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LUNAR GRAVIMETER - INTERIM STUDY REPORT

AUGUST 1970

CHARLES STARK DRAPER
LABORATORY

CAMBRIDGE, MASSACHUSETTS, 02139

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LUNAR GRAVIMETER STUDY

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ACKNOWLEDGEMENT

This report was prepared under DSR Project 55-42200, sponsored by the Manned Spacecraft Center of the National Aeronautics and Space Administration through Contract NAS 9-10749.

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ABSTRACT

This report covers the technical progress on the development of a gravimeter for a lunar traverse gravity experiment. The conceptual design of a simple, lightweight low-power instrument resulting from the initial study effort is described. Special sections detail the oven structure, thermal analysis, and the electronic subsystems. Preliminary specifications are included for the instrument, operation, and the experiment error budget.

The entirely self-contained gravimeter features a digital readout, simplicity in both mechanical and electrical design, and 1-milligal accuracy. Electronic power consumption has been minimized, and a 100 watt-hour battery contained within the instrument meets our requirements. This instrument concept considers a manually course-erected system using a 3-degree bubble level as a reference. The remaining alignment error is eliminated by an internal passive pendulum.

Compiled by Sheldon Buck
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1.0 INTRODUCTION

This preliminary report summarizes the progress made to date to define a Traverse Gravimeter experiment and its associated hardware. During this first three months of the program effort was directed to define a conceptual hardware design, a preliminary gravimeter specification, and an experiment error analysis with its associated operating criteria.

The following MIT personnel contributed material to this report.

John Harper Sheldon Buck Frank Siraco William Toth Robert Reid Glenn Mamon Henry Trantham George Bukow	CSDL
Dr. Charles Wing	MIT Dept of Earth and Planetary Sciences

In addition, Dr. Manik Talwani of the Lamont-Doherty Observatory of Columbia University contributed material.

2.0 CONCEPTUAL DESIGN

2.1 GENERAL

The basic concept of the lunar gravimeter described in this report was to take a proven gravity measuring technique successfully used on ship board gravity surveys and design an astronaut operated instrument around the same type of sensor and provide the required environmental protection. The relative gravity sensor, a vibrating string accelerometer, was selected because of its extremely low power consumption and small size.

The gravimeter consists of three major subassemblies (Ref. Figure 1): the transport case with its three support legs, the gravimeter housing, and the pendulous element. The gravimeter housing is coarsely erected or aligned with the local vertical by the astronaut by using one of two bubble levels as a reference. The pendulous element contains all the active working parts of the instrument and is self-leveling over a ± 3 degree range to within 7 arc minutes. The active components on this pivoted structure include: the accelerometer and its readout electronics, a double oven, a digital display, a memory device, and the power supply with its primary battery. The gravimeter is a self-contained instrument having no external requirements for power, recording or telemetry, or environmental protection for its operation.

2.2 TRANSPORT CASE

The transport case (see Figure 1) serves the following functions:

a. Carrying case
b. Stand for setting on the lunar surface
c. Cradle to allow inverting the gravimeter
d. Caging mechanism for the pendulum.

It is a thin wall metal spherical shell, split and hinged at its equator. The interior of the lower half is smooth to provide a seat for the six low friction buttons on the gravimeter housing. This allows coarse gravimeter alignment and also inverting gravimeter for initial and final traverse bias readings. The exterior of the lower half will have carrying handles and three sockets for accepting snap-on legs.

When the upper half is closed and latched, it locks the gravimeter housing in the transport case and cages the pendulum. The caging is accomplished by locking three bullet-nosed pins which are fastened to the pendulum and project through clearance holes in the gravimeter housing. This operation places one of the pivot assemblies under tension.

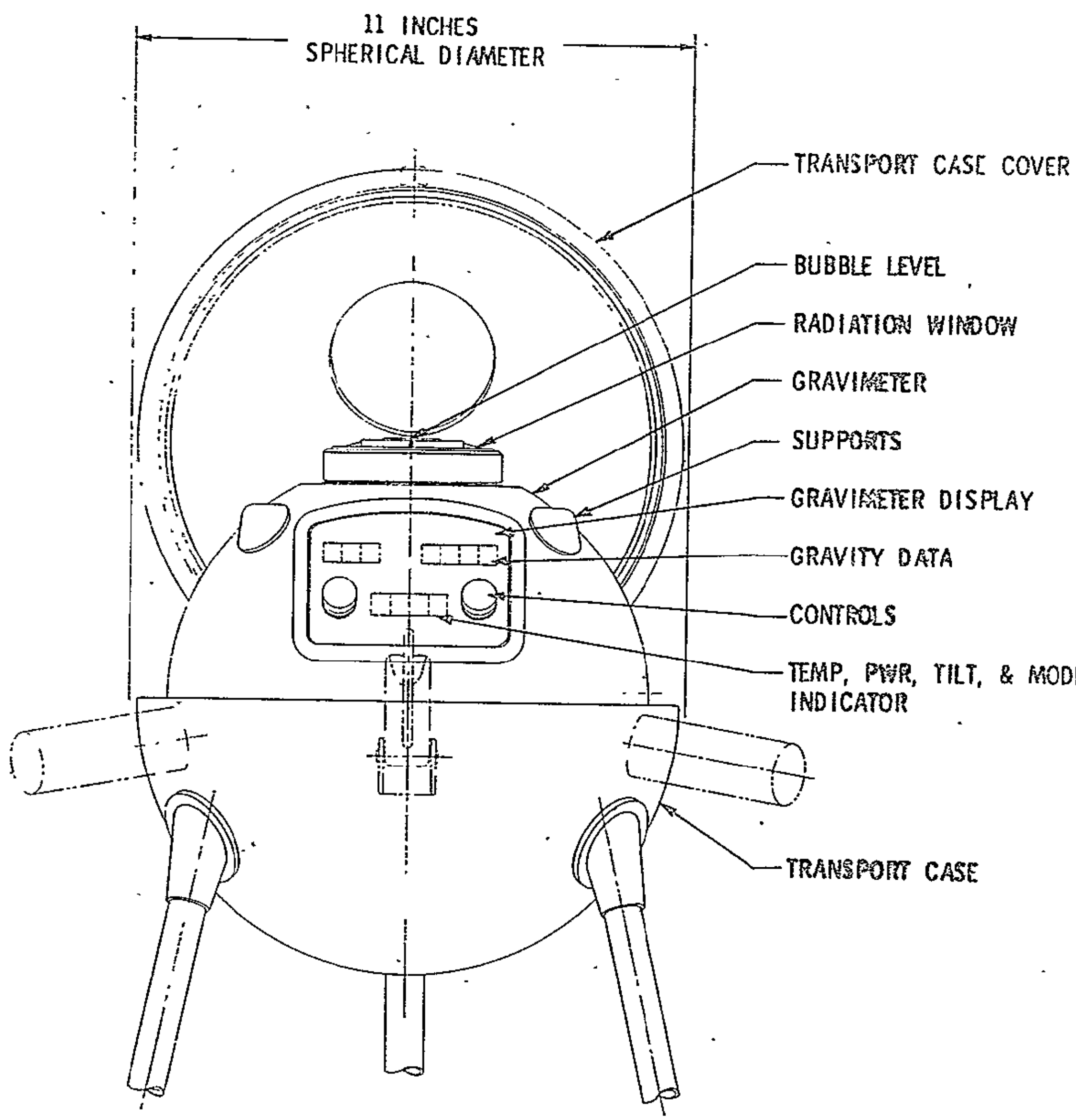


Fig. 1a Outline - Lunar Gravimeter - Front View

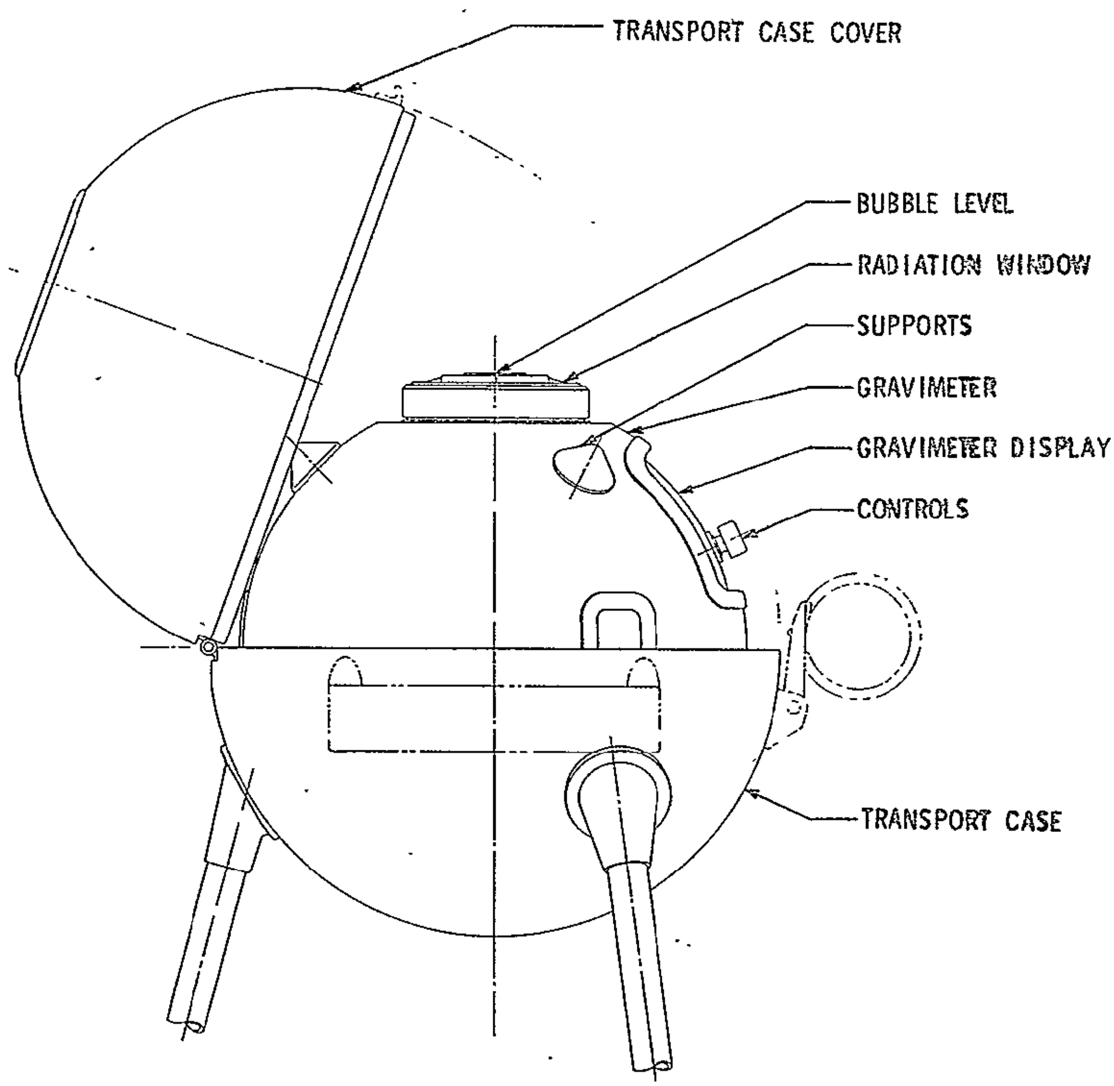


Fig. 1b Outline - Lunar Gravimeter - Side View

2.3 GRAVIMETER HOUSING

The gravimeter housing is the manually positioned, spherical member contained inside the transport case. This spherical housing has six conical teflon pads, three of which are supported by the lower hemisphere of the transport case (see Figs. 1 and 2). At the opposite end of the gravimeter housing is the other set of three stand-offs are used for support in the inverted or bias position. Two 2-axis bubble levels are located diagonally opposite each other on the housing as indicators of the input axis of the accelerometer which is inside. This housing must be positioned to within ± 3 degrees of the indicated vertical in order for the pivot assembly described in the next section (2.4.2) to function. Openings are provided in this housing are provided so that the operator can use the display and reach the switches located on the pendulous element.. Because no cable or flex wires bridge either of the two pivot assemblies, all of the instrument electronics and batteries are contained inside the gravimeter housing, i.e., mounted on the pendulous element.

2.4 PENDULOUS ELEMENT

2.4.1 General

The pendulous element, which is self-aligning within ± 3 degrees of vertical, contains the gravity measuring components of this instrument. Referring to Figure 2 we can see that the pendulous element is composed of pivot assemblies, the E frame, an intermediate oven, a precision oven, and the vibrating string accelerometer and its support.

2.4.2 Pivot Assembly

The basic gravimeter package is a passive pendulum with a pivot assembly at both ends. The upper pivot assembly is active while the lower pivot assembly is unseated, allowing 3 arc degrees of off-axis pendulum motion. When the instrument is inverted to take a bias measurement, the basic gravimeter package moves 1/2 inch axially and the pivot assembly roles are reversed.

The pivot assembly (see Figure 2) consists of a 3/8 diameter metal ball which pivots in a conical surface of an intermediate part which is captive with and moves axially with the ball. The outer surface of the intermediate part also is conical and mates with a corresponding seat in the housing. The center of gravity of the intermediate part coincides with the pivot ball center in order to eliminate disturbing torques from the unseated pivot assembly. Seals will be provided to exclude dirt from the pivot and retain the lubricant. A worst expected coefficient of friction of 0.02 will result in a tolerable pendulum misalignment of less than 4 arc minutes.

2.4.3 E Frame

2.4.3.1 General - The E frame serves as the electronics support and houses virtually all the electronic modules, the digital display, the battery package, and levels.

Maintaining the temperature of the E frame within reasonable limits is achieved by dissipating the oven and electronics heat by the E frame radiator. A thermal radiation heat sink is provided at either end of the pendulous elements near the pivot assemblies. A more detailed description of the thermal control concept will be found in Section 4.0.

2.4.3.2 Electronic Modules - The electronic modules consist of an analog package, two digital packages, and power supplies. The analog modules are: three temperature controllers, two VSA or string amplifiers, two oscillators, a mixer, a signal conditioner, and the level electronics. One digital module contains the drivers for the gravity measurement display, the temperature, power, level, and mode indicators, and the mode control logic. The second digital module contains the readout logic and memory electronics.

