T H E U N I V E R S I T Y O F M I C H I G A N

COLLEGE OF ENGINEERING
Department of Electrical Engineering
Space Physics Research Laboratory

Quarterly Status Report No. KQ-10

MEASUREMENT OF THE MOON'S ATMOSPHERIC PRESSURE

Prepared by

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TABLE OF CONTENTS

LIST OF FIGURES

I. INTRODUCTION

This is the tenth status report describing the research effort under contract NASw-133, and it covers the period March 1 through May 31, 1962. This contract calls for the construction, integration, calibration, and testing of a thermally controlled instrumented package which will be capable of measuring the atmospheric pressure on the moon. This package is to be incorporated in the Surveyor soft-landing lunar vehicle sometime in 1965.

II. RESEARCH EFFORT DURING THE PERIOD

Recent developments in the Surveyor program have set delivery schedules back by approximately one year, which means that tested prototype packages are to be delivered early in 1964. At this time, it is felt that the present design of the package should be carried through to at least two prototype units for testing and evaluation. Such a procedure would allow any changes or improvements required and/or desired, due to state-of-the-art advancements, to be made during the year prior to delivery with a better understanding of the hardware involved.

During the period covered by this report, a prototype design was essentially completed. Construction of at least one complete package will begin in June. An Interim Engineering Report for this package is included as the Appendix.

III. FINANCIAL REPORT
(Monthly Cost Breakdown)

Month	Wages Student	Wages Non-student	Overhead	Expendable Materials	Equip-ment	Travel
March	$ 939	$ 4,868	$2,903	$ 1,031	0	$ 551
April	346	5,679	3,012	7,844	$475	122
May	1,220	3,501	2,360	4,314	0	477
Totals	$2,505	$14,048	$8,275	$13,189	$475	$1,150

Grand Total $39,642

As of 31 May 1962, approximately $29,000 of the allotted funds remains.

APPENDIX

LUNAR ATMOSPHERE EXPERIMENT:

PROGRESS REPORT

1. INTRODUCTION

This report describes the progress to date on the instrumented package designed for use in the Lunar Atmosphere Experiment (see Fig. 1). The report has been prepared for the experiments, second design review and coordination meeting, held at the Jet Propulsion Laboratory (JPL) for the purpose of defining design concepts so that the package will be "as expected" and will be compatible with the Surveyor space vehicle.

The Lunar Atmosphere Experiment is being funded by NASA, Goddard Space Flight Center (GSFC), and is being performed in cooperation with GSFC, with N. W. Spencer (GSFC) as lead experimenter and D. R. Taeusch (U. of M.) as associate experimenter.

Progress is reported on the various components of the instrumented package in the following order:

Sensor Gauge and Magnet with Breakoff Device

High-Voltage Power Supply

Electrometer Amplifier

Instrument Container

Following these descriptions of progress to date, design concepts for thermal control are presented for review.

2. PROGRESS ON COMPONENTS OF THE INSTRUMENTED PACKAGE

2.1 SENSOR GAUGE AND MAGNET WITH BREAKOFF DEVICE

The sensor gauges and magnets (see Fig. 2) are being constructed by the Geophysics Corporation of America under contract with GSFC, NASA. Three flight model gauges have been delivered.

Each gauge has been operated down to the 10^{-12} mm Hg pressure range. A thin coating of Nickle 63 on the anode, which emits Beta particles, causes the gauge to start at these pressures within two minutes.

The gauge has been tested in vibration at the GSFC test facility. The anode electrical connection broke when the anode rotated during high-frequency vibration, but in all other respects the gauge seemed very rugged and remained vacuum-tight. Future gauges will be constructed so that the anode cannot rotate.

Figures 2, 3, and 4 show the gauge assembly, orifice, and breakaway cap. The weight of the entire assembly, including the breakoff pyrotechnics, is 4 lbs. The Geophysics Corp. has been asked to design a lighter magnet; its representatives feel this is possible and are now at work on the problem.

2.2 HIGH-VOLTAGE POWER SUPPLY

The high-voltage power supplies to be used in this experiment are being constructed by the Matrix Research and Development Corp. under contract with GSFC (see Fig. 5). Four flight units have been received to date.

2.2.1 Specifications

Input Voltage	29 volts ±2%
Mid-Value Output Voltage	Within range of 4000 volts ±2%
Regulation	±2% (all causes)
Number of Outputs	1
Load Current	0 to 10 μa
Operating Temperature Range	-50°C to +75°C
Input Current	Less than 25 ma at full load

Ripple	Less than 2.5 volts peak to peak
Oscillator Starting	—
Short Circuit Reliability	Short circuited output not damaging to any component
Weight	Approximately 12 oz
Size	4" x 2" x 1-11/16"
Vibration	"Mariner A" specifications
Non-Operative at Launch	—
Discharge Time Constant	3 hr minimum
Sterilization	125°C, 24-hr
Short Soak	-50°C and +75°C

2.3 ELECTROMETER AMPLIFIER

The sensor-gauge output-current detector for this experiment will be an electrometer amplifier designed and built under contract with GSFC by the Space Physics Research Laboratory of The University of Michigan. The amplifier unit will contain its own power converter, switching circuit, and syncramental motor for mechanical switching requirements (see Figs. 6,7,8, and 9). At the time it was announced that the Surveyor program would be delayed for six months, an amplifier was being temperature-compensated prior to construction and incorporation in the unit. After the delay had been announced, it was felt that attempts should be made to design a new amplifier which would be less temperature-sensitive and would consume less power than the old. The newly designed amplifier, now in the breadboard stage of development, is the one being reported here.

2.3.1 Specifications

Input Voltage	29 volts ±1% (operation) 22 volts +6, -4 (switching)
Input Current	10 ma (operation) 2.5 a - 10 ms (switching)
Ranges	7
Range Selection	Automatic with manual override

Size	Approximately 4" x 3" x 2-3/4"
Weight	Approximately 2 lbs
Operating Temperature	-50°C to +75°C
Sterilization	125°C, 24-hr

2.4 INSTRUMENT CONTAINER

The package which contains the various components for operation of the experiment will be a double-walled box, with outside dimensions of 6" x 7" x 8". The outer shell, a drawn aluminum box with .020" walls, will weigh .7 lbs. The inner box will weigh .8 lbs; it will be made of fiberglass, with inside dimensions of 4" x 5" x 6". Figure 10, preliminary drawing of the fiberglass inner container, shows details of the magnet-gauge assembly mounting and of the structural support members which mount to the outer skin. Figure 11, a preliminary drawing of the external shell, provides information on spacecraft interfaces.

3. LUNAR ATMOSPHERE EXPERIMENT: THERMAL CONTROL

3.1 INTRODUCTION

The purpose of this discussion is to describe the methods currently being considered for accomplishing thermal control of the Lunar Atmosphere Experiment instrument package. The thermal control problem is discussed for the two obviously separate conditions of lunar day and lunar night. The problem of lunar day temperature control is solved by adjusting the emissivity of the outer surface of the package such that a desired maximum equilibrium temperature will not be exceeded at lunar noon on the lunar equator. The problem of heat loss for conditions of lunar night is solved by using a double-walled container for the electronics and thermally insulating the inner wall from the outer skin.

Since this instrument package is suspended above the moon's surface and away from the spacecraft by a 15-ft boom, it is assumed that the package is thermally independent of the spacecraft.

3.2 LUNAR DAY TEMPERATURE CONTROL

Temperature control under conditions of lunar day requires that the external surface of the package be treated in such a manner that the input of power from the sun and the moon is balanced by an output radiating from the external surface at the desired equilibrium surface temperature. The power input from the sun will be approximately constant during the entire lunar day, but the input from the moon will vary approximately as the cosine of the sun's angle from the local normal. The maximum surface temperature of the moon on the equator at lunar noon is assumed to be 400°K.

