combine this operation with operation 3.6.3 when possible.

3.7.1 Tack Pt-10Rh thermocouple junction to closed end of heater. Wrap heater in one layer of tantalum foil. Place heater in vacuum furnace, pump pressure down to 1 x 10^{-5} mm Hg or less. Connect all thermocouples and power leads to feedthroughs.

A. Increase furnace temperature from room ambient to 2000°F gradually so that pressure does not rise above 4 x 10^{-5} mm Hg. (This generally takes approximately 1/2 hour.)

B. Hold at 2000°F ±25°F for 1 hour.

C. Check output of internal thermocouple against that of the external thermocouple.

D. Allow to cool down in vacuum to room temperature.

E. Examine sheath for cracks.

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Incoming Inspection of Resistance Jet Heaters with Inconel 600 Sheaths - CONTINUED	11-9-67	TP-0104-00-A

3.8 Repeat operations 3.3, 3.4, 3.5, and 3.6 except operations 3.6.3, 3.6.4, and 3.6.5.

3.9 Binocular inspect all heaters for sheath cracks, power lead wires for defects, and the thermocouple wires for defects. Measure power lead wires and diameter of thermocouples wires.

3.10 At request of Design Engineer, cut open one heater per lot with high-speed, miniature grinder for internal inspection.

3.11 Periodically this Test Procedure will be reviewed in view of the continued gathering of inspection data to determine if some of the operations can be limited to one-per-lot basis, specifically, operations 3.2.3, 3.6.3, and 3.8.

3.12 Attach copy of log sheet to each heater until actual fabrication into thruster.

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APPENDIX V

RESULTS OF INCOMING INSPECTION TESTS
ON A 24V. DC, 2000°F, Mo-50Re HEATER

LOG SHEET	(A) NUMBER LS- 0104-00-A
(B) PROGRAM RESISTANCE JET HEATER DEVELOPMENT	
(C) TITLE OF TEST Incoming Inspection of Resistance Jet Heaters with Inconel 600 Sheaths	(D) CONTRACT NO. Various - Resistance Jets
(E) LOCATION OF TEST	(F) TEST PROCEDURE USED TP-
(G) PART IDENTIFICATION NO. 119C2621, 119C2634, 47C143254, 47C143255 47C143256, 941D463	(H) PART SERIAL NO. 47C 143255

S/N 92 Project No.

	Value	Date	Initials
1. Purchase Order No. & Item Number	037-284812 Item #2	6-9-67	WBJ
2. Date received and P/O Number	037-284812 Item #2	8-4-67	WBJ
3. Lot Number			
4. S/N Applied	92	8-7-67	WBJ
5. Heater Description			
a. Drawing Number	47C 143255	8-7-67	WBJ
b. Voltage	24		
c. Wattage	20		
d. Sheath Material	INCONEL 600		
e. Element Wire Material	Mo-50 Re		
f. Power Leads Material	Mo-50 Re		
g. Core Material	MgO		
h. Fill Material	BN		
i. Thermocouple Material			
6. X-rays Made			
a. Number of Turns	88	8-14-67	WBJ
b. Active Heater Length	1 9/16	8-14-67	WBJ
c. Check Heater Wire Spacing Uniformity	OK	8-14-67	WBJ
d. Distance Between Closed End Plug & Last Turn of Element Wire	1/16	8-14-67	WBJ
e. Check that Core is Centered	OK	8-14-67	WBJ

TESTED BY	APPROVED BY
☐ PART ACCEPTED ☐ PART REJECTED	
SIGNATURE DATE	SIGNATURE DATE

	Value	Date	Initials
7. Mechanical Measurements Completed			
a. Outside Diameter			
1. End	0.3734"	8-14-67	WBZ
2. Mid-Point	0.3731"	8-14-67	WBZ
3. End	0.3729"	8-14-67	WBZ
b. Concentricity	0.001"	8-14-67	WBZ
c. Length of Power Leads	4 13/16"	8-16-67	WBZ
8. Hardness Measurement R_c	27.4 Rc	8-14-67	
9. Electrical Measurements Made			
a. Room Temperature	73°F	8-16-67	WBZ
b. Heater Resistance (3 readings)	9.10; 9.15; 9.10 Ω	8-16-67	WBZ
c. Thermocouple Resistance (3 readings)	-	-	-
10. Insulation Resistance Measurements			
a. Room Temperature	74°F	8-23-67	WBZ
b. At 50 v Resistance Between			
1. Power Lead and Outside Sheath	180×10^6 Ω	8-23-67	WBZ
2. Thermocouple Lead and Outside Sheath	-	-	-
3. Thermocouple Lead and Element Lead	-	-	-
4. Two Power Leads	-	-	-
c. At 500 v Resistance Between			
1. Power Lead and Outside Sheath	115×10^6 Ω	8-23-67	WBZ
2. Thermocouple Lead and Outside Sheath	-	-	.
3. Thermocouple Lead and Element Lead	-	-	-
4. Two Power Leads	-	-	-
d. 2000°F			
e. At ~~50~~ v Resistance Between			
1. Power Lead and Outside Sheath	0.13500×10^6 Ω (IMPEDANCE BRIDGE) $< 0.5 \times 10^6$ Ω on megohmeter	8-25-67	WBZ

	Value	Date	Initials
2. Thermocouple Lead and Outside Sheath	–	–	–
3. Thermocouple Lead and Element Lead	–	–	–
4. Two Power Leads	28.3 Ω	8-25-67	nbg
f. At. 500 v Resistance Between			
1. Power Lead and Outside Sheath	$< 0.5 \times 10^6$ Ω	8-25-67	nbg
2. Thermocouple Lead and Outside Sheath	–	–	–
3. Thermocouple Lead and Element Lead	–	–	–
4. Two Power Leads	–		–
11. Thermocouple Checked	–	–	–
12. Sheath Crack Examination	–	–	–
13. Mechanical Measurements Repeated			
a. Outside Diameter			
1. End	0.3734"	9-6-67	nbg
2. Mid-Point	0.3732"	9-6-67	nbg
3. End	0.3731"	9-6-67	nbg
b. Concentricity	0.001"	9-8-67	nbg
14. Hardness Measurement Repeated R_c	$< R_c$ 20	9-8-67	nbg
15. Electrical Measurements Repeated			
a. Room Temperature	73°F	9-8-67	nbg
b. Heater Resistance (3 readings)	9.12; 9.10; 9.13	9-8-67	nbg
c. Thermocouple Resistance (3 readings)	–	–	–
16. Insulation Resistance Measurements Repeated			
a. Room Temperature	73°F	9-8-67	nbg
b. At 50 v Resistance Between			
1. Power Lead and Outside Sheath	183×10^6 Ω	9-8-67	nbg
2. Thermocouple Lead and Outside Sheath	–	–	–
3. Thermocouple Lead and Element Lead	–	–	–
4. Two Power Leads	–	–	–

	Value	Date	Initials
c. At 500 v Resistance Between			
1. Heather Element Lead and Outside Sheath	$125 \times 10^6\ \Omega$	9-8-67	[illegible]
2. Thermocouple Lead and Outside Sheath	–	–	–
3. Thermocouple Lead and Element Lead	–	–	–
4. Two Power Leads			
17. Wire Diameter Measurements			
a. Power Lead	.035	9-8-67	[illegible]
b. Thermocouple Wire	–	–	–

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APPENDIX VI

FUEL TANK DESIGN - MATERIAL

1. SUMMARY

This report is a formal follow-up to my previous verbal recommendations of ASA-6061 alloy as the most desirably aluminum alloy for the tank shells of the proposed anhydrous ammonia storage tanks for the LODGE-M Satellite. Heat-treatable and non-heat-treatable aluminum alloys were considered.[1] The heat-treatable ASA-6061 alloy was selected for use in a gaseous and liquid anhydrous ammonia environment in the temperature range 32° to 120 F. ASA-6061 alloy was chosen because it:

a. Has good mechanical properties in the aged condition.
b. Is compatible with anhydrous ammonia.[2]
c. Is readily TIG welded.
d. Is fabricable into the desired shapes.
e. Is commercially available in sheets of the desired thickness.

ASA-6061 design properties of interest are given in the attached Data Sheet. Recommendations for fabrication and purchasing of tanks are given below.

2. FABRICATION REQUIREMENTS

Recommended procedure for forming spherical tanks from sheet[3] is as follows:

a. Perform metal spinning or hydroforming of sheet in the "O" (soft) condition into hemispheres:
b. Solution treat and quench hemispheres:
c. Rework hemispheres to remove any distortion caused by quench:
d. Weld hemispheres:
e. Artificially age hemispheres to T-6 condition:

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[1] The two classes of alloys differ primarily in that the heat-treatable alloys will age harden if properly treated In the fully aged condition, the heat-treatable alloys generally have considerably more strength than the non-heat-treatable ones.

[2] Care must be taken to avoid the presence of iron or iron oxide in the aluminum ammonia system, see Section III of the Data Sheet.

[3] Selection of starting sheet thickness to obtain a desired final tank wall thickness depends on 1) the ease of spinning or hydroforming; 2) the ease of welding the sheet; 3) the ease of chemical milling the sheet, and 4) the final wall thickness tolerances required. Assuming a final wall thickness of approximately 0.25", a starting sheet thickness of approximately 0.063" would probably be satisfactory. This issue should be clarified with vendor or vendors performing the tank fabrication.

It is desirable to spin or hydroform thicker sheet than the final desired thickness and to chemically mill all but welded area to final desired wall thickness. (It is easier to chemically mill before welding.) The extra material at the weld will provide added strength in this weakest area. To maintain good dimensional tolerances when chemical milling, it is recommended that not more than approximately 0.050" (in depth) of material be removed. Removal of material to this depth can be controlled to within approximately ± 0.002"[4] by chemical milling.

R. A. Ekvall
Physical Metallurgy

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[4] Private communication with R. C. Cowan, North American Aviation, Columbus Division, Columbus, Ohio (PH: BE 1-1851)

APPENDIX VII

FUEL TANK DESIGN - STRENGTH, CAPACITY, WEIGHT, AND HEAT TRANSFER

1. PROPELLANT REQUIREMENT

The total propellant requirement for the Dodge-M system was determined as follows:

a. Station Changing

Thrust Level, F = 200 micropounds
Specific Impulse, I_{sp} = 200 seconds
Duration, T = 0.25 years 0.787×10^7 seconds

$$\text{Propellant Required} = \frac{F \times T}{I_{sp}}$$

$$= \frac{200 \times 10^{-6} \# \; 0.787 \times 10^{7} \text{ seconds}}{200 \text{ seconds}}$$

Propellant Required = 7.87 lbs

b. Orbit Correction

Required Thrust - 200 #/sec
Specific Impulse, I_{sp} = 200 sec

$$\text{Propellant Required} = \frac{200 \# \text{ sec}}{200 \text{ sec}}$$

Propellant Required = 1.0 lbs.

c. Station Keeping

$$\text{Thrust Levels, } F = \begin{matrix} 3 \text{ micropounds} \\ 6 \text{ micropounds} \end{matrix}$$

Specific Impulse, I_{sp} = 150 seconds
Duration T = 1 year = 3.15×10^7 seconds

$$\text{Propellant Required} = \frac{F\ T}{I_{sp}}$$

$$= \frac{9 \times 10^{-6} \text{ lbs} \times 3.15 \times 10^{7} \text{ sec}}{150 \text{ sec}}$$

Propellant Required = 1.89 lbs

d. Total Propellant Required

Station Changing	=	7.87
Orbit Correction	=	1.00
Station Keeping (2 Units)	=	3.78
		12.65 # of NH_3

2. VOLUMETRIC REQUIREMENTS OF SUPPLY TANK

a. The volume of the total ammonia required to perform the station changing, orbit correction, and station keeping mission requirements is as follows:

Total Wt. of NH_3	=	12.65 lbs.
Leakage Allowance	=	Zero
Density of NH_3 @ 100°F	=	36.4 lbs/ft^3

$$\text{Volume of } NH_3 = \frac{12.65 \text{ lbs}}{36.4 \text{ lbs/ft}^3}$$

$$\text{Volume of } NH_3 = 0.347 \text{ ft}^3 = 600 \text{ in}^3$$

b. The volume of a 11.0 inch sphere is

$$\text{Volume} = \frac{\pi D^3}{6} = \frac{\pi\, 11.0^3}{6}$$

$$\text{Volume} = 698 \text{ in}^3$$

c. Ullage

$$\text{Ullage} = \text{Volume of tank} - \text{volume of } NH_3$$

$$\text{Ullage} = 698 - 600 = 98 \text{ in}^3$$

$$\%\ \text{Ullage} = \frac{98}{698} = 14\%$$

d. The location of the ammonia liquid-vapor interface at 1-G conditions before launch can be found for a 14% ullage equal to 98 in^3 by solving the equation for a spherical segment:

$$V = 1/3\pi\, h^2\, (3\, r-h)$$

where h = distance from liquid-vapor interface to top of supply tank
r = radius of the storage tank - 5.5 inches
V - ullage volume 98 in^3

Solving the above equation for h, we obtain a vapor-liquid interface loaction 2.61 inches below the vapor outlet.

3. STRESS ANALYSIS

a. Proof Pressure

The maximum internal operating pressure the supply tank will be subjected to during the normal anticipated service is 211.9 psi (saturated vapor pressure at 100°F). Based on a proof pressure 1.5 times the operating pressure, the required wall thickness for a 11.0 inch diameter sphere fabricated from 6061-T6 aluminum having a 0.2% offset yield strength of 35,000 psi is:

$$t = \frac{PD\ (FS)}{4\ Sy}$$

$$t = \frac{(211.9)\ (11.0)\ (1.5)}{4 \times 35{,}000}$$

$$t = 0.0249 \text{ inch}$$

b. Burst Pressure

The minimum wall thickness required for a burst pressure 3 times the operating pressure based on an ultimate strength of 42,000 psi for 6060-T6 aluminum alloy is:

$$t = \frac{PD \times FS}{4 \times \sigma(\text{Ultimate})}$$

$$t = \frac{(211.9)\ (11.0)\ (3.0)}{4 \times 42{,}000}$$

$$t = 0.042 \text{ inch}$$

Since the wall thickness calculated to meet the burst pressure requirement is higher than the wall thickness required for the proof pressure, the recommended minimum wall thickness of the sphere is 0.042 inch thick.

c. The welding of 6061 aluminum components in the heat treated T4 or T6 condition results in reducing the yield and ultimate strength in the heat affected zone. This is primarily a function of the welding speed. If the weld is made rapidly enough, the heat effected zone can be aged to develop a strength almost equal to the strength of fully heat treated material (6061-T6). However, since some uncertainty does exist in the ability to obtain 100% recovery, the wall thickness at the joint between the two hemispheres was increased to twice the thickness of the unaffected wall. This results in a circumferential band 1-inch wide and 0.084 inch thick.

d. During the filling operation of the tank with ammonia, the tank may be evacuated with a one atmosphere external pressure. The critical external pressure that can cause failure by elastic instability can be calculated by

$$P = \frac{2E\ t^2}{r^2 \sqrt{3\ (1 - \mu^2)}}$$

E = Young's modulus of Elasticity

μ = Poisson's Ratio

t = Wall thickness

r = Radius of sphere

$$P' = \frac{2 \times 10 \times 10^{6} \times (0.042)^{2}}{5.5^{2} \sqrt{3 (1 - .33^{2})}}$$

$$P' = 708 \text{ psi}$$

It is, therefore, safe to evacuate the sphere under a one atmosphere external pressure.

4. HEAT TRANSFER

The heat required to vaporize the liquid ammonia during operation of the thrusters will be transferred from the vehicle enclosure walls to the tank by thermal radiation and conduction through the tank support structure. The operating temperature of the tank will be a function of the temperature of the environment and the rate of vaporization of the ammonia. The primary heat transfer requirement is that the temperature of the tank be maintained for an ammonia saturation pressure sufficiently high to deliver vapor at the thruster at the design pressure. A tank temperature as low as 22°F with a saturation pressure of 50 psia would be sufficient to operate the thrusters at the design thrust.

The heat transfer rate required for the station changing operation which has a maximum flow rate of 1×10^{-6} lbs/sec is equal to:

$q = W \Delta H$

W = Mass flow rate, 1×10^{-6} lbs/sec

ΔH = heat of vaporization, 508 BTU/hr at 70°F

$q = 1 \times 10^{-6}$ lbs/sec at 508 BTU/hr

$q = 5.08 \times 10^{-4}$ BTU/sec

$= 1.83$ BTU/hr.

The rate of heat transfer to the 11-inch diameter tank in a 70°F environment is shown in Figure VII-1. At the design flow rate of 1×10^{-6} lbs hr., a temperature difference of 17°F between the tank and the environment will be required to maintain an equilibrium thermal balance of 1.8 BTU/hr. The tank temperature will operate at 52°F with a saturation vapor pressure of 92 psia.

5. ESTIMATED WEIGHT OF PROPELLANT TANK

Tank

Shell		1.64
Center Bank		0.08
Cones (Screen)		0.43
Cones Support		0.06
Vapor Tube		.01
Teflon Disk		.05
	Total	2.27 lbs

Propellant

Station Changing	7.87
Orbit Correction	1.00
Station Keeping	3.78
Total	12.65

Total weight of Tank and Propellant 14.92 lbs.

