AD-B193 901    LUNAR CAPSULE (SEISMOMETER) SYSTEM RA 3 4 AND 5    1/1
FUNCTIONAL SPECIFICATION(U) AERONUTRONIC FORD CORP
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AD-B193 901

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SPACE SYSTEMS OPERATIONS





# FUNCTIONAL SPECIFICATION



# LUNAR CAPSULE (SEISMOMETER) SYSTEM
## RA 3, 4, AND 5



Prepared for: Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California

Under Contract: JPL Contract N-21453
(Issued under NASA Contract NASw-6)

Prepared by: Lunar Systems

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16 November 1961

AERONUTRONIC
A DIVISION OF Ford Motor Company
FORD ROAD / NEWPORT BEACH, CALIFORNIA

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RA-345-4-140
15 November 1961



# AERONUTRONIC DIVISION
# FORD MOTOR COMPANY

FUNCTIONAL SPECIFICATION

LUNAR CAPSULE (SEISMOMETER) SYSTEM
RA 3, 4 & 5



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## 1.0 SCOPE

1.1 This specification shall serve to define the configuration and the functional performance for the lunar capsule system which forms a portion of the NASA Ranger Program. This system shall primarily consist of that equipment carried on the Ranger spacecraft specifically for the purpose of separating and landing a lunar landing sphere on the surface of the moon.

## 2.0 APPLICABLE DOCUMENTS

2.1 The following documents shall be a part of or be applicable to this specification:

SPECIFICATIONS

Aeronutronic Division, Ford Motor Company

(1) LCS-001 Environmental Specification for Condition I
(2) LCS-002 Environmental Specification for Condition II
(3) LCS-003 Environmental Specification for Condition III
(4) LCS-004 Environmental Specification for Condition IV
(5) LCS-005 Sterilization Specification

DRAWINGS

Aeronutronic Division, Ford Motor Company

No. 800000
No. 800037
No. 803211

MILITARY SPECIFICATION

MIL-E-5272(C) Environmental Testing

OTHER

Ford Motor Company Subcontract SC-05850
Ford Motor Company Subcontract SC-

## 3.0 REQUIREMENTS

### 3.1 Description

The primary purpose of the lunar capsule system shall be to land and instrumented landing sphere on the surface of the moon after being boosted through a launch trajectory and controlled in space on an impact trajectory with the moon. The major assemblies of this system are:

(1) Lunar landing sphere, complete with a survival sphere
(2) Retrorocket
(3) Altimeter assembly
(4) Spin motor assembly
(5) Support and separation structure
(6) Altimeter support and erection assembly
(7) Power sequencing assembly
(8) Wiring harness assembly
(9) Retrorocket thermal shield
(10) Retrorocket/landing sphere; interconnect

### 3.2 Performance

In order that the primary objective of the lunar capsule may be fulfilled, the following functional events shall take place but not necessarily in the following order:

An altimeter shall be carried as a part of the spacecraft, but in a stowed position so to fit within the vehicle nose fairing. The altimeter shall, after erection, turn on and warmup, monitor the distance to the lunar surface from the approaching spacecraft, and then provide a signal at a predetermined fuzing altitude. The fuzing signal shall initiate the subsequent events. The altimeter shall act as a pulsed radar and shall signal the proper altitude when the return pulse falls within the range gate. The altimeter-generated signal shall initiate both a spin-up system and a separation system so that a solid propellant retrorocket motor may decelerate the lunar landing sphere to impact velocity. The impact velocity will permit survival during the impact of the survival sphere assembly. Spin-up of the assembled retrorocket motor and lunar landing sphere is requ red to provide spin stabilization to permit the proper velocity vector control during retrorocket motor performance. After burnout of the fixed-impulse retrorocket motor, the power and sequencing assembly shall cause separation of the empty motorcase from the lunar landing sphere.

The lunar landing sphere shall consist primarily of an impact limiter assembly surrounding an integral survival sphere but with attachment flanges on each end (axially) of the limiter. The lunar landing sphere shall be attached by a clamp device to the retrorocket and at the opposite end provides support for an omniantenna which shall previously have been deployed from the forward position on the sphere so that the following events may occur.

After impact, the survival sphere shall cause a seismometer to be oriented along the local vertical axis and the capsule transmitting antenna to be erected into proper position. The seismometer and associated sphere electronics shall obtain seismic data from the lunar surface, condition and transmit these data to the earth using as a part of the communication link the NASA (JPL) DSIF. (See Figure 3.1 for trajectory).

## 3.3 Physical Characteristics

### 3.3.1 General

Figure 3.2 illustrates the subassemblies and components which make up the lunar capsule system. Drawing numbers and specification numbers are shown.

### 3.3.2 Weight

The total weight of the lunar capsule system shall not exceed 325.0 pounds.