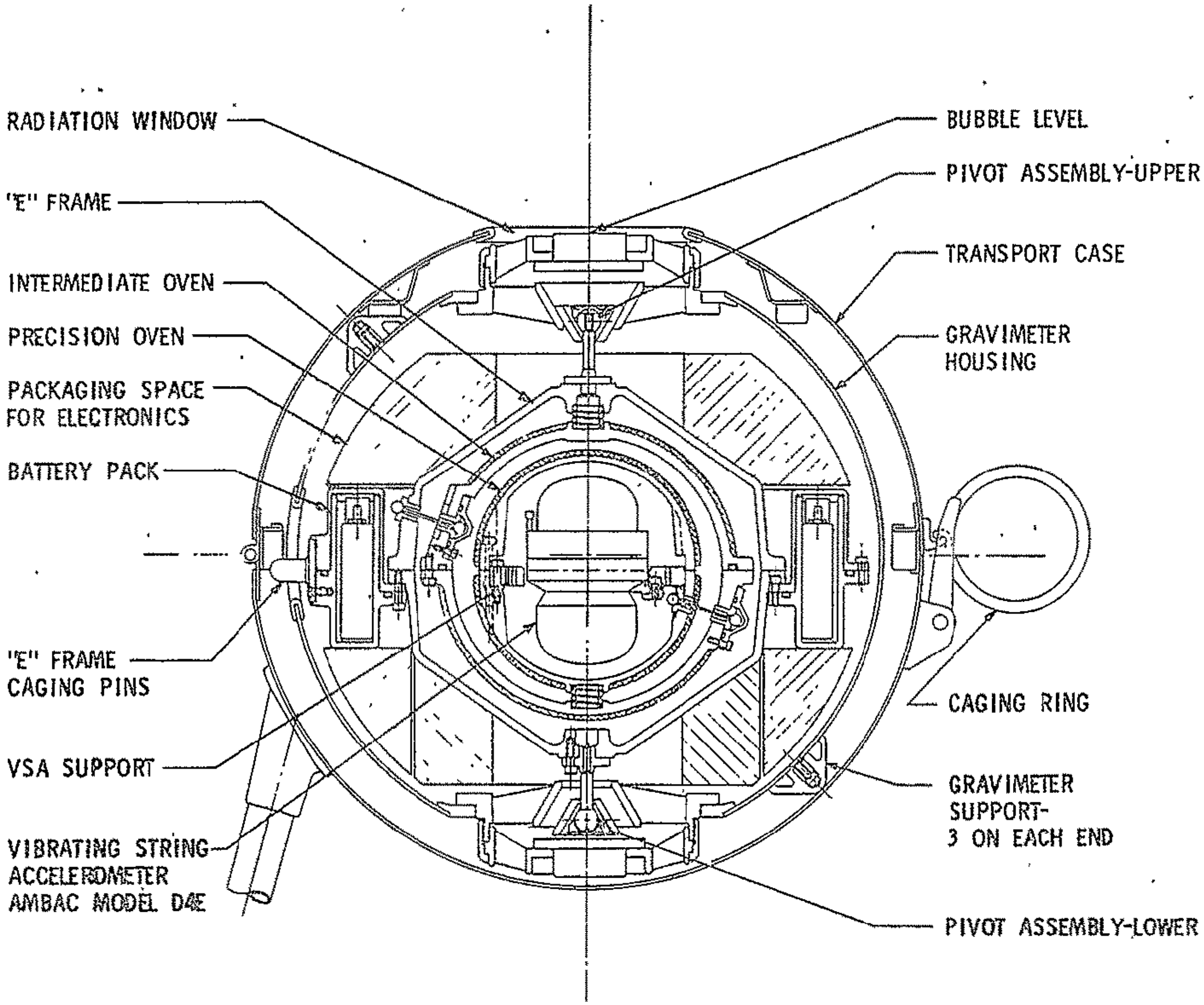


Fig. 2 Cross Section - Lunar Gravimeter

An active memory will have its own permanently connected battery/power supply. The main power supply feeds all circuits except for the memory. Details of the electronic systems are explained below.

2.4.3.3 Digital Display - On the digital display panel (see Figure 1) are a seven digit decimal display of the gravity measurement, a four letter display of the instrument operating status, and, two of the mode switches. The seven digit display of the divided VSA period represents the gravity measurement. This period will be approximately 30 seconds in the normal gravity measuring position and therefore produce a quantization level of 0.1 milligals. With the gravimeter in the inverted or bias measuring position, the VSA period of 10 seconds will give a quantization of 0.3 milligals. This seven digit binary coded decimal display is divided into an initial group of three digits followed by a group of four digits. Except for initial transfer and bias readings only the last four decimals need be transmitted for gravity data during a traverse. The 4 letter display of instrument operating status operates as follows:

First position	Temperature C - cold H - hot
Second position	Battery P - low
Third position	Measurement O - on
Fourth position	Level L - tilt

Temperature	Battery	Measurement Mode	Level
C or H	P	O	L
Normally Clear	Normally Clear	on when period measurement taking place	on whenever level indicates greater than 7 min.

2.4.3.4 Battery Pack - The battery power supply consists of eighteen 1 1/2 volt silver zinc primary cells connected series-parallel. The nine volt battery pack is capable of delivering 75 watt-hours of power. The eighteen primary cells which weigh 1.9 lbs were selected over secondary cells because of their higher energy density. The cells are potted in a hermetically sealed annular magnesium casting (see Figure 2) creating a readily removable battery assembly. The hermetic seal is effected by o-ring flanged joints screwed together to allow manual activation of cells by the addition of potassium hydroxide electrolyte shortly before installing the gravimeter on the LM. A wet life of 10-25 days is expected. Individual cells have normally closed low pressure relief valves and the battery case has a redundant relief valve.

If further study indicates that more power is required, 15 cells of the next larger size could be grouped in an annular casting of approximately the same diameter with 1/4 inch increase in height. The cells would weigh 2.4 lbs and interconnected to form a 7.5 volt, 100 watt-hour battery. The E frame also supports the magnesium battery case with its silver-zinc primary cells.

2.4.3.5 Level - Depending upon whether the pendulous element is operated on its normal or inverted pivot, one of two two-axis levels is used to monitor the alignment of the accelerometer input axis with the local vertical. When the astronaut initiates a gravity measurement, the levels provide a display signal whenever the input axis deviates by more than 7 arc minutes from vertical. Moreover, when the input axis remains within 7 arc minutes for more than 2 seconds, a gravity measurement is automatically taken. If this condition is not reached in 2 minutes time, a gravity measurement is taken by providing an over-ride signal and the existence of vertical deviation is indicated on the display.

The level selected for this instrument is a two degree of freedom simple pendulum whose mass constitutes the moving portion of two orthogonally-mounted differential transformers. The phase-sensitive output signal is proportional to the tilt angle about the two sensing axes.

2.4.4 Oven Assembly

The VSA is housed within two concentric spherical shells which are mounted inside the E frame (See Figure 2).

The VSA is mounted on the inner edge of a stainless steel support ring containing interrupted circumferential slots to provide a high thermal impedence between the inner and outer edges. The outer edge of the support ring is mounted at the equator of the inner spherical shell (precision oven). The precision oven is suspended inside the outer spherical shell (intermediate oven) by three 0.040 diameter stainless steel wires and a compression spring to provide high thermal impedence. Wires are threaded at both ends and screwed into steel balls. The intermediate oven is similarly suspended within the E frame. The location of wires and springs are discussed in Section 2.7.

The spherical shells are magnesium castings with parting lines at the equator. Heater blankets are intimately bonded to their outer surfaces. Steel bushings are provided for seating the balls of the wire suspension. O-rings are provided at all penetration points of the intermediate oven to provide a hermetic seal.

With the aid of suitable fixtures, the following assembly procedures are anticipated. The lower halves of the precision and intermediate ovens are pre-assembled with the three wire assemblies and compression spring. The upper halves of the intermediate oven and E frame are similarly pre-assembled. The VSA is mounted to its support which is then fastened into the lower half of the precision oven. The upper half of the precision oven is then assembled. This assembly is then fastened to the pre-assembled intermediate oven and E frame half shells. Note that the wire suspensions need not be dismantled should it be necessary to replace the VSA.

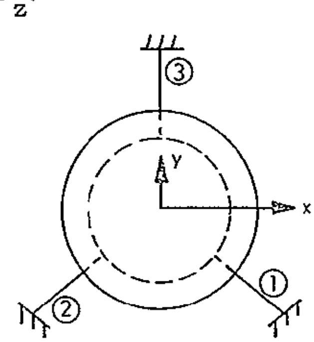
2.4.4.1 VSA Thermal Isolation - Structural Analysis - The VSA is isolated from the external environment by a pair of concentric spherical shells which are each suspended by 3 preloaded wires. In this system the unit is securely positioned with a minimum of physical connection to the outside.

2.4.4.1.1 Inner Shell - The inner shell has radial support wires located at latitude 19.43° ($\sin^{-1} 1/3$) South, longitude 0, 120 East, and 120 West. Nominal wire specifications are:

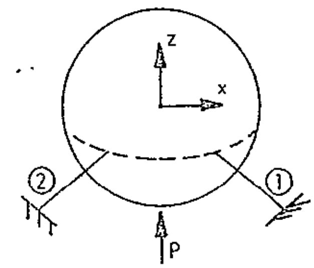
Length	-	0.875"
Diameter	-	0.040"
Material	-	Series 300 Stainless steel, hard drawn, 185×10^{3} psi UTS min.

A 50# preload is supplied by a compression spring at the South Pole. Wire load is equal to the preload for the particular geometry chosen and stress in each wire is 40,000 psi. Assuming a set of coordinates as shown gives the following expression for the wire tensions for body forces F_x, F_y, and F_z.

(Z is the VSA input axis)



$$\begin{bmatrix} T_1 - P \\ T_2 - P \\ T_3 - P \end{bmatrix} = \begin{bmatrix} -0.615 & +0.354 & +1 \\ +0.615 & +0.354 & +1 \\ 0 & -0.708 & +1 \end{bmatrix} \begin{bmatrix} F_x \\ F_y \\ F_z \end{bmatrix}$$



The first possible mode of failure considered is either breaking or slacking of the wires. In the z direction the limits are +160 g's (breaking) and -66 g's (slackening) assuming a 3/4# total inner package weight. The smaller of these is above the limits for the VSA.

The spring constants are 1.35×10^4 #/in. in the z direction and 5.40×10^4 in.the x and y directions. There is no cross coupling between the axes.

Small angular deflections will give rise to restoring torques by displacing the direction of the wire tension away from the cg of the inner package and produce only second order charges in wire tensions or preload. The stiffnesses are 454 in #/rad for z rotation and 284 in #/rad for x and y.

2.4.4.1.2 Outer Shell - The wire suspension of the outer shell is identical to the inner one except that is is upside-down. The translational stiffnesses are the same 1.35×10^4 and 5.40×10^4 #/in. but the angular spring constants are greater because the tension loads are applied at a greater radius. Their values are:

$$K_x = K_y = 534 \text{ in \#/rad}$$

$$K_z = 855 \text{ in \#/rad}$$

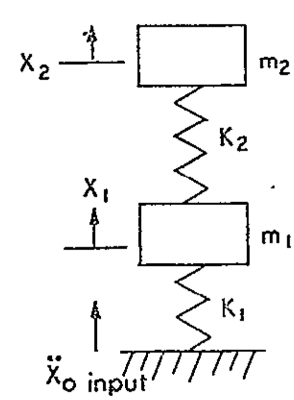
2.4.4.1.3 Dynamic Response - Because there is no cross coupling between axes for either translational or rotational motion each axis may be treated as a two degree of freedom system going from outer case to inner shell.

K_1 = outer set of wires

K_2 = inner set of wires

m_1 = outer shell

m_2 = inner VSA package and shell



The responses are:

Translational z direction

$K_1 = 1.35 \times 10^4$ #/in.

$K_2 = 1.35 \times 10^4$ #/in.

$m_1 = 2.03 \times 10^{-3}$ # sec^2/in.

$m_2 = 2.34 \times 10^{-3}$ #sec^2/in.

$f_1 = 338$ Hz

$f_2 = 925$ Hz

Translational x and y

$K_1 = 5.40 \times 10^4$ #/in.

$K_2 = 5.40 \times 10^4$ #/in.

$m_1 = 2.03 \times 10^{-3}$ # sec^2/in.

$m_2 = 2.34 \times 10^{-3}$ # sec^2/in.

$f_1 = 676$ Hz

$f_2 = 1850$ Hz

Angular z

$K_1 = 855$ in. #/rad.

$K_2 = 454$ in. #/rad.

$I_1 = 6.85 \times 10^{-3}$ # in. sec^2

$I_2 = 2.88 \times 10^{-3}$ # in. sec^2

$f_1 = 38.6$ Hz

$f_2 = 91.4$ Hz

Angular x and y

$K_1 = 534$ in. #/rad.

$K_2 = 281$ in. #/rad.

$I_1 = 6.85 \times 10^{-3}$ # in. sec^2

$I_2 = 3.18 \times 10^{-3}$ # in. sec^2

$f_1 = 32.4$ Hz

$f_2 = 64.6$ Hz

2.4.5 Vibrating String Accelerometers

The potential for gravity measurements with accelerometers has recently been proven in the Bell Aerosystems sea gravimeter and the MIT surface ship gravimeter, (Wing, 1969).

In a sense, gravimeters are but a special type of accelerometer. In particular, they are high sensitivity, narrow range (1g ± 0.001g), accelerometers. Most inertial guidance accelerometers have low sensitivity and wide range (± 50g). Accelerometers which may be considered for lunar gravimetry have both high sensitivity and wide range.

Due to the application for which they are designed, inertial guidance accelerometers have many attractive features for lunar gravimetry: wide range, digital readout small size and weight, and most important, a method of calibration at low accelerations. Many accelerometers may be tilted a full 90° and measure only the sensitive axis component of gravity ($g \cos \theta$) to within ± 1 mgal. When the range is restricted to that encountered in gravimetry, the stability of accelerometers is comparable to that of quartz gravimeters. For example, the stabilities (allowing for linear drift correction) required for lunar gravimetry (0.1 mgal/12 hr., 0.5 mgal/week) are presently met by at least four accelerometers.

The best accelerometer, as judged on the basis of all of the factors important in lunar exploration is probably the American Bosch Arma vibrating string accelerometer, followed closely by the Bell Aerosystems DVM (Digital Velocity Meter).

