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APOLLO EXPERIENCE REPORT - LUNAR MODULE DISPLAY AND CONTROL SUBSYSTEM

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16. Abstract

The lunar module display and control subsystem equipment is described, with emphasis on major problems and their solutions. Included in the discussion of each item is a description of what the item does and how the item is constructed. The development, hardware history, and testing for each item are also presented.

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LUNAR MODULE DISPLAY AND CONTROL SUBSYSTEM

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SUMMARY

The lunar module display and control subsystem provides interface between the astronauts and the spacecraft. During a mission, this subsystem enables the crew to accomplish the following:

1. Manual subsystem operation under normal or contingency conditions

2. Safe shutdown of equipment

3. Monitoring of conditions of subsystems and reserves of propellant and energy

4. Recognition of malfunctions and incipient hazards

5. Adjustment or selection of alternate subsystem operating modes

This paper describes the equipment required to accomplish these functions successfully. Considerations regarding the philosophy, general requirements, configuration, and selection of the types of displays and controls are not discussed in this report.

INTRODUCTION

The lunar module (LM) display and control subsystem consists of a variety of equipment. All controls are of two basic types: (1) switches and (2) variable controls. Displays are of four basic types: (1) moving pointer, (2) fixed pointer, (3) numeric, and (4) status indicators. Included within the display subsystem is interior and exterior vehicle lighting. Interior lighting, used to illuminate the cabin, consists of floodlights, electroluminescent panels, luminescent switch tips, and utility lights. Exterior lighting, consisting of a docking target, docking lights, and a tracking light, is used by the astronaut to visually guide and orient the spacecraft to achieve a successful docking.

The equipment in this subsystem is generally located on panels within the spacecraft. Interconnection among the various equipment is accomplished by cables and "black boxes." These black boxes include control assemblies, and time-delay relays (which are also part of this subsystem), and other relay, resistor, and diode assemblies.

The subsystem equipment is discussed, with emphasis on major problems and their solutions. Included in the discussion of each item is a description of what the item does and how the item is constructed.

DEVELOPMENT

Specifications were developed by the prime contractor to meet the basic system and specific subsystem requirements. The specifications were submitted to industry for competitive bidding and proposals. A manufacturer for each item was selected on the basis of the manufacturer's response to the specifications. To ascertain that each selected component would meet design requirements and to ensure adequate system operation, a test program was established. The test program for each component included the following.

1. Design feasibility tests — Tests performed on a breadboard model to establish confidence in the electrical concepts and to serve as an aid in selecting components.

2. Design verification tests — Tests performed on a working model to evaluate the design by comparing actual performance with design predictions, while determining the effect of varying environmental conditions.

3. Qualification tests — Tests performed on actual flight-type hardware to ensure its ability to meet design and mission requirements under varying environmental conditions. Testing is generally performed on two separate items, one of which is used for design limit tests and the other for endurance tests.

4. Acceptance tests — Tests performed on each flight item to ensure its ability to meet technical requirements. This testing includes vibration and temperature tests, in addition to normal functional tests. Successful completion of this test indicates that the item is ready for delivery.

5. Preinstallation tests — Tests are performed on a completed subassembly, usually to the panel level, just before spacecraft installation. This testing is performed to ascertain that the components are installed properly and were not damaged during subassembly fabrication.

The subsystem test program, in addition to various system-level tests, provided the components required for an operational LM spacecraft. In some cases, additional qualification testing (Δ qual) was required to meet changing or new requirements. Generally each component required at least one Δ qual because of changing vibration acceptance test requirements.

HARDWARE HISTORY

The LM display and control subsystem consists of a variety of components. These components and the quantity used on each vehicle are presented in table I. The LM cabin control and display panels are shown in figure 1.

TABLE I. - LUNAR MODULE DISPLAY AND CONTROL COMPONENTS

Components	Specification number	Quantity per vehicle
Controls		
Switches:		
Toggle	LSP-350-802	140
Rotary	LSP-350-803	10
Pushbutton	LSP-350-808	8
Variable:		
Potentiometer	LSP-350-604	14
Synchro	LSP-350-606	2
Display		
Moving pointer:		
Meter	LSP-350-801	18
Cross pointer	LSP-350-305	2
Thrust to weight	LSP-350-306	1
Fixed pointer:		
Range indicator	LSP-350-307	1
Numeric:		
Helium pressure/temperature	LSP-350-201	1
Propellant quantity	LSP-350-401	1
Mission timer	LSP-355-316	1
Event timer	LSP-355-315	1
Status:		
Flag	LSP-350-804	32
Component caution	LSP-350-806	20
Caution and warning display	LSP-350-809	4
Interior light		
Floodlights	LSP-340-410	20
Utility light	LSP-340-412	1
Electroluminescent panels	LSP-340-402	--
Switch tips	LSP-340-404	140
Exterior light		
Self-luminous exterior	LSP-340-406	67
Docking light	LSP-340-412	5
Tracking light	LSP-340-411	1
Miscellaneous items		
Time delay	LSP-350-202	2
Control assembly	LSP-350-308	3
Overlays	LSP-340-407	--