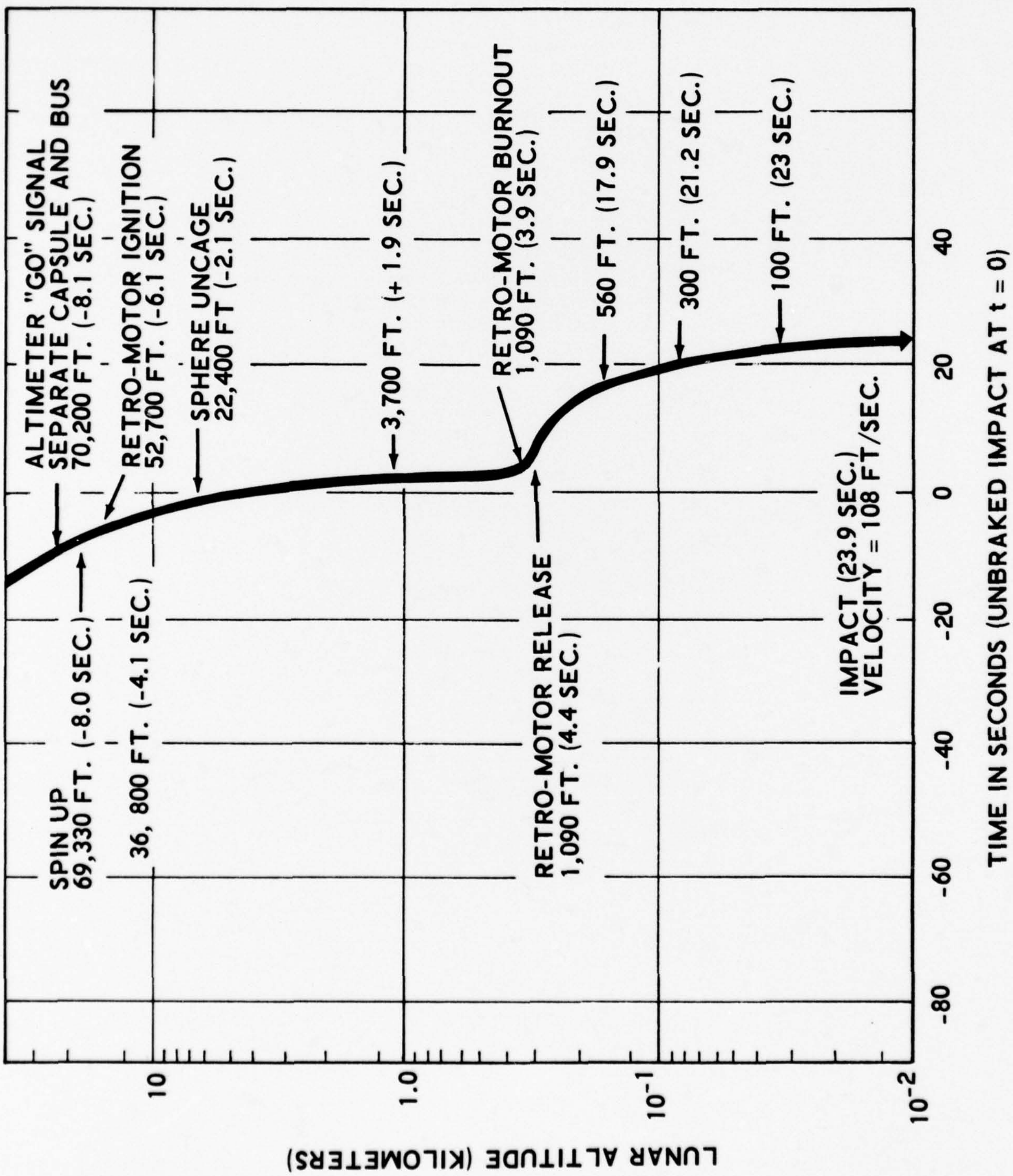


FIGURE 3.1 TERMINAL TRAJECTORY

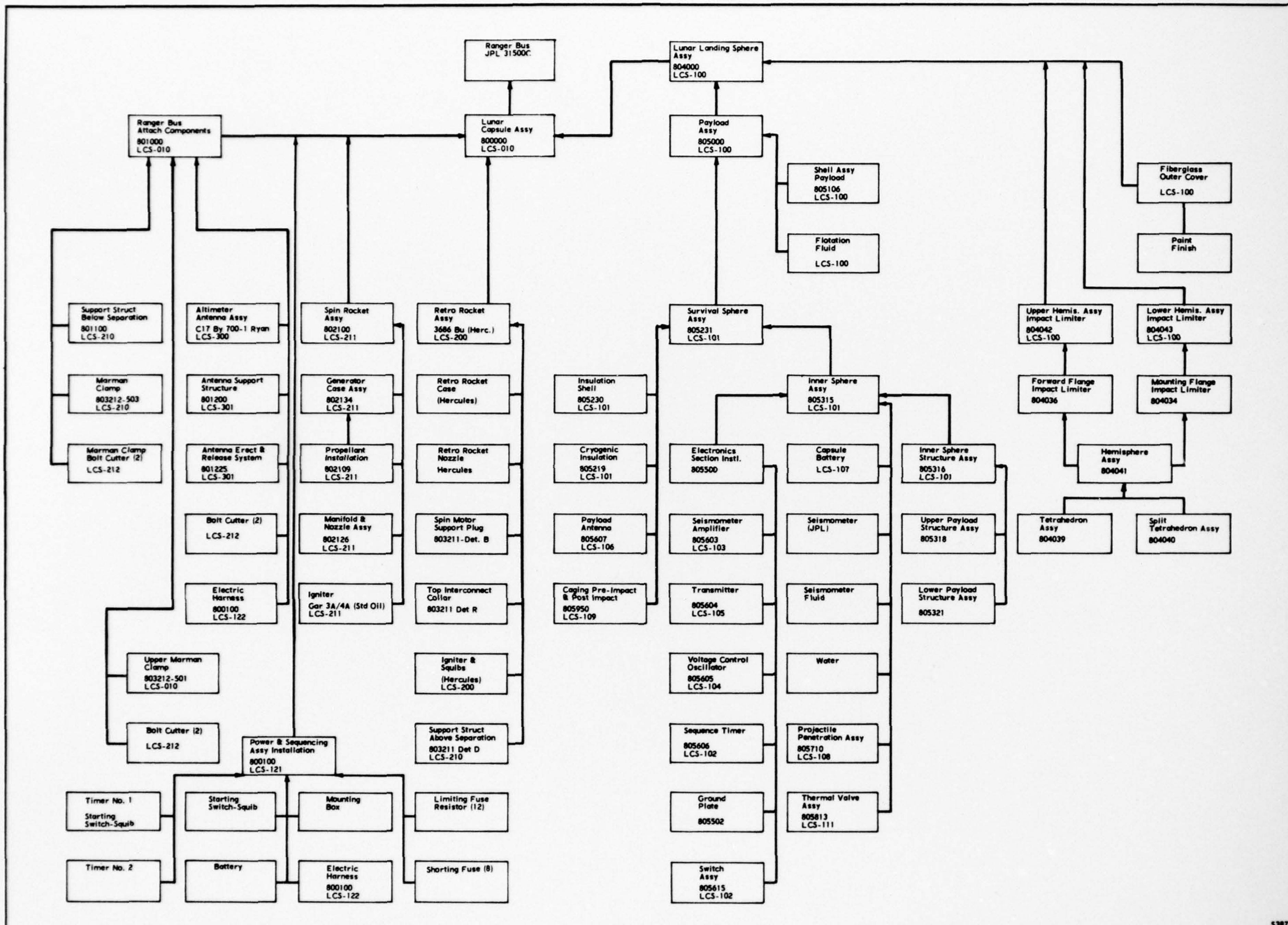


FIGURE 3.2 LUNAR CAPSULE ASSEMBLY FLOW CHART

3.3.3 Center of Gravity Location

The center of gravity of the entire system as installed (altimeter in stowed position) on the spacecraft bus shall lie within 30 inches of the bus-support assembly interface. The misalignment of the thrust axis of the rocket motor and the bus longitudinal axis (as defined by the bolt-circle surface) shall not exceed 0.1 degree. Excluding the altimeter and its support structure, the capsule center of gravity shall not be displaced more than 0.03 inch from the retrothrust axis.

3.4 Lunar Landing Sphere

The lunar landing sphere consists of that portion of the Ranger vehicle which is released from the Ranger spacecraft (and boost vehicle), spin stabilized, decelerated to a landing velocity not to exceed (for most trajectories) 200 feet per second and which, after landing, places a seismometer in position for obtaining seismic data from the lunar surface for an interval of approximately niney (90) days. The lunar landing sphere shall also condition and transmit the seismometer data obtained during this time interval to the earth utilizing as a part of the communication link the NASA (JPL) Deep Space Instrumentation Facility.

The configuration shall consist of the following:

(1) Impact limiter complete with outer cover, attach flanes, and inner shell.

(2) Survival sphere complete with outer shell, structure, antenna and transmitter, seismometer and associated electronics, venting and temperature control systems, and caging and uncaging systems.

(3) Floatation fluid to support the survival sphere within the impact limiter inner shell.

The primary functional requirements of the lunar landing sphere are to:

(1) Provide impact survival and structural support of the survival sphere.

(2) Accomplish erection to the local gravity vector of the antenna and simultaneously emplace the seismometer in position.

(3) Provide insulation and temperature control against the lunar environment of the seismometer and associated data-communication system.

(4) Provide for venting of the floatation fluid.

(5) Provide the structural support for the spacecraft omniantenna and to support the landing sphere on the retrorocket throughout the launch environment.

### 3.4.1 Survival Sphere

The survival sphere is a major portion of the Ranger lunar landing sphere. It is that portion which, by utilizing the impact limiter and floatation fluid, is caused to survive accelerations attendant to impact with the lunar surface. The survival sphere shall obtain seismic data from the surface of the moon, condition, and transmit those data utilizing the NASA (JPL) DSIF at Goldstone Lake.

The configuration shall consist of the following:

(1) An outer shell
(2) A thermal insulation layer
(3) An upper assembly containing the electronics subassembly
(4) An antenna block
(5) A lower assembly containing a seismometer, battery, water storage, venting, and caging devices.

The primary functional requirements of the survival sphere are to:

(1) Provide the structural support for the antenna, electronics, power supply, and seismometer.

(2) Provide thermal insulation so that lunar temperatures do not cause the internal temperature to vary beyond the limits 32 to 80°F. The thermal control valve, which vents water vapor during the lunar day, also forms a functional portion of the temperature control system.

(3) Provide the means for erecting the antenna to the local gravity vector within the floatation fluid.

(4) Expel the floatation fluid by venting through two internally punctured holes (venting system) but without wetting the thermal insulation layer sufficiently to detract from its functioning. The venting action must also vent the floatation fluid from within the seismometer.

(5) Provide both a pre-impact uncaging and a post-impact caging of the survival sphere within the impact-limiter inner shell.

Physical characteristics of the survival sphere shall be as follows:

(1) Total weight of the survival sphere shall not exceed 49.0 lb.

(2) A positive displacement of the center of buoyancy and the center of gravity of the survival sphere shall exist in the approximate amount of 0.5 in. The center of buoyancy shall be so located so as to place the antenna in the uppermost position of the sphere while floated in the floatation fluid.

(3) Figure 3.3 shows the components and subassemblies of the survival sphere. Drawing numbers and specification numbers are included.

3.4.1.1 <u>Sequence Timer</u>

The function of the Sequence Timer shall be to program the sequential events within the survival sphere immediately after the time of lunar impact. A block diagram of the sequence timer is shown in Figure 3.4. The sequence timer begins operation near the time of lunar impact when turned on by an external 25G acceleration switch. Simultaneously, the acceleration switch also fires squibs to uncage the inner sphere.

After lunar impact, the sequence timer fires two (2) parallel squibs to cause sphere caging. Next, it fires two (2) parallel squibs to produce sphere venting of the floatation fluid. At thirty (30) minutes from start, the sequence timer closes two (2) switches for actuation of the seismometer centering motor. It also disconnects itself from the battery power supply at this time.