The vibrating string accelerometer (VSA) consists ideally of a pair of single vibrating strings back to back. As in Figure 3, separate but equal masses, joined by a soft isolating string, are suspended by identical beryllium copper strings. Let the accelerometer input vector coincide with the gravity vector and let g represent the acceleration of gravity. Expressing the frequency of each string as a MacLaurin's series:

$$F_1 = K_{01} + K_{11}\, g + K_{21}\, g^2 + K_{31}\, g^3 \ldots$$
$$F_2 = K_{02} - K_{12}\, g + K_{22}\, g^2 - K_{32}\, g^3 \ldots$$

The negative terms in series 2 are a result of the inverse relationship between gravity and tension on the lower string. Subtracting the series to obtain the difference frequency:

$$\Delta F = F_1 - F_2 = (K_{01} - K_{02}) + (K_{11} + K_{12})\, g + (K_{21} - K_{22})\, g^2 \ldots$$

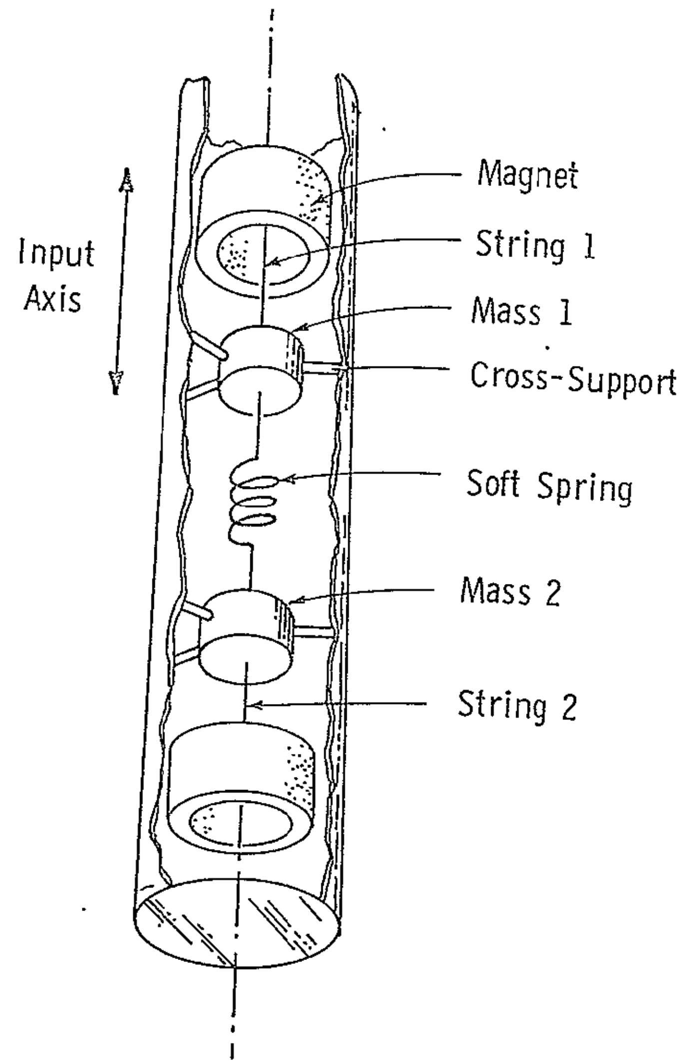


Fig. 3 VSA Schematic

If the strings were identical, all even order terms would be zero. The short term stability of this relationship is about 1 mgal/24 hrs., and the predictability is about ± 0.03 mgal/24 hrs. Details of how the VSA difference is processed is explained in Section 2.5.2 below.

The double string VSA shares all the desirable features of the single string. It can be seen that combining two oppositely oriented strings, however, results in several improvements which are very important in lunar gravimetry: 1. end to end support, combined with cross supports, eliminates cross-coupling, thus allowing calibration at 1/6g on the tilt table, 2. the support system eliminates the need for damping or clamping and unclamping, 3. the difference frequency between the two strings is a nearly linear indication of gravity.

During the first three months of the program, effort was spent in determining if the D4e vibrating string accelerometer would be satisfactory.

Contacts were made with Arma, Kearfott, and Holloman AFB to obtain background information. Kearfott was very helpful as they had performed a qualification test on the D4e using environments one would encounter for a ballistic missile application. On the basis of these tests CSDL has concluded that the instrument may be satisfactory for the gravimeter application provided additional qualification testing to the exact apollo launch environment does not identify any problem areas.

Since receiving seven units in mid-June, CSDL has been conducting a preliminary screening with the following results

1. Three units, Arma serial numbers 23, 27, and 48 show promise of meeting the gravimeter requirements. The one sigma bias stabilities obtained on these instruments whose temperature is being controlled to within an estimated 0.01°F peak to peak were

S/N 23	less than 0.16 micro g
S/N 27	less than 0.22 micro g
S/N 48	less than 0.37 micro g

S/N 23 and 27 presently meet the required 0.5 micro g two sigma specification (see following page). S/N 48 does not presently meet that specification, but until a more precise knowledge is available on temperature stabilities, temperature gradients, and voltage stabilities presently being obtained in the current test set-up, the unit should not be eliminated as a possible candidate.

2. The ability of the VSA to perform well across low level vibration; e.g., on ocean going gravimeters indicates its suitability to perform within specification of the lunar traverse. Further testing is required on the instruments to verify this capability.

3. No information presently on hand indicates the effects of low level shocks (due to handling) on the VSA performance. During the test program these effects will be investigated and suitable instrument shock mounting will be employed as required.

4. Test results obtained on the STAFF program indicate that the VSA can withstand launch vibration and acceleration environments and still meet the lunar transfer requirement. This capability will be verified during the test program.

In view of the above, and assuming a temperature control requirement of $\pm\ 0.02^{o}F$ peak to peak, CSDL believes that the program can proceed with good confidence. Since a total of six acceptable units are required to fully support the overall program (including three flight qualified, one qualification, and two reliability* units) the four additional GFE units to be obtained are a necessity.

As a total of eleven D4e accelerometers will be furnished CSDL, 7 from ERC inventory and 4 from Marshall Spaceflight Center, the expectation is that sufficient units will be available to support the program. At this time it is not contemplated that additional units will be purchased from an industry source. The long lead time, 8-12 months, and high cost, approximately $500K, seem unrealistic at this time if flight equipment is required by late 1971. If for some unknown reason the available D4e instruments prove unsatisfactory an alternate approach will be investigated.

2.5 ELECTRONIC SUBSYSTEMS

2.5.1 General

Electronic subsystems for the Lunar Gravimeter experiment consist of: VSA Signal and Readout Electronics, Temperature Control Electronics, Pendulum Electronics, and Power Supply Electronics. Beyond required accuracy and reliability considerations, design of these functions in their various modes, was aimed at minimizing the power and the time needed to perform each function. A natural advantage of this approach is the attendant decrease in weight of the energy source.

*Units with performance slight outside the flight specification will still be suitable for reliability testing.

Also considered was the convenience with which the astronaut could operate the equipment and be assured of obtaining good data with a minimum of training.

2.5.2 VSA Signal and Readout Electronics

2.5.2.1 General - Obtaining a readout in display and memory from the VSA string frequencies involves several stages of signal processing. Basically, this design approach processes the VSA difference frequency into a time gate, or period, which is a function of the difference frequency and which is subsequently displayed and stored. By mixing and filtering the outputs of the VSA oscillator amplifiers, converting the difference frequency from a sine wave to a pulsed waveform to drive the readout logic, and counting it down to an appropriate frequency, the VSA difference frequency is processed as a gate and its pulse width is measured. A convenient time gate is provided by the countdown of the frequency appearing at the output of the signal conditioner. The resulting time gate is used to gate a high frequency clock. This gated clock is counted in a 7 decade BCD counter where outputs are strobed into a storage register. Outputs from this register feed a visual display and a memory, but the register will retain the data until the astronaut initials a new measurement cycle. Display readings may be taken anytime after the completion of a data cycle unless a new cycle has been initiated. A level by-pass feature is powered so that a reading may be taken after a fixed time interval even if there is no level indication.

In the signal-generation-to-readout chain are the functions of: oscillator amplifiers, mixer, signal conditioner, logic, display and memory.

2.5.2.2 Oscillator - Amplifiers - The VSA oscillator amplifiers as designed by ARMA have been evaluated as to power dissipation and temperature stability. The ARMA design consists of four silicon transistors and two transformers such that the amplifiers total power consumption is one watt. Most of this power is consumed in the biasing of the transistors. It was found that by changing only three biasing resistors, the amplifiers were modified to accept a 15 volt supply instead of a 28 volt supply, thereby reducing the total power consumption to 250 mw. To minimize temperature sensitivity, precision low temperature coefficient resistors can be used in critical portions of the amplifier. The driving circuit for the VSA, which simply consists of two diodes and a resistor, will be packaged with the VSA in the internal oven. By holding the temperature of these components constant, there should be a large attenuation of the VSA's frequency drift with temperature.

A new design employing low power operational amplifiers is being investigated. It is anticipated that the power consumption can be reduced even further.

2.5.2.3 Mixer - To obtain the difference frequency between the two outputs of the oscillator amplifiers, a simple ring demodulator is used as shown in Fig. 4. A 0.1 μf capacitor across the output is used to filter out the sum frequency and to pass only the difference frequency.

2.5.2.4 Signal Conditioner - To drive logic, the sine wave output of the mixer must be converted to a pulse type waveform. The sine wave output of the mixer is further filtered by an R-C integrator and the input signal is clamped by two back to back silicon diodes. The signal is fed to a National semiconductor LM 111 comparator, the output of which is a pulsed waveform whose amplitude is 5 volts and whose frequency is the same as that of the sine wave produced by the mixer.

2.5.2.5 Logic - The difference frequency of the VSA which appears at the output of the signal conditioner must be quantized properly and put into a proper form before the data can read into display and/or permanent memory.

The difference frequency will be about 130 Hz at 980 gals (earth) and about 20 Hz at 162 gals (moon). The necessary quantization to obtain 0.1 mgal resolution is therefore about 1 part in 10^7 on earth and 1 part in 1.62×10^6 on the moon.

The proposed readout method involves the use of the VSA output signal into a time gate. A convenient time gate is provided by the countdown of the frequency appearing at the output of the signal conditioner. The resulting time gate is used to gate a high frequency clock.

Figure 5 shows the proposed conceptual logic drawing. Figure 6 shows timing waveforms. The main purpose of the control logic is to generate a gate whose pulse width is a direct function of the VSA difference frequency. This gate is then used to gate a clock to a counter whose output is then strobed into a register. The register will hold the data until a new word is strobed into it. Register output feeds the visual display and memory.

One method of reading out the VSA would be to count zero-crossovers of the difference frequency. Since the 20 Hz output of the VSA will yield 20 zero-crossings per second, to obtain 1.62×10^6 zero crossings, a readout time in excess of 1000 minutes would be needed. This is obviously impractical. If we say the readout time should be limited to 20 or 30 seconds, and if we assume we could improve the resolution of the VSA by multiplying the VSA frequency (e.g., a phase-locked multiplier) we would have to multiply by about 4000. Assuming a stable phase-locked multiplier can be built with a ratio of 4000, the disadvantages are the following:

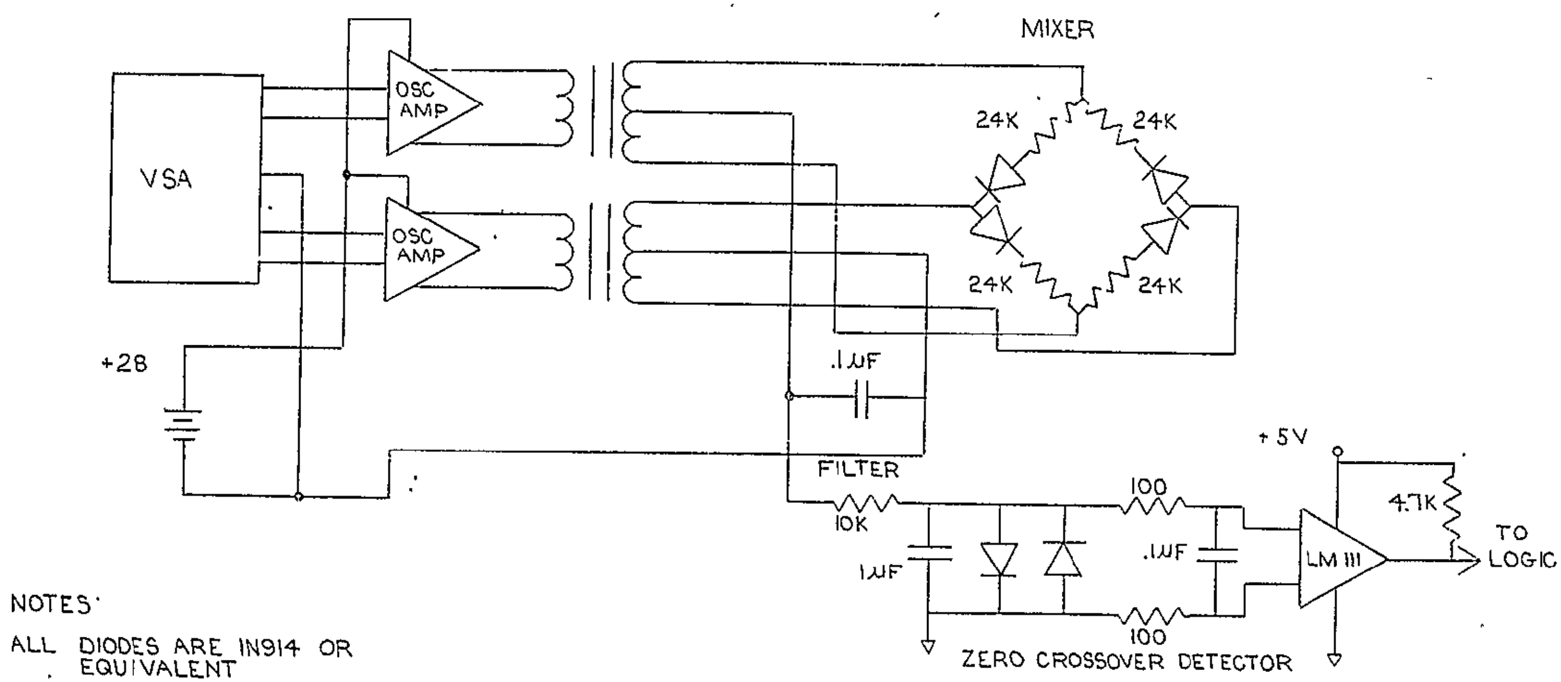


Fig. 4 VSA-OSC Amplifiers, Mixer and Zero Crossover Detector

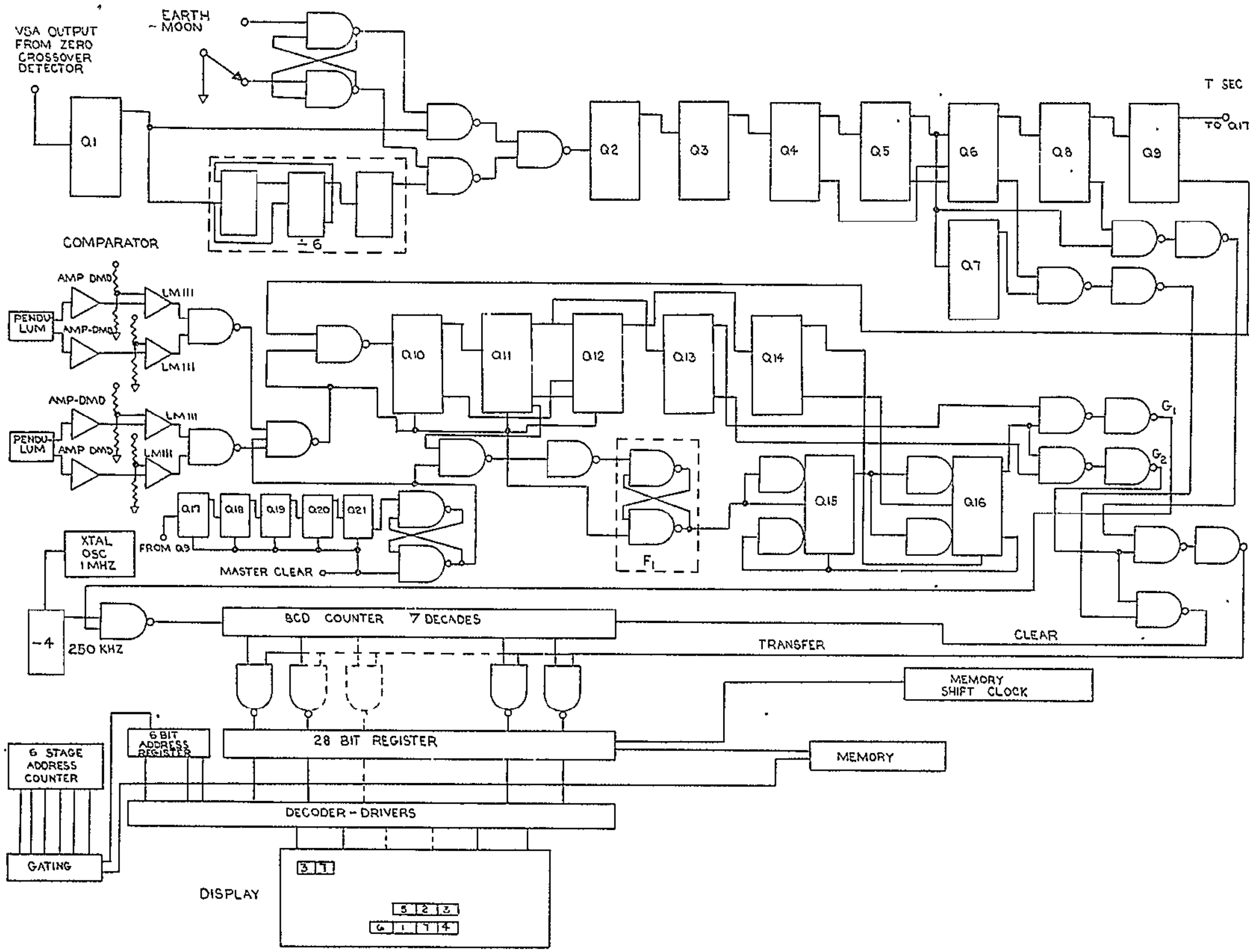


Fig. 5 Lunar Gravimeter Readout Design Logic

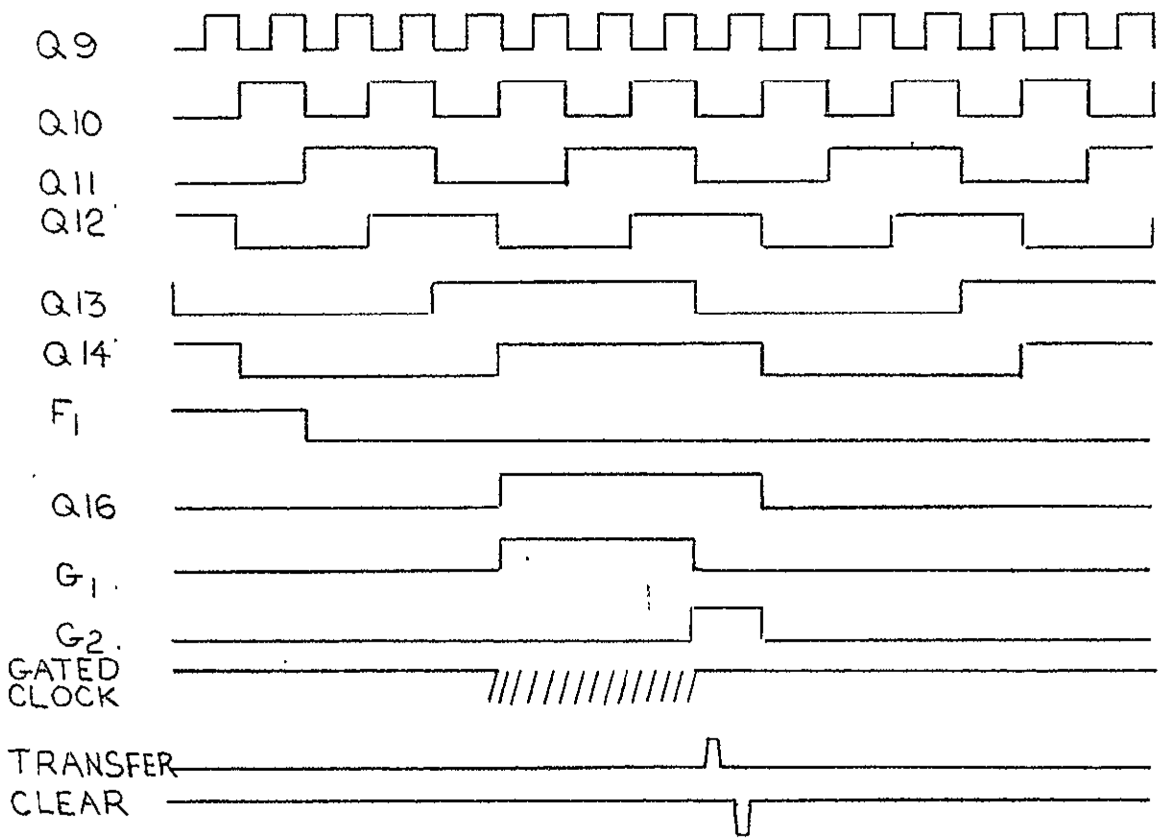


Fig. 6 Timing Waveforms

a. A considerable amount of circuitry, and hence power, would be required for a phase locked multiplier.

b. Its dynamic range would have to be large enough to account for drift as well as bias shifts in the VSA.

c. The multiplier would probably require a separate voltage controlled oscillator of the proper frequency to use on the earth where the VSA has an output of 130 Hz.

d. The usual problems of a phase-locked loop (e. g. , phase jitter, loss of lock, etc) would threaten the design.

In the proposed method, the output of the VSA signal conditioner feeds flip-flop Q_1 which divides the frequency in half. (See Fig. 5). Q_1 output feeds a divide by six circuit. Arrangements are such that the gating for the clock of Q_2 will accept either the output of Q_1 or the output of the divide by six circuit. The divide by six circuit is provided so that the frequency of the signal toggling Q_2 will be approximately the same on earth as it will be on the moon (VSA frequency is about 6 times as high on earth as on the moon).

Q_2 output is further counted down in Q_3, Q_4, Q_5, and Q_7. Q_6 has the same frequency as Q_5 but lags by 90°. Q_6 output is divided by two in Q_8. This 90° phase shift is provided so that by combining the outputs of Q_5 and Q_7 and Q_6 and Q_8, appropriate timing pulses are generated which can be used for transferring data and clearing the counter.

Q_8 output is divided by two in flip-flop Q_9. Q_9 output is further divided by Q_{10} and Q_{11}, but only under certain conditions. Q_{10} and Q_{11} will toggle if the gravimeter is within seven arc-minutes of level.

A two axis pendulum will sense the deviation from vertical. (Actually, two pendulums will be used. One will indicate level for the normal position; the other will be necessary for the inverted position, when bias measurements are made.) Analog outputs are fed to threshold detectors whose threshold will be such that a logical "1" will be obtained if the level indicates the system is more than seven arc-minutes off level. A logical "0" will be obtained if the level indicates the system is less than seven arc-minutes. This control signal is fed to the reset lines of flip-flop Q_{10} and Q_{11}. It is also used to inhibit the clock to Q_{10} and as a reset to R-S flip-flop F_1. The threshold detector uses LM 111 comparator circuits similar to the signal conditioner circuit. Level indication functions as follows:

The clock to Q_{10} is a square wave, with a period of T seconds. If the indicated level is less than seven arc-minutes, Q_{10} and Q_{11} will begin to count. If the gravimeter is swinging with an amplitude of greater than seven arc-minutes, Q_{10} and Q_{11} will be reset as it swings out of the null range, and the counting must start over. When Q_{11} becomes a "1", flip-flop F_1 is set. Since the time that the first indication that the level is less than seven arc-minutes occurs asynchronously with respect to the clock to Q_{10}, the time it takes for Q_{11} to switch from a "0" to a "1" has an uncertainty of T seconds (one period of the waveform). F_1 will be set, therefore, between 2T and 3T seconds after a null level condition is indicated. Since the period of the pendulum's swinging is expected to be about 2 seconds, Q_{10} and Q_{11} will be reset every time the pendulum swings out of the null region. Thus the level indication must remain under seven arc-minutes for a minimum of 2T seconds. A pendulum bypass mode is also provided. Q_9 is also fed to a 5 bit counter. The last flip-flop will therefore change states after 16T seconds. The output of the divide by 32 counter is fed to an R-S flip-flop which will "set" when the divide by 32 output changes state. A "master clear" signal will initialize the counter and flip-flop upon the application of power.

Measurement of gravity, however, does not yet begin. When Q_{11} or the "divide by 32" circuit enables flip-flop F_1, a logical "0" is sent from the Q output of F_1 to Q_{15} and Q_{16}. Flip-flop F_1 will remain in this state unless an off-level indication from the state detector is obtained. Q_{15} and Q_{16} act as a gate selector. Q_{16} output is one gate selected from a train of pulses fed to Q_{16}. The train of pulses to clock Q_{16} is obtained from Q_{14}. The Q output of Q_{10} is fed to Q_{12} to provide a waveform in quadrature with Q_{11}. Q_{14} divides Q_{12} by 2. Looking at Fig. 3, it is seen that the output of Q_{16} is a gate of width 4T seconds occurring 3T seconds after F_1 goes to "0". Q_{16} is used to begin the gravity measurement. Therefore the total time from the first indication of level until the beginning of the measurement is 5T seconds to 6T seconds. The measurement ends at the termination of Q_{16} which is 4T seconds. Total time for a measurement is thus 9T seconds to 10T seconds.

Q_{11} output is divided by 2 in Q_{13}. By gating Q_{16} with Q_{13} and Q_{13} respectively, its output is separated into two gates of 3T seconds and T seconds. The 3T second gate, G1, is used to gate the crystal oscillator into the decade counter. The T second gate, G2, is used to gate the outputs of transfer and clear pulses generated earlier in the countdown stage.

Data from the decade counter is strobed into the storage register by the transfer pulse. The clear pulse will reset the decade counter to zero to permit the counting to start over for the next word.

Outputs of the storage register will drive decoder-drives and, in turn, drive the display.

Biases of the VSA's obtained from NASA are approximately 5 to 10 Hz. With a scale factor of 127 Hz/g, the frequency range VSA that must be measured will be 11 Hz to 31 Hz on the moon (-1g and +1g measurements) and 117 to 137 Hz on the earth. Gate width for counting is $3/4 \times 512/f$. On the moon the gate width will range from 12 to 35 seconds. On earth the gate width is $3/4 \times 512/f \times 6$. Gate widths to be measured will range from 17 seconds to 24 seconds. Measurement quantization of 0.1 mgal per bit is desired. This will determine the clock frequency. Also it is well to limit the number of stages in the decade counter. A seven decade counter will not overflow if the clock frequency is 250 kHz since a gate width of 40 seconds would be required before the counter will overflow. For earth measurements the longest gate width is 24 seconds. On the moon the longest gate width is 35 seconds.

The worst quantization error on earth will be for the 17 second gate reading or approximately 0.25 mgal. On the moon the worst quantization will be 0.054 mgal.

A table below indicates the quantization and dynamic ranges for a 250 kHz clock and 7 decade counter.

Earth	Quantization	Dynamic Range
17 sec	0.25 mgal	2500 gals
24 sec	0.16 mgal	1600 gals
Moon		
12 sec	0.054 mgal	540 gals
35 sec	0.018 mgal	180 gals

After level indication is obtained, the total read time is

Earth	Reading
17 sec	51 to 56.7 seconds
24 sec	72 to 80 seconds
Moon	
12 sec	36 to 40 seconds
35 sec	105 to 116.7 seconds

It is anticipated that the Texas Instrument 54L series of TTL logic will be used in this application. This low power family consumes one milliwatt per gate and 4 milliwatts per flip-flop. For the display drivers and storage registers, there are some low power medium scale integration (MSI) functions made by Signetics and Fairchild which might prove somewhat advantageous over the TI devices.

2.5.2.6 Display - A prime candidate for the display is the Monsanto MAN-1 seven segment gallium arsenide display. This device provides considerable brightness at moderate power. Although relatively simple and apparently suitable for this application, no effort has yet been undertaken to evaluate any of these units or to ascertain their reliability. If problems arise with the solid-state devices, the alternative display will be the electro-luminescent type used in the various portions of the Apollo display systems. Although the EL display consumes less power and is more familiar to the field, the solid state display offers the advantage of higher brightness and the use of a low voltage DC supply. The EL display requires a high voltage AC supply.

2.5.2.7 Memory - For a permanent storage mechanism, a magnetic core memory or active semiconductor memory is being considered. The semiconductor memories are highly attractive in that they are small, low power and fairly simple to integrate electronically with the rest of the system. Disadvantages are that with the exception of Read-Only Memories (ROM's) power must be kept on continuously to prevent losing the data. Most ROM's must be preprogrammed and are therefore unsuitable. Radiation, Inc. makes a 512 bit pre-programmable memory in which nichrome wires are burned out in the programming. Here the disadvantages are that the burning of the nichrome wire requires a 100 millisecond high current ramp waveform and that there is no fool-proof method of pretesting the actual memory that will be flown. Semiconductor random access memories can perform the required job, but will require the maintenance of power on at all times, until the device is read back on earth.

Core memories are probably the most reliable. A memory meeting the requirements can be built for less than a watt of power and occupying only a few cubic inches of space. A core memory, however, is much more expensive than a semiconductor memory for this application. In terms of power, size and economics a semiconductor memory consisting of several IC chips employing MOS technology is probably the most attractive.