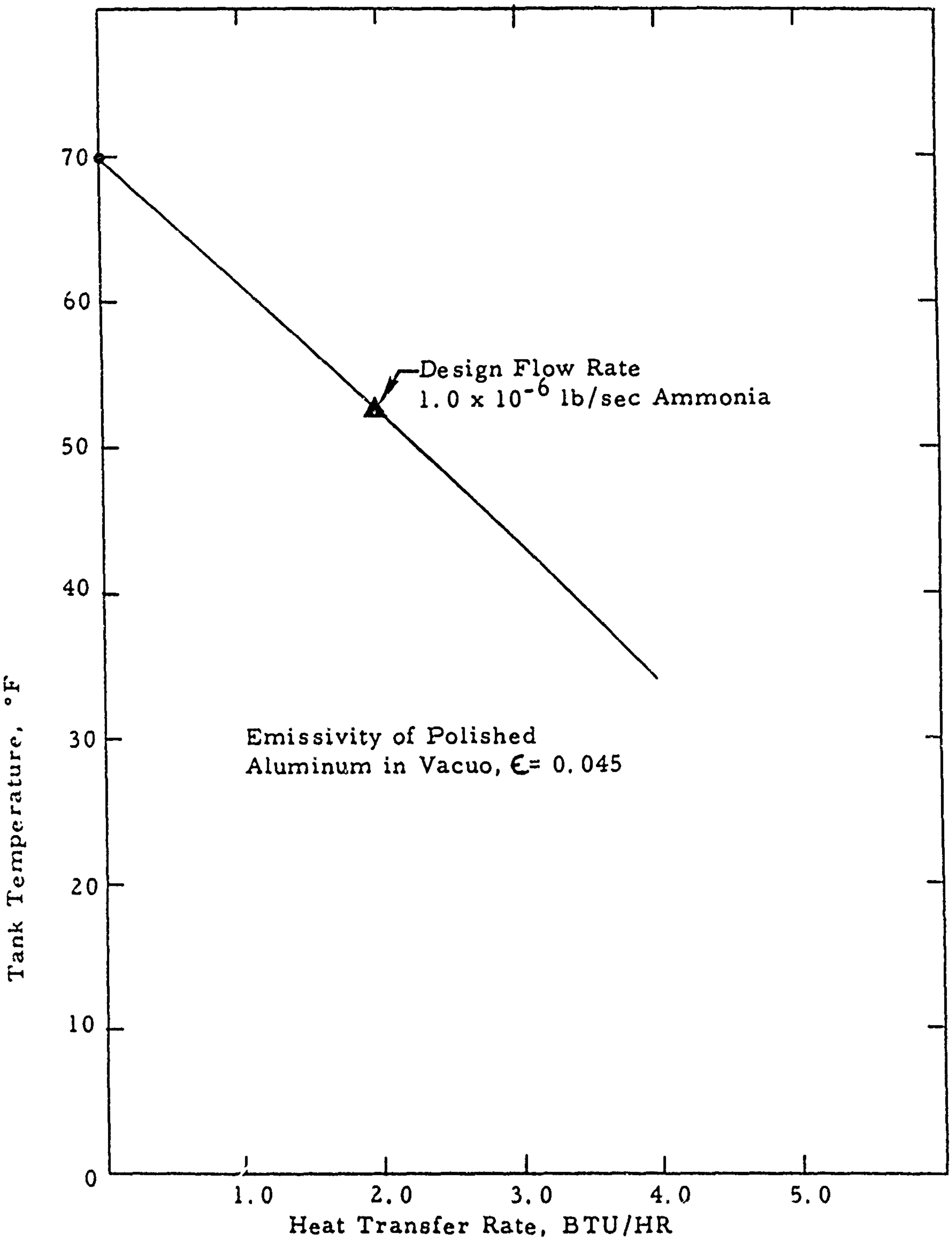

Figure VII-1

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APPENDIX VIII

QUALIFICATION TEST PROCEDURE
FOR
SOLENOID VALVES

(DESIGNED TO G.E. SPECIFICATION 04-0017-00-A)

QUALIFICATION TEST PROGRAM FOR SOLENOID VALVES

AT SPPS/EVENDALE

1.0 DEFINITION

A series of tests listed and described below comprising a Qualification Test Program for solenoid valves procured for the project. This test program applies to solenoid valves specified by the engineer in charge of this program.

2.0 LIST OF TESTS IN ORDER OF PERFORMANCE

2.1 High Potential Test, per Paragraph 4.1.

2.2 Physical Examination Test, per Paragraph 4.2.

For convenience, the following group of tests is referred to as "Functional Tests."

2.3 Proof Pressure Test, per Paragraph 4.3.

2.4 Static Leakage Test, per Paragraph 4.4.

2.5 Pressure Drop and Gas Flow Test, per Paragraph 4.5.

2.6 Response Test Time, per Paragraph 4.9

2.7 Pull-in Voltage Test, per Paragraph 4.6.

2.8 Drop-out Voltage Test, per Paragraph 4.7.

2.9 Coil Resistance Test, per Paragraph 4.8.

2.10 Insulation Resistance Test, per Paragraph 4.10.

2.11 Power Consumption Test, per Paragraph 4.11.

2.12 Temperature Rise Test, per Paragraph 4.12.

For convenience, the following group of tests is referred to as "Environmental Tests."

2.13 High Temperature Test, per Paragraph 4.13.

2.14 Low Temperature Test, per Paragraph 4.14.

2.15 Post-Temperature Functional Test, repeat "Functional Tests" 2.3 through 2.12.

2.16 Sinusoidal Vibration Test, per Paragraph 4.16.

2.17 Random Vibration Test, per Paragraph 4.17.

2.18 Acceleration Test, per Paragraph 4.18.

2.19 Post-Vibration and Acceleration Functional Test, repeat "Functional Tests" 2.3 through 2.11.

2.20 Life Cycle Test, per Paragraph 4.15.

2.21 Cracking Pressure Test, per Paragraph 4.19.

2.22 Burst Pressure Test, per Paragraph 4.20.

3.0 TEST FACILITIES, CONDITIONS AND REPORTS

The facilities used in conducting the required tests shall provide the following:

3.1 Test Conditions

Unless otherwise specified, functional tests shall provide the following conditions:

(a) Temperature 75°F ± 10°F.
(b) Relative humidity: 90% or less.
(c) Barometric pressure: 28 to 32 inches of Hg.

3.2 Measurements and Tolerances:

All measurements shall be made with instruments whose accuracy has been verified and which are calibrated periodically. Maximum allowable tolerances on measuring instruments shall be:

(a) Temperature: ± 3°F
(b) Pressure (above 1 psia): ± 5%

3.3 Chamber Volume:

The solenoid valve being tested plus associated test equipment shall not exceed 50% of the internal volume of the test chamber.

3.4 Temperature Definition

For temperature tests at pressures greater than 10^{-1} torr, the temperature referred to is ambient air temperature. In these tests, the heat sources shall be such that radiant heat is not directed on the solenoid valve. For temperature tests under vacuum (less than 10^{-4} torr), temperatures referred to are mounting surface temperatures and radiation exchange surface temperatures.

3.5 Test Reports

Functional and environmental test data shall be reported to the engineer in charge no later than one week following completion of tests. All data shall be recorded in a hard cover log book.

3.6 Axis Definition

The major axes of the solenoid valve are defined on the specification control drawing.

3.7 Leakage

All leakage measurements shall be made with a helium mass spectrometer using helium gas as the pressurizing medium. When proper test conditions are established, leakage shall be monitored but not recorded until the leakage rate has been established and remained constant for at least ten minutes. Sensitivity of the detector is to be 5×10^{-10} std cc/sec of air.

3.8 Shaker Fixture Solenoid Valve Resonance Test

Prior to performing the vibration test, if an identical solenoid valve and fixture have not been previously vibrated in the machine to be employed for the vibration test, one solenoid valve shall be subjected to vibration in each of its three orthogonal axes through the frequency range from 5 to 2000 cps. If any alterations are made in the machine, fixture or solenoid valve, or if a different machine or fixture is employed, this test shall be repeated. The solenoid valve shall be attached to the test fixture through the attachment points in such a manner as to provide no additional stiffness or restraint other than at the attachment points. The vibration resonance shall be monitored through the entire frequency range in three orthogonal directions at the attachment point of the valve to the fixture. Instrumentation shall be connected but the valve shall not be pressurized during the resonance test. The amplitude reading normal to the direction of excitation shall be no greater than 50% of the maximum level in the direction of excitation. The maximum and minimum amplitude readings taken in the direction of shake at the attachment points shall not differ from each other by more than 50%. The solenoid-valve-fixture-table combination shall be altered until these conditions are satisfied. The frequency sweep shall be manually controlled to insure that all resonances observed at the attachment points are fully excited. Resonance frequencies, bandwidth, and amplification of all resonances shall be varied with frequency such that the maximum amplitude at the attachment points does not exceed the following:

Frequency	Amplitude/Force Level
5 to 20 cps	0.2 inches peak to peak
20 to 2000 cps	0.125 g's

4.0 TEST DESCRIPTION

4.1 High Potential Test - A high potential test will be conducted by applying a potential of 1000 volts a.c. rms, 60 cycles per second mutually insulated parts for a period of one minute. The solenoid valve shall show no evidence of mechanical or electrical failure. A rubber stamp bearing the markings "HIPOT TESTED OK" shall be applied to the solenoid valves. Completion of this test shall be noted on the data sheets submitted with the solenoid valves. Completion of this test shall be noted on the data sheets submitted with the solenoid valves. This test shall not be repeated. This test shall be conducted in accordance with MIL-STD-202.

4.2 Physical Examination - Each solenoid valve shall be inspected for conformance to the Specification Control Drawing plus inspection for workmanship, dimensions, finish, and any defects. Record the results of this inspection.

4.3 Proof Pressure Test - A pressure of 350 psig shall be applied to the inlet and outlet ports (connected together) for a period of one minute: see Figure VIII-1. There shall be no evidence of distortion or damage after removal of the proof pressure. There shall be no evidence of leakage when tested per Paragraph 4.4. Record satisfactory completion of the test, the post-test inspection, and leakage test.

4.4 Static Leakage Test

4.4.1 The solenoid valve shall be pressurized with helium to 200 psig (one ma coil current) and the leakage at the solenoid valve outlet measured by a helium mass spectrometer; see Figure VIII-2. The leakage rate shall not exceed 1×10^{-6} std cc/sec of air. Record the leakage measured.

4.4.2 The solenoid valve inlet and outlet, connected together, shall be pressurized with helium to 200 psig (no coil current) and the leakage measured by a helium mass spectrometer; see Figure VIII-2. The leakage rate shall not exceed 1×10^{-6} std cc/sec of air.

4.5 Pressure Drop Test - The test set-up for measuring pressure drop is shown in Figure VIII-3. The pressure drop across the solenoid valve shall not exceed 5% with an inlet pressure of 3.35 psia, a flow rate of 1×10^{-6} lb/sec, and a gas inlet temperature of 75 $\pm$ 10°F. The differential pressure shall be measured by water manometer or differential pressure gage and recorded. Nitrogen gas shall be used for this test.

4.6 Pull-In Voltage Test - With 200 psia applied to the inlet port of the solenoid valve and the outlet port open to ambient pressure, solenoid voltage shall be increased from 0 volts until full actuation of the poppet occurs. The test set-up per Figure VIII-4 (Response Time Test) may be used for this test. Full actuation must occur at a voltage no greater than 18 volts. Record the voltage at which full actuation occurs. Repeat this test at 50, 100, 150, and 250 psia.

4.7 Drop-Out Voltage Test - With 200 psia applied to the inlet port of the solenoid valve and the outlet port open to ambient pressure, solenoid voltage shall be decreased from 18 volts (minimum) until the poppet is fully closed. The test set-up per Figure VIII-4 (Response Time Test) may be used for this test.

4.7 Drop-Out Voltage Test - Continued - Full closing of the poppet shall occur at a voltage no lower than 6.0 volts. Record the voltage at which full closing occurs.

4.8 Coil Resistance Test - The electrical resistance of the solenoid coil shall be measured by Wheatstone Bridge or equivalent method. The coil resistance shall be 250 ohms minimum* at 75°F (or corrected to 75°F). Record the measured resistance and ambient temperature.

4.9 Response Time Test - The opening response time of the solenoid valve is defined as the time difference between application of voltage to the solenoid coil and full travel of the poppet as indicated by the current trace (electrical response) or by increase of outlet pressure to 63% of the pressure at steady state flow (pneumatic response). These relations are shown in Figure VIII-5.

The closing response time of the solenoid valve is defined as the time difference between removal of voltage from the solenoid coil and full travel of the poppet as indicated by the voltage trace (electrical response) or the first indication of pressure decay (pneumatic response). These relations are shown in Figure VIII-6.

The test set up for measuring response time is shown schematically in Figure 4. Valve inlet pressure for this test shall be 200 psig and the applied voltage shall be 21 volts. Coil voltage, coil current, and outlet pressure shall be recorded on an oscillograph (or equivalent recorder) having a known chart speed, or having a 60 cycle voltage trace recorded as a standard time reference.

Under the stated test definitions and conditions, the opening response time shall not exceed 100 milliseconds and the closing response time shall not exceed 100 milliseconds. Neither the electrical response time nor the pneumatic response time shall exceed the specified time limit. Either method may be used, but the same method shall be used for both opening and closing response times and the method used shall be indicated in the test data. Three readings each shall be taken and the data recorded. Each group of three readings (opening or closing) shall repeat within 10%.

4.10 Insulation Resistance Test - The resistance between the leads of the solenoid and the grounded case shall be 50 megohms minimum at 500 volts d.c. The leads may be connected together for this test. Record the insulation resistance measured.

4.11 Power Consumption Test - Using a suitable voltmeter and ammeter, apply 28 volts d.c. to the solenoid coil. Record voltage, current, and watts

4.12 Temperature Rise Test - With 28 volts d.c. applied to the solenoid coil, record temperature rise of body of valve adjacent to the coil after exposure of 8 hours. Vacuum tank pressure not greater than 10^{-6} torr.

* 250 ohms applies only to Carleton's Model 1809001-3C Valve.

4.13 High Temperature Test - The solenoid valve shall be placed in a thermal chamber and heated to the temperature specified below. At least one valve from each test lot shall be monitored for temperature by means of a thermocouple located at the mounting surface (see Specification Control Drawing). The temperature shall be maintained for eight (8) hours after the component has reached the required temperature. The test shall be performed at ambient pressure and with no (other) pressure applied to the solenoid valve.

Qualification test temperature: + 100°F

4.14 Low Temperature Test - The solenoid valve shall be placed in a thermal chamber and cooled to the temperature specified below. At least one valve from each test lot shall be monitored for temperature by means of a thermocouple located at the mounting surface. The temperature shall be maintained for eight (8) hours after the component has reached the required temperature. The test shall be performed at ambient pressure and with no (other) pressure applied to the solenoid valve.

Qualification test temperature: 0°F

4.15 Life Cycle Test - The solenoid valve shall be connected to an ammonia gas supply and the inlet pressurized to 20 psia. By means of an automatic actuator control, operate the valve for one second "on" and one second "off" duty cycle for a total of 100,000 cycles. Periodically, as specified in the tabulation below, the valve cycling shall be interrupted for a static leakage test per Paragraph 4.4.

4.15.1 After 1,000 total cycles - perform leakage test.

4.15.2 After 3,000 total cycles - perform leakage test.

4.15.3 After 6,000 total cycles - perform leakage test

4.15.4 After 18,000 total cycles - perform leakage test

4.15.5 After 30,000 total cycles - perform leakage test

4.15.6 After 50,000 total cycles - perform leakage test

4.15.7 After 70,000 total cycles - perform leakage test

After the completion of 100,000 cycles, the valve shall be subjected to the "functional" tests listed below in the order shown:

a. Static leakage test, per Paragraph 4.4.

b. Pressure drop and gas flow test, per Paragraph 4.5.

c. Response time test, per Paragraph 4.9.

d. Pull-in Voltage test, per Paragraph 4.6.

e. Drop-out voltage test, per Paragraph 4.7.

f. Coil resistance test, per Paragraph 4.8.

g. Insulation resistance test, per Paragraph 4.10

h. Power consumption test, per Paragraph 4.11.

4.16 Sinusoidal Vibration Test - The fixture and test equipment used to perform this test shall have been tested in accordance with the requirements of Paragraph 3.8 and proven satisfactory. The solenoid valve shall be vibrated as described below:

3.0 g zero to peak from 5 to 2000 cps applied in each of the 3 mutually perpendicular axes.

Vibration to be limited to 1/4 in. single amplitude.

Sweep rate =2 octaves/minute.

4.17 Random Vibration Test

Roll-off 20 cps at 3 db/octave
Flat - 20 cps to 480 cps at 0.2 g^2/cps
Increase from 480 cps to 600 cps at 3 db/octave
Flat - 600 to 1000 cps at 0.25 g^2/cps
Roll-off above 1000 cps at 12 db/octave
Overall amplitude = 17.2 g rms
Duration = 3 minutes in each of 3 mutually perpendicular axes

4.18 Acceleration Test - The solenoid valve shall be mounted on a centrifuge or a spin table and exposed to a steady acceleration force of 9 g's for a period of 3 minutes in both directions along each of its three perpendicular axes.

4.19 Cracking Pressure Test* - Apply pressure to the inlet port of the solenoid valve and gradually increase it until the measured leakage is 10 cc per hr. The pressure applied to the inlet port shall not exceed 300 psia. The differential pressure across the valve for the leakage value of 10 cc/hr shall not be less than 300 psid. There shall be no voltage applied to the solenoid during this test. Record the pressure at which cracking, if any, occurs.

4.20 Burst Pressure Test* - The inlet and outlet of the valve, connected together, shall be pressurized to 600 psig and the pressure maintained for 5 minutes. The valve shall not rupture at this pressure. The valve is not required to operate after this test.

*NOTE: Check witr the engineer in charge before proceeding with this test.