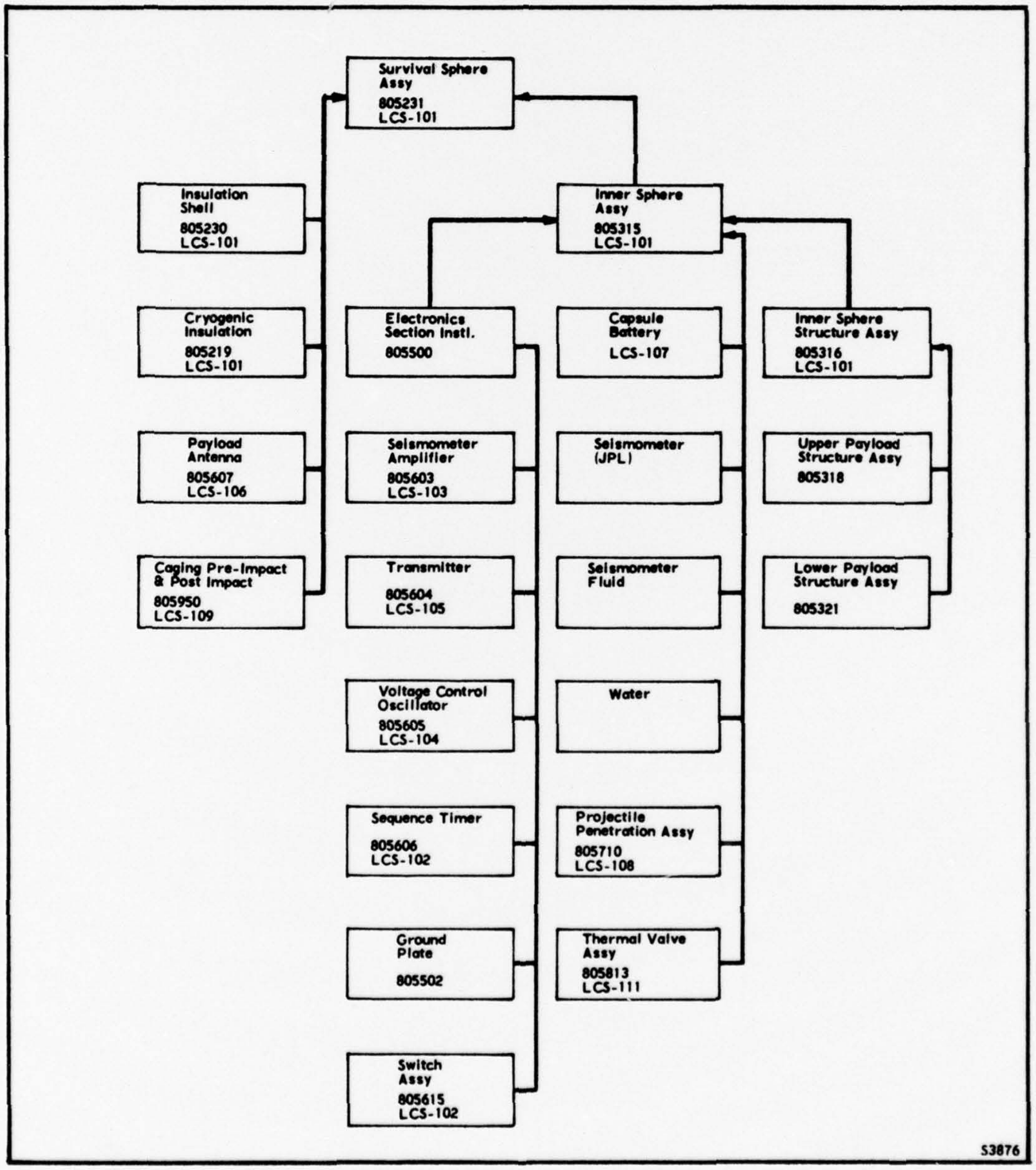


FIGURE 3.3 SURVIVAL SPHERE ASSEMBLY FLOW CHART

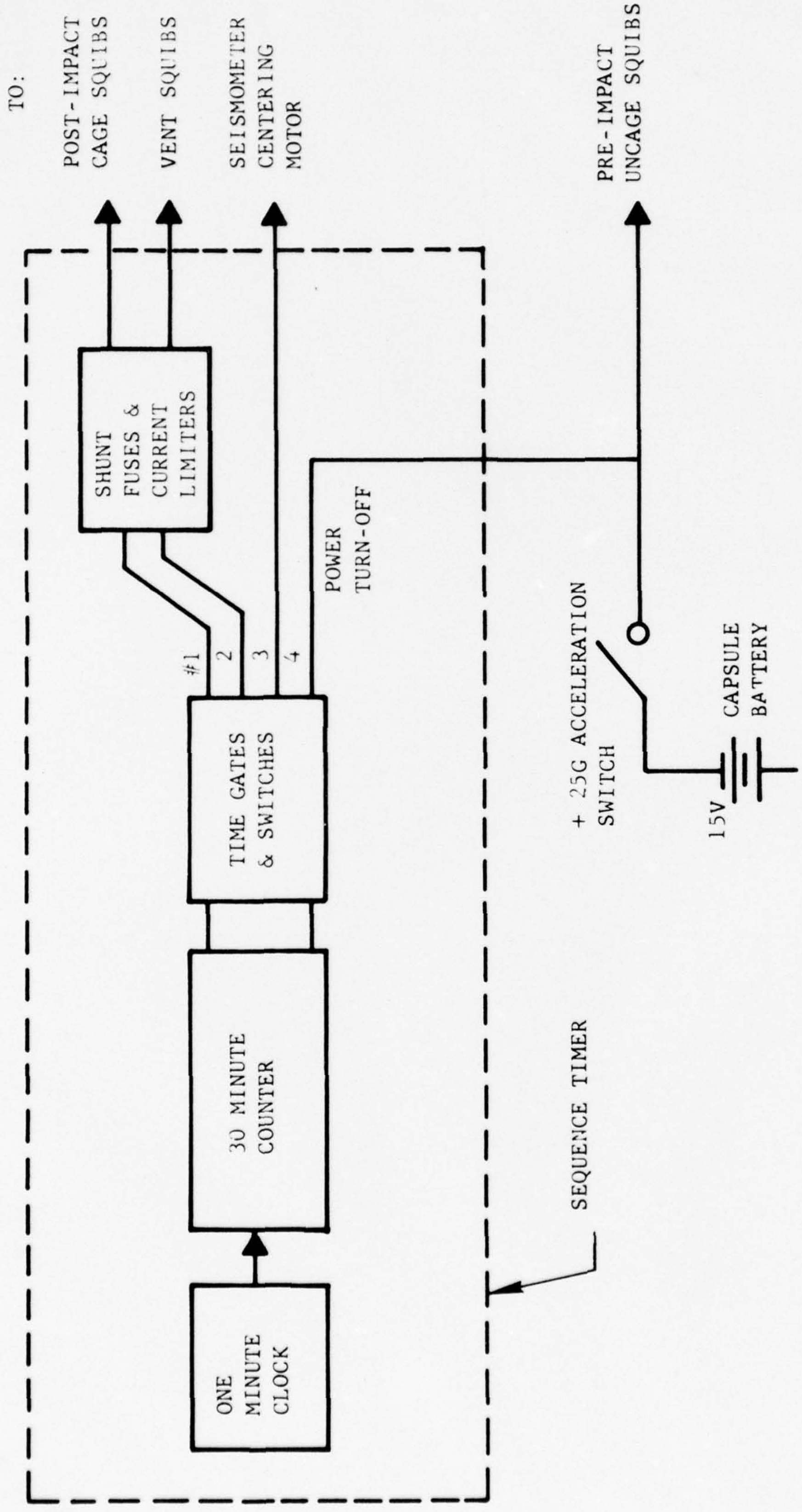


FIGURE 3.4 FUNCTIONAL DIAGRAM OF SEQUENCE TIMER

The timing interval of the clock which drives the timing counter is 1.0 (± 20%) minutes. The outputs of the timing counter are at the following nominal times:

(1) 15 Minutes - post impact caging squibs.

(2) 16 Minutes - sphere venting squibs.

(3) 30 Minutes - seismometer centering motor switches.

(4) 30 Minutes - power disconnect switch.

The squibs are protected from accidental firing by a paralleled one (1) ampere fuse.

At a battery voltage of fifteen (15) volts and with a current limiting resistance of 10 ohms, the sequence timer will furnish a squib firing current of not less than 0.8 ampere. A short circuit condition at the squib after it has been fired does not draw over 1.5 amperes for more than 1.0 seconds. Such a short circuit does not adversely affect the operation of the sequence timer.

The sequence timer closes two (2) switches for actuation of the seismometer centering motor.

### 3.4.1.2 Seismometer Amplifier

This subsection describes the functional performance of the seismometer amplifier which is part of the telemetry equipment in the lunar capsule.

The seismometer amplifier is a part of the equipment landed on the moon and used to telemeter seismic information to the earth. It shall receive its input signal from a seismometer transducer, amplify and condition this signal, and given an output signal to a voltage-controlled subcarrier oscillator. The basic functions are:

(1) Provide amplification of the seismometer signal so that the output is at the correct level for modulating the VCO.

(2) Provide amplitude compression so that an input signal whose amplitude is 31.6 times the reference level is reduced to a level no more than ten (10) times the reference level. (accuracy ± 2 db)

(3) Provide attenuation of signals whose frequencies are below 0.05 cps and above 5 cps.

(4) Provide amplitude clipping compatible with the requirements of the telemetry link.

3.4.1.2.1 <u>Electrical Characteristics</u>

The electrical characteristics are as follows:

(1) <u>Input Impedance</u>

The amplifier is a three-terminal differential-input device. The input resistance from either active input terminal to the third terminal is 1,000 (+0%, -40%) ohms.

(2) <u>Common-Mode Rejection</u>

The common-mode rejection of a signal applied simultaneously to both input terminals exceeds 30 db.

(3) <u>Voltage Gain</u>

The high level voltage gain of the amplifier is such that an input signal of 17 microvolts peak to peak at 0.5 cps produces the maximum output signal (clipping level). The low level differential gain is 150,000 ± 3 db.

(4) <u>Frequency Filtering</u>

The amplifier provides a loss-pass and a high-pass filter. The low-pass cut-off is at 5.0 cps (-3 db, ± 1 db) and provides approximately 18 db per octave attenuation. The high-pass filter cut-off is at 0.05 cps (-3 db, ± 1 db) and nominally provides 6 db per octave attenuation. The frequency response between 0.05 and 5.0 cps is flat to within ± 2 db of the 0.5 cps frequency.

(5) Output Voltage

The amplifier provides an ac output signal of one volt peak-to-peak at the clipping level. In addition, it provides a dc output signal which centers the vco to approximately 560 cps.

(6) Linearity

The amplifier when operated with the Ground Adapter gives an amplitude linearity error not greater than $\pm$ 10% of full-scale from a best straight line drawn through zero signal.

(7) Noise

The overall noise at the output of the amplifier, when measured in a 0 to 5 cps bandpass, does not exceed 1.0 microvolts peak-to-peak when referred to the seismometer input terminals. The measurement is taken over a typical 40 second period.

(8) Power

The seismometer amplifier operates from a power source of 6 to 7.4 volts dc with a maximum current not exceeding 5 milliamperes and a power source of 15 to 18.50 volts dc with a maximum current not to exceed 0.5 milliamperes.

3.4.1.3 Voltage Controlled Oscillator

This subsection establishes the performance requirements of the Voltage Controlled Oscillator (VCO), which is a part of the telemetry equipment in the survival sphere of the Lunar Capsule.

3.4.1.3.1 Functional Description

The voltage-controlled oscillator shall form a part of the telemetry link to transmit seismic information from the moon to the earth. The VCO shall accept an input signal from a seismometer amplifier and change this information into a frequency-modulated signal, which shall be used to modulate a transmitter.

3.4.1.3.2 <u>Input Characteristic Requirements</u>

The VCO shall have the following input characteristics:

(1) Input Impedance - 450 K or greater

(2) Modulation Sensitivity - 0 to +3 volts dc for $\pm$ 7 1/2% minimum frequency deviation

(3) Input Intelligence Frequency Response - flat within 1 percent from dc to 8.4 cps

3.4.1.3.3 <u>Output Characteristic Requirements</u>

(1) Output Voltage - minimum of 0.3 v rms into a load of 25 K ohms.

(2) Output Frequency - band 2 per IRIG Document 103-56

Center Frequency = 560 cps
Lower Frequency = 518 cps (-7 1/2 percent)
Upper Frequency = 602 cps (+7 1/2 percent)

(3) Amplitude Modulation - less than $\pm$ 1 db with combined frequency deviation, power supply variation, and temperature variation.

(4) Phasing - a positive incrasing input signal shall result in an increasing output frequency.

(5) Harmonic Distortion - less than 1 percent.

(6) Linearity - the curve of output frequency versus input voltage shall be linear to 1 percent of bandwidth when compared with the best straight line drawn through the curve.

(7) Frequency Stability - frequency drift due to the combined effects of temperature (+32$^{o}$ to + 80$^{o}$F). voltage changes, and aging shall not exceed 1.5 cps.

3.4.1.3.4 <u>Physical Requirements</u>

The VCO shall conform to Drawing 805580 in regard to dimensions, mounting provisions, weight, connections, and finishes.

3.4.1.3.5 Power Requirements

The VCO shall operate from a power supply of 6V $\pm$ 0.1 volts dc. The power supply current required by the VCO shall not exceed 1.0 milliampere.

3.4.1.4 Transmitter

This subsection establishes the design and performance requirements of the transmitter within the survival sphere of the lunar capsule.

3.4.1.4.1 Functional Description

The transmitter shall function as an element in the FM/PM tlemetry link to transmit seismic data from the moon to the earth. Other electronic elements of the data communication system are the voltage-controlled oscillator, the seismometer amplifier, and the antenna.

3.4.1.4.2 Modulation Requirements

The transmitter shall be phase modulated by an input signal from a VCO. The modulation characteristics shall be:

(1) Modulation sensitivity - not more than 0.3 v required for 1.4 radinas of phase deviation

(2) Input impedance - not less than 25K ohms shunted by not more than 500 picofarads

(3) Modulation frequency - 518-602 cps
560 cps, nominal

(4) Maximum input signal - 1 volt peak-to-peak

(5) Modulation distortion - not to exceed 2 percent from linear operation.

3.4.1.4.3 Output Requirements

The transmitter shall have a radio frequency output signal with the following characteristics:

(1) Frequency -
RA-3 960.150 ($\pm$ .002 percent) megacycles at 75$^{\circ}$F
RA-4 960.250 ($\pm$ .002 percent) megacycles at 75$^{\circ}$F
RA-5 960.150 ($\pm$ .002 percent) megacycles at 75$^{\circ}$F

Frequency change due to temperature shall not exceed 400 cps/$^{o}$F over the temperature range of 32$^{o}$ to 80$^{o}$F. Permanent carrier frequency shift due to the lunar impact shall not exceed 3,000 cps.

(2) Output power - not less than 50 milliwatts effective radiated power by the matching antenna 45$^{o}$ off axis as described in Section 3.4.1.5.

(3) Output impedance - matched to antenna to give a VSWR of not more than 1.5.

(4) Spurious output - radiated signals at all other frequencies to be at least 20 db below unmodulated carrier signal level.

(5) Phase stability - The phase stability of the numodulated RF carrier expressed as the peak phase error developed in an automatic phase control receiver having a normalized phase transfer function which is:

$$H\ (S) = \frac{1 + \frac{3}{4B_L}\ S}{1 + \frac{3}{4B_L}\ S + 1/2 \left(\frac{3}{4B_L}\right)^2 S^2}$$

where

$$2B_L = \frac{1}{2\ \ j} \int_{-J\infty}^{+J\infty} \left[H\ (S)\right]^2 \quad ds = 20 \text{ cps}$$

shall be less than 0.5 radians.

3.4.1.4.4 Power Requirements

The transmitter shall operate from a battery power supply with the following characteristics:

(1) Voltage - 15 to 18.5 volts

(2) Voltage modulation - 1 volt p-p at 560 cps

(3) Maximum current - 80 ma.

3.4.1.5 Antenna

This subsection establishes the performance requirements of the antenna which is used within the lunar capsule survival sphere to transmit seismic data from the moon to the earth.