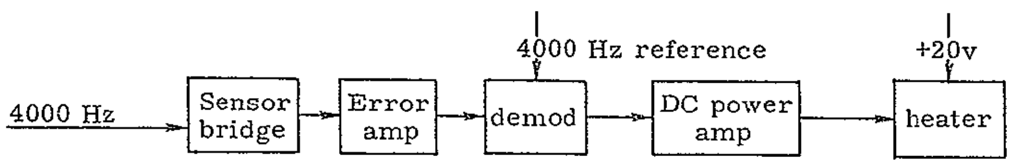
2.5.3 Temperature Regulation Electronics

2.5.3.1 General - Because of the sensitive thermal nature of the VSA as well as some of its supporting electronics, a reliable precision temperature regulation system is necessary. A "room-within-a-room" design is anticipated such that the innermost room, containing the most temperature sensitive elements, is most closely thermally controlled. Both proportional and "bang-bang" methods of control have been considered. Proportional control appears to be somewhat superior at this point.

2.5.3.2 Temperature Regulation Subsystem - Three parts of the system require temperature regulation: (1) the VSA units (called the Precision Oven) requiring control to $\pm$ 0.01°F and operating at about 125°F, (2) an oven containing certain parts of the electronics including the crystal oscillator and the VSA amplifiers requiring control to $\pm$ 1.0°F and operating at a temperature slightly above 100°F, and (3) an intermediate oven between (1) and (2) operating at about 100°F and requiring control to perhaps 5°F. It may be possible to eliminate oven (3) if tight enough control can be obtained in (1).

At present it is tentatively planned to make all three ovens proportional although it may be found possible to make some of them "bang-bang," especially oven (2) which requires somewhat more power (1.5 watt max) than the other, each of which requires about 0.2 watt max. Bang-bang regulators have the advantage of simplicity and efficiency but thermal inertia is always of concern in such a design; another disadvantage is that they are prone to generating electrical interference which may be picked up by other parts of the system.

If proportional regulators are used a typical block diagram of one is shown below:



The 2 volt-4000 Hz excitation is obtained from a tap on the 4000 Hz power supply. The error signal from the bridge is about 3 mv/$^{\circ}$F. It is amplified by an operational amplifier with a gain of about 1000, demodulated, and further amplified by the DC power amplifier which passes DC through a heater coil from the power supply. It can be seen that such a proportional system exactly balances the dissipation of the controlled structure. It is estimated that a thermal feedback factor of 400 to 500 can be obtained from the various gains and scale factors.

2.5.4 Pendulum Electronics

2.5.4.1 General - As a condition of best data, the gravimeter must be level within specific limits. By means of signals generated from pendulum, display will indicate level condition to the astronaut. Four sets of pendulum electronics are required per gravimeter, one each for x and y in each of the $\pm$ 1g gravimeter orientations.

2.5.4.2 Pendulum Electronics Subsystems - The pendulums will be energized with 3 volts, 4 kHz. The x and y outputs of each pendulum will be amplified and rectified. The DC outputs will trigger the comparator circuits whose outputs will be applied to the readout logic.

2.5.5 Power Supplies

2.5.5.1 General - Power supplied for the gravimeter experiment will be self contained and utilize a battery source. For efficiency, the number of voltage levels were minimized and the values of those levels also minimized. A major challenge of the experiment is to reduce overall required power as much as possible.

2.5.5.2 Power Supplies Subsystem - The voltages required for the system are:

(1) +5 volts DC regulated at 1200 ma,
(2) -5 volts DC regulated at 16 ma,
(3) +20 volts DC unregulated at 50 ma,
(4) 15 volts DC regulated at 17 ma,
(5) 3 volts AC regulated at 1 ma,
(6) 2 volts AC regulated at 6 ma,
(7) 10 volts AC regulated at 3 ma.

The largest component of power drain is the +5 volts. In order to minimize conversion losses a battery was chosen whose voltage does not drop below 7 volts near the end of its life and is estimated to be about 9 volts when new; thus the maximum loss in this regulator's face transistor is about 4.8 watts while the minimum is 2.4 watts. These powers are for peak load on the +5 regulator output. Sixteen ma of the +5 volts supplies operational amplifiers while the 1200 ma is for the logic and display circuits.

The other voltages are obtained by conversion and further regulation. The highest practical conversion frequency was chosen to minimize size and weight of components. The choice was dictated primarily by pendulum excitation which can be a maximum of 400 cps. This frequency is generated by a Colpitts oscillator. A good sine wave is taken from the oscillator tank and amplified to produce the required AC voltages: +3v for pendulum excitation; -2v for temperature control bridge excitation, and 10v for temperature control demodulators.

A distorted waveform is taken from the oscillator's collector, dipped into a square wave and used to key transistor switches. These switches obtain their supply voltage directly from the battery since it is not necessary for the amplitude of the output square wave to be regulated.

The output of the transistor switches is transformed to the necessary voltage levels and rectified. One output is +20 volts unregulated at 50 ma for temperature control heaters. Another 17 ma of this voltage is regulated down to 15 volts for the VSA amplifiers. The other output is -8 volts at 16 ma; it is regulated down to 5 volts for the operational amplifiers.

A block diagram of the power supply is shown in Figure 7.

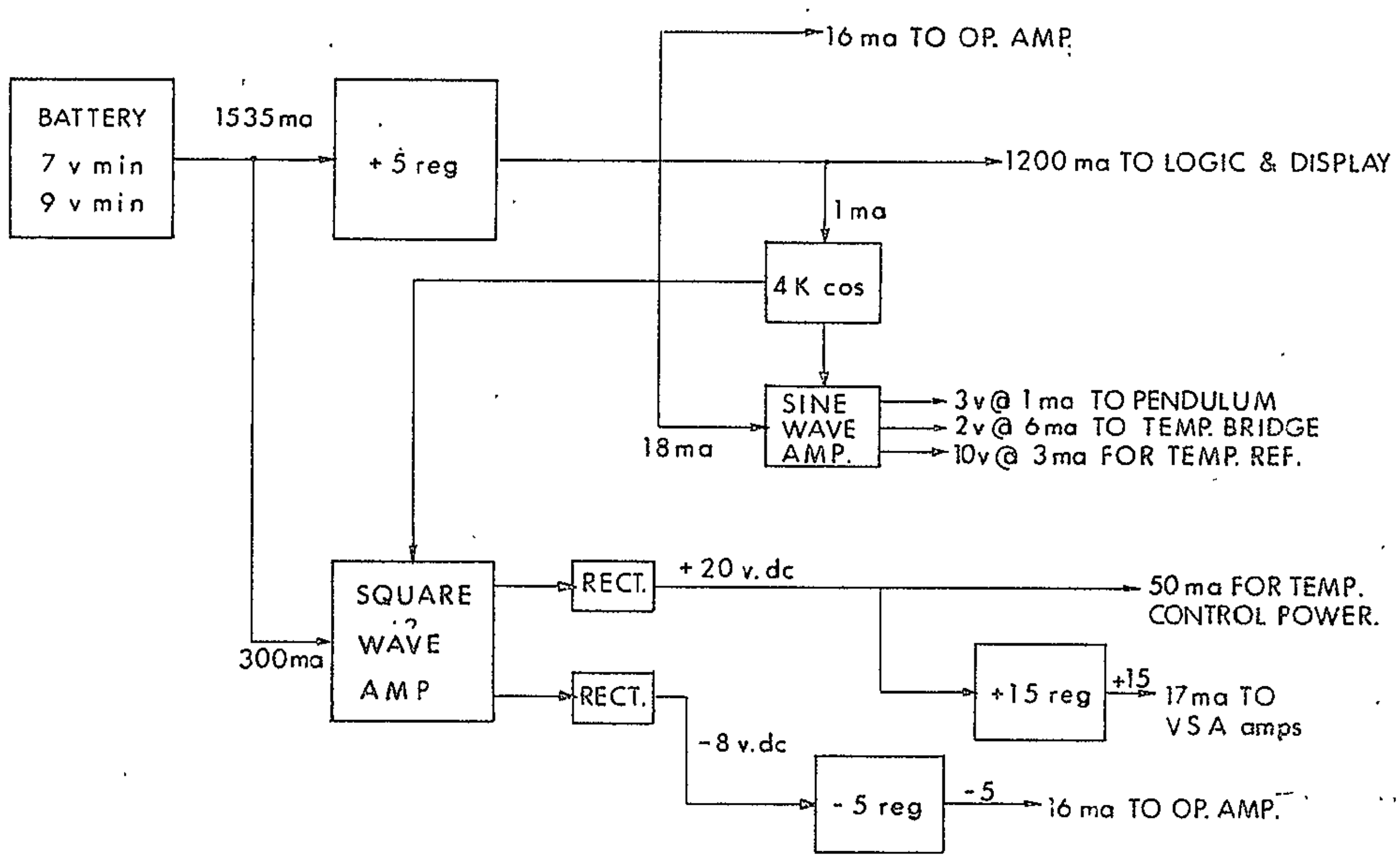


Fig. 7 Power Supply

3.0 MODES OF OPERATION

3.1 GENERAL

Careful consideration of necessary modes of operation was required to accomplish important objectives of the design -- minimum power and astronaut convenience. Modes were selected and designed such that long-time conditions such as "standby" made minimal demands upon the limited available power. It is anticipated that a time-line mode/power schedule will show favorable results.

3.2 MODES

There will be four primary modes of operation. They are "Off," "Standby," "Measure" and "Display". During the "Off" mode only the temperature control system will be on to keep the VSA at its desired control temperature.

In "Off," only the temperature control electronics will be on. All ovens will be regulated. The peak power in this mode will be 1 watt..

In "Standby," the VSA oscillator-amplifiers, the crystal oscillator, pendulums and their associated electronics will be turned on. In addition, small sections of the logic will be energized. If a memory is used an address counter must be turned on. Part of the readout logic will be energized also. The readout logic is turned on so that upon the initiation of a measurement cycle, a reset pulse will be generated which will initialize certain sections of the logic to prevent a premature reading count to be taken. Peak power in Standby is 1. 5 watts.

In the "Measure" mode the additional logic power will be turned on. This includes primarily the 7 decade BCD counter and the 28 bit storage register. After the gravimeter is levelled, the astronaut will activate the "Measure" button. Immediately a "Master Clear Pulse" is generated to initialize the readout logic. A measurement will proceed in accordance with the sequence of events explained above. For the first 15 seconds after the Measure mode button is pressed, two status lights will be displayed. The status light will display "H" or "C" if the temperature is too hot or too cold. Another light will display a "P" if the battery power is too low. An "□" will be displayed on a third light from the time a measurement is initiated until the reading has been completed. The astronaut should not activate the "Display" mode until the "□" has disappeared. An "L" will appear on a fourth light whenever the level indication is such that the pendulums are not within seven arc-minutes of level. Besides indicating when the pendulum has levelled, it also indicates if the measurement was taken under the pendulum bypass mode. As soon as the "□" has disappeared the astronaut can activate the Display button. The Display button will apply power to the 7 numerical display light and their decoder drivers. The Display mode will present the numerical display for only 15 seconds to minimize the possibility of leaving the display power on too long. The Display button can be pressed any number of times as long as a new measurement cycle has not been initiated. Total power in the "Measure" mode is about 3.5 watts. In the Display mode the peak power when all segments are on is 10 watts. One possible time-power profile is shown in Figure 8.

Figure 8a illustrates instrument usage of the entire 8 day period from the time of manually activating the battery. Note a total time of 12 hours during which a total of 64 measurements are taken.

Figure 8b illustrates instrument power profile during one typical measurement cycle. This cycle is shown to take 5 minutes, with 5 minute travel or wait time before another measurement. This particular cycle can obviously be shortened, however, it is taken as typical in order to show worst conditions assumed for battery power usage.

Under the condition shown in Figure 8, a total battery power of 62 watt hours is utilized over the 8 day period.

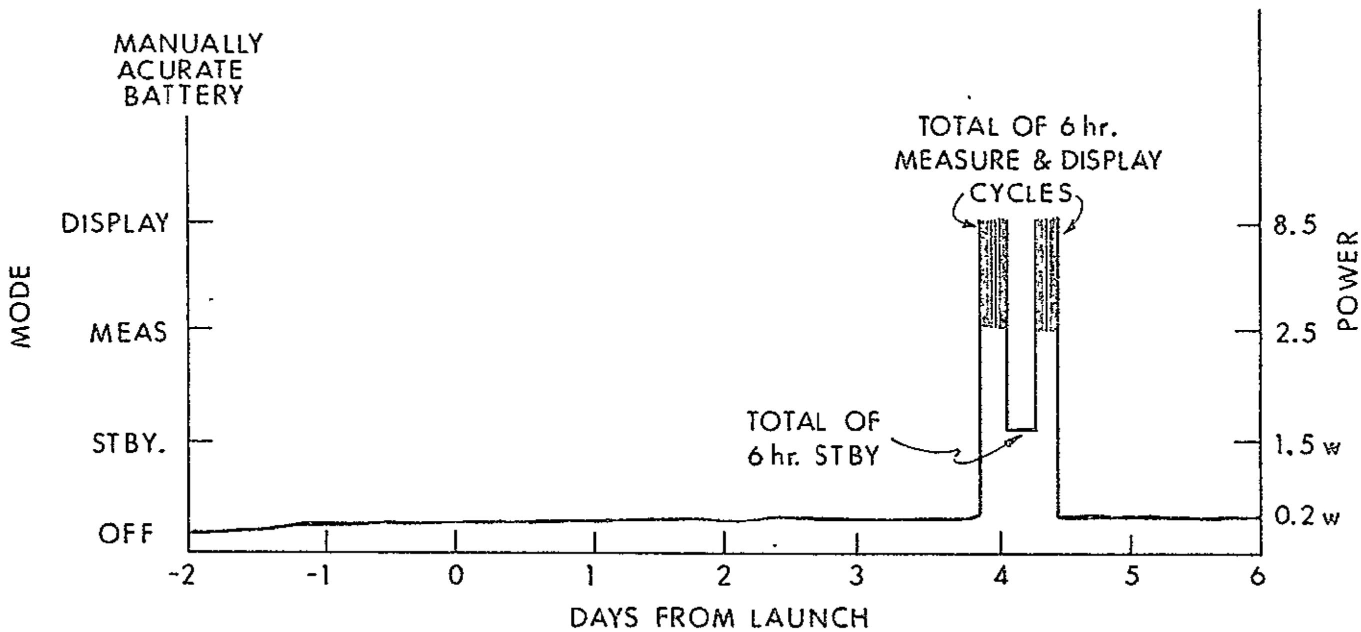


Fig. 8a Time-Power Profile (Nominal)