Among the many ways of adjusting emissivities and absorptivities, the most economical for our present purpose is that of using a base material of low emissivity which is light, inexpensive, and easy to work with (aluminum, for example) and then coating the required area with a material whose absorptance-to-emittance ratio is low. One such material is the "white paint" available from Hughes Aircraft. The area will thus radiate the amount of power required to maintain the desired temperature. In adjusting emissivities and absorptivities we assume the following values:

Area of Outer Shell	292 in.^{2}
Emissivity of Aluminum (e)	.10
Absorptivity of Aluminum (a)	.15

Emissivity of White Paint (e_p)	.956
Absorptivity of White Paint (a_p)	.244
Area Seen by the Sun (20% of total area)	58.4 in.^{2}
Area Seen by the Moon (50% of total area)	146 in.^{2}
Maximum Temperature of the Package	75°C = 348°K

We also assume that the moon will not see any of the "white paint". The extent of the area assumed to be seen by the sun is only approximate, as is the constancy of the area during lunar day. The emissivities given above are "low temperature" values, based on the assumption that these values remain constant for the temperature range in which we are working. The absorptivities given above refer to the absorptance of the material to solar radiation, since the absorptance of the material to low-temperature radiation will be equal to its low-temperature emissivity. Let it be emphasized that all these values are only approximate, the purpose of this report being to explain the method rather than exact design requirements.

Since we want to know how large the area of "white paint" must be in order to maintain the package temperature at 75°C on the lunar equator at lunar noon, we set the power input equal to the power output, with the package temperature at 75°C, and set the "white paint" area as the unknown quantity.

The sun's contribution is:

$$\text{Power input} = [a\,(.20A - A_p) + a_pA_p]\,S$$

where

$$A = \text{total area of outer shell} = 292 \text{ in.}^2$$

$$A_p = \text{area of paint (to be solved for)}$$

$$S = \text{solar constant} = .9\ \frac{\text{Watts}}{\text{in.}^2}$$

$$a = \text{absorptivity of aluminum} = .15$$

$$a_p = \text{absorptivity of paint} = .244$$

$$\therefore (\text{Power in})_{sun} = (7.884 + .085\ A_p) \text{ Watts}$$

The moon's contribution is:

$$\text{Power input} = e \times .50A \times K \times T_m^4 \quad \text{(Stefan-Boltzman equation)}$$

where

$$e = \text{low temperature emissivity of aluminum} = .10$$

$$K = \text{Stefan-Boltzman constant} = 3.66 \times 10^{-11} \; \frac{\text{Watts}}{\text{in.}^2 \, {}^\circ K^4}$$

$$T_m = \text{Temperature of moon's surface} = 400^\circ K$$

$$\therefore (\text{Power in})_{\text{moon}} = 13.68 \text{ Watts}$$

The power output of the package is:

$$\text{Power output} = e\,(A - A_p)\,KT_p^4 + e_p A_p K T_p^4$$

where

$$e = \text{emissivity of aluminum} = .10$$

$$e_p = \text{emissivity of paint} = .956$$

$$A = \text{total area of package} = 292 \text{ in.}^2$$

$$A_p = \text{area of paint (to be solved for)}$$

$$K = \text{Stefan-Boltzman constant} = 3.66 \times 10^{-11} \; \frac{\text{Watts}}{\text{in.}^2 \, {}^\circ K^4}$$

$$(\text{Power out})_{\text{package}} = (15.71 + .460\,A_p) \text{ Watts}$$

Setting the power input equal to the power output for equilibrium, we have:

$$7.884 + .085\,A_p + 13.68 = 15.71 + .460\,A_p$$

or

$$A_p = 15.6 \text{ in.}^2$$

Therefore, only 16 square inches of paint is required to hold the temperature of the package below 75°C during the period of maximum heat input from the external environment. More paint could be used for safety and for achieving a lower temperature; however, it is probably desirable to have as high a maximum temperature as the electronics will allow, in order to: (a) store as much heat as possible for lunar night operation, thus minimizing power requirements for heating during lunar night, and (b) keep the package warm enough in case it is not located on the lunar equator, for the moon's contribution to the power input would then be decreased.

It is of interest here to compute the equilibrium temperature of the package when the sun is on the horizon at the day-to-night terminator. At this time, according to the assumptions being made by Hughes Aircraft, the moon's temperature will be 169°K.

The sun's contribution is assumed to remain constant, i.e., 9.21 Watts. But the moon's contribution will now be:

$$\text{Power input} = .10 \times 146 \times 3.66 \times 10^{-11} \times 8.16 \times 10^{8}$$

$$(\text{Power in})_{\text{moon}} = .436 \text{ Watts}$$

Therefore, by setting this power input equal to the Stefan-Boltzman equation for the package, we get:

$$9.21 + .44 = 9.65 = (.10 \times 276.4 + .956 \times 15.6)\, 3.66 \times 10^{-11}\, T_p^4$$

$$\therefore T_p = 281°\text{K at day-night terminator.}$$

As was stated previously, these temperatures are equilibrium values. The computations do not take into consideration either the heat supplied by the power input for operation of the experiment or the heat capacity of the package. The latter will cause a time delay in attaining or approaching the computed temperature, but is thought to be small during lunar daytime since the sun moves with respect to the moon roughly 12 angular degrees per earth day. These facts are considered in Section 3.4 of this report, where equations for the problem of package temperature versus time are presented for the 29 earth-day lunar cycle. The solution to the problem will be reported soon.

3.3 LUNAR NIGHT HEAT LOSS

The objective of temperature control under conditions of lunar night is to reduce the heat loss from the package as much as possible so that minimum power is required in order to maintain the minimum operating temperature of the package. A double-walled container is used, the outer shell being treated as described in Section 3.2. The Inner container, which holds the electronics and the Redhead gauge, is made of fiberglass with approximately 1 sq in. of structural support material between the inner container and the outer shell. NRC-2 super insulation is used in the separation (approximately 1 inch wide) between the inner container and the outer shell.

Heat loss through wire power leads is decreased by using a 3-inch insert made of low-thermal-conductivity metal, such as Constantan, for each lead.

Thus the heat losses are as shown in the following sections.

3.3.1 Structural Supports

The thermal conductivity of material to be used is $5 \times 10^{-3} \frac{\text{Watt in}}{\text{in.}^2\ ^\circ\text{C}}$. Since the material we are using is approximately 1" x 1" x 1", we have:

$$(\text{Power loss})_{\text{supports}} = 5 \times 10^{-3} (T_p - T_s)$$

where T_p = internal package temperature

T_s = outer shell temperature

3.3.2 Insulation Material

The conductance of the insulating material (NRC-2) which is used between the inner and outer shells is reported to be $1 \times 10^{-6} \frac{\text{Watt in}}{\text{in.}^2,\ ^\circ\text{C}}$. If we assume an effective conductance area which is approximately 200 sq in., we get the following power loss through the insulation:

$$(\text{Power loss})_{\text{Insulation}} = 2 \times 10^{-4} (T_p - T_s)$$

3.3.3 Power Input Leads

Three inches of constantan wire will be inserted in each input lead between the inner and outer shell. It has been found that 16-mil wire is strong enough for this purpose and has a resistance of .28 ohms, which is tolerable. The thermal conductivity per wire is given by:

$$\text{Conductance (Watts)} = \frac{A\Delta T}{L} K$$

where

$$K = .554 \frac{\text{Watt in}}{\text{in.}^2,\ ^\circ\text{C}}$$

$$A = \pi(.008)^2 = 2.01 \times 10^{-4} \text{ in.}^2$$

$$\Delta T = T_p - T_s$$

$$L = 3 \text{ in.}$$

$$\therefore \text{Conductance per wire} = 3.71 \times 10^{-5} (T_p - T_s) \text{ Watts.}$$

The total conductance for input leads assuming 15 leads is consequently,

$$\text{Conductance} = 5.57 \times 10^{-4} (T_p - T_s) \text{ Watts}$$

3.3.4 Breakoff Device Flange and Redhead Gauge Orifice

The breakoff device flange will have an area of approximately 3 sq in., and will be radiating to space with an emissivity of approximately .05. The orifice will have an area of .11 sq in., and will be radiating into space with an emissivity of 1, the emissivity of a blackbody. Because the flange and Redhead gauge tubulation do not touch the outer shell or come in contact with the super insulation, there is no thermal short to the outer shell.