3.4.1.5.1 Functional Description

The antenna and its dielectric material shall constitute the top portion of the survival sphere. The antenna shall be capable of surviving lunar impact and afterwards capable of transmitting data to the earth. The antenna feed shall be designed to provide low thermal conductivity to the inner sphere. The antenna shall be right-hand circularly polarized. The antenna beam cross section shall be circular about the vertical axis with the major portion of the radiated energy within 45 degrees of the vertical axis.

3.4.1.5.2 Electrical

The antenna shall be used at nominal frequency of 960 megacycles.

The input impedance shall be nominally 50 ohms to match the transmitter described in section 3.4.1.4. The VSWR shall be less than 1.5 to 960 megacycles when drivere from a 50 source.

The on-axis radiation shall be right-hand circular with an axial ratio of less than 1.0 db.

The gain of the right-hand circularly (RHC) polarized radiation component of the antenna shall be measured with the antenna within an impact limiter. The RHC polarized component gain measured anywhere within 45 degrees of the vertical axis shall not be less than 0 db relative to an isolated isotropic antenna.

3.4.1.6 Battery

This subsection defines the performance requirements of the lunar capsule battery.

3.4.1.6.1 Functional Description

The battery shall be mounted within the survival sphere of the lunar capsule. Its principal function shall be to provide power to the electronic subassemblies within the sphere after the time of lunar impact.

3.4.1.6.2 Electrical Requirements

3.4.1.6.2.1 Cell Configuration

The battery shall consist of two (2) modules as shown in Figure 3.5. Each module shall contain an isolated 3-volt, 6-volt, and 9-volt section. The leakage resistance between any cells shall not be less than 500K.

3.4.1.6.2.2 Capacity

The minimum ampere-hour capacity of the battery during a niney (90) day discharge period shall be as shown below:

| | Per Modul (amp-hr) | Per Battery (amp-hr) |
|---|---|---|
| 3-volt output | 28.8 | 57.6 |
| 6-volt output | 18.9 | 37.8 |
| 9-volt output | 30.6 | 61.2 |

When the battery sections are connected as shown in Figure 3.6, the minimum ampere-hour capacities shall be as shown below:

| | Discharge Current (milliamperes) | Capacity (amp-hr) |
|---|---|---|
| 6 volt-output | 46 | 66.6 |
| 9 volt-output | 61 | 61.2 |

Not more than two (2) charge-discharge cycles shall be required to achieve the minimum required ampere-hour capacity.

3.4.1.6.2.3 Voltage

The following voltage limitations shall apply to the battery when operated as shown Figure 3.6.

| | Maximum (v) | Minimum (v) |
|---|---|---|
| 6-volt output | 7.40 | 5.80 |
| 9-volt output | 11.10 | 8.70 |

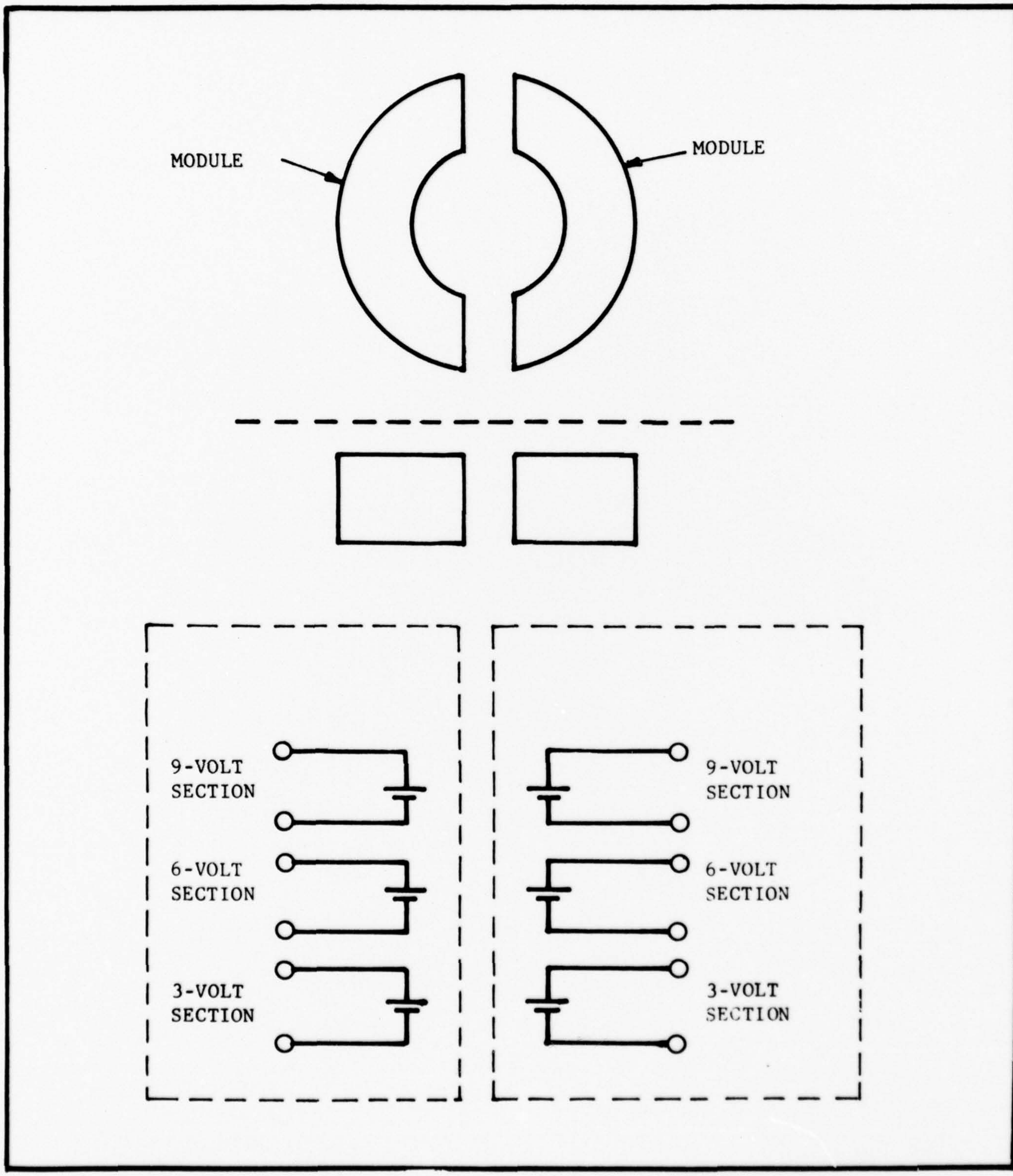


FIGURE 3.5 BATTERY CONFIGURATION

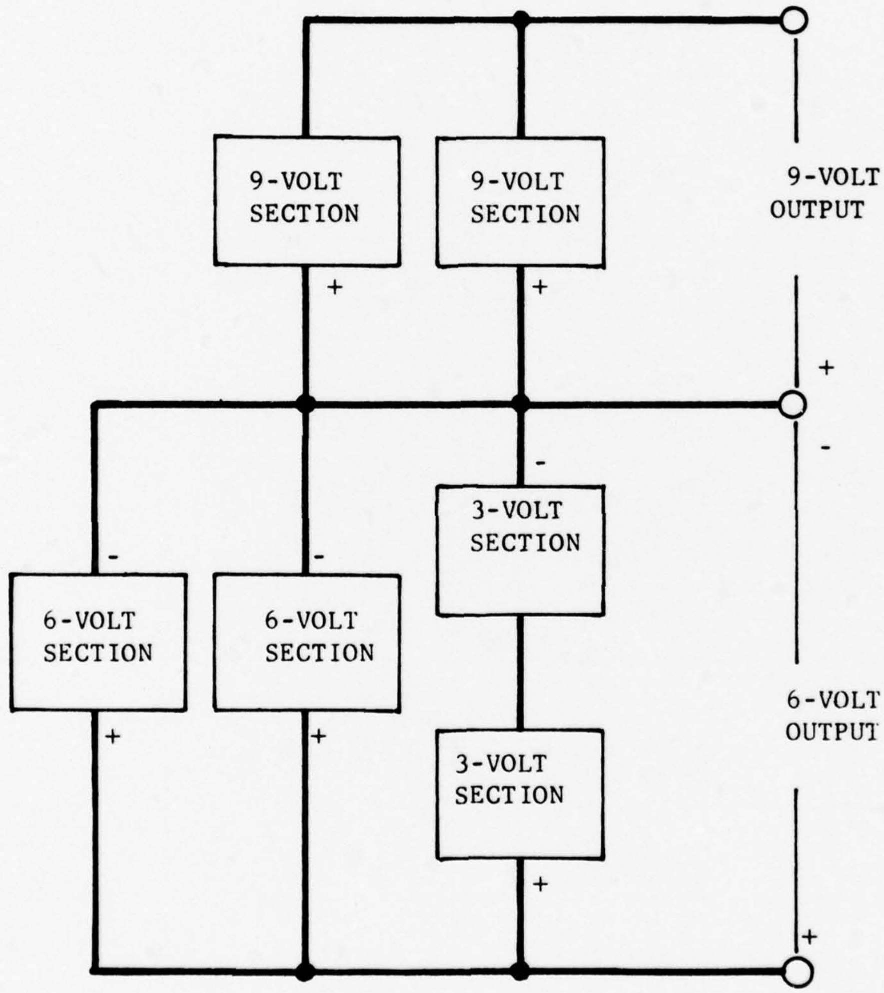


FIGURE 3.6 BATTERY SECTION CONNECTIONS

3.4.1.6.2.4 Storage

The battery shall be capable of storage in the charged state at a temperature of 25°C for a period of thirty (30) days without a loss of more than three (3) percent of its ampere-hour capacity.

3.4.1.6.2.5 Output Impedance

The output impedance of the battery shall not exceed 0.2 ohm.

3.4.1.6.2.6 Current Transients

The battery shall be capable of furnishing a momentary current of ten (10) amperes for firing ordnance squibs. Such a current shall not exist for over two (2) seconds nor occur more often than once per minute.

3.4.1.6.2.7 Battery Charging Procedure

Each module section shall be individually charged according to the table below. There shall be an automatic voltage cut-off device to terminate the charge at the limiting voltage. The voltage of each section shall be recorded continuously during the charging time.

| | PER MODULE | | |
|---|---|---|---|
| | 3v Section | 6v Section | 9v Section |
| Nominal discharged open-circuit voltage (v) | 3.20 | 6.40 | 9.60 |
| Constant charge current, maximum value (amp) | 0.42 | 0.36 | 0.42 |
| Nominal charge time (hr) | 99 | 59 | 90 |
| Maximum charging voltage limit (v) | 3.90 | 7.80 | 11.70 |
| Nominal charged open-circuit voltage (v) | 3.70 | 7.40 | 11.10 |

3.4.1.6.2.8 Accelerated Discharging Procedure

This procedure shall be followed for discharging a battery in approximately fifty (50) hours. There shall be an automatic voltage cut-off device for terminating the discharge at the limiting voltage. The voltage of each section shall be recorded continuously during the discharging time. The table below gives the nominal and limiting discharge parameters for each section of a battery module.

| | PER MODULE | | |
|---|---|---|---|
| | 3v Section | 6v Section | 9v Section |
| Nominal charged open-circuit voltage (v) | 3.70 | 7.40 | 11.10 |
| Constant load resistance, minimum value (ohm) | 4.2 | 12 | 12 |
| Nominal discharge time (hr) | 50 | 50 | 50 |
| Minimum discharging voltage limit (v) | 2.90 | 5.80 | 8.70 |
| Nominal discharged open-circuit voltage (v) | 3.20 | 6.40 | 9.60 |

3.4.1.7 Venting System

This subsection defines the configuration and performance requirements for the survival sphere venting system which forms a part of the lunar landing sphere.