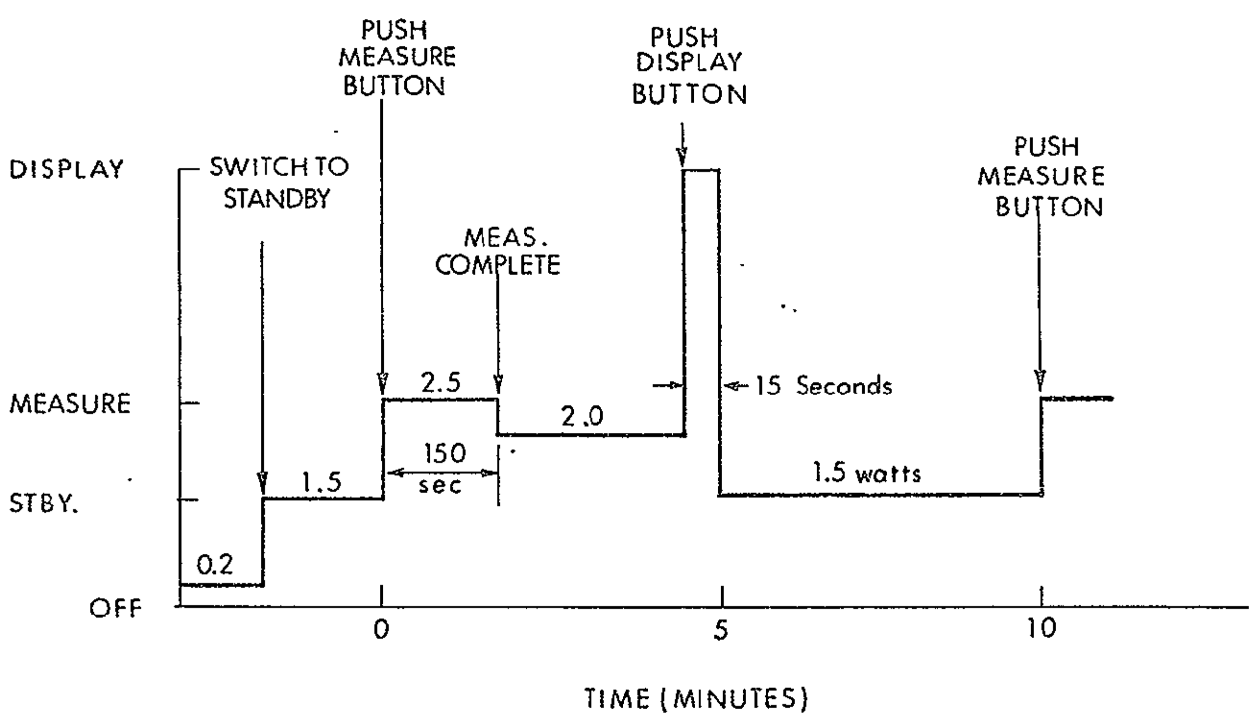


Fig. 8b Detail of Measure and Display Cycle

4.0 THERMAL DESIGN

4.1 GENERAL

The object of the thermal design herein described is to provide a suitable thermal environment for components of the Lunar Gravity meter during trans-lunar and lunar stay periods. The most thermally sensitive element is the VSA. It is mounted within concentric ovens to provide a maximum isolation from external disturbance. Thermally sensitive electronic components are placed within a small oven which is then mounted on the E frame. Non-sensitive electronic components are placed directly on the E frame. All internal heat is collected by the E frame and subsequently dissipated by radiation surfaces on the E frame. The mechanical configuration is illustrated in Figure 2.

An external cover and concentric housing provide the thermal interface between the inner structure and external environments.

During periods of high internal power dissipation the outer cover will be open. This will allow sufficient direct radiation to space from the E frame radiator to maintain desired temperatures.

Battery power must be conserved to assure an 8 day useful life and the capability for 64 gravity measurements. This is achieved by operating the instrument in the OFF mode during idle periods. A typical time line for power usage is shown in Figure 8.

Feasibility of the design approach is demonstrated by analysis of a simple thermal analog having 6 nodes.

4.2 DESCRIPTION OF ELEMENTS

4.2.1 VSA and Precision Oven (P-Oven)

The VSA is mounted directly in the (P-Oven) as shown in Figure 2 This oven is controlled in temperature to within $\pm$ 0.02°F of a nominal temperature of 125°F. This assembly has a thermal time constant of several hours.

4.2.2 Intermediate Oven (I-Oven)

The precision oven is mounted within the intermediate oven. The intermediate oven serves to reduce external gradients, or hot spots, and is heated only if its temperature falls below some preset value. This effectively limits the range within which the precision oven must operate.

4.2.3 Electronics Frame (E Frame)

The intermediate oven is mounted within the Electronics Frame. The E frame serves the dual purpose of mechanically mounting the batteries and heat dissipating electronics, as well as provide a radiating surface to reject internal heat dissipation. Electronic and display modules use the E frame as their heat sink. E frame temperature is expected to operate between 25°F and 110°F, depending upon operating mode, and time duration. Certain components mounted on the E frame require temperature control. These are the string amplifiers and possibly the crystal oscillator. These devices are mounted within their own oven operating at 135°F and using the E frame as a heat sink. Radiant heat transfer to space, directly from the E frame radiator, occurs through a web structure attached to the housing. This web structure supports the E frame via a ball pivot assembly.

4.2.4 Housing

The housing and E frame are thermally separated by a fairly high thermal resistance, with a vacuum space existing between them under normal operating conditions. At this writing the major heat rejection mechanism is the radiating surface at the top of the E frame. It is quite possible that heat rejection via the housing could be useful, however, this situation must be carefully analyzed since low housing temperatures would have to be guaranteed at all times.

4.2.5 Case

The case provides an interface between itself and the sun, moon, space, and location. Its temperature depends upon a heat balance involving all of these variables as well as operating mode and time duration. Case temperatures, in examples illustrated later, go from -20°F to +104°F.

4.3 THERMAL ANALYSIS

4.3.1 Analog Nodes

An estimate of thermal conditions existing during various modes of operation is made using simplified analog techniques involving heat balances at the nodal points. Consider the system with the following nodal points:

- (P) Precision oven, including VSA
- (I) Intermediate Oven
- (E) Electronics Frame
- (H) Housing
- (C) Cover, consisting of Top (C_T) and Bottom (C_B)
- (C/O) Oven containing string amplifiers and crystal oscillators

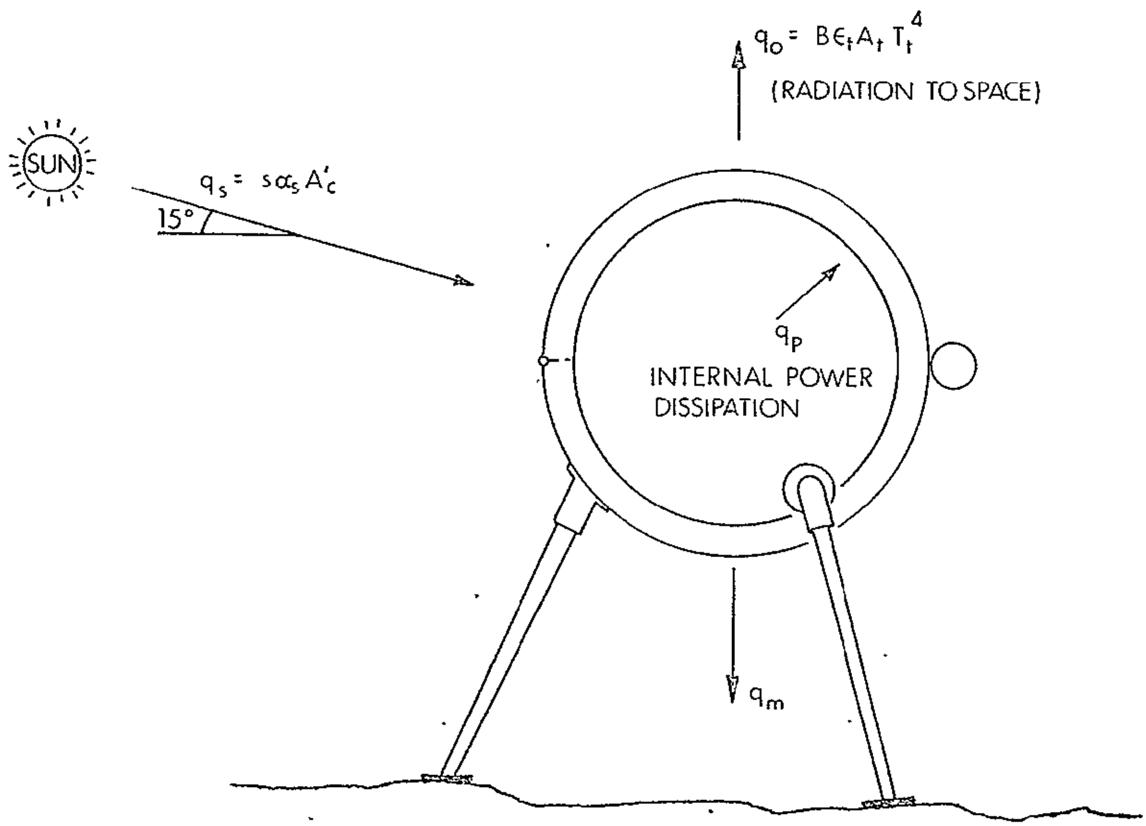
Energy in the form of electrical power enters the system at the (P), (I), (E), and (C/O) nodes. Energy exchange with the Lunar surface and the Sun will occur at the (C) node.

Radiation to Space will occur at the (E), (C) and (H) nodes at various times.

Use of simple resistance-capacitance analog techniques is restricted to estimating thermal conditions. More accurate results must await complete digital steady state and transient analysis.

4.3.2 Heat Balance of Cover and Housing Assembly

Consider first a heat balance involving the Housing (H) and Cover (C) assembly.



$q_m = B\,(F_\epsilon F_A)_b A_b \left[T_b^4 - T_m^4\right]$

(RADIATION TO MOON)

q = heat flow, watts

s = solar constant (0.9 watts/in^2)

α_s = solar absorbtivity

F_ϵ F_A = product of emmissivity factor and geometrical area factor controlling radiation heat transfer between surfaces.

Subscripts

b	=	bottom
t	=	top
c	=	cover
ci	=	inside of cover
s	=	solar
H	=	housing
m	=	moon

In the steady state $\sum$ Energy-in $= \sum$ Energy-out thus:

$$q_P + \alpha_s S A^1{}_C = B \epsilon_t A_t T_C{}^4 + B (F_\epsilon F_A)_b A_b \left[T_C{}^4 - T_m{}^4 \right]$$

In writing this equation it is assumed that energies are evenly distributed throughout the shells by conduction along shell walls as well as by radiation and conduction paths between (H) and (C). This is a design constraint readily achieved by selection of wall thickness and internal surface finish.

also:

$$A_t = A_b = 2 A_C{}^1 = 2\pi r_c{}^2$$

and,

$$(F_\epsilon F_A)_b = \frac{1}{\frac{1}{\epsilon_b} + \frac{1}{\epsilon_m} - 1} = \epsilon_b$$

The heat balance for the combined cover and housing assembly reduces to:

$$\left(1 + \frac{\epsilon_b}{\epsilon_t}\right) T_c{}^4 = \frac{q_P}{B \epsilon_t A_t} + \frac{S}{2B} \frac{\alpha_s}{\epsilon_t} + \frac{\epsilon_b}{\epsilon_t} T_m{}^4$$

The desired result is obtained by setting $t_b = t_t = \alpha_s$

$$T_C^4 = \frac{1}{2} \frac{q_P}{B \epsilon_t A_t} + \frac{S}{2B} + T_m^4$$

The significance of the result is that if design constraints noted in the above derivation are followed, namely that (a) the Cover-Housing assembly distribute and exchange heat such that each is a single node.
(b) the ratios $\frac{\epsilon_b}{\epsilon_t}$ and $\frac{\alpha_s}{\epsilon_t}$ remain constant with a value near unity. Then we find cover temperature to be practically independent of the surface characteristics.

The relative influence of heat inputs due to internal power dissipation, solar energy, and the moon is shown below for the OFF mode with $\epsilon_t = \alpha_s = 0.2$ and a Lunar surface temperature of 40^oF as specified for the sunlit Lunar surface in Reference 1 with sun angle of 15^o.

$$T_C^4 = \frac{1}{2} \left[16 + 1280 + 625\right]$$

$$T_C = 97^oF.$$

Consider the situation with the instrument again on the sunlit lunar surface, but no direct solar input to the cover.

$$T_C^4 = \frac{1}{2} \left[16 + 0 + 625\right]$$

$$T_C^4 = -35^oF$$

It should be noted that the lunar surface provides a stabilizing influence for the situations illustrated above. In the first instance, the cover tends to be cooled by a radiation exchange to the lunar surface. In the second instance the cover tends to be warmed due to heat exchange from the moon's surface.

Similarly, in the "STANDBY" mode, in sunlight (cover closed) the cover temperature is estimated to be 104^oF. If the cover is shaded from direct sunlight, its temperature will drop to -20^oF.

Reference 1. TIR 727-S-8293 (S) GE/ASD Report

4.3.3 Thermal Resistance Between Shells

Central to the thinking in terms of temperature regulation during various modes of operation is the consideration of power usage. Power is at a premium since internal batteries must be depended upon for all phases of the 8 day voyage. Accordingly, the OFF mode only allows for precise temperature control of the VSA - P-oven assembly. In order to do this economically, a high thermal resistance must be postulated between the P-oven and the external environment. There is a limit to how high one should go in selecting this thermal resistance, however, for practical reasons of building the instrument. It is also undesirable to achieve resistances so high that time constants for initial warm-up become prohibitive. Insufficient cool-down rate can make thermal control difficult. A comfortable compromise results with a thermal resistance of $150^{\circ}F/watt$ between the (P), (I), and (E) structures. A similar value is selected between the (E) and (H) shell.

Thermal resistances are computed for the operating situation of a vacuum between shells. This is assured at all times between the (P) and (I) ovens by sealing and evacuating this area. Other areas will be evacuated once in space, or in simulated test or storage environments. The thermal resistance between shells consists of a parallel combination of heat flow paths. Consider the transfer between (P) and (I) ovens.