The power radiated from flange and orifice is:

$$(\text{Power loss})_{\text{flange and orifice}} = .05 \times 3 \times 3.66 \times 10^{-11}\, T_p^4 + 1 \times .11 \times 3.66 \times 10^{-11}\, T_p^4$$

$$(\text{Power loss})_{\text{flange and orifice}} = 9.52 \times 10^{-12}\, T_p^4$$

The total power losses are:

$$(\text{Power loss})_{\text{lunar night}} = \underbrace{5 \times 10^{-3} (T_p - T_s)}_{\text{Structural}} + \underbrace{2 \times 10^{-4} (T_p - T_s)}_{\text{Insulation}} + \underbrace{5.6 \times 10^{-4} (T_p - T_s)}_{\text{Input leads}} + \underbrace{9.52 \times 10^{-12}\, T_p^4}_{\text{Flange}}$$

T_p = temperature of internal package

T_s = temperature of outer shell

It is of interest to compute what the heat loss will be when the temperature of the internal package is -50°C, the minimum operating temperature. For this computation we assume the external shell temperature to be 140°K. (According to this assumption, approximately .5 Watts is transmitted to the outer shell from the inner package.)

$$\therefore (\text{Power loss})_{\text{lunar night}} @ T_p = 233°\text{K}) = 5 \times 10^{-3} (83) + 2 \times 10^{-4} (83) + 5.6 \times 10^{-4} (83) + 9.52 \times 10^{-12} \times 2.47 \times 10^{9} = .502 \text{ Watts}$$

This value is well within the limits of the limit postulated by Hughes Aircraft.

3.4 TEMPERATURE OF THE MOON'S SURFACE

Now that the thermal requirements for the package have been theoretically satisfied, it is possible to set up the equation for package temperature versus time. We note that:

$$(\text{Net power input})_{\text{internal package}} = \frac{dT_p}{dt} C_p$$

where

T_p = temperature of internal package

C_p = heat capacity of internal package

t = time

Therefore:

$$-\frac{dT_p}{dt} C_p = \underbrace{5 \times 10^{-3} (T_p - T_s)}_{\text{Structural}} + \underbrace{2 \times 10^{-4} (T_p - T_s)}_{\text{Insulation}} + \underbrace{5.6 \times 10^{-4} (T_p - T_s)}_{\text{Input Leads}}$$

$$+ \underbrace{9.52 \times 10^{-12} T_p^4}_{\text{Flange}} - (\text{Electrical power in})_{\text{average}}$$

This equation becomes:

$$-\frac{dT_p}{dt} C_p = 5.76 \times 10^{-3} (T_p - T_s) + 9.52 \times 10^{-12} T_p^4$$

$$- (\text{Electrical power in})_{\text{average}}$$

In this equation, T_s, the temperature of the outer shell, is given by the solution of:

$$-\frac{dT_s}{dt} C_s = \underbrace{1.57 \times 10^{-9} T_s^4}_{\text{Radiating to Space}} - \underbrace{5.34 \times 10^{-10} T_m^4}_{\text{Heat from Moon}} - \underbrace{5.76 \times 10^{-3} (T_p - T_s)}_{\text{Heat from Inner Package}}$$

$$- (\text{Sun's Contribution})_{\text{lunar day}}$$

Thus we have two simultaneous differential equations with two unknowns, T_p and T_s. The electrical power input term refers to the power input for operation of the experiment. For the moment, this is assumed to be 2 Watts for 5 minutes every hour, which averages out to .167 Watts. The sun's contribution to the heat input will be assumed constant for the lunar day, at 9.2 Watts.

The heat capacity of the package, C_p, is approximately .75 $\frac{\text{Watt hr}}{°C}$ and the heat capacity of the shell, C_s, is .067 $\frac{\text{Watt hr}}{°C}$.

The moon's surface temperature, T_m, is approximately as shown in Fig. 12, with the maximum lunar day temperature on the equator at lunar noon assumed to be 400°K, and the minimum lunar night surface temperature assumed to be 119°K. The maximum-minimum values in the figure are those which the Jet Propulsion Laboratory has suggested for thermal design. An explicit expression for this equatorial temperature variation has been developed for use in the above differential equations. This expression is:

$$T_m^4 = 10^8 f(t)$$

where f(t) is given by:

for $\quad 0 < t < 3.64 \qquad f(t) = 1.988$

for $\quad 3.64 < t < 1.77.30$

$$f(t) = - (6.48) + (262.48) \sin[(8.8595)\ 10^{-3}\ t]$$

for $\quad 177.30 < t < 346.72$

$$f(t) = 5.1 + (250.9) \sin[(8.859)\ 10^{-3}\ t]$$

for $\quad 346.72 < t < 709.20$

$$f(t) = \frac{15.73}{[(0.5076)\ t - 175.6]^{0.3955}}$$

and $\quad [f(t) + 709.20)] = f(t)$

This value for the temperature of the moon's surface follows closely the values determined theoretically by F. B. Bjorklund of Hughes Aircraft Company and is felt to be adequate for our thermal design.

The solutions to the previous equations are not yet available, but will be reported in the near future. Once these solutions have been derived, a theoretical time for the lunar-night heat input required for maintaining operating temperature can be determined.

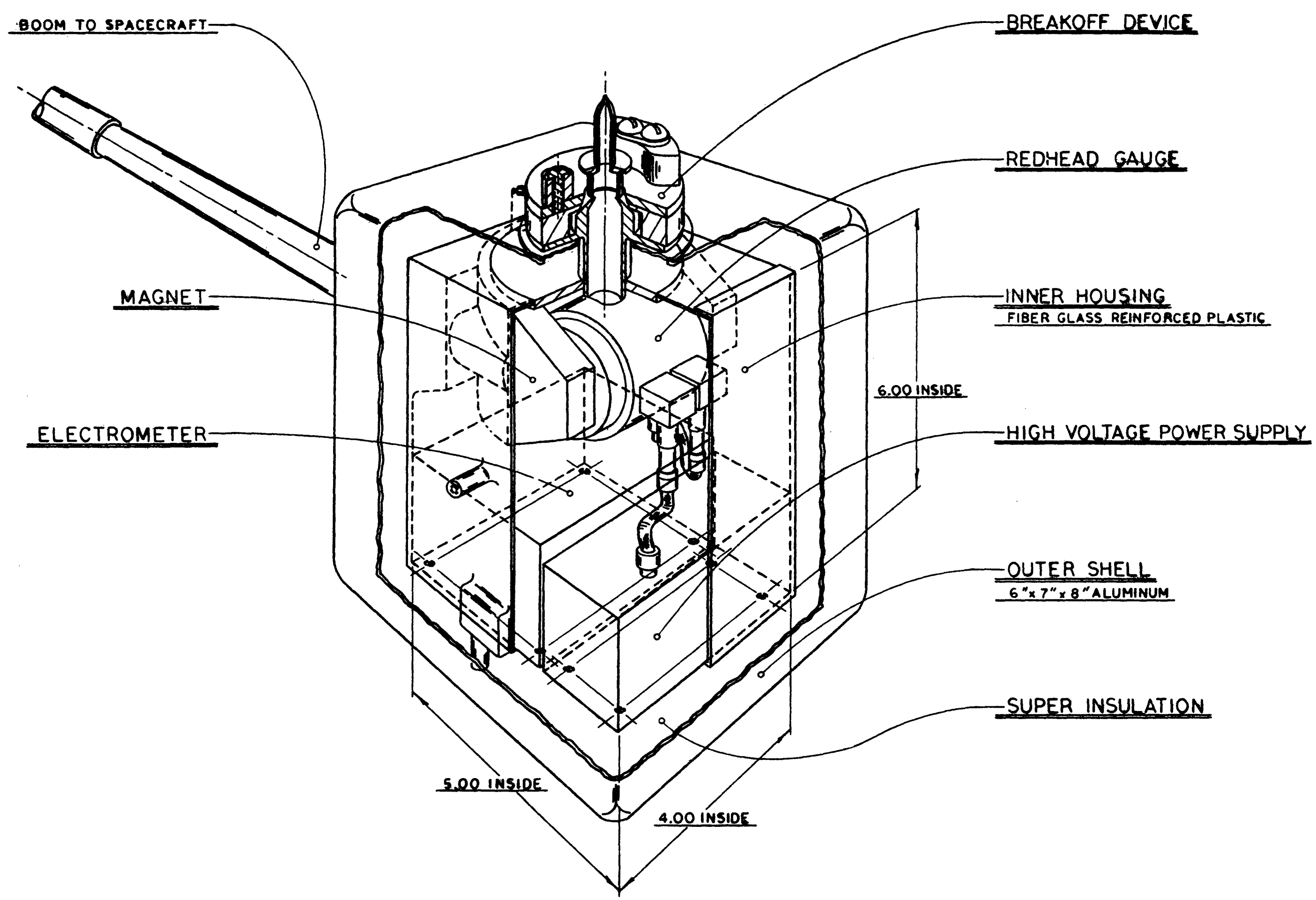

Fig. 1. Instrumented package (lunar atmosphere experiment).