3.4.1.7.1 Description

This assembly is to fire one or more projecties through the impact limiter to allow venting of fluid to the outside after lunar impact.

The assemblies shall perform satisfactorily under the conditions expected to exist immediately after lunar impact and be able to remain operable under all environmental conditions encountered during storage, shipment, handling, stand-by, preflight, and flight.

(1) Barrel Strength and Defiguration

The ultimate strength of the barrels shall be at least 60,000 psi.

(2) Action Time

The ignition shall occur in less than 50 milliseconds, as measured from the initiation of the firing current to the penetration through the impact limiter when the venting assembly is fired with 1.0 ampere.

(3) Squib Characteristics

The squib shall have a reliability of 0.929 at the 95% confidence level when fired with a 1.0 ampere current.

(4) Resistance

The squib bridge wiring shall contain a resistance of $2.7 \pm 0.6$ ohms.

(5) Insulation

The resistance between the bridge wire and the metallic portion of the barrel shall be 1.0 megohm or greater.

(6) Ignition

The charge shall fire when a current of 1.0 ampere or more is applied to its terminals for a dureation of 15 milliseconds. It shall not fire when a current of 0.10 ampere or less is applied five minutes.

3.4.1.8 Caging System

This subsection defines the configuration and performance requirements for the survival sphere caging systems which form a part of the lunar landing sphere.

3.4.1.8.1 Description

The caging and uncaging devices incorporated in the lunar capsule are located in the inner sphere and will functionally cage the floating inner sphere during the full duration of early flight. The device will uncage the sphere just prior to lunar impact and cage again after lunar impact with a time delay sufficient for the floating sphere to rotate to its proper orientation before actuation.

3.4.1.8.2 Performance

The assemblies shall perform satisfactorily under the conditions expected to be existent during the time of operation in flight and be able to withstand all environmental conditions expected to be encountered during storage shipment, handling, stand-b , preflight, and lunar impact to the operational period of the assemblies.

(1) Pin Strength and Distortion

The ultimate strength of the indexing pins shall be at least 50 lb in compression. The pins shall be able to withstand a bending force of 50 in-lb with less than 0.002 inch permanent deformation.

(2) Action Time

The pin motors shall function in less than 15 milliseconds, as measured from the initiation of the firing current to the full extension or retraction of the pins when fired with 1.0 ampere.

(3) Cartridge Characteristics

The cartridge shall fire with a reliability of better than 0.93 at the 95% confidence level, when energized with a 1.0 ampere firing current.

(a) Resistance

Each unit shall contain a single bridge wire, the resistance of which shall be 2.0 $\pm$ 0.6 ohms.

(b) Insulation

The resistance between the bridge wire and the metallic portion of the assembly shall be 1.0 megohm or greater.

(c) Ignition

The charge shall fire when a current of 1.0 ampere is applied to its terminal for a duration of 15 milliseconds. It shall not fire when a current of 0.18 ampere is applied for five minutes.

3.4.1.9 Intertia Switch Device

A minimum weight inertia switch is required as a sequency event initiator in the payload package of the Lunar Capsule mission. The desired weight is .75 oz. and shall not exceed 1.0 oz.

3.4.1.9.1 Functional Performance

(1) The switch must not close when subjected to:

(a) Sustained longitudinal acceleration of 0 to 12 G's in any direction.

(b) Impulsive acceleration, with time integral up to 60 ft/sec., of up to 100 G's.

(2) The switch can close when subjected to 20 G's longitudinal acceleration for more than one second in the perferred orientation.

(3) The switch must close when subjected to 30 G's longitudinal acceleration for more than one second in the perferred orientation.

(4) Following the above, the switch must withstand 3000 G's acceleration in any direction for fifteen milliseconds without shorting to ground.

(5) The switch contacts may re-open, if convenient, (or stay closed) after the acceleration drops to 12 G's.

3.4.1.9.2 Electrical

The contact arrangement shall be normally open with single pole-single throw ability. The switch shall have a current carrying capacity of eight amperes for 100 milliseconds at 28 volts D.C. The leakage current limits across open contacts shall be less than ten microamperes at 28 volts D.C. The switch shall have two terminals or wires that are isolated from ground.

3.4.1.10 Thermal Control Valve

This subsection establishes the requirements on performance, design, fabrication and test for the thermal control valve. The thermal control valve forms a part of the landed-capsule temperature control system.

3.4.1.10.1 General Performance Requirements

(1) The general performance requirements are as follows:

(a) The term 'thermal control valve' as used herein is defined to consist of an assembly of an aluminum cap, aluminum body, inlet, and outlet tube.

(b) One thermal control valve per lunar capsule is required.

(c) This thermal control valve will be placed in a water vapor vent line between internal tank and external vacuum.

(d) Thermal control valve is to be fully operable within the temperature range 32°F to 100°F.

(e) A valve open orifice of 0.060-in. diameter is to be maintained.

(f) Valve closed leak area should be less than 0.10 percent of valve open area.

(g) The valve will support no loads other than that resulting from its mass during acceleration. The valve is not required to function during High G acceleration.

(h) Pressure downstream of valve will be hard vacuum. Upstream pressures will be less than 1 psia.

(i) Life cycle of valve 90 days during that time 93.0 cu. in. water is converted to vapor and exhausted at temperatures noted in Paragraph 6.6.

(j) Valve weight to be minimum consistent with satisfaction of function.

### 3.4.1.11 Position Switch

This subsection defines the performance requirements of a position switch assembly for use in the survival sphere of the lunar capsule.

#### 3.4.1.11.1 Functional Description

The position switch assembly shall consist of two isolated single-pole single-throw switches which are controlled on earth by the attitude of the survival sphere. The switches will be used to keep the sphere electronics turned off during shipping, storage, and handling. The switches will be used to turn on the electronics prior to launch and to keep the electronics on during lunar flight.

3.4.1.11.2 Switching Operations

The switch assembly will be connected within the sphere as shown in Figure 3.7. The operation of the switches can best be defined by describing the necessary switching sequence when the unit is installed within a survival sphere.

3.4.1.11.2.1 Ground Operation Sequence

(1) Assume the switches are "on" when installed within a sphere (Figure 3.8a).

(2) The switches shall turn "off" when the sphere is rotated beyond $100^{\circ} \pm 30^{\circ}$ from vertical. (Figure 3.8b).

(3) The switches shall turn "on" when the sphere is rotated to within $45^{\circ} \pm 30^{\circ}$ from vertical. (Figure 3.8c).

3.4.1.11.2.2 Flight Operation Sequence

The switch assembly within the sphere shall be mounted upright and be turned "on" before launch. The switches must remain "on" during the launch, space transit, and midcourse maneuver conditions.

3.4.1.11.2.3 Electrical Requirements

The electrical requirements are as follows:

(1) The switch contacts shall be isolated from each other and the case by a resistance of not less than 3 megohms.

(2) The switches shall be capable of conducting continuously a current of 0.1 ampere. The switches shall be able to break this current in a 20-volt resistive circuit.

(3) The maximum closed resistance of a switch shall not exceed 0.2 ohms.

(4) The maximum leakage current across open contacts shall not exceed 5 microamperes at 15 volts DC.