(a) Each supporting tensile wire; heat transfer by conduction

$$R_C = \frac{L}{kA} \quad {}^{\circ}F/\text{watt}$$

using: 0.025" diameter piano wire ($K = 0.635 \frac{\text{watts}}{\text{in} - {}^{\circ}F}$

and 1" length

$$R = 3200^{\circ}F/\text{watt per wire}$$

(b) Copper conductors; heat transfer by conduction using: #30 wire 0.01" diameter and $K = 5.36$ watts/in - $^{\circ}F$ and 1" length

$$R = 2400^{\circ}F/\text{watt per wire}$$

(c) Heat transfer by radiation; the equivalent heat transfer resistance between surfaces can be found by the approximate relation:

$$R\, F_{\epsilon}\, F_A = \frac{705 \times 10^2}{T/100^3} \qquad \frac{{}^{\circ}F - in^2}{watt}$$

for concentric spheres use

$$F_c \, F_A = \frac{1}{\frac{1}{\epsilon_1} + \frac{A_1}{A_2} \frac{1}{\epsilon_2} - 1}$$

using:

$$E_1 = E_2 = 0.02$$

$$T = 579^{\circ}R \text{ (average temperature)}$$

we find

$$R = 762^{\circ}F/\text{watt}$$

for a spherical surface area of 50 in^2

(d) Composite thermal resistance between (P) and (I) oven. Assuming a total conducting path equivalent to 4 steel wires in parallel and 8 copper conductors in parallel, with the radiation exchange of (C); the composite resistance is

$$R_{(P) - (I)} = 170^{\circ}F/\text{watt}$$

It is reasonable, therefore, to use a figure of $150^{\circ}F$/watt for calculations to follow.

4.3.4 E Frame Radiator

This surface is designed to radiate the average amount of heat collected by the E frame at a design point temperature of $75^{\circ}F$. Heat pulses due to short duration peak demands (display cycle for example) are integrated by the thermal mass of the structure and ultimately radiated over some longer time period.

The average power during use on the moon is derived from Figure 3. A dissipation of 1.9 watts is necessary over the 12 hour use period. A radiating surface of 10 in^2 with emissivity of 0.8 will radiate this power at $50^{\circ}F$. For optimum thermal control and battery conservation this radiating area must be modulated according to actual power usage.

The heat radiated from the surface for a particular condition is estimated from

$$q_{0E} = 0.352 \times 10^{-3} \, F_c \cdot F_A \, A \left(\frac{T}{100}\right)^4 \text{ watts}$$

The assumption has been made that heat is not received by the radiator from external sources such as the sun or from the web structure supporting the E frame pivot assembly. Sun baffles and highly reflective webs are postulated at this writing.

4.4 THERMAL PERFORMANCE

4.4.1 Power and Temperature Estimates

(1) OFF — Only the P-oven is temperature controlled ($125^{o}F \pm 0.02$)

Trans Lunar; assume case temperature of $50^{o}F$; power required is 0.16 watts

On Moon; in sunlight; case temperature $97^{o}F$ requires 0.05 watts

On Moon; case shaded; case temperature $-35^{o}F$; power required 0.34 watts

(2) STANDBY — All analog electronics on and electronics oven controlled to $125^{o}F$. System awaiting a manual "measure" command. Power usage estimate 1.5 watt, with the cover closed and dissipating all heat.

In Sunlight; cover temperature $104^{o}F$ E frame $75^{o}F$

(3) MEASURE — Total cycle as described in Figure 3 average power usage 1.9 watts, with peak power 8.5 watts during 15 second DISPLAY. Cover is open, with E frame radiator dissipating all heat.

In Sunlight; estimate of cover temperature $90^{o}F$ E frame at $75^{o}F$.

4.4.2 Precision Oven Warm-Up Time ($75^{o}F$ - $125^{o}F$)

The initial warm-up rate will be

$$\frac{\partial T}{\partial t} = \frac{q}{m C_P}$$ where q = applied power

m = mass of material heated

c = specific heat

The P-oven material is magnesium, with a volume of 5.25 in^3, a weight of 0.35# and specific heat of 0.24

If $q_{max} = 10^{-3}$ Btu/sec (1 watt)

$$\frac{\partial T}{\partial t} = 0.011^{o}F/\text{second}$$

4.4.3 VSA Warm-Up

The accelerometer is mounted with in the P-oven with structure having a thermal resistance of about $50^{\circ}F/watt$. The VSA weighs 0.5# with an average specific heat of 0.14. This results in a first order time constant of 1 hour. It is expected that this time delay, coupled with the P-oven time lag will require a 24 hour warm-up to assure properly controlled VSA thermal conditions.

5.0 EXPERIMENT DESIGN

The scientific investigator, Dr. Charles G. Wing of the Massachusetts Institute of Technology in conjunction with Dr. Manik Talwani of Columbia University, has prepared the material which is presented in this section. Paragraph 3.1 of the Statement of Work for Contract NAS9-10749 was used as a guide. The material to be presented is as follows:

5.1 Preliminary Operational Criteria and Procedures
5.2 Experiment Error Budget
5.3 Elevation Control Requirements
5.4 Experiment Data Requirements

It should be kept in mind that this material is of a preliminary nature in that an instrument design has not been finalized, nor have astronaut and spacecraft interfaces been defined.

5.1 PRELIMINARY OPERATIONAL CRITERIA AND PROCEDURES

5.1.1 Introduction

The preliminary procedures for performing the lunar gravimeter traverse/transfer experiment are set forth below. They are comprised of:

a. A gravimeter sequence defining the operation of the instrument from T-2 days through completion of the mission,

b. procedure for making a gravity measurement,

c. description of the traverse experiment criteria,

d. description of the transfer experiment criteria.

5.1.2 Gravimeter Sequence

The general time-line for duties associated with the gravimeter experiment is given in the table below.

TABLE I

Time	Tasks
2 days prior to launch	1. Disconnect gravimeter from launch site test equipment and install charged primary batteries.
	2. Perform pre-launch bias and scale factor calibration
	3. Secure instrument and place in spacecraft container
	4. Install instrument in spacecraft and secure for launch and transfer to Moon

Time	Tasks
EVA1	1. Remove instrument from spacecraft and deploy on Moon's surface. Place instrument in "Standby".
	2. Perform a "Normal" mode gravity measurement and report results.
	3. Perform an "Inverted" mode gravity measurement and report results.
	4. Secure Instrument and place on Rover vehicle.
	5. During each stop during a traverse repeat Step 2.
	6. At completion of Traverse perform Steps 2 and 3.
	7. Place unit in "OFF" mode. Secure until next traverse.
Each Succeeding EVA	1. Deploy instrument on Moon's surface and place in "Standby" mode.
	2. Perform a "Normal" and "Inverted" mode gravity measurement.
	3. Repeat Steps 4 through 7 for the remainder of the EVA.
End of Mission	1. Remove internal memory and secure for return to Earth. This assumes that a memory can be incorporated in the instrument design without compromising mission requirements or astronaut work load.

The sequence for each traverse EVA is identical. At the beginning of each EVA, power is switched to "Standby". The first gravity station is the lunar base station bias calibration. Each subsequent station consists of a normal-mode measurement. The last station is a bias calibration reoccupation of the lunar base station. The procedures for making normal-mode and bias calibration measurements are described below. Subsequent to the last measurement of the EVA, power is switched to "OFF".

After all measurements have been made, the internal memory is removed from the instrument and stowed in the LM.

5.1.3 Making a Gravity Measurement

5.1.3.1 Normal Mode - For a normal mode gravity measurement, the astronaut will remove the instrument from the rover, deploy it on the moon's surface, raise the instrument cover, level to within ± 3°, and actuate the "measure" switch (may be coincident through interlock control). The instrument is now automatic, leaving the astronaut free for other duties. After approximately two minutes, the astronaut returns and reports the measurement to earth, also noticing any monitor lights. The instrument cover is replaced (automatically switching the instrument to "standby") and the instrument is returned to its rover storage position.

5.1.3.2 Bias-Calibration Mode - The bias calibration or inverted mode of operation, is the same as the normal-mode with the exception that the instrument is inverted $180^{\circ} \pm 3^{\circ}$ and an upside-down measurement is made. Both measurements are reported to earth and recorded.

5.1.4 Traverse Criteria

The following criteria must be met for a successful traverse experiment:

1. The gravimeter instrumental errors must be less than 1.0 mgal after correction for linear drift over a period of up to 5 hours. The allowable error is described by a function:

$$G = \pm (0.5 + 0.1T) \text{ milligal}$$

where T = hours elapsed time from LM base station

2. The elevation control must be accurate to ± 14 meters relative to the datum plane over the entire traverse. A goal for interstation elevation is ± 2 meters.

3. The horizontal coordinates of each gravity station relative to the LM landing site must be known with an error limit which increases linearly from ± 10 meters at the LM site to ± 25 meters at the furthest station from the LM.

4. The minimum number of gravity stations per traverse is 5. The maximum number per traverse is 20 or 30 depending on whether 2 or 3 traverses are made.

5. Along a straight line projection from the LM to the furthest gravity station, the station-spacing is: 0.5 km minimum; 1.0 km maximum.

6. Elapsed time between each measurement must be known to within 7.0 minutes.

5.1.5 Transfer Criteria

The following criteria must be met for a successful transfer experiment:

1. Initial calibration of the accelerometer terms K_1 and K_3 must be accurate to ± 12 ppm and $\pm 30\ \mu g/g^3$ when performed at Earth g.

2. The shift of K_1 and K_3 (aside from a linear drift correction) from prelaunch calibration to lunar measurement must be less than ± 24 ppm and $\pm 30\ \mu g/g^3$.

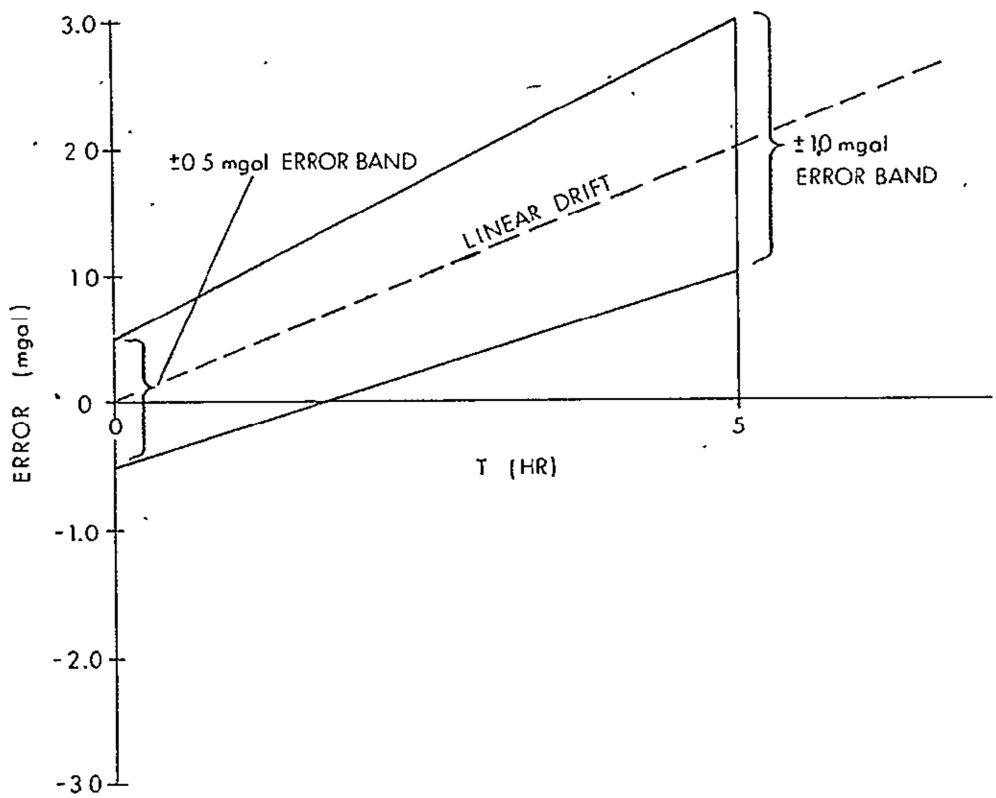


Fig. 9 Instrument Error Budget

3. The instrument performance due to all error sources other than those of (1) and (2) must be 4.0 mgal.

4. An error of relative selenocentric distances for adjacent LM landing sites of less than ± 40 meters must be achievable at some time in the future.

These criteria and procedures have been developed on the basis of a conceptual design. Needless to say they will be refined to reflect further design effort, assignment of a lunar landing site, spacecraft/rover interface definitions and astronaut work load as defined by this and other experiments.

5.2 EXPERIMENT ERROR BUDGET

This section specifies the experiment error budget for the Lunar Traverse Gravimeter Experiment. The experiment involves two independent parts using the same instrument. Due to different requirements, the error budgets are prescribed separately for the traverse and transfer portions. The total allowable experiment errors are computed on the basis of expected gravity signal and minimum acceptable signal to noise ratios. Individual errors within the total error limits are distributed according to best estimates to maximize probability of meeting all criteria. These error allocations are subject to redistribution as the experiment develops. All errors are to be considered 2 sigma.

5.2.1 The Traverse Measurement

5.2.1.1 Objectives - The principal purpose of the gravity traverse experiment is to detect density differences below the Moon's surface along the traverse. These differences may be due to the contrast between the lunar dust and rocks or to the contrast between different kinds of rocks. The traverse gravity experiment can be thought of as a valuable extension of surface geology experiments, particularly in deciding whether contacts between different kinds of rocks extend to great depths.

5.2.1.2 Method - A traverse base station will be determined at the beginning of each traverse (probably coincident with the ± g measurement). Relative gravity values will then be obtained along the traverse at points preselected for their geological interest.

5.2.1.3 Station Spacing - The required station spacings depends upon the minimum wave length gravity anomaly we are attempting to resolve. An estimated minimum of 5 stations per traverse is required for the experiment to have any value. The maximum number of stations will not exceed 60 per mission regardless of the number of traverses.

5.2.1.4 Maximum Error - For the gravity experiment to be of value the "noise" due to measurement error should be significantly lower than the expected total variations of gravity along the traverse. Any estimates of this total variation of gravity are subject to considerable uncertainty. An estimate that we adopt for the present discussion is 30 mgal. It is felt that the largest error tolerable in the reduced-free air gravity is ± 3 mgal. However, the largest part of this error will probably arise from the error in relative elevation of the gravity stations.

The total maximum error due to all instrumental causes should not be larger than ± 1.0 mgal.

5.2.1.5 Traverse Error Budget (established at Earth g) -

A. Total instrumental error from all causes ± (0.5 + 0.1T) milligal, where T is elapsed hours from LEM base station. Assuming a traverse time of 5 hours the maximum error is ± 1.0 milligal.

B. Free-air elevation reduction error caused by maximum elevation error with respect to datum plane of ± 14 meters ± 2.8 milligal.

C. Offset error due to maximum drift rate of 1 milligal per hour and timing error of 6 minutes ± 0.1 milligal.

D. Error due to horizontal coordinate error of ± 10-25 meters in presence of maximum regional gradient of 5 milligal per kilometer ± 0.1 milligal

Total Error 3.0 milligal rms

5.2.2 <u>The Transfer Measurement</u>

5.2.2.1 Objective - The transfer gravity measurement falls out as a bonus in the course of the traverse experiment. An accuracy value of ± 6 mgal from instrument causes alone is assigned to the transfer experiment.