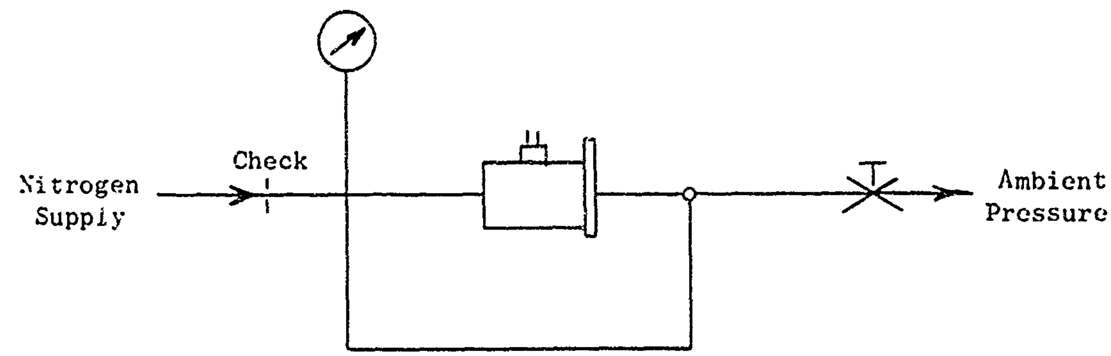

Figure VIII-1 Proof Pressure Test

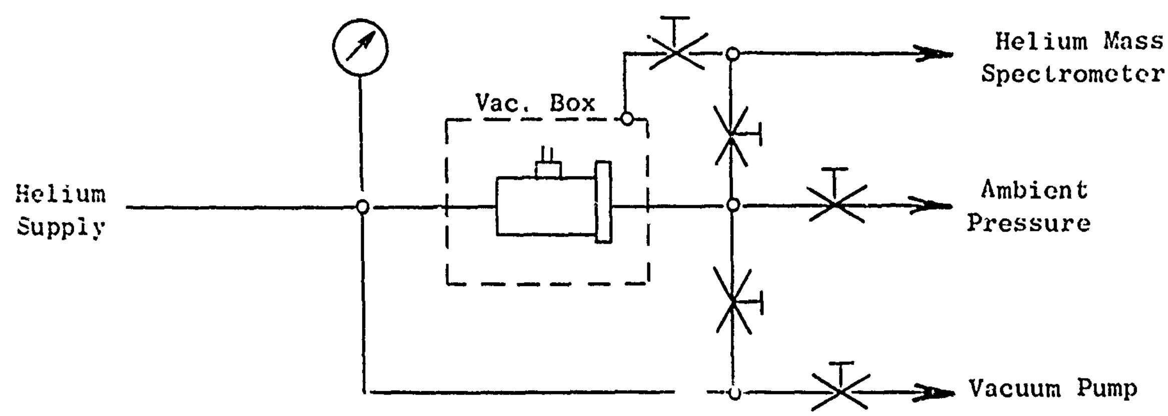

Figure VIII-2 Static Leakage Test

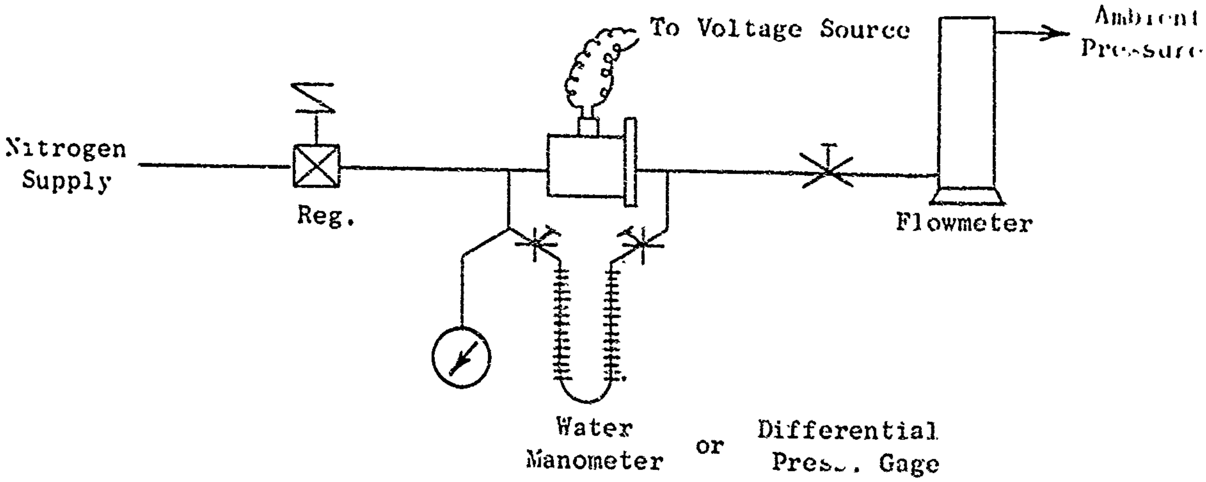

Figure VIII-3 Pressure Drop Test

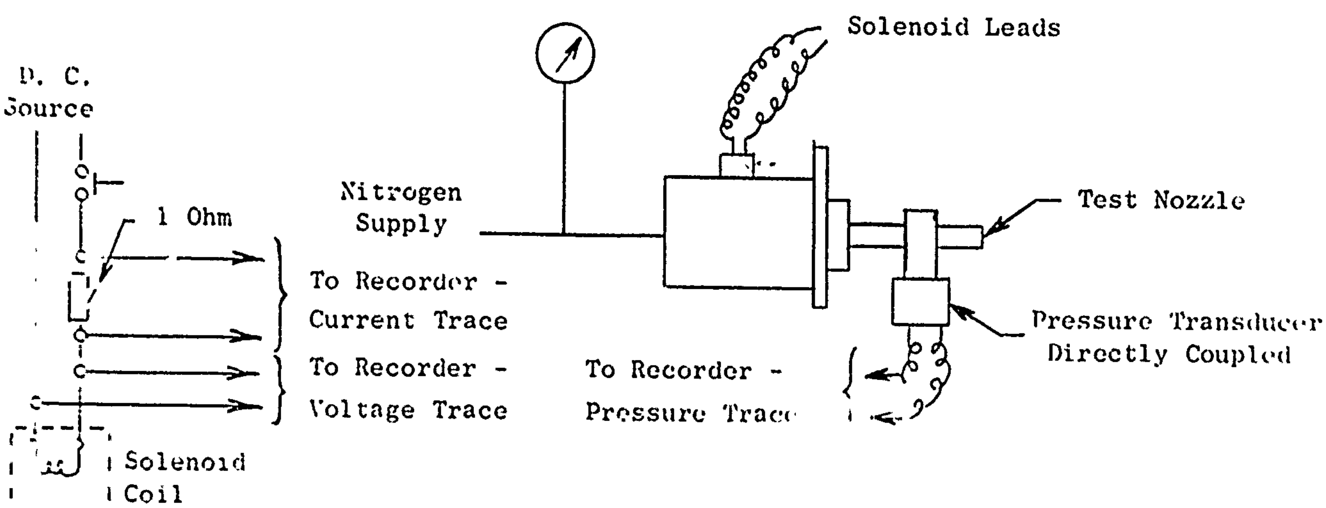

Figure VIII-4 Response Time Test

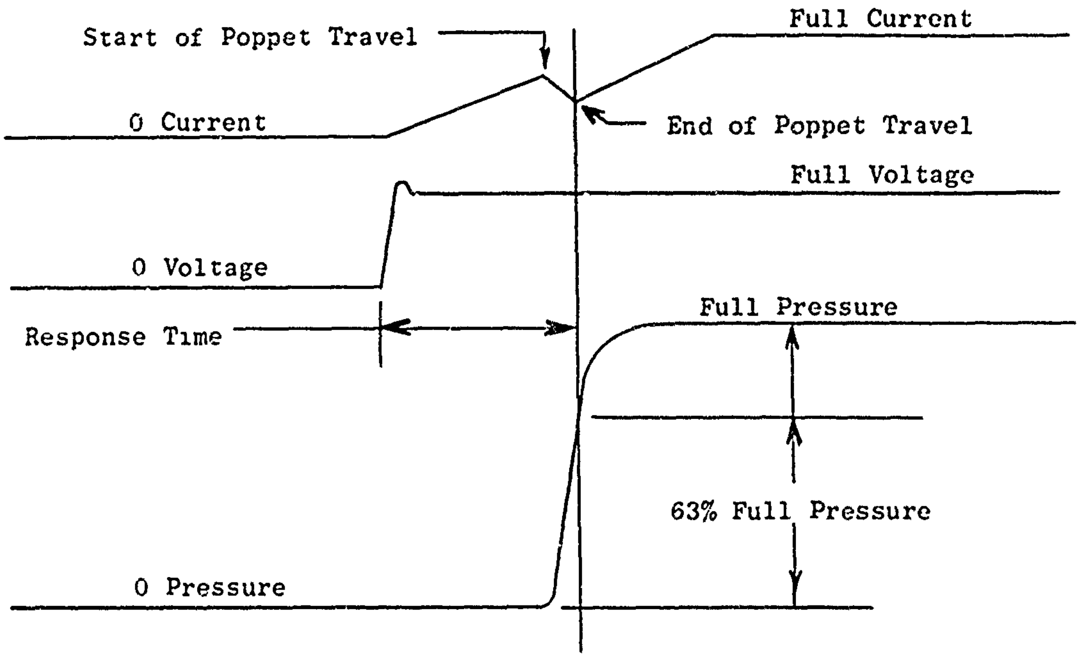

Figure VIII-5 Opening Response Time

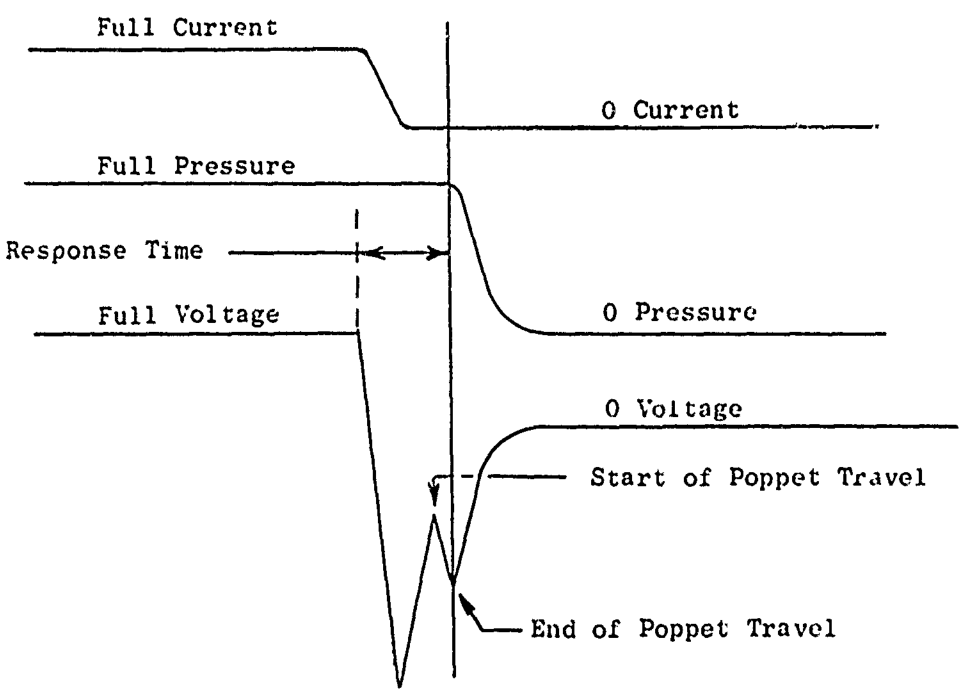

Figure VIII-6 Closing Response Time

APPENDIX IX

QUALIFICATION TEST PROCEDURE
FOR
PRESSURE REGULATORS

(DESIGNED TO G.E. SPECIFICATION 04-0018-00-A)

1.0 PRESSURE REQUIREMENTS

1.1 System Definition - The regulator must meet the requirements of this specification when used in a system consisting of the following: tank; tubing of significant pressure drop; explosive valve with 10 millisecond poppet opening with a .01 in^2 effective area orifice; subject regulator; downstream volume of 1/2 to 1 in^3; solenoid valve; and orifice.

1.2 Inlet Pressure Shocks - The function of the unit shall not be impaired as a result of repeatedly actuating the upstream shut-off valve which has an orifice diameter of .01 inches, and an opening time of 10 ms with 220 psia inlet pressure to the solenoid valve. In addition, the outlet pressure shall not exceed 3.3 psia for station keeping and 14.7 psia for station changing during the transient following actuation of the solenoid valve.

1.3 Regulated Pressure

1.3.1 Station Keeping - For the flow range of 0 to 7 x 10^{-8} lbs/sec and inlet pressures between 30 and 220 psia, the outlet pressure shall be maintained with the limits of 3.3 psia $\pm$ 1% for all environmental conditions down to 30 psia inlet pressure.

1.3.2 Station Changing - For the flow range of 0 to 1 x 10^{-6} lbs/sec and inlet pressures between 30 and 220 psia, the outlet pressure shall be maintained within the limits of 14.7 psia $\pm$ 1% for all environmental conditions down to 30 psia inlet pressure.

1.4 Lockup Pressures - Slow lockup is defined as the outlet pressure obtained when slowly reducing flow from a regulating condition with a hand valve until a no flow condition exists. Surge lockup is defined as the outlet pressure obtained when rapidly reducing flow as by a solenoid valve (10 millisecond poppet movement) from a regulating condition until a no-flow condition exists. After any flow up to 7 x 10^{-8} lbs/sec for station keeping and 1 x 10^{-6} lbs/sec for station changing is slowly or suddenly interrupted with a volume between the unit and the shutoff valve as specified in Paragraph 1.1, the outlet pressure shall not be more than 0.03 psia for station keeping and 0.147 psia for station changing above the regulated pressure at the minimum flow specified for regulation.

1.5 Transient Response - The outlet pressure shall be within the requirements of paragraph 1.3 within 1.0 second after the effect of a flow demand within the flow range specified herein is realized at the outlet. Pressure shall not exhibit peak deviations in excess of 2 psi nor shall the average pressure exceed 0.33 psia for station keeping and 1.47 psia for station changing from the performance specified in Paragraph 1.3. After the 1.0 second, there shall be no periodic or aperiodic oscillation or wander.

2.0 QUALITY ASSURANCE PROVISION

2.1 Classification of Tests - The component shall be subjected to the following classes of tests.

2.1.1 Acceptance Tests - Tests performed to assure conformance to the drawing and this specification. (See Paragraph 2.4)

2.1.2 Qualification Tests - Tests performed to establish the suitability of design, production processes and procedures. (See Paragraph 2.5)

2.2 Test Procedures (All gas flow is to be ammonia.)

2.2.1 Test Quantities - The quantities of regulators required for test are as follows:

2.2.1.1 For Qualification Tests - One (1) sample representative of prime hardware.

2.2.1.2 For Acceptance Tests - These tests shall be performed on all units submitted for acceptance.

2.2.2 Test Conditions - Unless otherwise specified, tests shall be conducted under the following ambient conditions:

a. Temperature — 77 ± 18°F

b. Relative Humidity — 90% maximum

c. Barometric Pressure — 30 ± 2 inches of mercury

2.2.3 Test Facilities

2.2.3.1 Chamber Volume - The unit under test shall not exceed 50% of the environmental test chambers volume.

2.2.3.2 Heat Source - Environmental test heat sources shall be so located that heat will not be directly radiated on the unit under test unless so specified in the individual test.

2.2.3.3 Mounting Provisions - The component shall be attached to test fixtures through the attachment points employed to mount it in service, in such a manner as to provide no additional stiffness or restraint to the unit at other than the attachment points.

2.2.3.4 Temperature Changes - Unless otherwise specified, temperature changes shall not exceed 1.8°F (1°C) per second.

2.2.4 Measurements and Tolerances

2.2.4.1 Measurements - All pertinent signal and environmental inputs to the unit under test and all pertinent performance parameters shall be measured and recorded during all applicable tests. To the maximum extent possible, measurements shall be made in terms of standard units rather than arbitrary dial, indicator, or control settings.

2.2.4.2 Test Conditions - Unless otherwise specified, the maximum tolerances on test conditions shall not exceed the following:

a.	Temperature	± 2.0°C (± 3.6°F)
b.	Humidity	+ 5,-0% (relative)
c.	Pressure when measured by manometers	± 5%
	When measured by ion gages	± 10%
d.	Acceleration	± 10%
e.	Vibration amplitude	
	Sinusoidal	± 10%
	Random	Per Paragraph 2.5.2.4
f.	Vibration frequency	± 2%
g.	Shock amplitude	± 10%
h.	Shock duration	± 10%
i.	Time (except shock duration)	± 5%

2.2.4.3 Tolerance Ratio - Whenever possible, a ratio of not less than 10 to 1 shall be maintained between the tolerance of the measured parameter and the tolerance of the measurement. The tolerance of the measurement shall include basic instrument accuracy and instrument-use errors such as resolution, repeatability and parallax.

2.2.4.4 Calibration - All test instruments shall be under the control of a calibration plan. The plan shall specify the frequency of calibration, accuracy of the calibration standards, and maintenance of calibration records. The calibration records shall be available for General Electric Quality Control inspection at any time.

2.3 Test Documentation

2.3.1 Performance Records - Records shall be made of all data necessary to determine compliance with this specification. These data shall provide criteria for checking satisfactory performance of the unit during testing. Test data shall be recorded before, during and after each test as specified herein. The data shall include, but not be limited to, the following:

a. Date of test.
b. Test program.
c. Type of test.
d. Name, drawing number, serial number, and applicable equipment specification of the unit under test.
e. Test specification and applicable paragraphs.
f. Identification of each parameter measured, specification limits on the parameter, and actual parameter measurement of reading.
g. Name and location of manufacturer of unit under test.
h. Name and location of testing agency.
i. Name of individual conducting test and names of any Engineering, Quality Control, or customer witnesses.
j. Operating time during test including number of operating cycles.
k. Interruptions and causes.
l. Failure report numbers on each failure report written.

m. List of test instruments and equipment including manufacturer's names, model numbers, and identification serial numbers.
n. Calibration data including next calibration date and accuracy of equipment.
o. Record of visual examination performed before, during, and after tests.
p. Copies of all applicable vendor certified data.

2.3.2 Test Data Reports - Test reports, providing the data required by Paragraph 2.3.1 shall be submitted.

2.3.2.1 Qualification Test Data Reports - Test data, in accordance with Paragraph 2.3.1 and 2.3.2 shall be submitted at the completion of each component test program. The test reports shall be distributed to the General Electric SPPS, Design Engineer, and Quality Control Engineer.

2.3.3 Failure Reporting - All failures as defined in Paragraph 2.3.5 shall be reported.

2.3.4 Failure Analysis Reporting - When required, failure analysis shall be performed.

2.3.5 Failures - Any reading, indication, or measurement which is not within the limits specified by this specification when the inputs and environmental factors are within tolerances; and deterioration or corrosion which could in any way prevent the unit under test from meetings its operation requirements; any loose, bent, cracked, or otherwise damaged or improperly adjusted parts, or any evidence of poor workmanship shall constitute a failure.

2.3.5.1 Qualification Test Failures - Testing shall be suspended, and the failure shall be reported immediately to the responsible SPPS design engineer and Quality Control Engineer. The determination whether to repair and retest or reject shall be at the discretion of the Design Engineer.

2.3.5.2 Acceptance Test Failures - Units under test, in which failures are detected during individual tests, shall be rejected. The responsible SPPS Design Engineer and Quality Control Engineer shall be notified. A failure report shall be submitted within 24 hours. Testing of other units in the lot may continue. Rejected units shall be stored in a bonded storage area for inspection.

2.4 Acceptance Test

2.4.1 Visual Examination - Each component shall be inspected to determine compliance with the applicable GE drawing.

2.4.2 Performance Test - Each component shall be tested per the following.

2.4.2.1 Inlet Pressure Shock Test - Perform in accordance with Paragraph 1.2.

2.4.2.2 Proof Pressure Test - With the outlet shut off so there is no flow demand on the unit, apply an inlet pressure of 350 psig for one minute.

Pressurize the inlet and outlet to 350 psig. for one minute. At the end of tests there shall be no evidence of leakage or failure in the regulator. This test shall be performed only once on each unit prior to environmental test.

2.4.2.3 Flow and Slow Lockup Test - At an inlet pressure of 220 psia, read the outlet pressure for flows of 0, 1 x 10^{-8}, 7 x 10^{-8}, 1 x 10^{-8}, 0 lbs sec for station keeping and 0, 0.5 x 10^{-6}, 1 x 10^{-6}, 0.5 x 10^{-6}, 0 lbs sec for station changing. Pressure shall be in accordance with Paragraphs 1.4 and 1.3.

2.4.2.4 Response and Fast Lockup Test - With the system as defined in Paragraph 1.1, make transient recordings of the response of the unit to downstream solenoid valve actuations (in the opening and closing directions) for inlet pressures of 220, 100, and 30 psia and the flow of 7 x 10^{-8} lbs sec for station keeping and 1 x 10^{-6} lbs/sec for station changing with a downstream volume of 1/2 to 1 in.^{3}. The test shall meet the requirements of Paragraphs 1.3, 1.4, and 1.5. Ambient pressure shall be reduced to less than 1 x 10^{-6} mm Hg just prior to the actuation of 220 psia and shall be maintained below this pressure for the remaining actuations.

The recording shall show performance for at least 5 seconds. The response of the transient recording equipment shall be such as to have an attenuation of no more than 3 db at 600 cps.

2.4.2.5 Leakage Test - With the outlet shut off, apply an inlet pressure of 220 psia. Establish a flow of at least 7 x 10^{-8} lbs/sec for station keeping and 1 x 10^{-6} lbs/sec for station changing through the unit by partially opening the downstream valve for approximately one minute. Shut off the downstream valve slowly. Read the outlet pressure. Maintain these conditions for 24 hours. Read the pressure after 24 hours. Repeat at 30 psia. This test shall be performed with a known volume on the outlet for the purpose of calculating leakage, which shall not exceed 5 x 10^{-10} std cc sec through the seat. Reset pressure to 220 psia and check combined leakage past body and relief valve. Leakage shall not exceed 5 x 10^{-10} std cc/sec when checked with helium. Repeat with 30 psia.

2.4.2.6 Temperature Test - The component shall be placed in a test chamber and the internal temperature lowered to 0°F and maintained for 4 hours. While the component is at 0°F, conduct a performance test per Paragraphs 2.4.2.1 and 2.4.2.3 through 2.4.2.5. The internal chamber temperature shall then be returned to room ambient and the component shall be removed and subjected to a visual examination.

Repeat for 125°F.