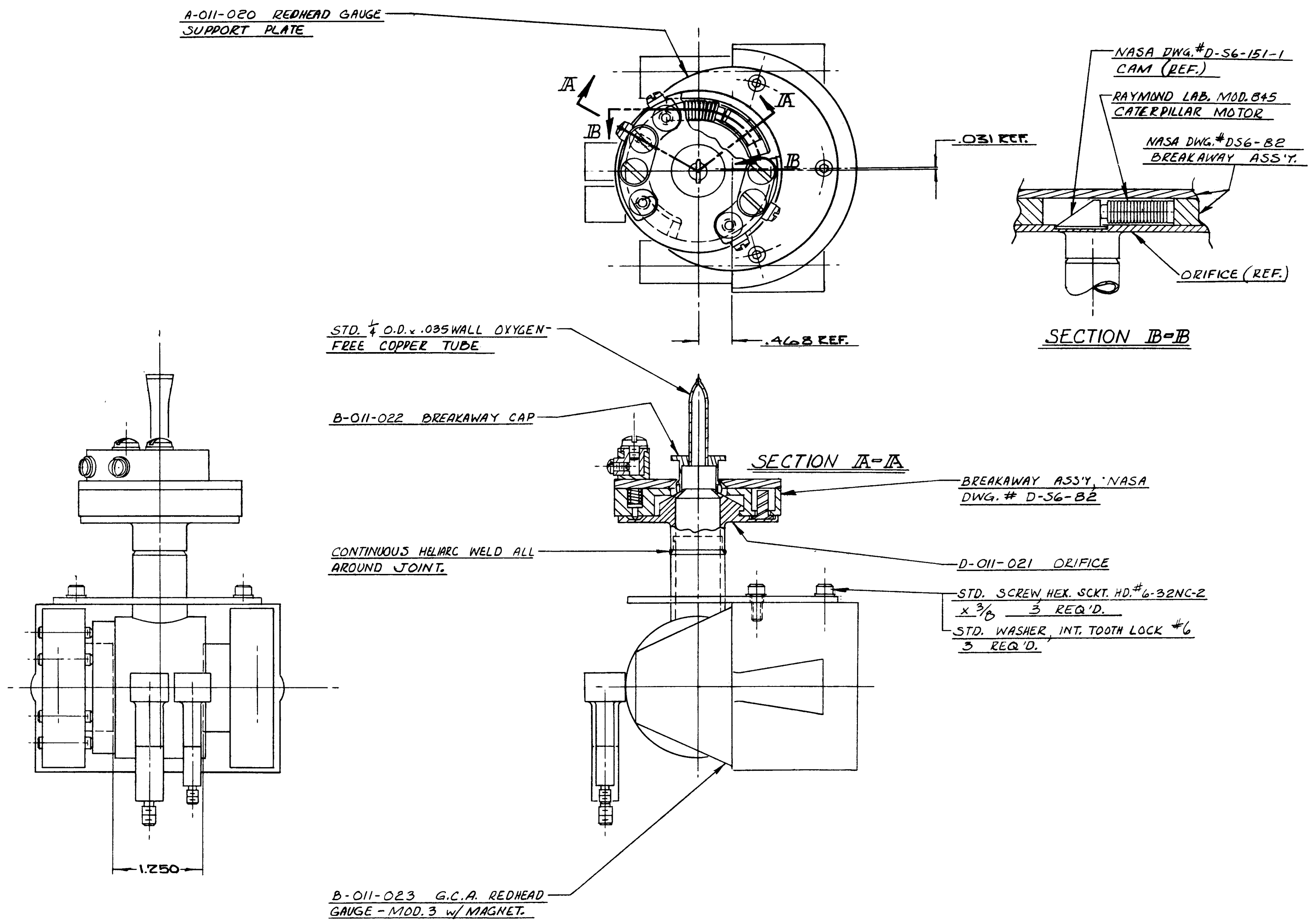

Fig. 2. Redhead gauge assembly.

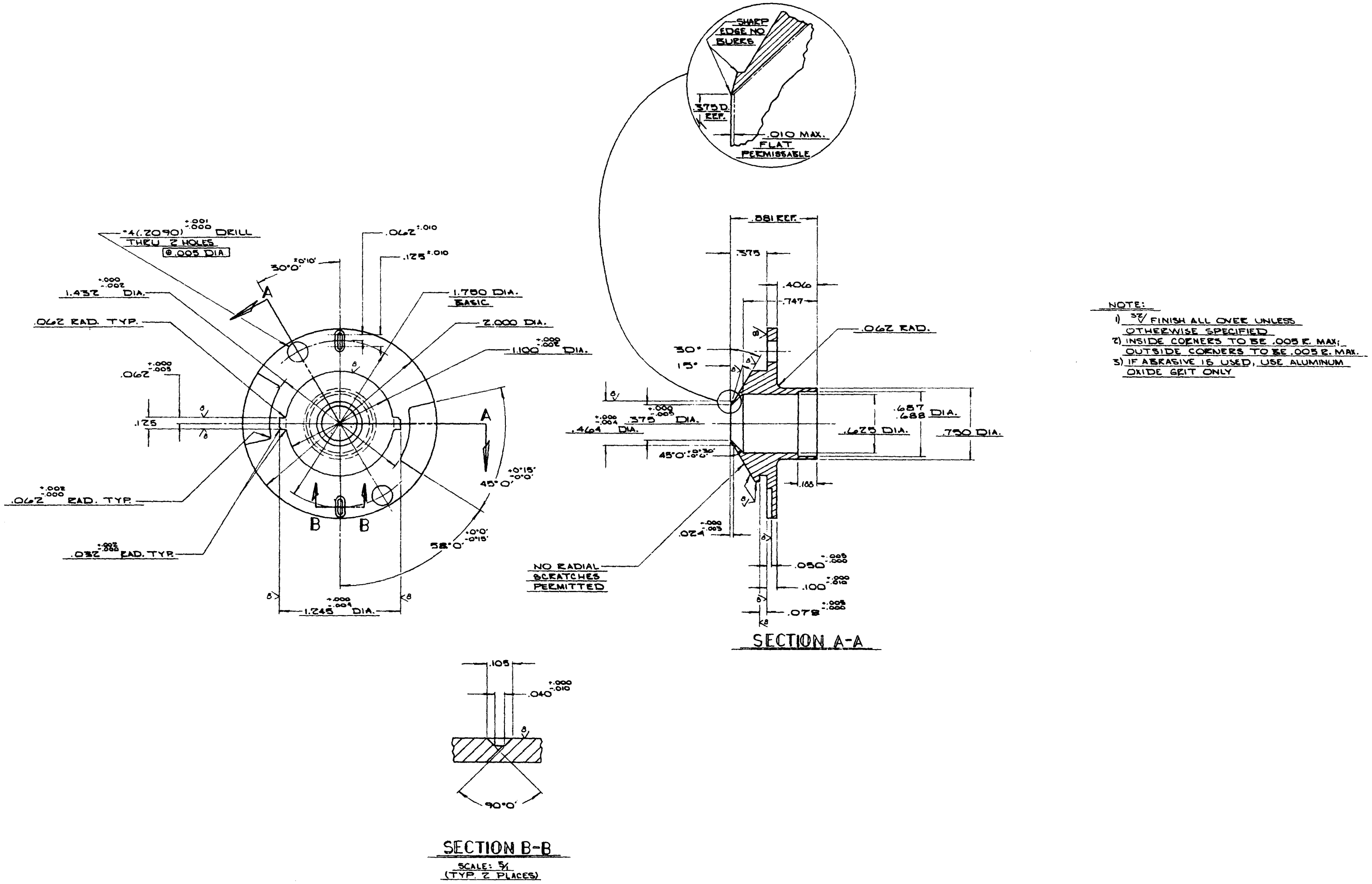

Fig. 3. Orifice for redhead gauge assembly.