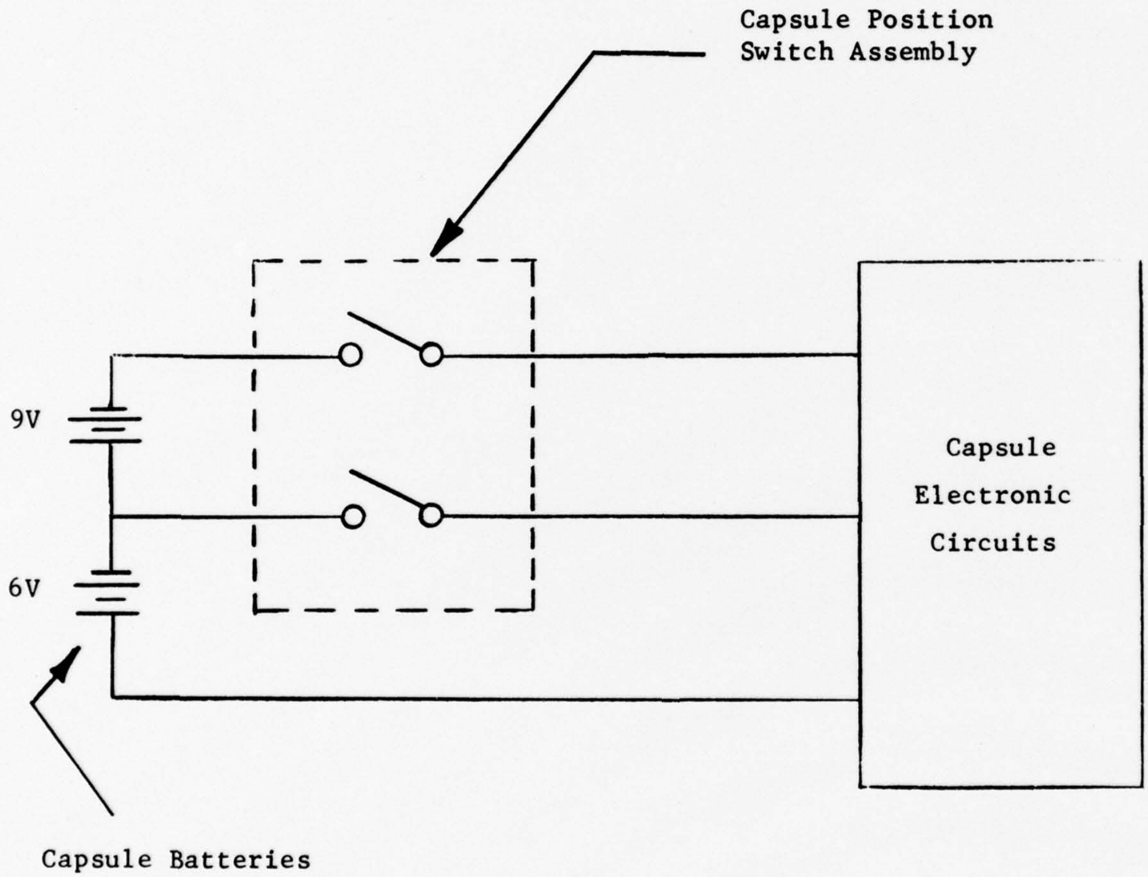


FIGURE 3.7 SWITCH CONNECTIONS

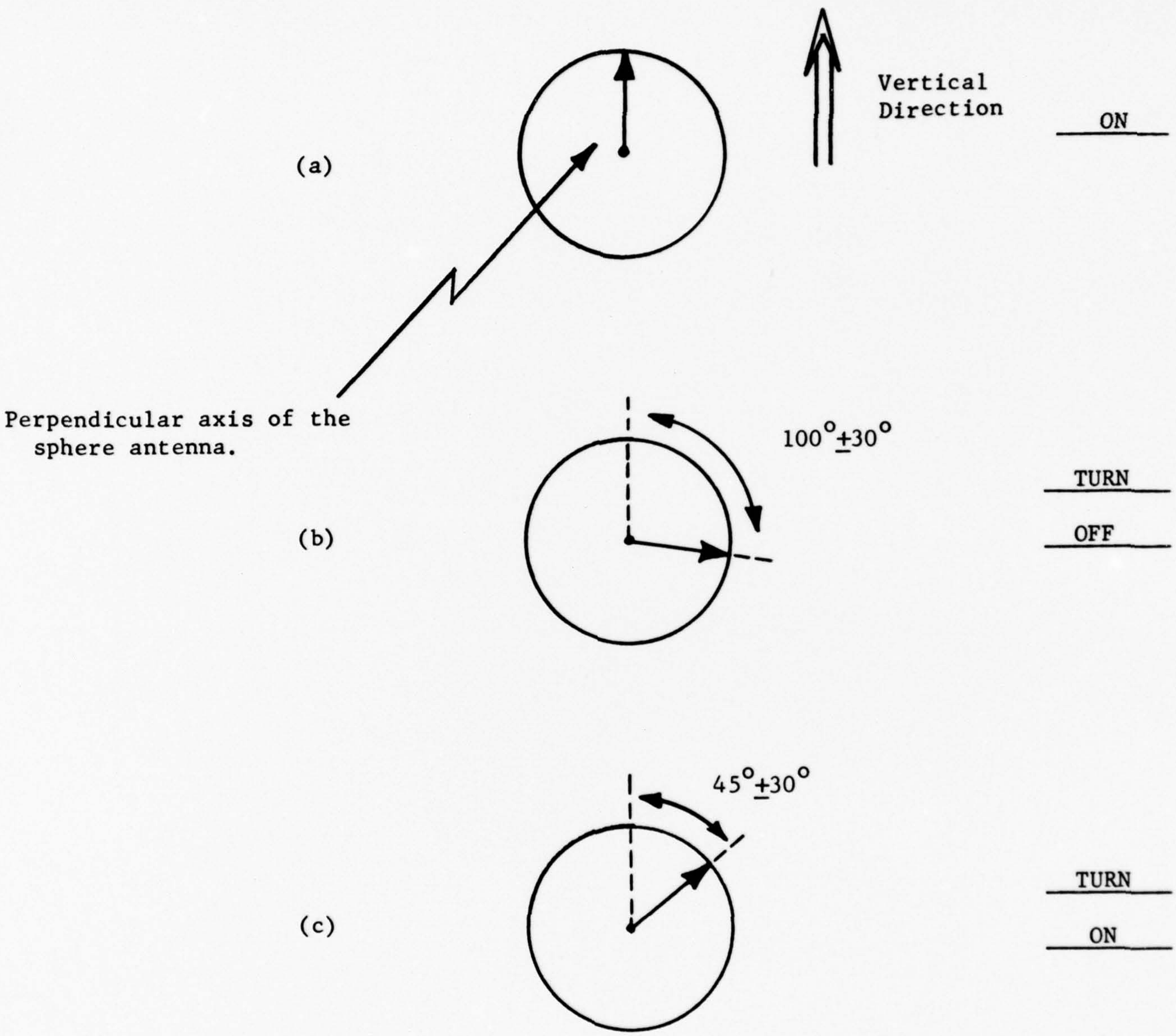


FIGURE 3.8 GROUND OPERATION SEQUENCE

3.4.1.12 Starting Timer

This subsection describes the functional performance requirements for the lunar capsule starting timer, which forms a part of the lunar capsule system.

3.4.1.12.1 Description

The lunar capsule starting timer is to operate during launch and boost phase of the Ranger mission to fire two squib switches, each of which will then shunt the position switches.

3.4.1.12.2 Functional Requirements

The function of the starting timer shall be to fire two squib switches when an inertia switch remains closed for a sufficient length of time.

(1) Time Interval

With the inertia switch closed, the starting timer shall complete its timing cycle in 1.5 $\pm$ 0.5 seconds.

(2) Reset

The starting timer shall reset to its prestart condition within 400 milliseconds after removal of supply voltage.

(3) Supply Voltage

The starting timer shall operate satisfactorily with a supply voltage of 12 to 18.5 volts.

(4) Load

The load consists of two National Northern Type M7 squib switches, each with a 10-ohm resistor in series with it. Maximum current required will be 2.3 amperes for 25 milliseconds. Leakage current through each squib switch bridge before firing shall not exceed 1.0 milliampere.

(5) Self-Disconnect

The starting timer voltage supply shall be carried through the inertia switch and the normally-closed contacts of one of its squib switches so that when the squib switch is fired, even though the inertia switch remains closed, the circuit will be removed from the battery.

### 3.4.2 Impact Limiter

This ection presents the description and functional specification for the lunar capsule impact limiter.

The impact limiter shall be fabricated, for the most part, of a crushable balsa wood transparent to radio frequency energy and surrounding an inner shell containing the floatation fluid and survival sphere. An outer cover shall be provided which is adequate to contain the balsa wood under impact, either normal or oblique, with the lunar surface at velocities up to two hundred (200) feet per second. An interconnect member, adequate to attach the sphere to the retrorocket throughout the flight environment, shall be attached at one end (along the thrust axis) and shall provide a means of separating the landing sphere from the rocket after rocket motor burnout. This separation shall be accomplished by use of a clamp device activated in flight from the power and sequencing assembly. A similar clamp device and attach fitting shall be provided on the opposite end, along the thrust axis, to support the spacecraft omni-antenna. The separation of this omni-antenna from the sphere is activated by the spacecraft. Upon impact, the impact limiter material crushes, absorbing the kinetic energy of impact and limiting the acceleration imparted to the survival sphere to a value permitting survival.

### 3.4.3 Floatation Fluid

This section presents the functional specifications and description of the landing sphere floatation fluid.

The floatation fluid is a mixture of Freon tf and Freon 114B2 with specific gravities of 1.57 and 2.16, respectively, and mixed in the correct proprotion to achieve the same specific gravity as the survival sphere. During impact the fluid supports the ball hydrostatically, thus preventing point loadings of the survival sphere from contacting the inner shell of the impact limiter. After impact, the fluid supports the survival sphere until caging.

## 3.5 Power and Sequencing Assembly

This subsection defines the configuration, the functional performance, and the test qualification requirements for the power and sequencing assembly which forms a part of the lunar capsule system.

### 3.5.1 Description

The power and sequencing assembly shall consist of a battery power source, two pairs of parallel squib switches, three delay timers, an arming G-switch and associated protective circuitry for sequencing the terminal events just prior to lunar impact. (Figure 3.9)

### 3.5.2 Functional Requirements

The functions of the power and sequencing assembly shall be:

(1) An arming G-switch isolates the battery supply from the timer and squib circuits until after launch acceleration exceeds a prescribed minimum value.

(2) Receive an electric signal from the altimeter on the lunar bus indicating a specific lunar altitude has been reached. This signal will actuate the squib switches which provide power to initiate operation of the delay timers.

(3) After a fixed delay, a solid state switch is closed and provides power for firing squibs to ignite the spin motor. (Spin Motor Firing Timer)

(4) After a delay determined by the temperature of the retrorocket propellant, close a solid state switch to fire two retrorocket squibs. (Retrorocket Firing Timer)

(5) After a specified fixed delay from the initial event, close a solid state switch to fire two squibs, which shall cause the retrorocket to separate from the capsule. (Retrorocket Release Timer)

The power and sequencing assembly shall survive during lunar flight to bus separation and shall operate during spin-up and retrorocket firing except for:

(1) The spin motor firing timer need not survive spin-up and retrorocket firing and

(2) The retorrocket firing timer need not survive retrorocket firing.

The assembly need not survive the lunar impact.

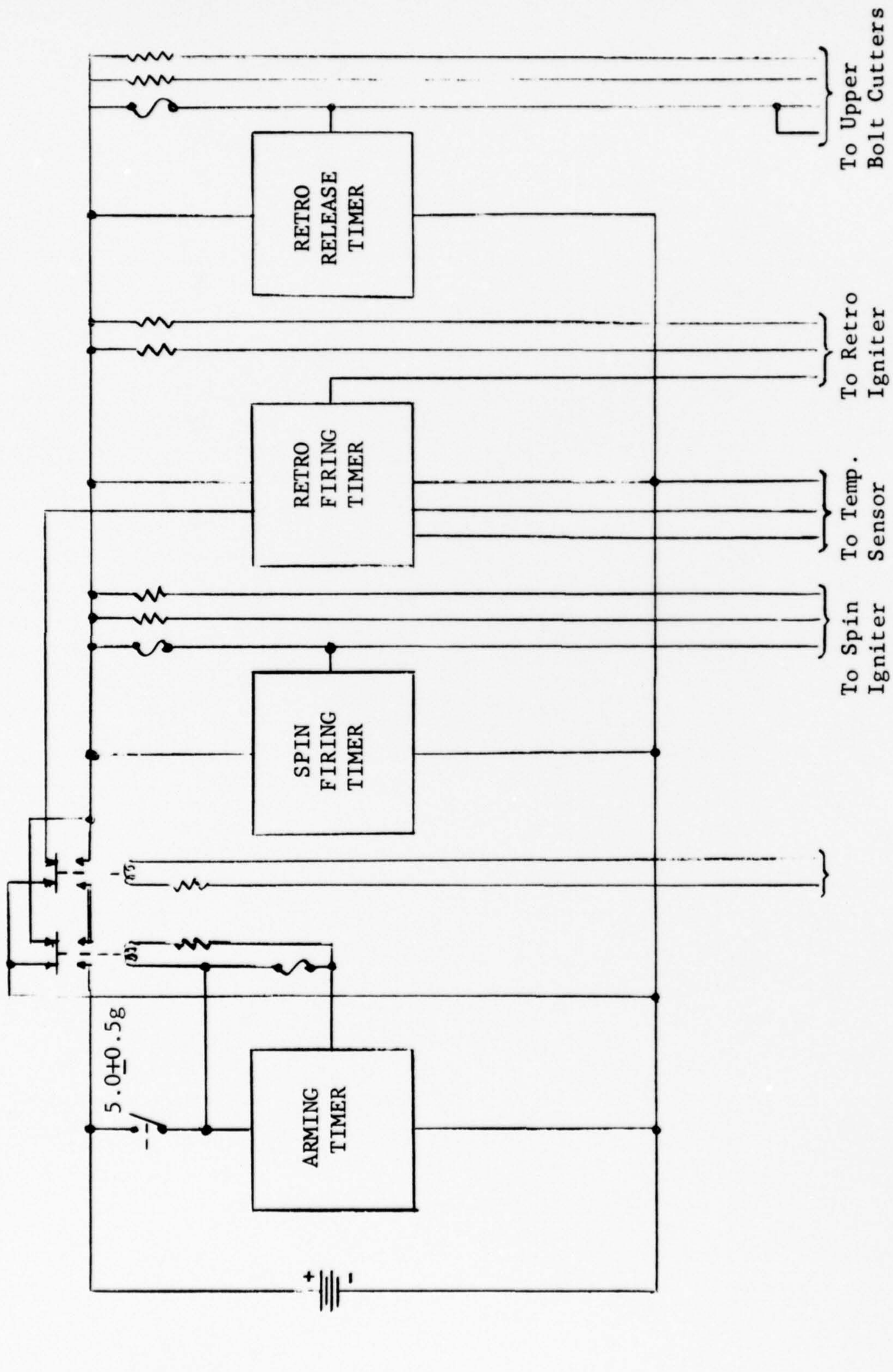


FIGURE 3.9
POWER AND SEQUENCING ASSEMBLY
BLOCK DIAGRAM

The power and sequencing assembly shall be self-contained, including batteries to operate the switching devices and actuate the specified squibs. Figure 3.8 illustrates the general functions of the assembly.