2.5 Qualification Tests - Qualification tests shall be conducted per the following:

2.5.1 Acceleration - The component shall be placed in an acceleration apparatus and subjected to acceleration of 9 g for three minutes in each direction along each of three mutually perpendicular axes.

2.5.2 Vibration Test - The component shall be subjected to sinusoidal vibration in each of three mutually perpendicular axes. The test levels shall be in accordance with the following schedule. Vibration input control shall be in accordance with Paragraph 2.5.2.1. At the completion of the procedure outlined in Paragraphs 2.5.1 and 2.5.2 the component shall be subjected to a performance test 2.4.2.1 and 2.4.2.3 through 2.4.2.5 and a visual examination.

a. Random Vibration

Roll-off 20 cps at 3 db/octave
Flat - 20 cps to 480 cps at 0.2 g^2/cps
Increase from 280 cps to 600 cps at 3 db/octave
Flat - 600 to 1000 cps at 0.25 g^2/cps
Roll-off above 1000 cps at 12 db/octave
Overall amplitude = 17.2 g rms
Duration = 3 minutes in each of three mutually perpendicular axes

b. Sinusoidal Vibration

3.0 g zero to peak from 5 to 2000 cps applied in each of 3 mutually perpendicular axes
Sweep rate = 2 octaves/minute, maximum amplitude - 1/4 in.

2.5.2.1 Vibration Input Control - The specified vibration inputs shall be monitored and maintained at the component mounting points. For large components, where a variation of vibration input exists between mounting points, the mounting point with the lowest vibration level shall be taken as the control point. Transverse motion (crosstalk) shall be monitored at the component mounting points and shall be limited to not more than 100% of the applied vibration.

2.5.2.2 Equalization - The above specified power spectral density input to the component shall be within $\pm$ 3 db in the frequency range of 20 - 300 cps and within + 6 db and -3 db in the frequency range of 300-2000 cps using analyzing filters with a maximum permissible filter band width of 50 cps. The overall g rms level in the range of 20-2000 cps shall be maintained within $\pm$ 10% of the nominal g rms level specified.

The attenuation below 20 cps and above 2000 cps shall be greater than 12 db/octave. The following analyzer characteristics shall be reported for each test:

a. Filter Bandwidth
b. Integrator Time Constant
c Amplitude Accuracy

2.5.2.3 Temperature Stabilization - The temperature of a component shall be considered to be stabilized when three consecutive temperature readings taken at 15 minute intervals at the most centrally located accessible mass are within 5°F (2.8°C) of the specified temperature. Alternatively, the temperature may be considered to be stabilized when the duration of exposure to the chamber temperature is not less than four (4) hours.

2.5.3 Humidity - The component shall be installed with the ports capped in a test chamber in accordance with Paragraph 2.5.3.1. Prior to starting the test, the chamber temperature shall be between +68°F and +100°F (+20°C and -38°C) with uncontrolled humidity. The temperature and relative humidity shall then be gradually raised to +160°F (+71°C) and 100 percent respectively over a period of 2 hours. These conditions shall be maintained for a period of not less than 6 hours. With the relative humidity maintained at 95 percent the chamber shall then be gradually reduced to between +68°F and +100°F (+20°C and +38°C) over a period of not less than 16 hours. The component shall be nonpressurized and nonoperating during this test period. At the conclusion of the test, the component shall be removed from the test chamber and returned to room ambient conditions. Excess moisture may be removed only by turning the component upside down and wiping external surfaces. Connectors may be dried with an air blast. The component shall be subjected to a performance test per paragraphs 2.4.2.1 and 2.4.2.3 through 2.4.2.5 and a visual inspection within one hour after completion of the last humidity cycle.

2.5.3.1 Chamber - The test chamber and accessories shall be so constructed and arranged that condensate does not drip on the component under test. The chamber shall be vented to the atmosphere to prevent builoup of vapor pressure. Relative humidity shall be determined by dry bulb/wet bult thermometer comparison. The wet bulb thermometer shall be installed at the internal mouth of the air inlet duct. The air velocity flowing across the wet bulb shall not be less than 900 feet per minute. Provisions shall be made for controlling the flow of air throughout the internal test chamber volume where the velocity of air shall not exceed 150 feet per minute. Distilled or deionized water having a pH between 6.5 and 7.5 at +77°F (+25°C) shall be used to obtain the specified humidity.

2.5.4 Endurance - 3000 cycles

The regulator shall be connected to an ammonia gas supply with a solenoid valve provided in the line downstream of the regulator. By means of a disc actuator, cycle the regulator by activating the solenoid valve to a duty cycle of one second "on" and one second "off". The cycling will be interrupted for a leakage test per paragraph 2.4.2.5 at the intervals indicated below:

2.5.4.1 After 1,000 total cycles - perform leakage test.

2.5.4.2 After 2,000 total cycles - perform leakage test.

2.5.4.3 After 3,000 total cycles - perform leakage est.

2.5.5 Burst Pressure Test - The regulator shall be subjected to 600 psig pressure at the inlet port. The test shall not result in rupture of the exterior surface. Rupture of "O" rings or internal parts shall not constitute a failure.

2.6 Disposition of Samples - Components that have been subjected to the qualification tests shall be considered expended.

APPENDIX X

RESULTS OF QUALIFICATION TESTS ON "C" SOLENOID VALVES

S/N 27 - Regular Size Solenoid Valve
S/N 3 - Miniature Solenoid Valve

INTRODUCTION

All the qualification tests except the vibration and acceleration tests were conducted at the SPPS laboratory. The latter tests were conducted at GE Philadelphia Mechanical System Laboratory on February 10 through 14, 1967. The test results are obtainable from Mechanical Systems Laboratory Report 8277-SPPS-119 dated 2/16/67. (Reference 2)

Two solenoid valves were used: a regular size solenoid valve 28 VDC, P/N 1809001-30, S/N 27, and a miniature solenoid valve 21 VDC, P/N 1958001-3, S/N 3. The applicable GE source control drawings are 142B1562 for the regular size valve and 142B1580 for the miniature valve. GE SPPS Specifications 02-0063-00-D and 04-0001-00-A apply to both valves in general, while GE SPPS Specification 04-0017-00-A applies specifically under the Y-39 program requirements. It should be noted that the miniature valve S/N 3 magnetic coil was damaged after the completion of the qualification tests and it is presently not suitable for usage.

The qualification tests were conducted in accordance with the Y-39 Qualification Test Program for Solenoid Valves, issued on December 2, 1966. The tests are being reported in the order of their performance.

PRE-VIBRATION AND ACCELERATION TESTS

4.1 <u>High Potential Test</u>

1000 volts AC 60 cps test ran in accordance with MIL-STD-202 for one minute. O.K.

4.2 <u>Physical Examination</u>

Dimensions - O.K.
Finish - O.K.
Defects: S/N 3: no visual defects
S/N 27: lock-wire had to be re-done; electrical leads were not pig-tailed.

4.3 <u>Proof Pressure Test</u>

S/N 27 and S/N 3: 350 psig, 60 sec N_2 gas used. No visual damage.

4.4 <u>Static Leakage Test</u>

	Internal Leakage	External Leakage
S/N 27:	4.4×10^{-7} cc/sec	not detectable
S/N 3:	1.8×10^{-8} cc/sec	not detectable

4.5 <u>Pressure Drop</u>

S/N 27: 0.0368 psi
S/N 3: 0.0326 psi

4.9 Response Test Time

	Time To Open	Time To Close
S/N 27	29 msec	48 msec
S/N 3	50 msec	40 msec

4.6 Pull-in Voltage Test

	Voltage S/N 27	Voltage S/N 3
200 psia	17.50	12.0
50 psia	12.80	9.50
100 psia	14.50	10.50
150 psia	16.00	11.50
250 psia	18.00	12.00

4.7 Drop-out Voltage Test

at 200 psia:

S/N 27: 7 volts - valve fully closed

S/N 3: 6.5 volts - valve fully closed

4.8 Coil Resistance Test

S/N 27: 250.4 ohms

S/N 3: 800.3 ohms

4.10 Insulation Resistance Test

S/N 27: 90 megohms @ 500 VDC

S/N 3: 90 megohms @ 500 VDC

4.11 Power Consumption Test

	Applied Voltage	Current	Power
S/N 27	28 VDC	109 ma	3.05 W
S/N 3	24 VDC	30 ma	0.72 W

4.12 Temperature Rise Test

S/N 27 Initial temperature - 76°F
Applied Voltage - 28 VDC

Elapsed Time, Hr.	Temperature, °F
0	76
1	155
2	158
3	158
4	158
8	158

S/N 3 Initial temperature - 77°F
Applied voltage - 28 VDC

Elapsed Time, Hr	Temperature, °F
0	77
1.5	123
4.5	123
8	123

ENVIRONMENTAL TESTS

4.13 High Temperature Test

	S/N 27	S/N 3
Applied voltage	28 VDC	24 VDC
Qualification test temp.	101°F	101°F

S/N 27 Elapsed Time, Hr	S/N 27 Temperature, °F	S/N 3 Elapsed Time, Hr.	S/N 3 Temperature, °F
0.5	152	0.75	106
2.5	193	4.50	139
4.5	194	6.50	139
8	194	8	139

4.14 Low Temperature Test

	S/N 27	S/N 3
Applied voltage	28 VDC	24 VDC
Qualification test temp	0°F	0°F
Cooled with	LN_2	LN_2

S/N 27 Elapsed Time, Hr.	S/N 27 Temperature, °F	S/N 3 Elapsed Time, Hr.	S/N 3 Temperature, °F
0	0	0	0
1	31	3	56
2	31	6	56
3	31	7	56
4	31	22	56

4.16, 4.17, 4.18 Vibration and Acceleration Test

The vibration and acceleration test conditions, set-up, data, and conclusions are contained in GE-Philadelphia Mechanical Lab System Report 8277-SPPS-119, dated 2/16/67. (Reference 2)

2.19 Post-Vibration and Acceleration Function Tests

4.3 Proof Pressure Test

S/N 27 1×10^{-6} std cc/sec leakage
S/N 3 no leakage detected

4.4 Static Leakage Test

No external leakage recorded with either valve.

4.5 Pressure Drop Test

S/N 27 0.037 psi

S/N 3 0.033 psi

4.9 Response Time Test

	Time to Open	Time to Close
S/N 27	30 msec	48 msec
S/N 3	40 msec	50 msec

4.6 Pull-in Voltage Test

S/N 27 17.5 volts to open @ 200 psia

S/N 3 12 volts to open @ 200 psia

4.7 Drop-out Voltage Test

S/N 27 6.9 volts to close @ 200 psia

S/N 3 6.5 volts to close @ 200 psia

4.8 Coil Resistance Test

S/N 27 250.2 ohms

S/N 3 800.0 ohms

4.10 Insulation Resistance Test

S/N 27 90 megohms @ 500 VDC

S/N 3 90 megohms @ 500 VDC

4.11 Power Consumption Test

S/N 27 2.65 watts

S/N 3 0.75 watts

4.15 Life-Cycle Test

Test gas - ammonia

	Cycles	Leak Rate cc/sec S/N 27	Leak Rate cc/sec S/N 3
4.15.1	1,000	1×10^{-6}	No leakage
4.15.2	3,000	1×10^{-6}	No leakage
4.15.3	6,000	1×10^{-6}	1.15×10^{-9}
4.15.4	18,000	1×10^{-6}	1.8×10^{-8}
4.15.5	30,000	1×10^{-6}	1.8×10^{-8}
4.15.6	50,000	1×10^{-6}	1.8×10^{-8}
4.15.7	70,000	1×10^{-6}	1.8×10^{-8}
	100,000	1×10^{-6}	1.8×10^{-8}

POST LIFE CYCLE FUNCTIONAL TESTS

4.15(a) Static Leakage Test

	External	Internal
S/N 27	None	1×10^{-6} cc/sec
S/N 3	None	1.8×10^{-8} cc/sec

4.15(b) Pressure Drop Test

S/N 27 0.040 psi

S/N 3 0.035 psi

4.15(c) Response Time Test

	To Open	To Close
S/N 27	30 msec	48 msec
S/N 3	40 msec	50 msec

4.15(d) Pull-in Voltage Test

S/N 27 12 volts @ 200 psia

S/N 3 17.5 volts @ 200 psia

4.15(e) Drop-out Voltage Test

S/N 27 6.9 volts @ 200 psia

S/N 3 6.5 volts @ 200 psia

4.15(f) Coil Resistance Test

S/N 27 250.3 ohms

S/N 3 800.1 ohms

4.15(g) Insulation Resistance Test

S/N 27 90 megohms @ 500 VDC

S/N 3 90 megohms @ 500 VDC

4.15(h) Power Consumption Test

S/N 27 3.05 watts

S/N 3 0.75 watts

4.19 Cracking Pressure Test

4.20 Burst Pressure Test

The solenoid valve qualification test program ends with the performance of these two tests. As noted in paragraph 4.20, the valves are not required to operate after the burst pressure test. In order to realize further usage from the valves, it was decided to defer the two tests to some appropriate time in the future.

APPENDIX XI

RESULTS OF QUALIFICATION TESTS
ON "C" PRESSURE REGULATOR

1.0 INTRODUCTION

The following test results were obtained from a partially completed test program which was designed for the purpose of qualifying a pressure regulator (1992001-3 S/N 1) for ammonia gas or liquid set for a nominal 15 psi outlet pressure. The applicable GE-SPPS drawing is the source control drawing no. 47C141387. The test program was designed per GE-SPPS Spec. 04-0018-00-A with specific reference to the DODGE-M project. The numbers in parenthesis refer to the paragraph numbers appearing in the test procedure.

2.0 ACCEPTANCE TESTS (2.4)

2.1 Visual Examination (2.4.1)

The regulator complied with the applicable GE drawing.

2.2 Inlet Pressure Shock Test (2.4.2.1)

Solenoid valve upstream of the regulator completely opened after 29 ms subjecting the regulator to 220 psia. Outlet pressure was found to be within $\pm$ 1% of pressure setting.

2.3 Proof Pressure Test (2.4.2.2)

(a) Outlet was shut off and the inlet side was subjected to 350 psig for one minute.

(b) The inlet and outlet ports were subjected simultaneously to 350 psig for one minute.

At the completion of these tests, the regulator pressure setting was 15.22 psi at a flow of 1×10^{-6} lb/sec.

NOTE: The regulator was cycled 60,000 times prior to these tests and still indicated a lock-up pressure of 15.22 psi.

2.4 Regulated Pressure (1.3.2)

Flow rate = 1×10^{-6} lb/sec
Lock-up pressure = 15.22 psi
0.049 psi fluxtuation during test
Test was of one hour duration

3.0 HUMIDITY TESTS (2.5.3)

(a) Uncontrolled humidity; 2 hours

(b) 100% humidity; 160°F, 6 hours

(c) 95% humidity; 70-100°F, 16 hours

The above constituted one cycle (24 hours). The cycle was repeated five times for a total test time of 120 hours.

4.0 QUALIFICATION TESTS (2.5)

4.1 Sinusoidal Vibration (2.5.2.6)

The regulator was mounted on a model C-126 vibration exciter and subjected, along each of the three axes, to the following specifications:

5-15.5 cps - 1/4 inch constant displacement

15-2000 cps - 3.0 g constant acceleration in three mutually perpendicular axes at a sweep speed of 2 octaves per minute.

Results of these three tests showed no visual damage to the pressure regulator.

4.2 Random Vibration (2.5.2.a)

After the sinusoidal excitation was performed, a random environment was applied to the regulator according to the following specifications:

Roll-off: 20 cps at 3 db/octave
Flat: 20 cps to 480 cps at 0.2 g^2/cps
Roll-up: from 480 cps to 600 cps at 3 db/octave
Flat: 600 to 1000 cps at 0.25 g^2/cps
Roll-off: above 1000 cps at 12 db/octave
Overall amplitute: 17.2 g rms
Test duration: 3 minutes

The results of these three tests showed no visual damage to the pressure regulator.

4.3 Acceleration (2.5.1)

The regulator was mounted on a Genisco Model E-185 centrifuge and subjected to an acceleration of 9 g's for 4½ minutes in each direction along each of the three mutually perpendicular axes. No visual damage was noticed.

4.4 Post-Vibration Tests

Lock-up pressure tests were conducted after the completion of the vibration and acceleration test. Nitrogen was used as the test gas. The regulator exhibited a gradual increase in the outlet set-pressure as follows:

Elapsed Time, hrs	Lock-up Pressure, psi
0	16.0
1/4	16.2
2	17.2
4	17.8
19	19.5

The pressure regulator was consequently shipped back to the manufacturer for inspection and cleaning. The regulator was placed on test again and the following results were obtained:

Elapsed Time, hr.	Lock-up Pressure, psi
0	15.4
1	15.4
2	15.4
4	15.4
24	15.4

4.5 Temperature Stabilization (2.5.2.3)

The regulator was mounted, using the four mounting holes, on the inside surface of a hot/cold test chamber designed to maintain set temperatures between 125°F and -30°F. This chamber was installed in SPPS Vacuum Facility VTF-3 (3 ft. diameter x 4 ft. long). Figure XI-1 is a schematic of the test set-up. With the exception of the Inlet Pressure Shock Tests (2.4.2.1) all testing was done with regulator under high vacuum ($\sim 5 \times 10^{-5}$ torr). For the Shock Tests, only the mechanical pump was used so that the interior of the regulator could be evacuated readily by dumping to the vacuum chamber. (These large quantities of gas would choke the diffusion pump.) Testing was all performed with nitrogen gas from a room temperature supply. Regulator outlet pressure was constantly being monitered on a Sanborn strip chart recorder to detect transients and was being read on a vacuum referenced mercury manometer for accuracy. Table XI-I is a summary of the data obtained. The first column contains the paragraph number and description of the test outlined in the test program. In all tests the transients were less than 0.5% of the stable pressure reading once that reading was reached.

During test 2.4.2.5 "0°F Leakage" at 220 psia inlet pressure, the regulator locked shut and could not be opened, even after warming up and releasing the pressure on the outlet and inlet. The test was halted at this point.

Tests not completed were:

2.4.2.5 0°F Leakage at 30 psia inlet pressure.
125° Leakage at 220 and 30 psi inlet pressure.

2.5.4 3000 cycles.

At this point the regulator was removed from the test setup. Visual examination showed that one of the Teflon slides in the damping mechanism was slightly cocked and was probably causing or contributing to the malfunction. Upon removal of the mechanism, the regulator immediately opened and the damping mechanism shaft cocked to one side by about 1/32 of an inch. However, it remains free of interference from any other component.