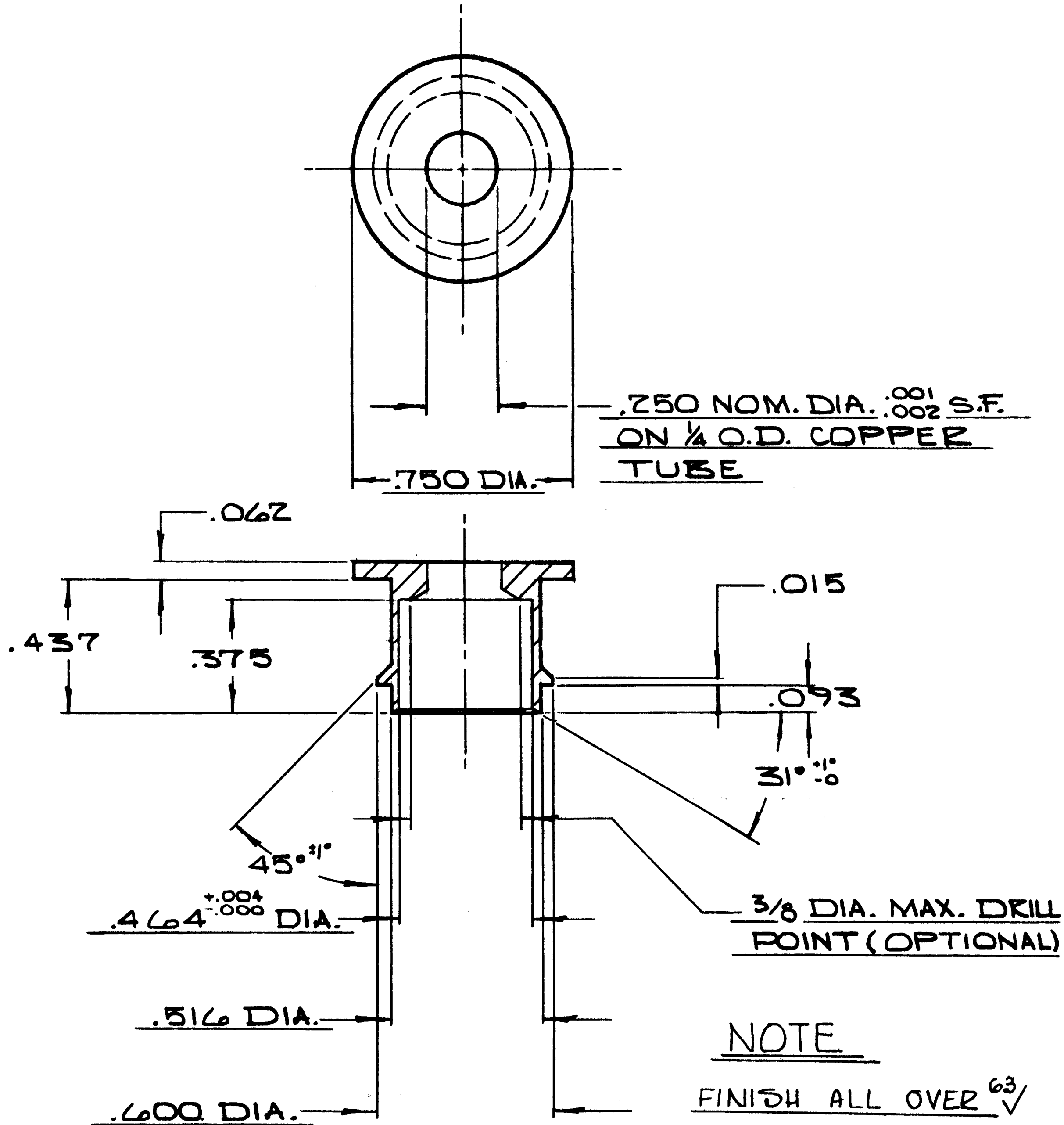

Fig. 4. Breakaway cap for redhead gauge assembly.

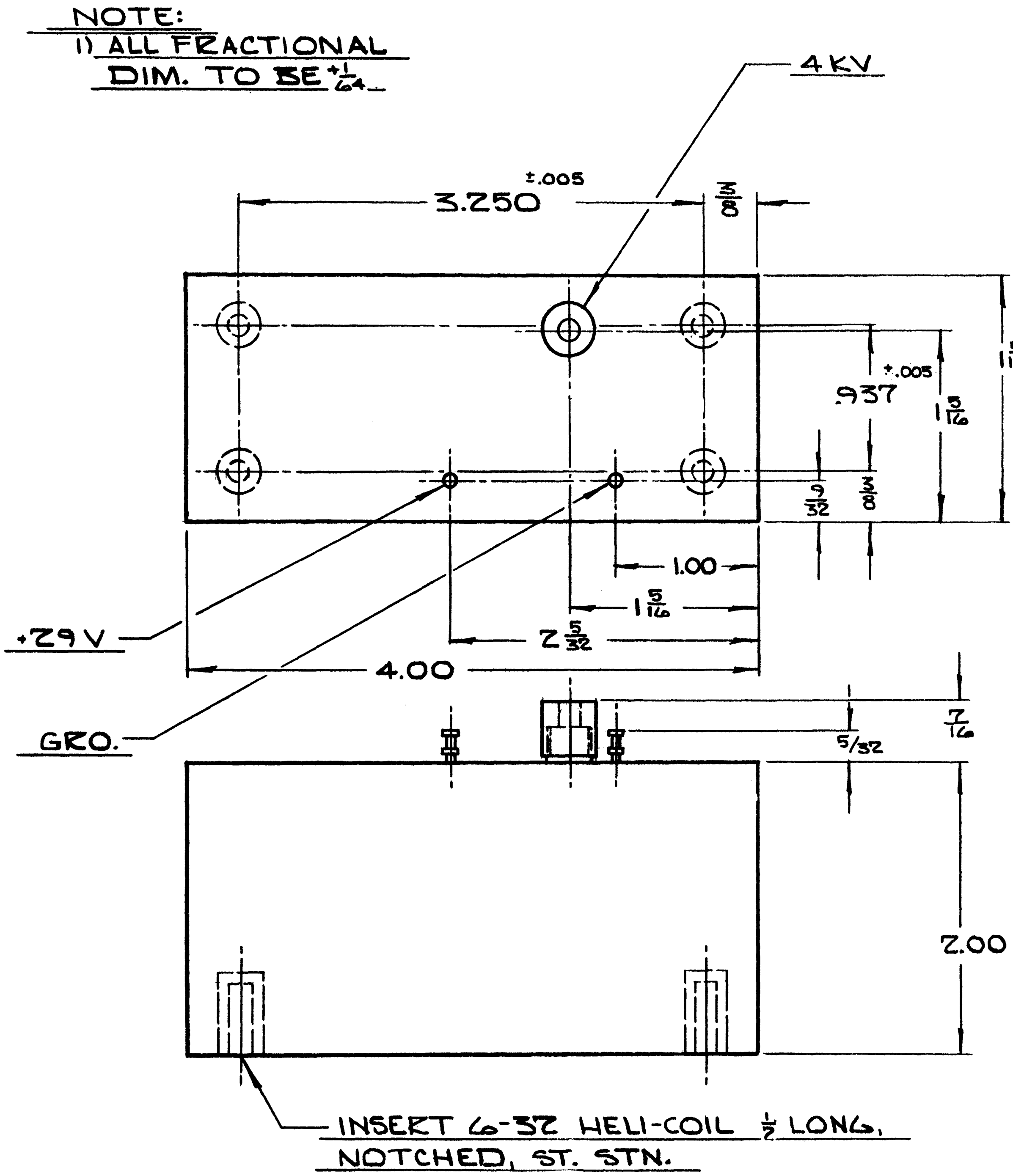

Fig. 5. High-voltage power supply for instrumented package.