## 3.3 Arming G-Switch

The arming G-switch is employed as a safety measure for prelaunch handling of spin and retro motors. This G-switch and associated integrating circuitry isolates the motor igniter squibs (and associated timing circuitry) from the battery supply until after the positive axial acceleration exceeds 5.0 $\pm$ 0.5 g's for a period of 1.5 $\pm$ 0.5 seconds. After the above acceleration limits have been exceeded, a pair of parallel squib switches connect the battery supply to the imput terminals of the altimeter-controlled switch. The G-switch and associated integrating circuitry shall not be activated by any vibration or shock which has an axial acceleration integral of less than 145 ft/sec.

The G-switch-controlled squib switches shall operate on a current of 0.8 ampere or more existing for not less than 25 milliseconds. Switch contact resistance after actuation shall not be greater than 0.1 ohm. Before actuation, the squib switches shall be protected by a shunt fuse of not more than 0.2 ohm which will open within 0.1 second with a current of 2 amperes, but will not open with a continuous current of 0.9 ampere or less. Each squib switch shall have a series limiting resistor which allows a minimum firing current of 0.8 ampere. These resistors shall provide protection against short circuits within the squib firing bridges by limiting the current to not more than 2.0 amperes per switch for not more than 1.0 second.

## 3.5.4 Altimeter Controlled Squib Switch

The altimeter fuzing signal shall actuate a pair of squib switches to initiate operation of the terminal events of the power and sequencing unit. Each switch shall operate on a current of 0.8 Ampere or more existing for not less than 25 milliseconds. Switch closure must take place within 10 milliseconds after application of the electrical signal. Switch contact shall not be greater than 0.1 ohm. Each squib switch shall have a series limiting resistor which allows a minimum firing current of 0.8 ampere. These resistors shall provide protection against short circuits within the squib switch firing bridge by limiting the current to not more than 2.0 amperes per switch for not more than 1.0 second. Before actuation, the altimeter-controlled squib switches will be afforded shunt protection by the normally closed contacts of the altimeter fuzing relay.

3.5.5 Spin Motor Firing Timer

The spin motor firing timer provides a fixed delay of 130 milliseconds between actuation of the altimeter-controlled squib switch and application of power to spin motor igniter squibs. The time delay error shall not exceed ± 20 milliseconds. The leakage current of the solid state switch which fires the spin motor squibs shall not exceed 1.0 milliampere before gating occurs. The spin motor ignition current shall be a minimum of 1.0 ampere per squib for not more than 100 milliseconds. Prior to actual firing, the spin motor ignition squibs shall be protected by a shunt fuse of not more than 0.2 ohm which will open within 0.1 second with a current of 2 amperes, but which will not open with a continuous current of 0.9 ampere or less.

3.5.6 Retrorocket Firing Timer

The retrorocket firing timer for firing the retrorocket squibs shall have a variable time delay which is determined by sensing the temperature of the retrorocket propellant. The nominal delay shall be 2.0 seconds (from actuation of the altimeter-controlled squib switch) at the predicted mean grain temperature 16°C. The delay shall increase with increasing temperature at the rate of 0.023 sec/°C. Provision shall be made in the design for calibration to a mean grain temperature between 11°C and 21°C and correction coefficient between 0.018 sec/°C and 0.028 sec/°C. Such calibration may be accomplished prior to final assembly. The required linear operating region for the variable delay shall be ± 12.5°C about the predicted mean grain temperature. The maximum allowable time delay error (3 sigma value) of the timer, including temperature sensing network error, shall not exceed ± 0.05 second at any temperature within the required range. The leakage current shall not exceed 1.0 ma before gating occurs.

Firing current for the retrorocket igniter shall be not less than 1.0 ampere per squib for not more than 100 milliseconds. Shunt protection for the retrorocket igniter squibs will be provided by an external short which opens at bus-capsule separation and to which connections shall be provided in the power and sequencing assembly.

3.5.7 Retrorocket Release Timer

The retrorocket release timer fir firing the squibs which separate the retrorocket from the capsule shall have a fixed time delay of 13 seconds after actuation of the altimeter-controlled squib switch. The maximum allowable time delay error shall not exceed ± 0.5 second. The leakage current of the solid state switch which fires the separation squibs shall not exceed 1.0 ma before gating occurs.

Firing current for the separation squibs shall be not less than 2.0 amperes per squib for not more than 100 milliseconds. Prior to actual firing, the separation squibs shall be protected by a shunt fuse of not more than 0.2 ohm which will open within 0.1 second with a current of 2 amperes, but which will not open with a continuous current of 0.9 ampere or less.

### 3.5.8 Power and Sequencing Assembly Battery

This section covers technical requirements of a sealed battery pack for use in the Power and Sequencing Assembly (PSA) of the lunar Capsule Subsystem, as part of the RANGER program.

### 3.5.8.1 Requirements

#### 3.5.8.1.1 General

The battery described herein is to be part of a Power and Sequencing Assembly. During a brief period of operation at the end of an earth-to-moon flight, it is to supply power for the operation of two timing devices as well as the initiation of four sets of ordnance devices. While the battery need not survive the rough lunar landing, it must operate after launch, boost, earth-to-moon transit and during spin and retro-thrust.

#### 3.5.8.1.2 Electrical

The battery is to deliver power for a period not to exceed 16.5 seconds after closure of a switch connecting the battery to its loads. Open-circuit voltage shall be no greater than 28 volts, and terminal voltage shall be no less than 14 volts at any time during operation. The estimated current drain is 30 milliamperer continuous to electronic circuitry plus three current pulses. The first pulse is of three amperer peak, occurs at the beginning of operation, and is of duration no greater than one second. The second pulse begins approximately two seconds after the beginning of operation and is of the same amplitude and duration as the first. The final pulse occurs at the end of the period of operation, is of 4 amperes peak, and is of duration not to exceed 100 milliseconds. It is desirable that the battery be rechargeable for test purposes through at least 5 cycles.

### 3.6 External Wiring System

This section defines the configuration and the functional performance, requirements for the lunar capsule wiring system which forms a part of the Lunar Capsule System.

### 3.6.1 Description

The external wiring system for the lunar capsule is composed of the necessary components (wires, cables, connectors, and junction box) which are used for transmission of electrical signals for controlling terminal events prior to lunar impact. A portion of the system is also used to provide telemetry information to the bus telemetry system.

The cables shall be composed of multiconductor shielded wire. The number of conductors in the various cables is depended upon the requirement of the particular cable. The inner conductors are seven-strand, size 26 copper wire with Teflon low voltage insulation. The shield is 30-gage silver plated copper mesh, with a Teflon wrap where required. Wiring which does not require shielding will be the same type of wire as used for the inner conductors in the shielded cable.

### 3.6.2 Performance

The external wiring system shall perform the function of interconnecting the following subassemblies: power and sequencing assembly, squib loads, junction box, bus, and altimeter. The functions which are to be performed through these interconnections are as follows:

#### 3.6.2.1 Control

(1) Altimeter Antenna Erection

(2) Altimeter Turn-On

(3) Bus Separation

(4) Power and Sequencer Assembly Turn-on

(5) Spin Motor Ignition

(6) Retrorocket Motor Ignition

(7) Retrorocket Motor Separation

#### 3.6.2.2 Telemetry

(1) Altimeter Antenna Erected

(2) Altimeter Signal Strength

## 3.7 Retromotor

This section shall serve to define the functional requirements and the configuration for the lunar capsule system retrorocket motor. The retrorocket motor is to be developed and produced by a contractor other than Aeronutronic in accordance with Ford Motor Company Subcontract SC-05850 which is hereby made a part of this specification.

### 3.7.1 Description

The retrorocket motor is required to serve as ; the prime mover of a vehicle weighing approximately 290 lb. The motor with suitable attach devices shall serve as a primary structural member of the lunar capsule system in such a way so that seperation from the space craft structure may occur and separation of the accelerated payload may also occur. The lunar capsule system shall include means of providing a retational velocity of approximately 300 rpm as a means of stabilizing the retrorocket/payload velocity vector. The retro-rocket shall also serve as the structural support for the spin system so that subsequent jettisoning of the spin system can occur.

### 3.7.2 Requirements

The retrorocket motor shall utilize solid propellants, provide a thrust level of approximately 5500 lb have a nominal burning time of approximately 10 sec, support a payload weight of approximately 92 lb mounted along the thrust axis on the head end of the motor, accelerate the payload in gravity-free vacuum space to approximately 8816 feet per second, meet the environmental requirements of Specification LCS-003, and be of nearly symmetrical configuration so that a condition of dynamic balance may be achieved.

## 3.8 Support, Spin and Separation System

This section shall serve to define the requirements for support of the retrorocket motor and lunar landing sphere assemblies and the requirements for spin and separation of those assemblies from the Ranger spacecraft. It shall specify the functional performance requirements of these subsystems.

### 3.8.1 Description

The lunar capsule support and separation assembly is required to attach the lunar capsule to the bus from which the capsule separates for the terminal maneuver. This assembly serves as the primary structural member for transmitting loads from the bus to the capsule. The separation device will detach the capsule from the bus upon receipt of a signal from the altimeter. The support structure is, in fact, fabricated in two portions: one portion attaches to and becomes a part of the retrorocket motor case and one portion which bolts to the spacecraft bus and, in flight, remains with the bus after separation. Separation is accomplished by releasing both or either one of two bolts which attach a marmon clamp across the separation joint.

To stabilize the retrorocket motor velocity vector to the attitude (provided by the spacecraft bus) required for landing, a spin motor is attached to the retrorocket motor/lunar landing sphere assembly and causes a rotational velocity of 285 rpm. The spin motor is mounted so that the spent motor case can be jettisoned shortly after retrorocket motor burnout. The initiation circuits for spin-up, separation, and the retrorocket motor shall be provided with arming circuits (either on the spacecraft or the lunar capsule assembly) so that inadvertent operation of these systems cannot occur.