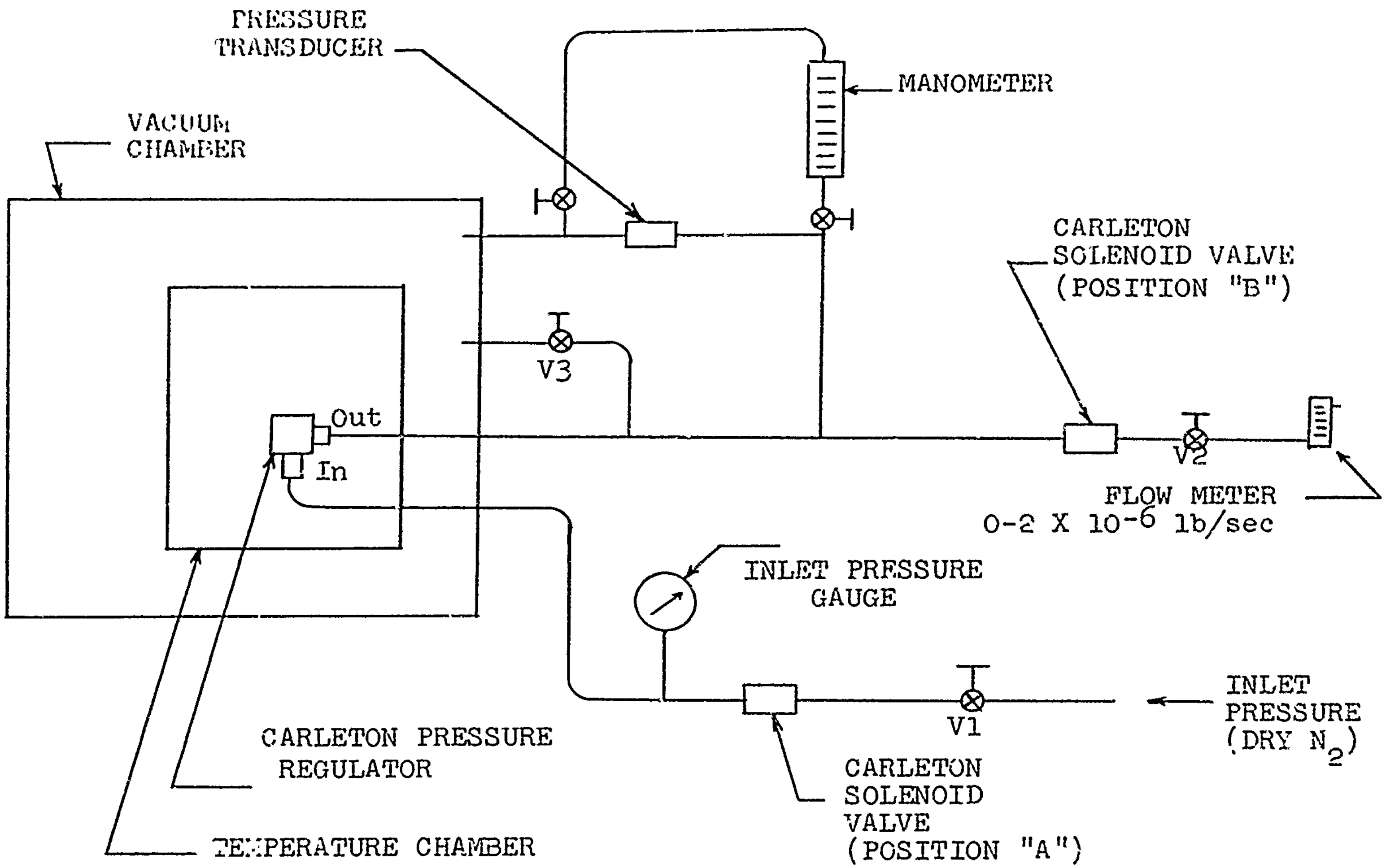
PRESSURE TRANSDUCER
MANOMETER
VACUUM CHAMBER
CARLETON SOLENOID VALVE (POSITION "B")
V3
Out
In
V2
FLOW METER
0-2 X 10-6 lb/sec
INLET PRESSURE GAUGE
INLET PRESSURE (DRY N2)
V1
CARLETON PRESSURE REGULATOR
CARLETON SOLENOID VALVE (POSITION "A")
TEMPERATURE CHAMBER

Figure XI-1

TABLE XI-I TEST DATA

Test	Regulator Temp. °F	Inlet Pressure Psia	Flow x 10^{-6} lb/sec	Stable Outlet Pressure (psia)*	Max. Variation from 15.2 psia %
2.4.2.1	R.T.	220	0	16.20	6.6
Inlet Pressure	0	220	0	16.35	7.6
Shock Test	125	220	0	15.76	3 7
2.4.2.3	R.T.	200	-	15.81	4.0
	R.T.	220	0.5	15.34	0.9
	R.T.	220	1.0	15.27	0.5
	R.T.	220	0.5	15.61	2.7
	R.T.	220	0	15.81	4.0
	0	220	0	15.98	5.1
	0	220	0.5	15.32	0.9
	0	220	1.0	15.27	0.5
	0	220	0.5	15.66	3.0
	0	220	0	16.01	5.3
	125	220	-	15.76	3.7
	125	220	0.5	15.27	0.5
	125	220	1.0	15.20	0.0
	125	220	0.5	15.56	2.4
	125	220	-	15.76	3.7
2.4.2.4	R.T.	220	1.0	15.20	0.0
Response	R.T.	200	0	15.69	3.2
& Fast	R.T.	100	1.0	15.20	0.0
Lockup	R.T.	100	0	15.74	3.6
	R.T.	30	1.0	15.22	0.1
	R.T.	30	0	15.74	3.6

Table XI-I Cont'd,

Test	Regulator Temp. °F	Inlet Pressure Psia	Flow x 10^{-6} lb/sec	Stable Outlet Pressure (psia)*	Max. Variation from 15.2 psia %
	0	220	1.0	15.29	0.6
	0	220	0	16.05	5.6
	0	100	1.0	15.29	0.6
	0	100	0	16.13	6.1
	0	30	1.0	15.29	0.6
	0	30	0	16.05	5.6
	125	220	1.0	15.61	-1.2
	125	220	-	15.61	2.7
	125	100	1.0	15.10	- .6
	125	100	0	15.69	3.2
	125	30	1.0	15.05	0.2
	125	30	0	15.56	2.6
			Time (hrs.)		
2.4.2.5 Leakage	R.T.	220	-	16.0	Supply valve and outlet valve closed
	R.T.	131	40	53.0	
	R.T.	220	-	16.0	Supply valve open
	R.T.	217	24	48.6	
	R.T.	30	0	15.0	Supply valve open
	R.T.	30	24	27.1	
	0	220	0	16.0	Supply valve open
	2	220	24	13.3	
	R.T.*	215	-	15.3	Supply valve open
	R.T.*	215	24	15.9	
	R.T.*	215	312	15.9	

* Damping mechanism removed.

The regulator without the damping mechanism was set up outside the vacuum chamber for a quick check on its operation. With an inlet pressure of 160 psig nitrogen the flow rate was varied between 0 and 1×10^{-6} lb/sec. The outlet pressure remained at $14.57 \pm .04$ psi throughout, even during lock up. At a flow of 1×10^{-6} lb/sec the inlet was closed and opened twice. Each time the outlet pressure rose to 14.48, 0.6% from the 14.57.

It appears that the damping mechanism, as well as being unnecessary, is also the cause of the problems encountered in the testing thus far. A leakage test lasting 13 days was performed with the regulator in vacuum without the damping mechanism. Dry nitrogen pressure of 200 psig was applied to the inlet. The outlet pressure with a volume of about 2 in.^{3} rose to 15.3 psia and overnight crept to 15.9 where it remained for the duration of the test.

Leak Rate Determination After Removal of Damping Mechanism

A rise in pressure of 0.1 psi or 0.6% of 15.9 would have been visible on the gauge. This rise would have meant a leakage of 0.6% of 2 in^3 or .012 in.^{3}. Therefore, for the 12 days at 15.9 psia the leak rate was less than .012 in ^{3}/12 days or 1.9×10^{-7} std cc/sec nitrogen.

APPENDIX XII

STATEMENT OF WORK

EXHIBIT "A"

12 SEP. 1966

STATEMENT OF WORK

RESISTOJET MICROTHRUSTOR SYSTEM DEVELOPMENT

I. INTRODUCTION

The contractor shall comply with the requirements set forth in this work statement to fulfill the following three-fold program objectives:

A. Provide the required ammonia propellant, resistance heated, microthrustor station keeping and station changing systems for the DODGE-M satellite.

B. Perform the necessary analytical and experimental research to provide a complete investigation of the capabilities of low power, low thrust thermal storage, resistance heated thrustors and their categorized mission applications.

C. Supply a twenty minute documentary color film coverage of all electric propulsion thrustor systems being used in conjunction with the DODGE-M satellite program.

II INTERFACE REQUIREMENTS

The contractor shall have responsibility for the resistojet microthrustor systems interface with the Johns Hopkins Applied Physics Laboratory (APL). Throughout the program the contractor shall provide interface information to the APL in the areas of weight, volume, power, thermal environment, command and control, and telemetry channels.

III. MICROTHRUSTOR SYSTEM RESEARCH

A. Flight Package Development: The contractor shall provide the required space flight qualified, resistojet microthrustor package which will be utilized on the DODGE-M satellite for the specified E-W station keeping and station changing capability. For clarification of the Phase A objectives, the following definitions are provided:

Thrustor: A thermal storage, resistance heated device which incorporates three nozzles or orifices which can be operated individually or simultaneously. A total of two thrustors are used for the entire flight package.

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STATEMENT OF WORK (Cont'd)

Station Keeping Systems: Consists of a propellant tank, an open-close explosive valve set, a relief valve, a filter, a pressure regulator, two pairs of solenoid valves in parallel (one pair for each nozzle).

Station Changing System: Includes a propellant tank, an open-close explosive valve set, a relief valve, a filter, a pressure regulator, two pairs of solenoid valves in parallel (one pair for each nozzle).

Flight Package: Consists of two station keeping systems and one station changing system and the necessary thermocouples and pressure Transducers (this package consists of two thrustors).

B. Systems Requirements: The contractor shall submit a detailed schedule and description for the microthustor testing program to AFAPL (APIE-1) for approval which is separate from a general over-all three phase program schedule which must also be supplied. This testing description shall identify and provide rational for tests to be conducted and specify the procedures for conducting these tests. Within this document, the contractor shall provide detailed specifications of the microthruster systems. These specifications shall include over-all physical characteristics such as weight, volume dimensions, etc., interface requirements such as power, voltage waveforms, telemetry command and control, etc., and independent operational characteristics such as thrust, specific impulse, propellant flow rate, etc. The system shall include at least the specifications given in the following paragraphs (C1 and C2). After approval of this document by the AFAPL (APIE-1), any changes must also be approved by AFAPL (APIE-1).

C. Thrustor Systems Design and Development:

1. Station Keeping System: The contractor shall provide two (2) completely redundant station keeping systems and two (2) identical back up systems capable of providing a total impulse of 300 lb-sec per system over the total mission duration. Each system shall be designed with the following nominal restraints:

(a) Thrust Level - 3, 6, or 9 micropounds

STATEMENT OF WORK (Cont'd)

(b) Specific Impulse - $>$ 140 sec

(c) Input power - $<$ 20 watts

(d) Total weight - $<$ 5 pounds (excluding only power, power and signal conditioning)

(e) Total Volume - $<$ 300 in^3

2. <u>Station Changing System</u>: A completely independent system and back up shall be furnished by the contractor each enabling one, 180 degree orbit maneuver and correction for an orbit injection velocity error of <u>approximately 7 ft/sec requiring a total impulse of nearly 1800 lb sec</u>. This system design shall be based upon the following nominal parameters:

(a) Thrust Level - 200 micropounds

(b) Specific Impulse - $>$ 200 sec

(c) Input power - $<$ 20 watts

(d) Total weight - $<$ 12 pounds (excluding power, power and signal conditioning)

(e) Total volume - $<$ 1800 in^3

3. <u>Prototype System Testing</u>: A complete system (or systems) shall be designed, fabricated, life tested, and flight qualified before the final flight hardware is built and acceptance checked. A life test of over 2000 hours shall be a goal with the system to approximate the actual space flight program. In addition, a resistojet system test shall be conducted in conjunction with an ion engine system in an appropriate facility at the Electro-Optical Systems, Inc. in Pasadena, California. The purpose of this test is to reveal any operational compatibility problems between the two systems. The resistojet contractor shall provide the necessary labor support during set-up and testing as required. The ion engine contractor will be responsible for the test program supervision.

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STATEMENT OF WORK (Cont'd)

4. "Zero G" Propellant Tank Test: Testing of a prototype "Zero G" propellant supply tank shall be performed aboard a KC-135 aircraft to determine the effects of liquid propellant vaporization. This work must be coordinated through the Air Force Aero Propulsion Laboratory (AFAPL-APIE-1).

5. Thrust Vector Effects: The Johns Hopkins Applied Physics Laboratory (APL) has established the requirement of all microthrustor subcontractors to know their respective engine thrust vector deviations from the actual mounting axis. A test shall be performed by the resistojet system contractor which ascertains the fulfillment of this requirement within a maximum of one (1) degree.

6. Hardware Requirements: The contractor must supply to the Air Force or other designated agency at least the following items developed under Phase A of this contract:

(a) Two complete flight packages of the resistojet microthrustor systems with acceptance checked components which will fulfill the requirements of the DODGE-M program.

(b) One complete set of spare active components as follows:

(1) Five NO, NC, explosive valve sets

(2) One station-keeping regulator

(3) One station-changing regulator

(4) Four solenoid valves

(5) Four heaters

(6) Two pressure relief valves for station-changing

(7) Two pressure relief valves for station-keeping

(8) Four line filters

(c) At least one complete flight package mock-up for use by the satellite design and development contractor for integration purposes. This may necessarily be the actual flight package or a non-operable facsimile.

STATEMENT OF WORK (Cont'd)

(d) One microthrust measuring system used in conjunction with the thrust vector testing phase.

(e) Two propellant tank fill systems

7. General Comments: Each flight system shall consist of all the necessary flight qualified thrustors, solenoid valves, pressure regulators, propellant supply tanks, safety valves and hardware to provide high reliability in mission achievement. Thrustor power conditioning will be included by JHAPL if it is found to be necessary, depending mainly upon the available power form aboard the DODGE-M satellite. The flight package(s) shall also contain all of the necessary instrumentation to continuously monitor and evaluate the operation of all three resistojet systems while in orbit. Cold gas operation of each thrustor system will be required if the engine heater should fail. Signal conditioning, command and control modules, and a ground control console will be provided for the resistojet package by Johns Hopkins Applied Physics Laboratory.

8. Yaw Control: The contractor shall be prepared to provide yaw control to the DODGE-M satellite in addition to the station changing and maneuvering capability, if the need arises. This program expansion shall be considered an increase in effort under the present contract and will be negotiated accordingly at that time. This is an interface problem which will be assessed very early in the program in order to minimize the work schedule reorientation.

IV. SYSTEM INTEGRATION AND LAUNCH

The contractor shall provide integration and launch support services to the prime contractor (APL) as necessary throughout the program. These services shall include:

A. Control console design and checkout

B. Supplying the data readout system requirements

C. Technical support during installation of the package aboard the satellite

D. Ground checkout just prior to launch

E. Technical support as necessary during thrustor system operation

STATEMENT OF WORK (Cont'd)

V. DATA REDUCTION AND REPORT WRITING

The contractor shall collect and reduce the data received from the satellite and report the results according to the standard Air Force technical report procedures.

VI. EXPLORATORY DEVELOPMENT

A. Mission Analysis: A study of the possible mission applications for low power (less than 20 watts), low thrust (1 to 500 micropound) resistance heated thrustors shall be performed. This will include but not be limited to the following:

1. Best propellant for each mission type

2. Total system weight and power comparisons for each mission with competitive microthrustor systems (i.e., subliming solids, ion engine, etc.).

B. Thrustor Development: The thrustors shall be of the thermal storage type which can be operated in either the continuous or pulsed [illegible] mode. Experimental and analytical investigations shall be performed to determine the compatibility of all engine components with the heater in a vacuum environment. At least three prototype microthrustors shall be designed, fabricated, tested and evaluated in the range from 1 to 500 micropounds; fulfilling, most efficiently, the greatest number of foreseeable missions within the specified power range. A [illegible] lifetime shall be demonstrated with each thrustor operating at the theoretical optimum heater temperature and propellant flow rate under a simulated space environment. Upon completion of the life tests, a metallurgical examination of all critical engine components shall be made and an extrapolated lifetime determined for each unit. During the experimental testing of these thrustors, at least the following parameters should be recorded:

1. Thrust

2. Engine chamber pressure

3. Heater temperature

4. Propellant temperature at the exit nozzle or orifice

5. Propellant flow rate

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6. Engine surface temperature

7. Heater power

The theoretical thrust variation as a function of nozzle throat or orifice diameter tolerance should be investigated to determine the limitations of thrust value accuracy. In addition, the variation of the thrust with ambient chamber pressure and expansion ratio or orifice diameter shall be studied. After the experimental data has been evaluated, three additional thrustors shall be fabricated, incorporating all of the necessary design changes indicated by the prototype test program. These thrustors shall be deliverable items to this laboratory.

VII. DOCUMENTARY FILM

The contractor shall supervise the production of a twenty-minute, color, narrated, documentary film covering the microthrustor aspects of the DODGE-M satellite program. There shall be approximately equal film time devoted to each of the three types of electric propulsion systems being developed for this satellite. The film should cover the development of the individual systems and include the launch. Approval of the film content and assemblage must be obtained from the AFAPL (APIE) before the final master copy is made. A total of ten (10) copies of this film will be required.

APPENDIX XIII

RESISTANCE JET SYSTEM INTERFACE DOCUMENT

RESISTANCE JET SYSTEM

INTERFACE DOCUMENT

This document represents the technical requirements and agreements between the General Electric Company Space Power and Propulsion Section and the Applied Physics Laboratory of the Johns Hopkins University regarding the resistance jet system interface with the DODGE-M Satellite.

GENERAL ELECTRIC COMPANY
Space Power and Propulsion Section

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AFAPL (APIE)

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JOHNS HOPKINS APL

November 25, 1966

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1.0 Introduction

The resistance jet propulsion system supplies thrust for the station keeping and station changing missions of DODGE-M using heated ammonia as propellant. The system includes the thrusters, propellant flow control elements, propellant tankage, power and signal conditioning, and certain pressure and temperature sensors. Thrust is supplied by solenoid valve actuation which permits propellant to flow through the selected thruster nozzle.

2.0 System Description

2.1 Pneumatic Subsystem

The pneumatic subsystem is shown schematically in Fig. 1. It consists of a propellant tank which supplies ammonia vapor to the inlets of three pressure regulators. The regulators are selected by actuating an explosive valve on ground command. Two regulators supply propellant to the east or west station keeping thrusters at a pressure of approximately 3 psia. The third regulator supplies propellant to both the east and west station changing thruster at a pressure of approximately 10 psia. Each thruster nozzle is supplied with two solenoid valves in series to reduce leakage probability. There are three nozzles in each of the two thrusters. The thruster body is heated by an electrical resistance element so that the propellant temperature is raised to approximately 2000°F before exhausting it into space. An ammonia decomposition catalyst is incorporated into the heated propellant flow passages to promote decomposition, thereby increasing specific impulse.