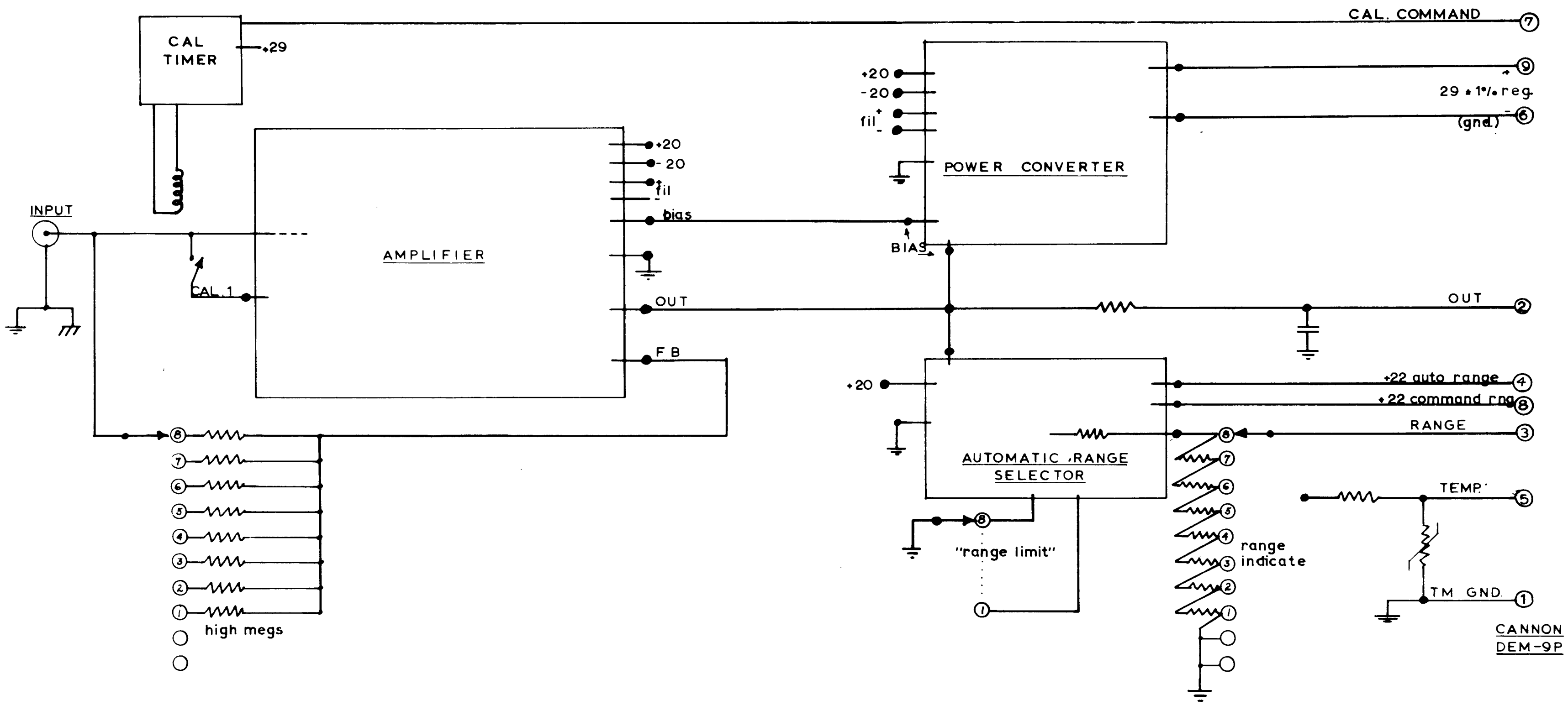

Fig. 6. Multirange electrometer amplifier (complete unit).

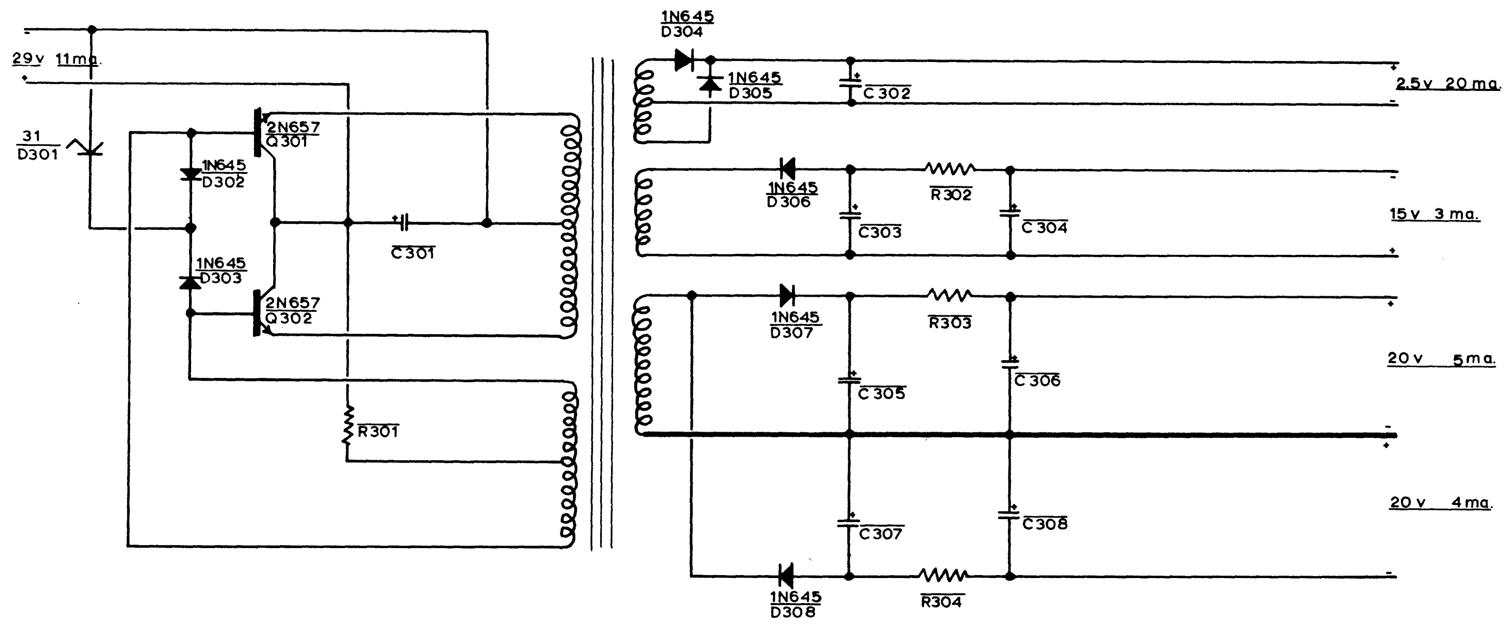

Fig. 7. Power converter for multirange electrometer amplifier.

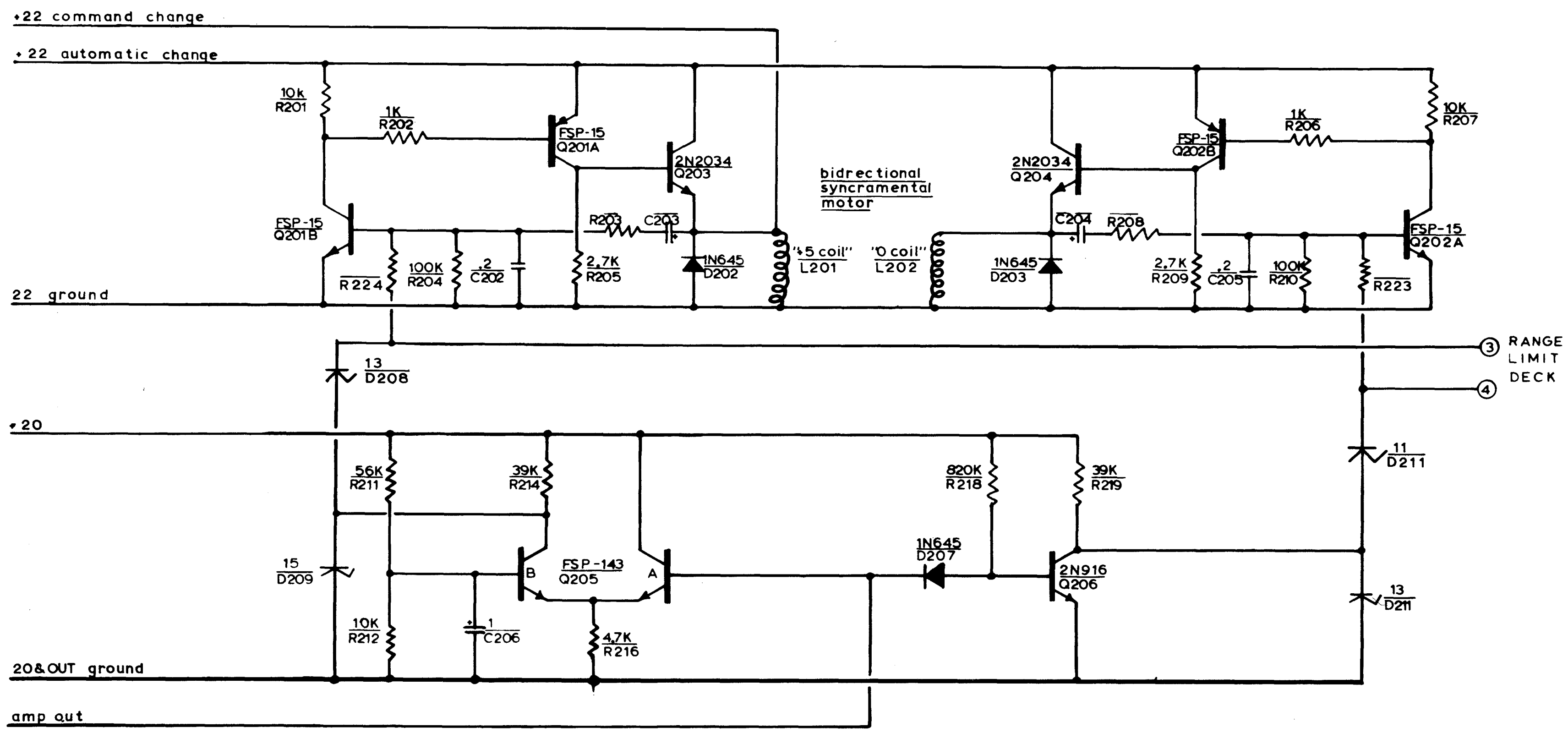

Fig. 8. Automatic range selector for multirange electrometer amplifier.