5.2.2.2 Method - The proposed gravimeter uses an inertial grade accelerometer as its gravity sensor. The accelerometer equation may be written:

$$f = k_0 + k_1 g + k_2 g^2 + k_3 g^3 \ldots$$

where f is output frequency.

Measuring in the normal (+g) position:

$$f_{+g} = k_0 + k_1 g + k_2 g^2 + k_3 g^3 \ldots$$

while in the inverted (-g) position:

$$f_{-g} = k_0 - k_1 g + k_2 g^2 - k_3 g^3 \ldots$$

Taking the sum and difference of the ± g measurements:

$$f_{+g} + f_{-g} = 2k_0 + 2k_2 g^2 + \ldots \tag{2d}$$

$$f_{+g} - f_{-g} = 2k_1 g + 2k_3 g^3 \ldots \tag{2e}$$

In the particular accelerometer, the k_2 and k_3 terms are accurately modeled from k_0 and k_1. In addition, k_2 and k_3 are very stable. Experience shows that k_0 is much more subject to unpredictable behavior than k_1. The experiment is therefore greatly enhanced by using equation (2e) to eliminate k_0.

The experiment consists of: A) measuring or modeling k_0, k_1, k_2, k_3 on earth over a tilt table range of ± 1g, B) assuming constant or predicting k_2 and k_3, C) extrapolating the value of k_1 from last calibration using previous drift history, and D) performing a ± g measurement to obtain lunar g through equation (2e Equation (2a) yields k_0 which is a good indicator of gravimeter performance. It is anticipated that ± g measurements will be performed at the beginning and end of eacl traverse as a monitor of instrument performance.

5.2.2.3 Transfer Error Budget (established at earth g) -

A.	Initial calibration error of accelerometer terms k_1 and k_3, ± 12 ppm ± 30 $\mu g/g^3$	± 2 milligal
B.	Shift in calibration of accelerometer terms k_1 and k_3 due to drift uncertainty, temperature, acceleration, vibration, and shock; ± 24 ppm and ± 30 $\mu g/g^3$	± 4 milligal
C.	Instrument performance other than errors of A and B	± 4 milligal
D.	Free-air elevation reduction error due to error in relative selenocentric distance of ± 40 meters	± 8 milligal
E.	Errors due to horizontal coordinate errors are insignificant	± 0 milligal
	Total Error	± 10 milligal rss

5.3 EXPERIMENT ELEVATION CONTROL REQUIREMENTS

Several factors have been considered in establishing experiment requirements for lunar elevation control. While the total variation in gravity along the traverse is not known, a working estimate of 30 milligals has been adopted. We feel that free-air gravity measurements determined to an accuracy of ± 3 milligals will certainly provide useful results. Attributing the entire inaccuracy to elevation inaccuracy used in the free-air reduction, elevation would have to be known at least to an accuracy of ± 15 meters.

The instrument uncertainty is estimated as ± 1.0 milligal with respect to the base station on the moon. However, because of the nature of the instrument, its accuracy is a function of time away from the base station. At neighboring gravity stations the relative accuracy of the gravity measurement may be as high as ± 0.5 milligal. We estimate, therefore, that an elevation accuracy of ± 2 meters (corresponding to a free-air correction of ± 0.4 milligal) will not seriously degrade the accuracy of the total measurement and hence is adopted as the goal of the elevation determination.

Elevation inaccuracies associated with a tilt of the datum plane are less serious than errors with respect to the datum plane because the corresponding gravity errors in the first case give rise to an apparent error in the regional gradient. These are less serious than residual gravity errors at each station.

If the entire traverse area is judged to be flat to within a few meters (apart from small craters, etc.) the requirement to obtain elevation will diminish considerably, provided care is taken to establish gravity stations away from obvious topographic highs or lows.

The maximum number of gravity stations for the total mission is planned to be 60. The maximum number of stations per traverse is 20 or 30 (depending on whether there are 2 or 3 traverses on which gravity measurements are made). We estimate that the minimum desirable number of gravity measurements per traverse is 5.

5.3.1 Summary

A. Estimated elevation accuracy requirement

1. Goal, ± 2 meters
2. Acceptable, ± 14 meters
   (tilt of the datum plane is less important than errors with respect to the datum plane).

B. Number of gravity stations for which elevation is required

1. desired maximum, 60 per mission i.e., 20 or 30 per traverse
2. minimum, 5 per traverse

5.4 EXPERIMENT DATA REQUIREMENTS

At the conclusion of each mission, within the constraints of NASA procedures and operational criteria, the principle investigator will require the data specified below. This will enable him to complete his investigation and prepare the required scientific reports relating to the experiment. It is envisioned that the scientific investigator or his representative will monitor the operation of the Gravimeter from Mission Control. During this process, a "quick look" real time evaluation of reported data will be made to determine

1. Whether the data being secured appears reasonable. If it is not, then a request would be made for a repeated measurement during a traverse stop provided astronaut work load is not infringed upon.
2. Determination of instrument performance, via astronaut reporting of status monitoring indicators contained on the display panel.

With the above it will be possible to determine if the instrument is performing within specification. If the decision is made that useful data is not being obtained, the experiment could be abandoned if desired.

5.4.1 Calibration of the Gravimeter

where $f = K_0 + K_1 G + K_0 G + K_2 G^2 + K_3 G^3$

and f = output frequency
K_0 = bias term
K_1 = scale factor
K_2, K_3 = non-linear terms

5.4.1.1 Final calibration values of K_0, K_1, K_2, K_3 established on the gravity sensor prior to instrument assembly.

5.4.1.2 Pre-launch values of K_0 and K_1 established on the gravimeter in normal and inverted mode operation two days prior to launch.

5.4.1.3 Predicted drift of K_0 and K_1 established over a span of at least 10 days with uninterrupted sensor temperature control at instrument operating temperature.

5.4.2 Lunar Measurements

5.4.2.1 Time-line of experiment from pre-launch calibration including instrument power modes, measurements, and status of monitor indicators.

5.4.2.2 Each lunar measurement

1. Raw data in Hertz as indicated by display or read from returnable memory.

2. Data in milligal, reduced by all known instrument corrections; include reduction computation.

3. Horizontal position relative to the LM landing site in two coordinates from ± 10 meters at the LM base to ± 25 meters ar the far end of each traverse.

4. Elevation relative to the datum plane to ± 14 meters over the entire traverse. The desired resolution for interstation elevation is ± 2 meters.

5. Time of measurement to within 1 minute.

5.4.3 Format

All data to be presented in hard copy or computer line print-out.

6.0 PRELIMINARY INSTRUMENT SPECIFICATION

6.1 INTRODUCTION

The purpose of this section is to specify the characteristics of a lunar gravimeter for use in the "Lunar Traverse Gravimeter" experiment. The experiment consists of two parts: 1) a gravity traverse wherein spot values of gravity relative to the LEM base station are obtained at preselected science stations along the rover or astronaut traverse, and 2) a gravity transfer wherein the value of gravity at the LEM landing site is obtained relative to the value at the earth launch site through an accurate calibration of the gravimeter sensor over the range 0 - 1 earth g.

The gravimeter is designed to be carried on the rover and deployed on the ground at each station. The instrument should easily be modified from rover to hand carried.

The proposed sensor is the AMBAC Lot D4e Vibrating String Accelerometer, hereinafter referred to as the VSA. The output of the VSA is a frequency proportional to input acceleration. The output equation may be expressed as a McLaurin's series:

$$f = K_0 + K_1 A + K_2 A^2 + K_3 A^3 + \ldots \qquad (1a)$$

where:

f	=	output frequency
A	=	acceleration along the input axis
K_0	=	bias term
K_1	=	scale factor
$K_2, K_3 \ldots$	=	non-linear terms

If we label the accelerometer position resulting in maximum output frequency for a fixed input acceleration as the "normal mode", we can define four modes of operation for the experiment.

a. $A = G_E$, normal earth mode
b. $A = -G_E$, inverted earth mode
c. $A = G_L$, normal lunar mode
d. $A = -G_L$, inverted lunar mode

We write the VSA output equation for the four modes:

$$f_{+GE} = K_0 + K_1 G_E + K_2 G_E^{\;2} + K_3 G_E^{\;3} + \ldots \qquad (1b)$$

$$f_{+GE} = K_0 - K_1 G_E + K_2 G_E^{\;2} - K_3 G_E^{\;3} + \ldots \qquad (1c)$$

$$f_{+GL} = K_0 + K_1 G_L + K_2 G_L^2 + K_3 G_L^3 + \ldots \qquad (1d)$$

$$f_{-GL} = K_0 - K_1 G_L + K_2 G_L^2 - K_3 G_L^3 + \ldots \qquad (1e)$$

Taking sums and differences:

$$f_{+GE} + f_{-GE} = 2K_0 G_E + 2K_2 G_E^2 \qquad (1f)$$

$$f_{+GE} - f_{-GE} = 2K_1 G_E + 2K_3 G_E^3 \qquad (1g)$$

$$f_{+GL} + f_{-GL} = 2K_0 G_L + 2K_2 G_L^2 \qquad (1h)$$

$$f_{+GL} - f_{-GL} = 2K_1 G_L + 2K_3 G_L^3 \qquad (1i)$$

The normal and inverted mode measurements required for equations (1f-i) together constitute a <u>bias calibration</u>.

Both K_2 and K_3 are small, stable, and predictable on the basis of K_0 and K_1. All of the VSA factors will be determined at a pre-launch calibration on earth. Thereafter, K_2 and K_3 will be assumed constant or determined by K_0 and K_1. Two days prior to launch, K_0 and K_1 will be updated through equations (1f) and (1g). After landing on the moon, a lunar gravity measurement (transfer) will be obtained using equation (1i). The traverse measurements will be obtained using a first-order approximation in equation (1j)

$$\Delta f_{+GL} = \Delta K_0 + K_1 \Delta G_L + \Delta K_1 G_L + \Delta K_1 \Delta G_L \qquad (1j)$$

The information displayed and recorded by the gravimeter is $\Delta f_{+GL}(t)$. $\Delta K_0(t)$ and $\Delta K_1(t)$ are both obtained by performing <u>bias calibrations</u> at the LEM base station before and after each traverse. The constant value G_L at the LEM station has been determined previously by the transfer measurement. The term $\Delta K_1 \Delta G_L$ is discarded.

The ability to perform lunar <u>bias calibration</u> is very important for the success of the experiment because it eliminates all dependence upon the stability of K_0. It is for this reason that the gravimeter is designed for operation in both the normal and inverted modes.

6.2 UNITS

The acceleration of gravity unit is the gal.

$$1 \text{ gal} = 1 \text{ cm/sec}^2$$
$$1 \text{ milligal} = 10^{-3} \text{ cm/sec}^2$$

⊕ 2

Therefore,

$$G_E - 980 \text{ gal}$$

$$G_L - 162 \text{ gal}$$

6.3 SPECIFICATIONS, GENERAL

6.3.1 Weight of Launch Package $\leq$ 25 Earth Lb.

6.3.2 Weight of Traverse Package $\leq$ 20 Earth Lb.

6.3.3 Volume of Launch Package $\leq$ 1.2 Cu. Ft.

6.3.4 Internal power sufficient for entire mission will be included. Power modes are: a) VSA temperature control, entire mission; b) VSA amplifiers, continuous during traverse; c) data electronics, during measurement only.

6.3.5 Two measurement modes shall be possible: normal and inverted.

6.3.6 Data quantization at lunar G:

$\leq$ 0.1 mgal, normal mode

$\leq$ 1.0 mgal, inverted mode

6.3.7 Full scale range $\geq 10^3$ gal, either mode.

6.3.8 The pendulous element shall be caged except during a measurement.

6.3.9 The pendulous element shall be passive and self-leveling from within $\pm 3^\circ$ from the vertical.

6.3.10 The measurement cycle shall be:

a) astronaut leveling to $\leq \pm 3^\circ$

b) astronaut initiate automatic operation by actuating "measure" control. (The pendulous element will be uncaged by the same action.)

c) A data sample will automatically be initiated after self-leveling to within $\pm$ 7 arc minutes or a lapse of 2 minutes time from uncaging, whichever comes first.

6.3.11 Astronaut controls shall include:

a) VSA POWER ON-OFF. Purpose - to conserve battery power during long periods of dormancy.

b) MEASURE ON-OFF. Purpose(s) - to initiate a gravity measurement and to cage pendulum element.

6.3.12 Visual displays shall include:

a) level bubble. Purpose - aids astronaut in leveling the instrument outer case.

b) level light. Purpose - indicates off-level of pendulous element of $\geq$ 7 arc minutes; also indicates measurement in progress.

c) battery light. Purpose - comes on when supply battery voltage reaches predetermined lower limit; indicates a known battery reserve for possible experiment schedule change.

d) temperature high/low. Purpose - indicates a potential loss of temperature control; allows astronaut to take preventive action (shade instrument if "high"; expose to sun if "low").

e) gravity. Purpose - display full scale gravity measurement for voice link communication to earth.

6.3.13 Provided the requirements of 3.1, 3.2, 3.3, and those of crew task and time line are not compromised, the instrument should contain internal data storage for a total of $\geq$ 64 full-scale gravity measurements in either mode.

6.3.14 The instrument shall be capable of operation in both lunar sun and shade.

The instrument with its levelling base shall be capable of operation on a slope not exceeding 15°.

6.3.15 The maximum acceptable gravimeter drift rate in any mode is 1 milligal per hour.

6.4 SPECIFICATIONS, TRANSFER ONLY

6.4.1 The initial earth calibration error of K_1 shall not exceed 12 ppm and 30 $\mu G/G^3$, 2σ.

6.4.2 The shifts of K_1 and K_3 due to the combined causes of drift predictability, launch vibration, shock, and steady acceleration from pre-launch calibration until the first bias calibration shall not exceed 24 ppm and 30 $\mu G/G^3$, 2σ.

6.4.3 The error from all sources other than referenced in 7.1 and 7.2 in determining G_L by the bias calibration method (equation 1i) shall not exceed 5 milligal, 2σ.

6.5 SPECIFICATIONS, TRAVERSE ONLY

6.5.1 Instrument repeatability under steady-state conditions shall be ≤ 0.2 milligal rms over a period of 4 hours after correction for a linear drift.

6.5.2 Instrument performance from all traverse error sources after correction for a linear drift and for a period of 5 hours shall be $\leq (0.5 + 0.1\,T)$ milligal rms, where T, in hours, equals elapsed time from first traverse measurement.

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MSC:	(11)

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