Pressure transducer and temperature sensor locations are shown in Fig. 1. The temperature sensor in the thruster body is a chromel-alumel thermocouple whose cold junction is located near the propellant tank. Cold junction temperature will be monitored with a thermistor supplied by JHAPL. The remaining temperature sensors will also be JHAPL supplied thermistors. The pneumatic system characteristics are as follows:

2 station keeping nozzles - 3×10^{-6} lb thrust $\pm$ 10%

2 station keeping nozzles - 6×10^{-6} lb thrust $\pm$ 10%

2 station changing nozzles - 200×10^{-6} lb thrust $\pm$ 10%

Station keeping specific impulse - 140 seconds minimum

Station changing specific impulse - 200 seconds minimum

Station keeping total impulse - approximately 600 lb-sec

Station changing total impulse - approximately 1800 lb-sec (including 200 lb-sec for possible orbit correction use)

Total subsystem weight - approximately 25 lb

Total subsystem volume - approximately 1 cubic ft

2.2 Power and Signal Conditioning Subsystem (PCS)

The PCS is shown in the block diagram of Fig. 2. Its function is to convert the nominal 40 volt 1000 cps power input of the satellite to current and voltage forms usable for valve actuation, heater operation, and amplification of thermocouple outputs. It performs these functions upon ground command. It does not contain the circuits for energizing the explosive valves or the pressure transducers. These will be supplied by JHAPL as discussed in paragraph 6. The PCS also includes test point connections for pre-launch checkout as well as appropriate telemetry connections for monitoring specific operations of the resistance jet propulsion system.

Total subsystem weight approximately (not presently known)

Total subsystem volume approximately (not presently known)

3.0 Space Allocation

3.1 Location

The propellant tank will be located near the center of mass of the satellite. Flow control components (i.e., valves, regulators, etc.) will be located behind access ports on the skin of the satellite. Thrusters will be located on the east and west faces of the satellite. The extent of protrusion beyond the skin is not presently known.

The PCS will be located within the vehicle in a location to be specified. The mounting requirements are discussed in paragraph 4.1.

3.2 Configuration - not applicable

3.3 Volume of Major System Components

Component	Quantity Required for System	Volume, each Rectangular Envelope Cubic Inches
System fuel tank	1	1331.00
Explosive valve, (N/C, N/O combination)	3	5.56
Filter	9	0.19
Pressure regulator	3	6.43
Pressure relief valve	3	0.19
Pressure transducer	11	2.25
Solenoid valve N/C	12	1.79
Thruster	2	25.00
Power & signal conditioner	1	*

NOTE: Consideration is being given to the mounting of six solenoid valves in each of two viewband boxes, each box equipped with electrical connector receptacles.

Total minimum system volume = 1500 cubic inches.

NOTE: This figure does not include:

(a) space required for component interconnecting ammonia gas tubing and fittings

(b) the power and signal conditioner package

(c) the increase in volume required when components are placed in each of two viewband boxes

4.0 Mounting Requirements

4.1 General

Components and subsystems will be held in place with removable fasteners. No additional requirements relating to this paragraph can be established until drawings are available for design review.

* Yet to be determined.

5.0 Weight Requirement

5.1 Weight Breakdown of Significant Components

Component	Quantity Required per System	Approx. Wgt., Ea., Lb.
System fuel tank	1	2.3
Fuel	1	12.7
Explosive valve N/C, N/O combination	3	0.56
Filter	9	Neg.
Pressure regulator	3	0.25
Pressure relief valve	4	0.06
Pressure sensor	11	0.14
Solenoid valve N/C	12	0.125
Thruster	2	0.75
Power & signal conditioner	1	*

NOTE: Consideration is being given to the mounting of six solenoid valves in each of two viewband boxes, each box being equipped with electrical connector receptacle.

Total minimum system weight= 21 lb.

NOTE: This figure does not include weight contributed by:

(a) component interconnecting ammonia gas tubing and fittings

(b) Power and signal conditioner package

(c) Viewband package enclosures

* Yet to be determined.

6.0 Power Requirements

6.1 AC and LC

6.2 Vehicle Power Characteristics

6.2.1 AC power, as described in JHAPL Spec S2P-S-003

AC voltage: $40 \, {}^{+\,20\%}_{-\,10\%}$ peak to peak

frequency: 976.41 cps

6.2.2 DC power

20 volts DC ${}^{+\,20\%}_{-\,10\%}$ from battery

± volts DC from telemetry system

6.3 Electric Propulsion System Power Requirements

Component	Power Form	Power Level, Watts	Type Load
Power & signal board	AC	*	Continuous
Four solenoid valves (worst case)	DC	14.24 4.00	Momentarily (100 ms) Continuously
Explosive valve N/O, N/C combination	DC	**	10 milliseconds
Thruster heater	DC	20	Continuously
Pressure transducer	DC	***	Intermittently

* Yet to be determined.

** That which is required to fire two primers simultaneously, 0.04 to 0.12 ohms per primer, 3 amperes per primer.

*** That corresponding to ± 1/4 volt DC applied across 5000 ohms.

7.0 Thermal Requirements

7.1 Required Thermal Environment

The temperature limits for any portion of the pneumatic subsystem between and including the solenoid valves and the propellant tank are 120°F maximum and 0°F minimum. The thermal environment for the PCS cannot be specified at this time. The restrictions noted are applicable to the operating condition. For the non-operative condition, the minimum temperature may be reduced to -40°F. The maximum temperature limit remains constant.

7.2 Heat Rejection to Spacecraft

During operation, the heat rejection to the spacecraft is as follows:

1 heater	20 watts
4 solenoid valves	4 watts
PCS	not known at this time

8.0 Commands

8.1 General. The command system controls the operation of the resistance jet thruster system. It is intended to make use of the command system's DPPT magnetic latching relay contacts to make and break the power circuits to the solenoid valves. Commands required are simple "on" or "off" operations.

8.2 Command Functions

Command	Description	Function
Make AC power available to resistance jet propulsion system.	Command system relay contacts close.	AC power is applied to power & signal conditioning package.
Remove AC power from resistance jet propulsion system.	Command system relay contacts open.	AC power is removed from power & signal conditioning package.
Heat up thruster in East station keeping system/East portion of station changing system.	Command system relay contacts close.	Heater in thruster is energized from DC power supply in conditioner.
Cool down thruster in East station keeping system/East portion of station changing system.	Command system relay contacts open.	DC power to heater in thruster is shut off.
Introduce ammonia gas into East station keeping system.	Command system relay contacts close, then open again.	Squib in N/C portion of explosive valve is fired, opening valve.
Cut off ammonia gas supply from East station keeping system.	Command system relay contacts close, then open again.	Squib in N/O portion of explosive valve is fired, closing valve.
Produce 3 micro-pound thrust from thruster in East station keeping system.	Command system relay contacts close.	N/C solenoid valves are energized and open. Heated gas passes through thruster 3 micro-pound orifice.

8.2 Command Functions (continued)

Command	Description	Function
Stop 3 micropounds thrust from thruster in East station keeping system.	Command system relay contacts open.	N/C solenoid valves are de-energized and close. Gas stops flowing thru 3 micropound orifice.
Produce 6 micropounds thrust from thruster in East station keeping system	Command system relay contacts close.	N/C solenoid valves are energized and open. Heated gas passes thru thruster 6 micropound orifice.
Stop 6 micropounds thrust from thruster in East station keeping system	Command system relay contacts open.	N/C solenoid valves are de-energized and close. Gas stops flowing thru thruster 6 micropound orifice.
Heat up thruster in West station keeping system/West portion of station changing system.	Command system relay contacts close.	Heater in thruster is energized from DC power supply in conditioner.
Cool down thruster in West station keeping system/West portion of station changing system.	Command system relay contacts open.	DC power to heater in thruster is shut off.
Introduce ammonia gas into West station keeping system.	Command system relay contacts close, then open again.	Squib in N/C portion of explosive valve is fired, opening valve.
Cut off ammonia gas supply from West station keeping system.	Command system relay contacts close, then open again.	Squib in N/O portion of explosive valve is fired, closing valve.
Produce 3 micropounds thrust from thruster in West station keeping system	Command system relay contacts close.	N/C solenoid valves are energized and open. Heated gas passes thru thruster 3 micropound orifice.
Stop 3 micropounds thrust from thruster in West station keeping system.	Command system relay contacts open.	N/C solenoid valves are de-energized and close. Gas stops flowing thru 3 micropound orifice.

8.2 Command Functions (continued)

Command	Description	Function
Produce 6 micropounds thrust from thruster in West station keeping system.	Command system relay contacts close.	N/C solenoid valves are energized and open. Heated gas passes thru thruster 6 micropound orifice.
Stop 6 micropounds thrust from thruster in West station keeping system.	Command system relay contacts open.	N/C solenoid valves are de-energized and close. Gas stops flowing thru thruster 6 micropound orifice.
Introduce ammonia gas into station changing system.	Command system relay contacts close, then open again.	Squib in N/C portion of explosive valve is fired, opening valve.
Cut off ammonia gas supply from station changing system.	Command system relay contacts close, then open again.	Squib in N/O portion of explosive valve is fired, closing valve.
Produce 200 micropounds thrust from East thruster in station changing system.	Command system relay contacts close.	N/C solenoid valves are energized and open. Heated gas passes thru thruster 200 micropound orifice/nozzle.
Stop 200 micropounds thrust from East thruster in station changing system.	Command system relay contacts close.	N/C solenoid valves are de-energized and close. Gas stops flowing thru thruster 200 micropound orifice/nozzle.
Produce 200 micropounds thrust from West thruster in station changing system.	Command system relay contacts close.	N/C solenoid valves are energized and open. Heated gas passes thru thruster 200 micropound orifice/nozzle.
Stop 200 micropounds thrust from West thruster in station changing system.	Command system relay contacts open.	N/C solenoid valves are de-energized and close. Gas stops flowing thru thruster 200 micropound orifice/nozzle.

9.0 Telemetry Channels

The list of parameters appearing below is incomplete.

Parameter	Channels & Spares, Number	Return Lines & Spares, Number	T/M Voltage Polarity	To Satellite Telemetry Circuitry			
				Output Voltage, Volts	Max. Output Voltage, Volts	Output Impedance, Ohms	Calibration Accuracy, %
Temp of thruster between 1950°F & 2050°F, East station keeping system/East portion of station changing system.							
Temp of thruster between 0°F & 2050°F, East station keeping system/East portion of station changing system.							
Temp of thruster between 1950°F & 2050°F, West station keeping system/West portion of station changing system.							
Temp of thruster between 0°F & 2050°F, West station keeping system/West portion of station changing system.							

9.0 Telemetry Channels (continued)

Parameter	Channels & Spares, Number	Return Lines & Spares, Number	T/M Voltage Polarity	To Satellite Telemetry Circuitry: Output Voltage, Volts	Max. Output Voltage, Volts	Output Impedance, Ohms	Calibration Accuracy, %
Current flowing in paralleled solenoid valve coils, 3 micro-pound thrust portion of East station keeping system thruster.							
Current flowing in paralleled solenoid valve coils, 6 micro-pound thrust portion of East station keeping system thruster.							
Current flowing in paralleled solenoid valve coils, 3 micro-pound thrust portion of West station keeping system thruster.							
Current flowing in paralleled solenoid valve coils, 6 micro-pound thrust portion of West station keeping system thruster.							

9.0 Telemetry Channels (continued)

Parameter	Channels & Spares, Number	Return Lines & Spares, Number	T/M Voltage Polarity	To Satellite Telemetry Circuitry: Output Voltage, Volts	Max. Output Voltage, Volts	Output Impedance, Ohms	Calibration Accuracy, %
Current flowing in paralleled solenoid valve coils, 200 micropound thrust portion of thruster, East station changing system.							
Current flowing in paralleled solenoid valve coils, 200 micropound thrust portion of thruster, West station changing system.							
Chamber pressure 3 micropound thrust portion of East station keeping system thruster			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	
Chamber pressure 6 micropound thrust portion of East station keeping system thruster.			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	

9.0 Telemetry Channels (continued)

Parameter	Channels & Spares, Number	Return Lines & Spares, Number	T/M Voltage Polarity	To Satellite Telemetry Circuitry			
				Output Voltage, Volts	Max. Output Voltage, Volts	Output Impedance, Ohms	Calibration Accuracy, %
Chamber pressure 3 micropound thrust portion of West station keeping system thrusters.			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	
Chamber pressure 6 micropound thrust portion of West station keeping system thrusters.			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	
Chanber pressure 200 micropound thrust portion of thruster, East station changing system			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	
Chamber pressure 200 micropound thrust portion of thruster, West station changing system.			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	

9.0 Telemetry Channels (continued)

Parameter	Channels & Spares, Number	Return Lines & Spares, Number	T/M Voltage Polarity	To Satellite Telemetry Circuitry: Output Voltage, Volts	Max. Output Voltage, Volts	Output Impedance, Ohms	Calibration Accuracy, %
Pressure, low pressure side of pressure regulator, East station keeping system			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	
Pressure, low pressure side of pressure regulator, West station keeping system			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	
Pressure, low pressure side of pressure regulator, station changing system			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	
Pressure, fuel tank			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	
Duplicate pressure fuel tank			+ and -	+0.25 to -0.25	+0.25 to -0.25	2500	

10.0 Electric Wiring

Data incomplete.

11.0 Ground Support Equipment

The ground support equipment will include necessary instrumentation for pre-launch checkout of the resistance jet propulsion system via a connector located at the base of the vehicle and accessible until just prior to launch.

In addition, the necessary apparatus for filling the propellant tank with ammonia will be constructed and considered part of the ground support equipment.

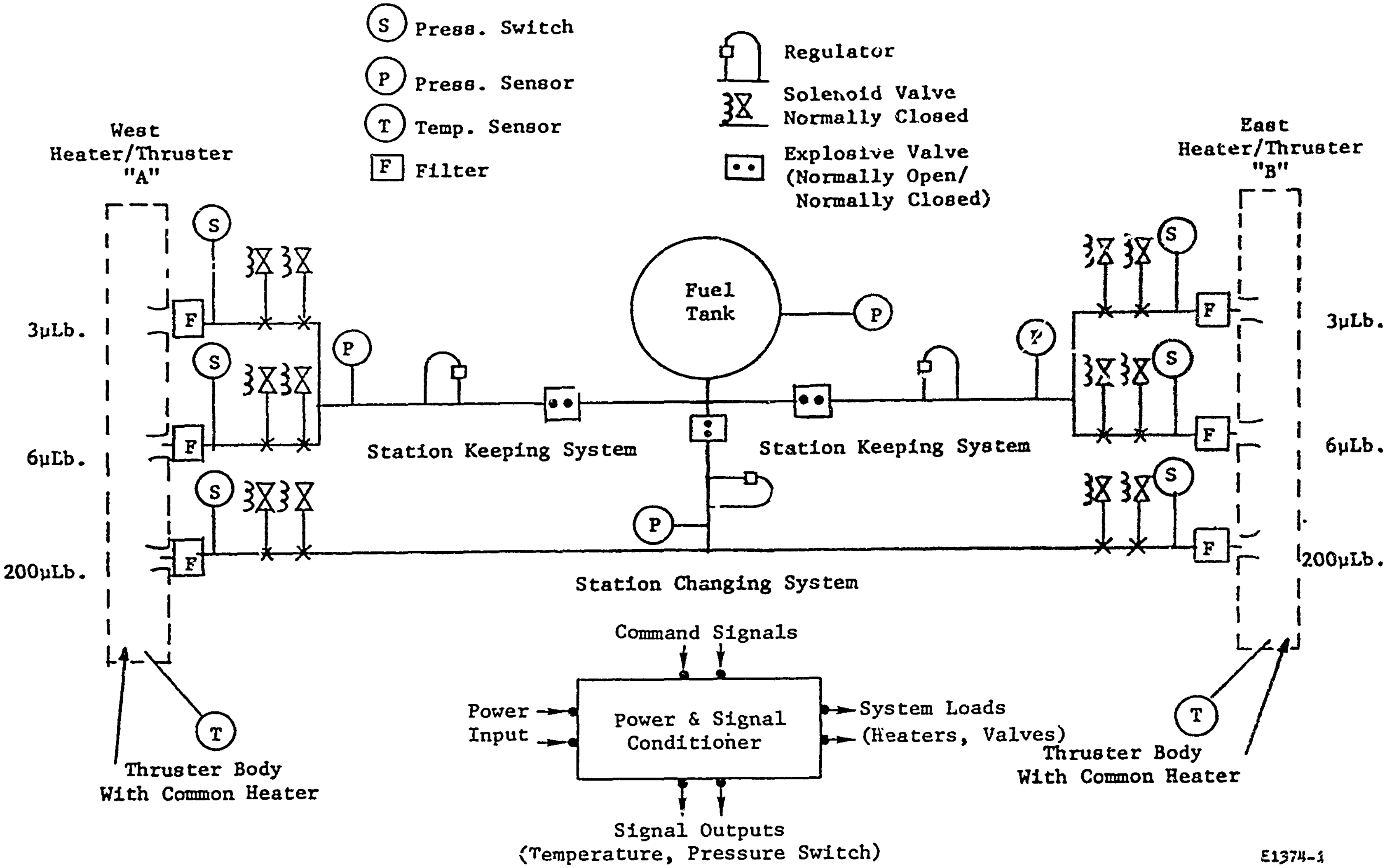

Figure 1. Resistance Jet Microthruster System, Schematic Diagram.

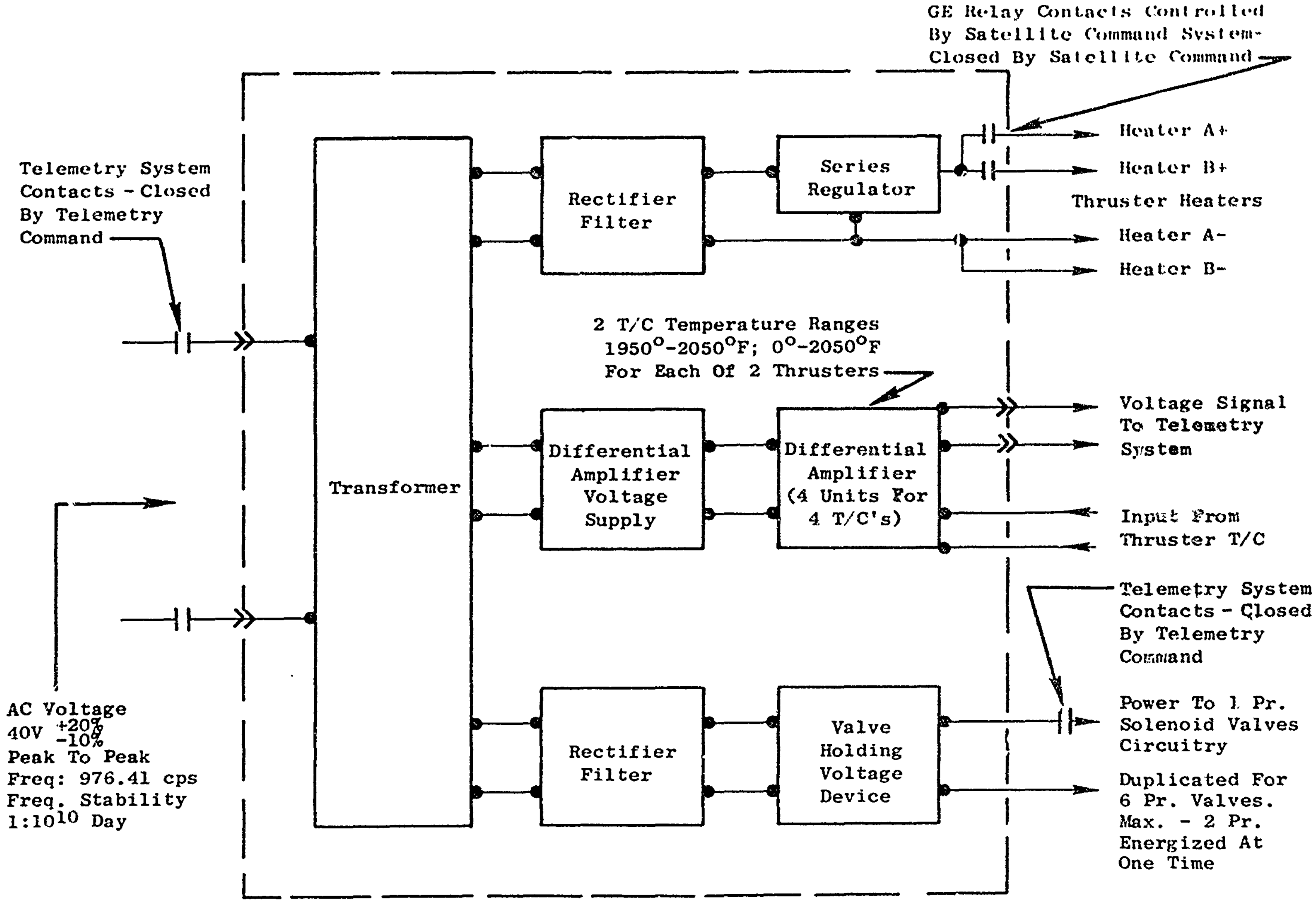

Figure 2. Function Block Diagram - Power & Signal Conditioner For Dodge "M" R.J. System

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APPENDIX XIV

STATEMENT OF WORK

EXHIBIT "A-1"

21 APRIL 1967

REVISED STATEMENT OF WORK

(U) RESISTOJET MICROTHRUSTOR SYSTEM DEVELOPMENT

I. INTRODUCTION

The contractor shall provide flight qualified ammonia propellant, resistance-heated, microthrustor packages capable of providing the required total impulse for station-keeping and station-changing of the DODGE-M satellite.