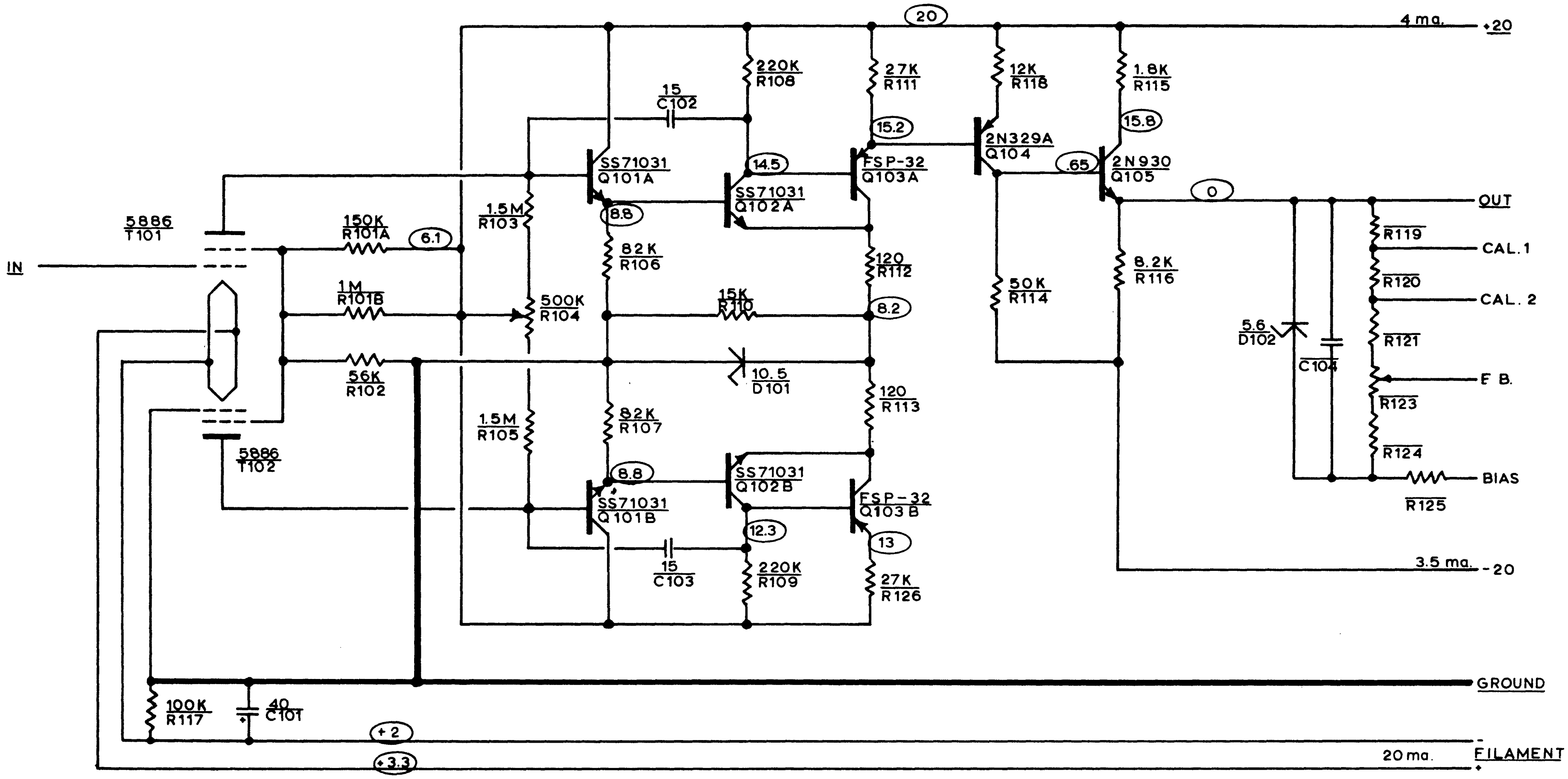

Fig. 9. Multirange electrometer amplifier (circuit diagram).

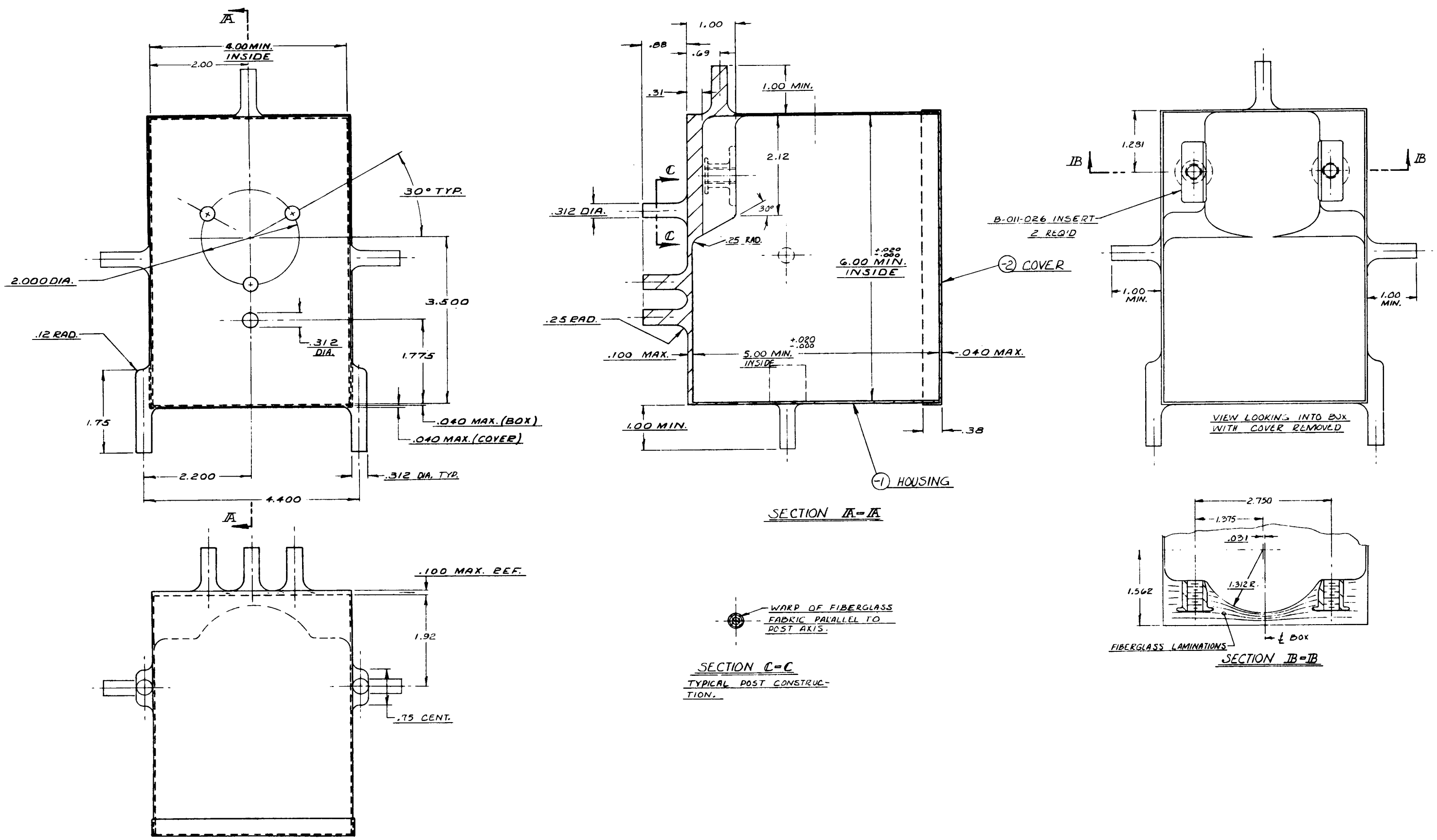

Fig. 10. Inner housing assembly for instrumented package.