### 3.8.2 Performance

The forward end of the support assembly, which lies forward of the separation plane, is attached to the rocket motor case continuously about the circumference of the case. The misalignment of the bus longitudinal axis (as defined by the bolt-circle of bus) and the thrust axis of the rocket motor shall not exceed 0.1 degree. Exposure to the environments of Specification LCS-001 shall not produce an additional misalignment which exceeds 0.1 degree. The separation process shall not produce a capsule angular rate which exceeds 0.04 radian/sec (this is equivalent to 10 mrad/sec 1 about each of two orthagonal axes).

### 3.8.3 Spin Motor

This subsection defines the configuration and the functional performance requirements, for the solid propellant spin motor which forms a part of the lunar capsule system.

#### 3.8.3.1 Description

The spin motor assembly is required to develop and deliver to the retromotor, lunar landing sphere assembly a torque to accelerate this assembly to a spin rate of approximately 285 rpm. The spin axis is to be coincident with the retrorocket thrust axis.

The spin motor unit will consist of one firing chamber with a solid propellant grain cast in place and an insertable igniter plus a manifold assembly to which is attached three exhaust nozzles. The nozzles will be canted so as to deliver forward impulse to the assembly as well as torque. Torque is delivered to the retromotor/lunar landing sphere assembly through the retromotor nozzle by slots cut into that nozzle through which the spin motor nozzles protrude.

3.8.3.2 Performance

The spin unit ignition is sequenced by a signal from the altimeter. The unit must ignite promptly and reliably upon receipt of the signal. The desired minimum probability of ignition is to be 0.995 at the 95% confidence level. Nominal performance requirements in the spin motor are as follows:

| | | |
|---|---|---|
| (1) | Spin torque impulse: | Approximately 70 ft-lb-sec |
| (2) | Thrust impulse: | Approximately 20 lb-sec |
| (3) | Duration: | Maximum of 1 sec |
| (4) | Alignment: | Motor design and alignment must be such that torque vector misalignment with respect to rocket centerline does not exceed 0.006 radians |
| (5) | Direction of rotation: | Clockwise, looking up the nozzle of retrorocket. |

3.8.3.3 Configuration and Physical Characteristics

The total weight of the spin motor system, in the flight readiness condition, complete with igniter, igniter lead, and attach devices shall not exceed 2.4 lb.

3.9 Thermal Radiation Shield

This section shall serve to define the configuration the functional requirements for the lunar capsule thermal radiation shield. The termal radiation sheild is intended primarily to supply shielding of the retrorocket motor during the lunar transit trajectory.

3.9.1 Description

The thermal radiation shield is to consist of a skirt-like assembly of low emissivity membranes surrounding the retrorocket and retrorocket support structure in such manner as to restrict the flow of heat, as thermal radiation, from the retrorocket assembly to space.

The thermal radiation shield is to be so disposed, at the time of release of the lunar capsule, as not to interfere with the release functions.

3.9.2 Performance

Effictiveness of the shield system is to be such that the heat loss rate of the retrorocket, when exposed to dark space, will be less than 6.8 Btu/hr.

3.10 Lunar Capsule Altimeter

3.10.1 Requirements

3.10.1.1 General

The altimeter desired is of the fuze type, to operate during the last protion of an earth-to-moon flight. It is to start operation upon receipt of a command signal and deliver a single output signal (designated "fuzing signal") at a prescribed altitude (designated "fuzing altitude"). The altimeter will not be separated from the bus and hence will be destroyed upon impact with the moon. It must, however, operate satisfactorily after experiencing launch, boost, and earth-to-moon transit conditions.

3.10.1.2 Performance

3.10.1.2.1 Starting

The altimeter shall start within one second after receiving a start signal voltage step of 18 to 30 volts. Seller shall inform ADF of the electrical characteristics of the altimeter start switch or circuit. Access to the activating device will be through two external terminals, both of which are isolated from chassis ground.

3.10.1.2.2 Frequency

No frequency of operation is specified, but the altimeter shall not cause any interference with other r-f equipment on the bus operating in the vicinity of 890 and 960 mc.

3.10.1.2.3 Measurement and Modulation

It is not the intent of this specification to define the modulation and measurement methods used to meet the performance requirements included in this specification.

3.10.1.2.4 Starting Altitude

The altimeter is to be started at a time before bus impact such that warm-up will be complete at an altitude of not less than 180,000 feet. Taking trajectory-time errors into account, the maximum altitude at which warm-up can be complete is 650,000 feet. Ranging shall not occur more than 15 seconds before completion of warm-up.

3.10.1.2.5 Descent Rate

During the time of operation of the altimeter, the average approach velocity to the moon will be between 8400 and 9400 feet per second.

3.10.1.2.6 Fuzing Altitude

The precise fuzing altitude is not yet known but is expected to be between 61,800 and 81,800 feet. The altimeter shall contain provision for laboratory variation of fuzing altitude over a range of ± 5 per cent.

3.10.1.2.7 Accuracy

The altimeter shall provide a range resolution accuracy such that the cumulative errors from all effects (range measurement, antenna boresighting errors, switching time, etc) shall result in a three-sigma measurement error of no greater than 500 feet at the fuzing altitude.

3.10.1.2.8 Warm-Up Time

Warm-up time shall be such as to minimize over-all altimeter weight.

3.10.1.2.9 Operating Time

The altimeter shall operate and be capable of delivering a fuzing signal for a period of not less than 65 seconds following warm-up.

3.10.1.2.10 Output Signal

At the fuzing altitude the altimeter shall supply the fuzing signal to the bus by connecting the return line of the two wire fizing line from a safe position available to the external circuit, to the 28 volt fuzing line thus completing the circuit. The fuzing connection thus made shall be capable of handling 10 amperes for one second.

3.10.1.2.11 Power Supply

The power supply, including batteries, converters, and regulators as necessary for altimeter operation shall be an integral part of the altimeter system.

3.10.1.2.12 Antenna

Development of the antenna is part of the altimeter development task. The antenna is to be erected at the beginning of spacecraft terminal maneuver (one hour before landing). The erection mechanism will attach to the bus structure. When the antenna is in the operating position, it will project over the side of the bus, and its beam axis must be parallel to the roll axis of the spacecraft. Details of the antenna erection system will be coordinated with ADF, who will maintain responsibility for the erection mechanism. The antenna beam axis shall be within $0.25^{o}$ of normal to a reference plane defined by three points on the antenna structure. Mounting of altimeter electronics on the antenna is permissible.

3.10.1.2.13 Testability

Provision shall be made for operating tests of the altimeter by application of external power and making a minimum of measurements. Self-test features and indicating devices are not desired, and special test equipment required is to be described to ADF. Test points shall be made available within the altimeter for monitoring of the video output.

3.10.1.2.14 Sensitivity

Sensitivity of the altimeter shall be that required by paragraphs shall be that required by the requirements of the preceding 13 paragraphs.

3.10.1.2.15 False Fuzing

The altimeter shall have a probability of false fuzing, from maximum starting altitude, of not greater than one percent.

3.10.2 Altimeter Battery

This subsection covers the technical requirements of a sealed battery pack for use in the Lunar Capsule altimeter.

3.10.2.1 Requirements

3.10.2.1.1 General

The battery desired is to be a sealed unit for use in a pulse-radar altimeter being developed by Wiley Electronics Company under LCS-300W. It is to provide power for a period of time up to two minutes in the last portion of an earth-to-moon flight. The battery will be destroyed upon impact with the moon, but it must perform satisfactorily after experiencing launch, boost, and earth-to-moon transit conditions.

3.10.2.1.2 Electrical

The battery is to operate for a period of two minutes. During this operating time, terminal voltage shall be not less than six nor more than nine volts. Current drain during operation is expected to be 4.5 amperes nominally and five amperes maximum. It is desirable that the battery be rechargeable for test purposes through at least 5 cycles.

3.10.3.1.3 Physical

The battery shall be of such a size and geometric configuration as to fit within a right parallelepiped 1-5/8" H x 1-1/8" W x 2-3/4" L, including terminal projections. Weight is to be the minimum consistent with the expected environmental conditions. The case is to be hermetically sealed. Electrical terminals shall be 4-40 x 3/16" studs, marked to identify polarity.

3.10.3.1.4 Environmental

Environmental conditions will be as described in LCS-001A, except that temperature limits are $+10^{o}C$ to $+70^{o}C$, and neither solar radiation nor radiation to $0^{o}R$ will be encountered.

3.10.3.1.5 Sterilization

The battery shall be rendered biologically sterile in accordance with LCS-005.

3.10.3.1.6 Shelf Life

The battery shall be capable of performance as specified in 3.2 after storage between $0^{o}C$ and $50^{o}C$ for 60 days. Re-charging before use is a permissible condition for meeting this requirement.

## 3.11 Altimeter Support and Erection System

This section shall serve to define the configuration and the functional requirements for the altimeter.

### 3.11.1 Description

The purpose of this structure is to support the altimeter antenna during all phases of the flight. The assembly is attached by four bolts at the bus interface. During launch and transit phases the antenna will be in a nested position so that the antenna assembly is within the shroud clearance envelope. At some time between the beginning of the bus terminal maneuver and altimeter turn-on, the antenna is erected so that its primary axis is parallel to the roll axis of the bus and its forward direction is toward the lunar surface. The signal for erection will be supplied by the bus. Erection shall be accomplished by spring torque working across the hinge joint. Provision shall be made to provide for carrying electrical signals to the altimeter in both the stowed and erected positions. Provision shall also be made to latch the deployed antenna in the erected position.

### 3.11.2 Performance

The altimeter support structure shall be capable of supporting the altimeter, with antenna for all flight loads including those of vibration and shock. Erection and release shall have a minimum desired reliability of 0.995 at the 95% confidence level. The electromechanical release device shall be redundant in order to meet this requirement. Deployment shall be accomplished in $5.0 \pm 0.5$ secs from the instant of release in the stowed position.

Alignment marks are to be provided on the antenna which define a plane at right angles to the electrical axis of the antenna within $0.25^{\circ}$. The plane represented by these marks shall be aligned with the roll axis of the bus within 0.35 degree on assembly. Exposure to the environments given above shall not cause these axes to be misaligned by more than an additional 0.35 degree. The effects of latch tolerances are included in the above misalignments. All disengage and engage latches must operate promptly and reliably. The disengagement signal will be provided by the bus.

### 3.11.3 Physical

The support and erection mechanism must hold the antenna (in nested position) so that this entire assembly is contained within the envelope shown in ADF Drawing No. 803211C and so that it does not interfere with the capsule and its support structure and the omni-antenna support assembly. The guide rails shall not interfere with the deployment motion and shall prevent contact between the Lockheed Shroud and the altimeter antenna at shroud separation. In the erected position, this assembly shall not interfere with, or the antenna's field of view be restricted by, any of the bus equipment. Attachment points consistent with antenna support points shall be provided. The total weight of this support and erection assembly shall be no greater than 2.72 pounds.

In the fully erected position, the support and erection assembly shall close a switch, which is considered a portion of the support and erection assembly to provide an electrical indication of the completion of erection. Resistance between switch terminals shall be not less than 500,000 ohms in the open condition and no more than 1 ohm in the closed condition.

END

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