II. INTERFACE REQUIREMENTS

The contractor shall have responsibility for the resistojet microthrustor systems interface with the John Hopkins Applied Physics Laboratory (APL). Throughout the program, the contractor shall provide interface information to APL in the areas of weight, volume, power, thermal conditions and the like.

III. MICROTHRUSTOR SYSTEM RESEARCH

A.. Flight Package Development: The contractor shall provide the necessary space flight qualified, resistojet microthrustor packages which will be utilized on the DODGE-M satellite for the specified E-W station-keeping and station changing capability. For clarification of the program objectives, the following definitions are provided:

1. Thrustor - A thermal storage, resistance-heated device which incorporates three nozzles or orifices which can be operated individually or simultaneously. A total of two thrustors are used per flight package.

2. Station-Keeping Systems - Consists of the centralized propellant supply tank, an open-close explosive valve set, filters, a pressure regulator, two pair of solenoid valves in parallel (one pair for each thrust level, 3 μ-lb and 6 μ-lb).

3. Station-Changing System - Includes the centralized propellant supply tank, an open-close explosive valve set, filters, a pressure regulator, two pair of solenoid valves in parallel (one pair for each 200 μ-lb thrust level).

4. Power and Signal Conversion Equipment - Includes all of the necessary electronics and hardware to enable complete control of the resistojet station-changing and station-keeping functions from

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REVISED STATEMENT OF WORK (Cont'd)

the ground and continuous monitoring of system parameters through the satellite's command and telemetry equipment.

5. Flight Package - Consists of two station-keeping and one station-changing systems and all of the necessary instrumentation, power and signal conversion equipment (this package includes two thrustors and one propellant storage vessel).

B. Systems Requirements: The contractor shall submit a detailed schedule and description for the microthrustor testing program to AFAPL (APIE-1) for approval. This testing description shall identify and provide rational for tests to be conducted and specify the procedures for conducting these tests. Within this document, the contractor shall provide detailed specifications of the microthrustor systems. These specifications shall include over-all physical characteristics such as weight, volume dimensions, etc., interface requirements such as power voltage waveforms, telemetry command and control, etc., and independent operational characteristics such as thrust, specific impulse, propellant flow rate, etc. The system shall include at least the specifications given in the following paragraphs (C1 and C2). After approval of this document by the AFAPL (APIE-1), any changes must also be approved by AFAPL (APIE-1).

C. Thrustor Systems Design and Development:

1. Station-Keeping System - The contractor shall provide two completely redundant station keeping systems and two identical back-up systems capable of providing a total impulse of 300 lb-sec system over the total mission duration. Two additional systems will be required as part of the prototype flight package. Each system shall be designed with the following nominal restraints:

(a) Thrust Level - 3, 6, or 9 micropounds

(b) Specific Impulse - $>$140 sec

*(c) Input Power - $<$22 watts

*(d) Total Fueled Weight - $<$5 pounds

(e) Total Volume - 100 in^3 (excluding only propellant tank)

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REVISED STATEMENT OF WORK (Cont'd)

2. <u>Station Changing System</u>: An independent system (aside from the centralized propellant tank) and back-up shall be furnished by the contractor each enabling one, 180 degree orbit maneuver and correction for an orbit injection <u>velocity error of approximately 7 ft/sec requiring a total impulse of nearly 1800 lb sec.</u> This system design shall be based upon the following nominal parameters:

(a) Thrust Level - 200 micropounds
(b) Specific Impulse - > 200 sec
* (c) Input Power - < 22 watts
* (d) Total Fueled Weight - < 14 pounds
(e) Total Volume - < 1500 in^3 (including propellant tank and thrusters)

* excluding power, signal and power conditioning

3. <u>Power and Signal Conditioning</u>: The contractor must convert the electrical power available on the satellite into the forms required by the resistojet microthrustor package. In addition, electronic circuitry will be necessary to condition the various instrumentation signals into a form acceptable by the vehicle telemetry system for ground transmission. All circuits will be constructed from components suitable for space operation. Verification of the capability of the power and signal conditioning package to sustain the launch into orbit will be made by subjecting the prototype unit to flight qualification tests equivalent to those conducted on the thrustor systems. Both of the final units will be acceptance tested before delivery is made. The design and development of this electronics package shall be based upon the following nominal constraints:

(a) Total Input Power - < 32 watts at nominal input voltage
(b) Total Weight - ≤ 5 lb
(c) Total Volume - < 150 in^3
(d) Power Conversion Efficiency - > 75% at nominal input voltage while supplying power to one thrustor heater, 2 pairs of solenoid valves and associated instrumentation.

These requirements are based upon an input power of the form and frequency specified in JHAPL Specification S2P-S-003 dated 12 Oct 1966. Input voltage shall be either 28 volts or 40 volts peak to peak with a variation of +20% to - 10%. A breadboard power supply having these characteristics shall be supplied by JHAPL at least one month before the Contractor's breadborad is tested. The telemetry requirements shall also follow the requirements of JHAPL Specification S2P-S-003.

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REVISED STATEMENT OF WORK (Cont'd)

4. Prototype Testing - A complete prototype flight package which includes all instrumentation power and signal conditioning shall be designed, fabricated, flight qualified and life tested before the final flight hardware is built and acceptance checked. A life test of over 2000 hours shall be a goal for the prototype package to approximate the actual space flight simulated operations.

5. Electro-Optical Systems Test - In addition, a resistojet system test shall be conducted in conjunction with both an ion engine and colloid engine system in an appropriate facility at the Electro-Optical Systems Company in Pasadena, California. The purpose of this test is to reveal any possible operational compatibility problems which may occur amoung the three systems. The test system to be used is a partial resistojet package consisting of flight-type components throughout except for the propellant storage vessel which may or may not be of flight quality. In its entirety this engine system shall include: (1) a propellant storage device (2) an explosive valve set (3) a station-keeping or station-changing regulator (4) two solenoid valves (5) a thrustor (6) a complete power and signal conditioning electronics package (7) the necessary filters (8) two pressure transducers (9) a pressure switch (10) a thrustor temperature sensing system (11) the complete wiring harness (12) a propellant fill system and propellant and (13) all of the necessary hardware to electrically and mechanically interconnect all components. The resistojet contractor shall provide the necessary labor support during set-up and testing as required. The ion engine contractor will be responsible for the test program supervision.

6. "Zero G" Propellant Tank Test - Testing of prototype "Zero G" propellant supply tanks shall be performed aboard a KC-135 aircraft to determine the resulting effects on liquid propellant vaporization and flow control. This work must be coordinated through the Air Force Aero Propulsion Laboratory (AFAPL/APIE-1).

7. Thrust Vector Effects - The Johns Hopkins Applied Physics Laboratory (APL) has established the requirement of all microthrustor subcontractors to know their respective engine thrust vector deviations from the actual mounting axis. A test shall be performed by the resistojet system contractor which ascertains the fulfillment of this requirement within a maximum of one (1) degree.

8. Hardware Requirements - The contractor must supply to the Air Force or other designated agency at least the following items developed under Phase A of this contract:

REVISED STATEMENT OF WORK (Cont'd)

(a) Two complete and acceptance qualified flight packages either of which will fulfill the requirements of the DODGE-M program.

(b) One complete set of spare active components as found necessary to assure the absence of delay in the replacement of a component which for any reason becomes unusable.

(c) At least one complete flight package mock-up for use by the satellite design and development contractor for integration purposes. This may necessarily be the actual flight package or a non-operable facsimile.

(d) One microthrust measuring system used in conjunction with the thrust vector testing phase.

(e) One portable or "suitcase" type testor unit of the control and checkout of all system functions.

9. General Comments - Each flight system shall consist of all the necessary flight qualified thrustors, solenoid valves, pressure regulators, propellant supply tanks, safety valves, power and signal conditioning electronics, and hardware to provide high reliability in mission achievement. The flight package (s) shall also contain all of the necesary instrumentation to continuously monitor and evaluate the operation of all three resistojet systems while in orbit. Cold gas operation of each thrustor system will be required if an engine heater should fail. All input power, a command and control system and the necessary telemetry channels will be provided for the resistojet package by the Johns Hopkins Applied Physics Laboratory.

10. Yaw Control - The contractor shall be prepared to provide yaw control to the DODGE-M satellite in addition to the station-changing and maneuvering capability, if the need arises. This program expansion shall be considered an increase in effort under the present contract and will be negotiated accordingly at that time. This is an interface problem which will be assessed early in the program in order to minimize the work schedule reorientation.

IV. SYSTEM INTEGRATION AND LAUNCH

The contractor shall provide integration and launch support services to the prime contractor (APL) as necessary throughout the program. These services shall include:

REVISED STATEMENT OF WORK (Cont'd)

A. Supplying the data readout system requirements.

B. Technical support during installation of the package aboard the satellite.

C. Ground checkout just prior to launch.

D. Technical support as necessary during thrustor system operation.

V. DATA REDUCTION AND REPORT WRITING

The contractor shall collect and reduce the data received from the satellite and report the results according to the standard Air Force technical report procedures.

APPENDIX XV

STATEMENT OF WORK

EXHIBIT "A-2"

7 FEB. 1968

DEPARTMENT OF THE AIR FORCE
HEADQUARTERS AERONAUTICAL SYSTEMS DIVISION (AFSC)
WRIGHT PATTERSON AIR FORCE BASE OHIO 45433

ASNKN-20 26 February 1968

SUBJECT: Contract F33615-67-C-1163

TO: General Electric Company
Missile and Space Division
Cincinnati, Ohio 45215

1. During the past sixteen (16) months of technical effort on subject contract there have been various changes made in the design and performance requirements of the harcware to be delivered. These changes have been brought about by mission redefinition and have created problems in various phases of the technical program. The attached Exhibit "A-2" dated 7 February 1968, contains firm requirements for all the areas in question and is to be considered the final changes in this program.

2. You are requested to submit a proposal of your best estimate to complete the program in accordance with the revised statement of work. This estimate shall be broken down by phase and will show the amount required to complete each phase.

3. In order to properly evaluate your proposal you are requested to submit a breakdown of funds already expended by phase and then indicate the funds required to complete the individual phases. These will then be consolidated on a DD Form 633-4 which will be completed and returned with your proposal.

4. A new contract milestone schedule will be required with a hardware delivery date reprogrammed to 1 March 1969. Any significant deviation from the costs negotiated on the basic contract and those already expended should be explained by the contractor.

5. Your reply is requested no later than 6 March 1968. Return to Aeronautical Systems Division, Attn: ASNKN-20/Mr. Ernest P. Cooper, Wright-Patterson AFB, Ohio 45433. If telephoning becomes necessary call Area Code 513, 255-2464.

ERNEST P. COOPER
Contracting Officer

2 Atchs:
1. Exhibit "A-2" dtd 7 Feb 68 on seven pages 68SEK-491

DD Form 633-4

REVISED STATEMENT OF WORK

RESISTOJET MICROTHRUSTOR SYSTEM DEVELOPMENT

I. INTRODUCTION

The contractor shall provide flight qualified ammonia propellant, resistance-heated, microthrustor packages capable of providing the required total impulse for station-keeping and station-changing of the DODGE-II satellite.

II INTERFACE REQUIREMENTS

The contractor shall have responsibility for the resistojet microthrustor systems interface with the Johns Hopkins Applied Physics Laboratory (APL). Throughout the program, the contractor shall provide interface information to APL in the areas of weight, volume, power, thermal conditions and the like.

III. MICROTHRUSTOR SYSTEM RESEARCH

A. Flight Package Development: The contractor shall provide the necessary space flight qualified, resistojet microthrustor packages which will be utilized on the DODGE-II satellite for the specified E-W station-keeping and station-changing capability. For clarification of the program objectives, the following definitions are provided:

1. Thrustor - A thermal storage, resistance-heated device which incorporates three nozzles or orifices which can be operated individually or simultaneously. A total of two thrustors are used per flight package.

2. Station-Keeping System - Consists of the centralized propellant supply tank, an open-close explosive valve set, filters, a pressure regulator, two pair of solenoid valves in parallel (one pair for each thrust level, 3 micropounds and 6 micropounds) and one thrustor.

3. Station-Changing System - Includes the centralized propellant supply tank, an open-close explosive valve set, filters, a pressure regulator, two pair of solenoid valves in parallel (one pair for each 100 micropounds thrust level) and two thrustors.

4. Power and Signal Conversion Equipment - Includes all of the necessary electronics and hardware to enable complete control of the resistojet station-changing and station-keeping functions from the ground and continuous monitoring of system parameters through the satellite's command and telemetry equipment.

REVISED STATEMENT OF WORK (Continued)

5. Flight Package - Consists of two station-keeping and one station-changing systems and all of the necessary instrumentation, power and signal conversion equipment (this package includes two thrustors and one propellant storage vessel but no electrical interconnecting harness).

6. Prototype Flight Package - Consists of two station-keeping and one station-changing systems and all of the necessary instrumentation, power and signal conversion equipment (this package includes two thrustors and one propellant storage vessel but used for flight qualification and life testing).

7. Flight Qualification Testing - Subjection of at least one complete prototype flight package by subassembly (each of 2 thrustors, power conditioning, and the propellant supply system), to qualification testing; including operation in a vacuum environment, vibration and "G" load testing to at least 100% of the specifications provided by JHAPL and final design approval following final component performance verification.*

8. Acceptance Testing - Subjection of each flight subassembly (each of 2 thrustors, power conditioning, and the propellant supply system) to a test procedure similar to that for qualification testing but much less stringent with the levels of vibration and "G" loading of 50% of the design specifications.*

B Systems Requirements: The contractor shall submit a detailed schedule and description for the microthrustor testing program to AFAPL (APIE-1) for approval. This testing description shall identify and provide rational for tests to be conducted and specify the procedures for conducting these tests. Within this document, the contractor shall provide detailed specifications of the microthrustor systems. These specifications shall include over-all physical characteristics such as weight, volume dimensions, etc., interface requirements such as power voltage waveforms, telemetry command and control, etc., and independent operational characteristics such as thrust, specific impulse, propellant flow rate, etc. The system shall include at least the specifications given in the following paragraphs (C1 and C2). After approval of this document by the AFAPL (APIE-1), any changes must also be approved by AFAPL (APIE-1).

* Refer to Par. 6.1 of the Johns Hopkins Applied Physics Laboratory Specification S2P-S-003 (dated October 12, 1966).

REVISED STATEMENT OF WORK (Continued)

C. Thrustor Systems Design and Development:

1. Station-Keeping System - The contractor shall provide two completely redundant station-keeping systems and two identical back-up systems capable of providing a total impulse of 150 lb-sec per system over the total mission duration. Two additional systems will be required as part of the prototype flight package. Each system shall be designed with the following nominal restraints:

a. Thrust Level - 3, 6, or 9 micropounds

b. Specific Impulse - 60 sec minimum

c. Input Power* - 15 watts maximum

2. Station Changing System - An independent system (aside from the centralized propellant tank) and back-up shall be furnished by the contractor each enabling orbit maneuverability and possible correction for orbit injection velocity error requiring a total impulse of at least 150 lb-sec. This system design shall be based upon the following nominal parameters:

a. Thrust Level - 100 micropounds

b. Specific Impulse - 100 sec minimum

c. Input Power* - 15 watts maximum

3. Flight Package Physical Limitations - The rectangular volume of the tank and flow control components shall not exceed 3600 in^3. A single thrustor shall be limited to a cylindrical volume of 30 in^3. The power and signal conditioning package rectangular volume shall not exceed a maximum of 150 in^3. A weight limit of 39 lb for the complete fueled flight package, including all interconnecting tubing (excluding only the wiring harness), must not be exceeded.

4. Power and Signal Conditioning - The contractor must convert the electrical power available on the satellite into the form required by the resistojet microthrustor package. In addition, electronic circuitry will be necessary to condition the various instrumentation signals into a form acceptable by the vehicle telemetry system for ground

* excluding signal and power conditioning

REVISED STATEMENT OF WORK (Continued)

transmission. All circuits will be constructed from components suitable for space operation. Verification of the capability of the power and signal conditioning package to sustain the launch into orbit will be made by subjecting the prototype unit to flight qualification tests equivalent to those conducted on the thrustor systems. Both of the flight units will be acceptance tested before delivery is made. The design and development of this electronics package shall be based upon the following nominal constraints:

a. Total Input Power - 20 watts maximum at nominal input voltage

b. Total Weight - 5 lb maximum

c. Power Conversion Efficiency - 75% minimum at nominal input voltage while supply power to one thrustor heater, 2 pairs of solenoid valves and associated instrumentation.

These requirements are based upon a satellite power with the following characteristics:

Type - AC

Frequency - 2400 Hz $\pm$5%

Voltage - 32V $\pm$5% (peak to peak)

Waveform - square wave

Power Allotment - 20 watts

The Johns Hopkins APL satellite contractor shall be responsible for supplying the necessary electrical signals and power to operate the explosive valves and provide for the pressure transducer requirements. The design of the telemetry portion of the conditioner shall follow the latest specifications from the satellite contractor. The design of the telemetry portion of the conditioner shall follow the requirements of the Johns Hopkins APL Specification S2P-S-003 (dated October 12, 1966).

REVISED STATEMENT OF WORK (Continued)

5. Tank and Components Support Structure - The contractor shall design, fabricate, assemble 4 tank and components support structures. At least 1 prototype structure, complete with the tank, flow control components and interconnecting tubing shall be put through flight qualification testing. The 2 complete flight subassemblies shall be acceptance tested along with the remainder of the flight subassemblies (power conditioning and propellant supply system).

6 Prototype Testing - A complete prototype flight package which includes all instrumentation, power and signal conditioning shall be designed, fabricated, flight qualified and life tested. A life test of over 2000 hours shall be a goal for the prototype package with a test procedure that closely simulates the actual space flight operations. Any components which fail during the life test should be carefully inspected. This test procedure will be developed in cooperation with APL.