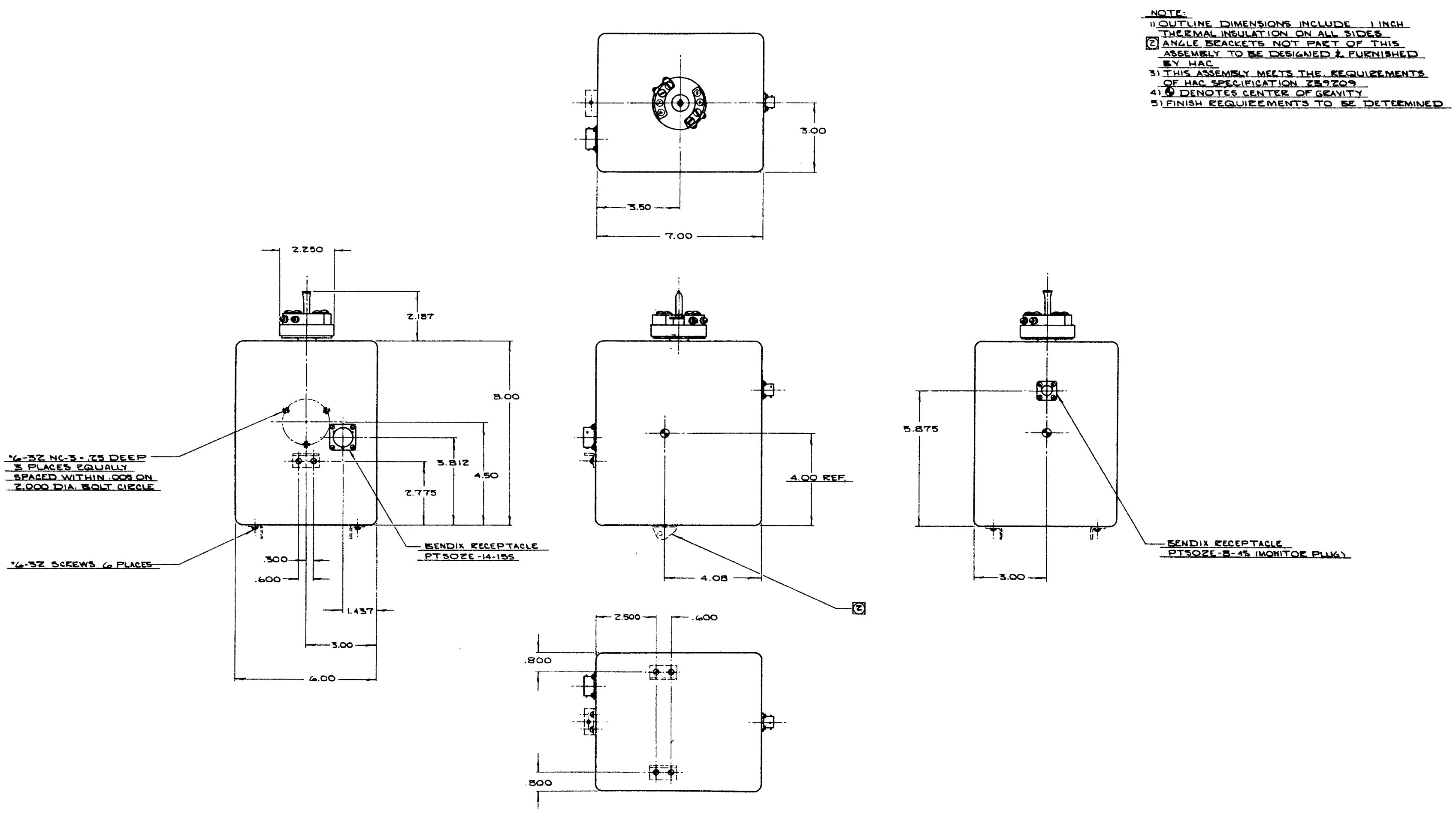

Fig. 11. Shell assembly for instrumented package.

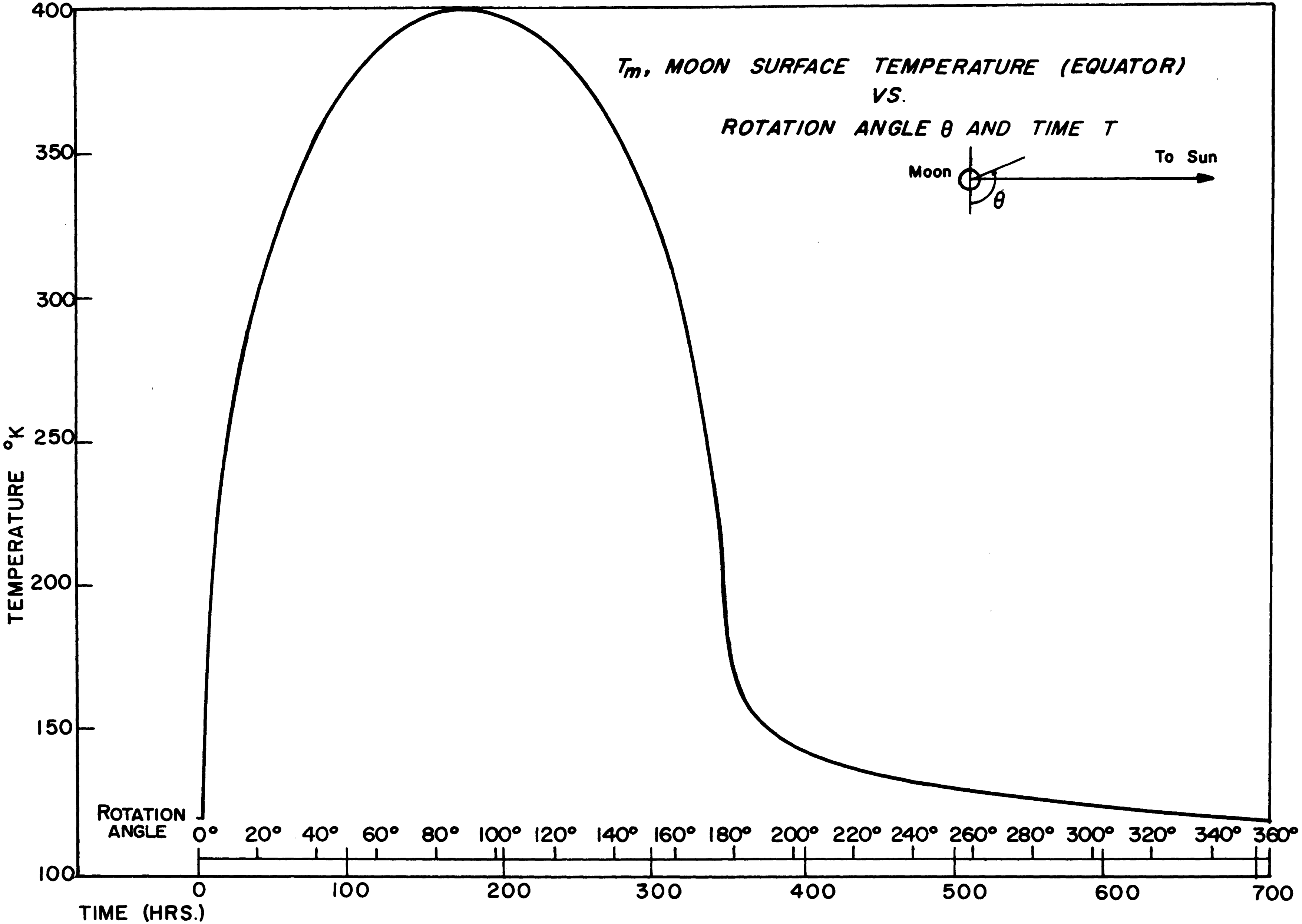

Fig. 12. Moon surface temperature vs. time and angle.