7 Electro-Optical Systems Test - In addition, a resistojet system test shall be conducted in conjunction with both an ion engine and colloid engine system in an appropriate facility at the Electro-Optical Systems, Inc. in Pasadena, California. The purpose of this test is to reveal any possible operational compatibility problems which may occur among the three systems. The test system to be used is a partial resistojet package consisting of flight-type components throughout except for the propellant storage vessel which may or may not be of flight quality. In its entirety this engine system shall include: (1) a propellant storage device; (2) an explosive valve set; (3) a station-keeping or station-changing regulator; (4) two solenoid valves; (5) a thrustor; (6) a complete power and signal conditioner; (7) the necessary filter; (8) two pressure transducers; (9) a pressure switch; (10) the complete wiring harness; (11) a propellant fill system and propellant; and (12) all of the necessary hardware to electrically and mechanically interconnect all components. The resistojet contractor shall provide the necessary labor support during set-up and testing as required. The ion engine contractor will be responsible for the test program supervision.

8. "Zero-G" Propellant Tank Test - Testing of prototype "Zero-G" propellant supply tanks shall be performed aboard a KC-135 aircraft to determine the resulting effects on liquid propellant vaporization and flow control. This work must be coordinated through the Air Force Aero Propulsion Laboratory (AFAPL/APIE-1).

9. Thrust Vector Effects - The Johns Hopkins Applied Physics Laboratory (APL) has established the requirement of all microthrustor subcontractors to know their respective engine thrust vector deviations from the actual mounting axis. A test will be performed by the resistojet system contractor which ascertains the fulfillment of this requirement within a maximum of one (1) degree.

REVISED STATEMENT OF WORK (Continued)

10 Hardware Requirements - The contractor must supply to the Air Force or other designated agency at least the following items developed under this contract:

a Two complete flight packages with acceptance tested components either of which will fulfill the requirements of the DODGE-II program.

b. At least one complete flight package mock-up for use by the satellite design and development contractor for integration purposes. This may necessarily be an actual flight package or the prototype flight system.

c Any active components remaining from the buildup of (a) and (b) above which may be used for the replacement of a component which for any reason becomes unusable.

d. One microthrust measuring system.

e. One portable or "suitcase" type testor unit for the checkout of all system functions.

f Two propellant tank fill systems.

11. General Comments - Each flight system shall consist of all the necessary flight qualified thrustors, solenoid valves, pressure regulators, propellant supply tanks, power and signal conditioning electronics, and hardware to provide high reliability in mission achievement. The flight package(s) shall also contain all of the necessary instrumentation to continuously monitor and evaluate the operation of all three resistojet systems while in orbit. Cold gas operation of each thruster system will be required if an engine heater should fail. All input power, a command and control system and the necessary telemetry channels will be provided for the resistojet package by the Johns Hopkins Applied Physics Laboratory.

IV SYSTEM INTEGRATION AND LAUNCH

(. The contractor shall provide integration and launch support services to the prime contractor (APL) as necessary throughout the program. These services shall include:

A. , Supplying the data readout system requirements.

REVISED STATEMENT OF WORK (Continued)

B. Technical support during installation of the package aboard the satellite.

C. Ground checkout just prior to launch.

D. Technical support as necessary during thrustor system operation.

V DATA REDUCTION AND REPORT WRITING

The contractor shall collect and reduce the data received from the satellite and report the results according to the standard Air Force technical report procedure.

VI. The contractor shall supply a monthly program expenditure breakdown by specific task areas, as outlined on the milestone chart. This will include at least, the programmed material and manpower costs for each task, the monthly acclamation and the cost remaining to complete each task.

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APPENDIX XVI

MICROTHRUSTER SYSTEM TEST SCHEDULE

Thrust Rig

After the thrust rig has been built up in vacuum facility, it will be examined to make certain that the effect of vacuum pump vibration, the effect of opening or closing the vacuum chamber door, and the effect of torsional twist of the cylinder making up the vacuum chamber itself are all minimal.

Tests will then be conducted:

a. Proving its suitability for the desired thrust range.

   This will be accomplished by applying an accurately known force to the system, waiting for it to stabilize out, and then applying a force in the opposite direction, representative of a thruster. The decrease in indicated force is a measure of the applied force. Values of force shall be selected to encompass the desired thrust range.

b. Determination of the accuracy of the thrust measurements.

   Because the thrust is produced through electrical force transducer and is indicated on a recording chart after the signal has been amplified, the accuracy of the force measurement is a measure of the accuracy of the associated electronic equipment. A check is readily made by applying dead weights to the force transducer and reading out the indicated weight on the recording chart.

c. Determination of the repeatibility of the thrust measurements.

   Using the straightforward technique described in Paragraph b above, the check on a value of force can be repeated as often as desired.

Preliminary Evaluation Thrusters:

After having determined the values of ammonia gas pressure for the station keeping and station changing thrusters, the initial testing of the preliminary evaluation thrusters will provide for a determination of the orifice size for the 3 and 6 micropound thrusters. Calculations will provide for selection of a range of orifice sizes to span the desired thrust levels. Testing will provide families of characteristic curves from which a

Preliminary Evaluation Thrusters - Continued

final orifice size can be selected. Such curves will include thrust vs. temperature, mass flow rate of ammonia gas vs. temperature and specific impulse vs. temperature. Orifices of the correct size will be procured and the testing repeated on them to prove their suitability.

Final testing of the preliminary evaluation thrusters will provide the final nozzle configuration for the 200 micropound thrust station changing application as well as providing for an evaluation of a catalyst.

It is planned that the preliminary evaluation thruster will be designed and built so that the interior portion upon which the orifice or nozzle is affixed will be readily removable from the outer or heat shield. Thus while one orifice is being tested, a second orifice can be mounted in position on the spare interior portion. The third interior assembly would be one containing the catalyst and would be held in reserve for the catalyst evaluation testing.

Ammonia Tank:

Testing of an actual ammonia tank is intended to qualify the tank structurally. Flight tests of transparent models in KC-135 aircraft are intended to prove that the internal design will provide for adequate vapor-liquid separation of the ammonia under zero-g conditions.

It is planned that as the first aluminum tank is being constructed, specimens of the parent metal joined with a weld that duplicates that of the peripheral weld will be subjected to tensile tests. The peripheral weld on the finished tank will then be subjected to x-ray inspection. After having been helium leak tested it will be pressure tested to 320 psig. The cleaning processes to which the interior will be subjected and the contamination tests will indicate whether or not the required cleanliness level can be attained. It will then be subjected to vibration and g-loading tests while 85% full of water (18.7 pounds) and pressurized internally with 150 psig of dry nitrogen. Upon completion of these tests, it will be drained of water and x-rayed to investigate the possibility of structural damage. Finally, it will be subjected to a "super-proof" internal pressure of 420 psig.

Ammonia Tank - Continued:

Transparent models of the ammonia tank, with the interior portions closely duplicated, will be mounted in a capsule for transport through zero-g flights in KC-135 aircraft. The interior of the capsule will be illuminated and motion picture cameras within it will photograph the change in liquid levels and the separation of liquid and gas within the tank. It is anticipated that if the tanks are mounted in their normal position and if they are no greater than 85% full of liquid, a gas bubble will always remain at the outlet part of the tank under zero-g conditions. Other interesting experiments could also be conducted, such as the liquid-gas separation behavior at levels less than 85% full of liquid and also when the tanks are at positions other than normally upright.

Solenoid Valve:

After an exhaustive survey of solenoid valves considered to be suitable for the Dodge-M microthruster propulsion system, it was decided that the Carleton Controls valve was the most suitable. Also it had been successfully qualified for flight as a part of an earlier program and experience with it had been satisfactory. Accordingly, it was selected as the prime candidate for the Dodge-M microthruster systems. However, qualification tests are to be rerun on the valve tests written around the particular environmental specification for Dodge-M components. A description of this testing program appears as Appendix "A".

Pressure Regulator:

Advance investigations of suitable ammonia gas pressure regulators suitable for use in the Dodge-M microthruster system had narrowed the choice down to the Carleton Controls regulator. Its small size and weight as well as in-house experience were influencing factors. One was selected to be subjected to qualification testing to examine its worthiness. A description of this testing program appears as Appendix "B".

Ammonia Tank Fill Valve:

A primary characteristic of an ammonia tank fill valve is zero leakage after closing. A survey will be made to examine the performance experience of both the so-called "quick-disconnect" type as well as the manually-operated, positive shut-off type.

The first type selected will be one that has had previous flight history. Qualification tests will consist primarily of long-term exposure to liquid ammonia interrupted by periodic opening, closing and leak checking.

Ammonia Tank Support:

If a decision is made that GE-SPPS is to supply a mounting means for the ammonia tank as well as structural members upon which this mount is to be attached, then a prototype supporting structure must be initially subjected to vibration and g-loading tests. The structure would be loaded with an ammonia tank containing a predetermined amount of water. A probable tank for this application could be the one that had already been qualified. If the design philosophy has proceeded to the point where gas flow components are also to be located on the supporting structure, masses representing these components may be mounted upon the structure so that actual loading conditions may be more nearly duplicated. The vibration and g-loading schedule would be the same as for all other components of the microthruster system.

Thruster Heater:

Although the design of a suitable heater for the thruster is at hand, one that has been proven by intensive testing to be suitable for the requirements of the flight thruster, efforts are constantly being directed towards a superior heater - superior in terms of larger operating life as well as the number of heat up and cool down cycles it can withstand. Such improved heaters are currently being tested in the GE-SPPS Electrical Propulsion Laboratory. Operating temperatures, input current and applied voltage are being measured. Their performance is being monitored and the decision as to which heaters to procure for the flight thrusters will be based largely upon this experience.

Flight Thruster:

After the flight thrusters have been fabricated, tests will be conducted to determine their system operating characteristics. These include power, thrust, and I_{sp} at temperature, warmup, cooldown, and vectoring. All measurements will, of course, be made in the vacuum tank with each thruster mounted in turn upon the thrust rig. Certainly on the first thruster a temperature profile will be made to investigate its thermal characteristics. Thermocouples located at pertinent points on its various surfaces will provide this information. Measurements, in total, will consist of ammonia gas pressure, thruster temperatures, thrust and heater current and voltage. The gas flow characteristics of the 3 and 6 micropound and the 200 micropound nozzles will have already been established.

Power and Signal Conditioner:

The breadboard power conditioner will be checked out using either simulated or actual loads for the solenoid valves and thruster heaters. Actual or simulated thermocouple signal inputs will be used to check out the temperature indicating portion of the circuit. The operation of the pressure switch can readily be simulated to activate the signal for transmission to the telemetry system. Such a breadboard will permit accurate detection and measurement of power losses within the circuitry.

The first or prototype power and signal conditioner will be the qualification unit. Its performance characteristics will be completely determined. After being subjected to the same vibration and g-loading schedule as the rest of the flight components, its performance characteristics will be re-examined. Measurements will be made to determine input power changes during warm-up and cool-down periods for the thruster. Added loading effect will be studied by operating the solenoid valves in sequence. The output of the thruster thermocouple circuitry into the telemetry system terminals will be monitored. Operation of the telemetry flag circuitry will also be monitored.

Each flight Power and Signal Conditioner will be subjected to performance tests to provide assurance that it meets its design specifications in every detail.

Suitcase Tester:

The suitcase tester is the instrument designed and built to check out the system without energizing the heaters. Because it is primarily a monitoring device, performance tests upon it will consist primarily of continuity and correctness of its circuitry. However, where command module signals are originated within the tester they will be checked for proper voltage and power levels.

Microthruster System Testing:

The prototype microthruster system will consist of a fuel tank, flow control components, and the thrusters as shown in Figure XVI-1. The power and signal conditioner will also be included. If it has been decided that the tank mounting and a mounting means for the flow control components is to be integrated, they will be assembled in that form. This system will be mounted in the vacuum tank with all electrical and instrumentation connections completed. After all the electrical wiring is checked for continuity and all pneumatic connections are determined to be leak tight, the suitcase tester (if available) can be connected into place and the system operation simulated with it using dry nitrogen gas in the system. With this operational test satisfactorily completed, an exercise involving the filling of the ammonia tank through the fill valves may be conducted. The heater in one or the other thruster would be energized. Power input into the power and signal conditioner would be monitored as well as the temperature of the thruster. After the thruster has stabilized at its operating temperature, ammonia gas can be released to the system by operating the explosive valves. The solenoid valves in each gas line leading to each orifice or nozzle of the thruster could then be opened in sequence. Operation of the various pressure switches would indicate that the respective solenoid valves had operated as desired. Changes in pressure level in the vacuum tank will indicate the passage of gas out the respective orifice or nozzle. Ammonia tank gas pressure and system gas pressures can be monitored by means of the pressure transducers. After each thruster is operated in turn and the performance of the systems monitored as described above, and the systems determined to be operable, the system is ready for its 2000 hour test. During this time, temperatures, pressures, and input power to the power and signal conditioner will be monitored and recorded as the thrusters are operated in accordance with a predetermined duty cycle.

Ultimately, the two flight microthruster systems will be built up in final form. The assembled portions will be checked for leaks. All components will have already been either acceptance tested or operationally tested, so no system test will be conducted.

NOTES

Figure XVI-1 of this Schedule is the same as Figure 1 of this Summary Report

Appendix "A" of this Schedule is the same as Appendix VIII of this Summary Report

Appendix "B" of this Schedule is the same as Appendix IX of this Summary Report

APPENDIX XVII

LISTS OF ISSUED DRAWINGS

1. MICROTHRUST MEASURING SYSTEM

Number	Title
246R734	Assembly
941D164	Upper Bracket
47C140241	Lower Bracket
142B1465	Dampener Plate
47C140243	Dampener Pot
47C140242	Instrumentation Frame
263E915	Test Chassis
47B115367	Instrumentation Arm
47C141405	Thruster Clamp
47B115368	Solenoid Clamp
142B1562	Solenoid Valve
47B115369	Electrode
142B1509	Clip
47B115300	Propellant Fitting
165A5280	Propellant Tube
47B115409	Cable Bracket Assy.
47B115411	Suspension Cable
142B1466	Support Rod
941D233	Thruster
47C141406	Chassis Table
142B1467	Plate Support
246R734	Assembly
47B115373	Pivot, Chassis
165A5283	Button, Pivot
165A5284	Screw, Adjusting
165A5285	Plate, Pivot
47C141407	Restoring Solenoid
47B115406	Stop Bracket
165A5291	Core Extension
47B115405	Solenoid Armature

2. PRELIMINARY EVALUATION THRUSTER

Number	Title
47B115341	Nozzle (.00724)
47B115348	Heat Shield Liner
47B115350	Cover Ring
47B115349	Outer Cover
165A5273	Cover End
165A5274	Heater Clip
941D233	Assembly, Nozzle Test Thruster
47B115344	Heater
47B115345	Flow Path Liner
165A5271	Liner End
165A5272	Supply Tube
47C141399	Thruster Body
47B115347	Containment Ring
47B115340	Orifice (.00100)

3. FLIGHT THRUSTERS

Number	Title
246R757	Assy, Flight Thruster
47B115464	Heater Unit
941D416	Body, Thruster
47C143221	Container, Screen
47C143950	Plate, Orifice
47B115469	Tube, Thruster
263E964	Flange, Thruster
47C143224	Tube Supply
47C143231	Connector, Electrical
941D417	Flange, HT Shield
47C143225	Cover, HT Shield
47C143226	Cover, Front
263E981	Heat Shield
47B115470	Beads, Fish Spine
47C143219	Thermocouple
47C143228	Extension
47C143229	Clamp, Thermocouple
47C143230	Cover, Extension

4. TANKAGE

Number	Title
941D169	Assy, 11 inch Diameter Ammonia Storage
47C140244	Tank, Upper Half
941D173	Cone Support
165A5754	Vapor Tube
142B1469	Nut
142B1469	Disk
142B1470	Centering Cross
47C140249	Screen Cone
165A5293	Fill Tube

5. AMMONIA GAS FLOW CONTROL

Number	Title
47B115388	Switch, Guage Pressure, Probe Type
47B115407	Valve, Axial Solenoid
47C141387	Regulator, Pressure
47C141413	Transducer, Pressure
47C141415	Valve, Explosive Operated
47C141416	In-Line Filter

6. PROPELLANT TANK SUPPORT AND FLOW CONTROL COMPONENT ASSEMBLY

Number	Title
263E973	Assembly, Y39 Propellant Tank and Flow Components Frame Structure
263E974	Assembly, Frame
941D426	Assembly, Tank Strut
941D427	Assembly, Tank Support
263E976	Assembly, Tank Strap
47B115473	Bracket
941D428	Manifold, Outlet
47C143259	Tube, Inlet Manifold
47C143258	Tube, Inlet Manifold
47C143257	Tube, Inlet Manifold
47C143243	Tube, Inlet
47C143247	Seal 1/4
47C143246	Strap, Component
47B115474	Spacer
263E974	Assembly, Frame Mounting Assembly
47C143248	Nut, Clinch
263E279G1	Assembly, Component Panel Assembly
47D172980	Panel .10 thk (Mg)
47B115478	Bracket, Press. Reg.
47B115479	Bracket, Elec. Conn.
47B115480	Body, Press. Sw

7. SPECIFICATIONS

Number	Title
01-0076-00-A	Heater Wire, Recrystallization - Resistant Platinum
01-0077-00-A	Boron Nitride Powder
01-0078-00-A	Swageable Magnesia Electrical Heater Cores
01-0079-00-A	Heater Wire, Mo-50Re
02-0202-00-A	Special Processing Requirements, Swaged Electrical Heaters
04-0013-00	Transducer, Pressure
04-0014-00	Valve, Combination, Explosive Actuated, Normally Closed/Normally Open
04-0015-00	Filter, Pneumatic (Ammonia)
04-0016-00-A	Relief Valve, (Pneumatic) Ammonia
04-0017-00	Valve, Solenoid Operated, Pneumatic (Ammonia), Coaxial, Low Pressure for Station Keeping and Station Changing Applications
04-0018-00	Valve, Pressure Regulator, Pneumatic (Ammonia), Station Keeping and Station Changing
04-0019-00	Power and Signal Conditioner

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11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
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13. ABSTRACT

This report is a summary of the work performed under Air Force Contract F 33615-67-C-1163 in the design, fabrication, and test of resistance jet microthruster systems for East-West station keeping and station changing duties on the Dodge-II Satellite. The thrusters were electrically heated and ammonia gas was used as fuel. For the station keeping function, thrust values of 3, 6, and 9 micropounds were supplied. For station keeping, 100 micropounds of thrust were provided, with a catalyst being used to assist in decomposition of the ammonia.

Included in this work was the design and fabrication of a torsion-wire device that was used for measuring thrust in the required low range.

Parabolic flights of KC-135 aircraft carrying plastic models of the microthruster system's fuel tank proved its suitability for delivering gas to the propulsion system under zero-g conditions. These experiments, showing equilibrium positions of the propellant, were recorded in slow motion on a 900 foot reel of color film.

Electrical designs were completed for a power and signal conditioner to convert satellite electrical power into the forms required by the microthruster system. A portion of this conditioner contained circuitry to condition the various instrumentation signals into a form acceptable to the vehicle telemetry system for earth transmission.

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