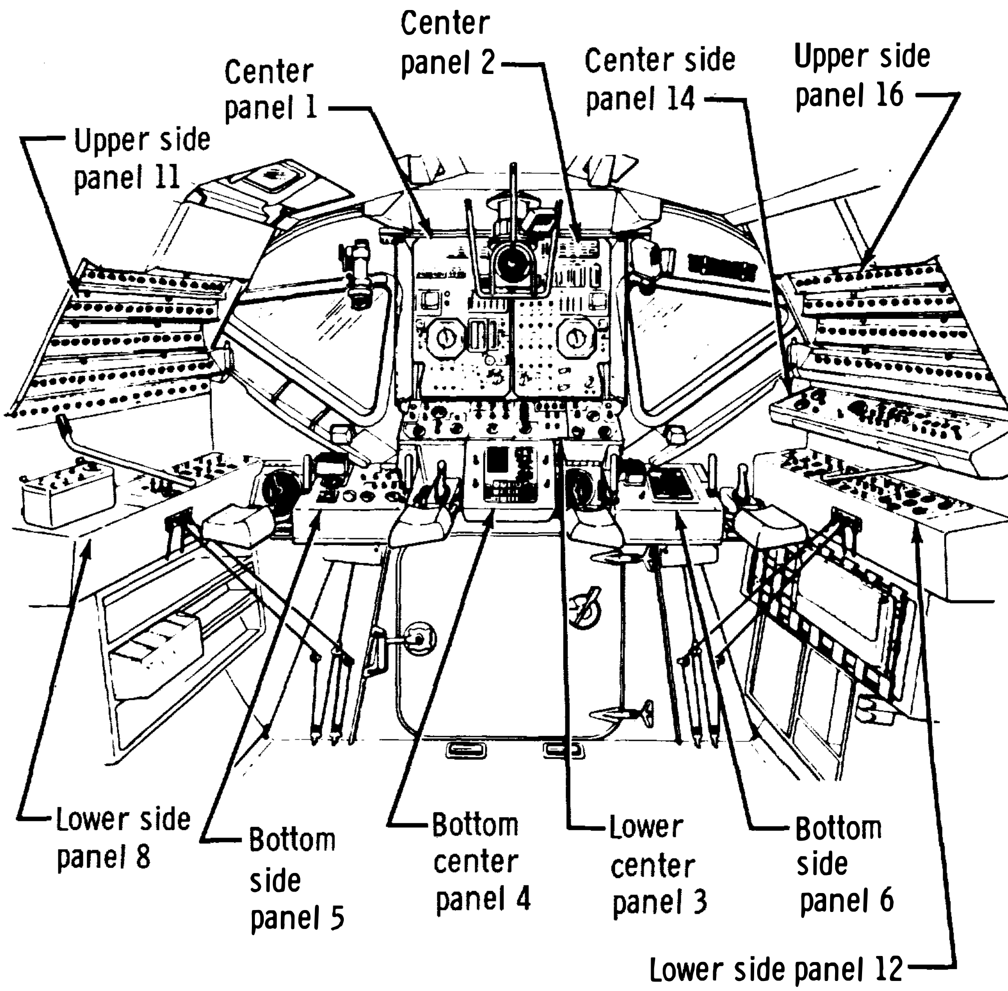


Figure 1. - Lunar module cabin control display panels.

Switches

Toggle switches. - Toggle switches are used to manually control the operation of the spacecraft. The LM contains approximately 140 toggle switches in 33 configurations. These configurations were selected to simplify system function and electrical operation. The types of toggle switches include momentary, maintain, and lock actuations, in either double- or triple-throw actions. Electrically, the configurations vary in size from one to eight poles.

Toggle switches are of two basic types, AT and LS. The AT-type switch contains hermetically sealed modules where electrical switching takes place and an environmentally sealed housing where mechanical actuation takes place. The LS-type switch has electrical and mechanical actuation taking place within a completely hermetically sealed case. Sealing is accomplished by using glass headers, a bellows, and a heliarc-welded case.

The toggle switch program was marked by numerous failures, each of a potentially serious nature. These failures occurred throughout the program from initial design through qualification testing, during manufacturing, and during vehicle testing. Major problems included the following.

1. Contact resistance
2. Lifetime limitations
3. Pressure sensitivity
4. Weld plate
5. Shorting
6. Hangup
7. Inverted contact button
8. Solder balls
9. Short springs

Solutions to these problems have been expensive in both time and money. In reviewing the program, it becomes obvious that many problems could have been averted. Tighter manufacturing controls should have been initiated early in the program by the switch manufacturer. Most of the corrective actions required modifications to normal switch handling or manufacturing procedures pertaining to small details. Examples of some of these actions are as follows.

1. Contact resistance has been a problem since the inception of the program. Improvements have been made by storing the contacts in a dry nitrogen atmosphere, mechanically cleaning the parts, and storing complete assemblies in a clean, dry area.

2. Shorting occurred in momentary switches when the contact arm whipped and made contact with the toggle return spring. The contact arm shaping was not properly controlled; therefore, a tolerance buildup could have caused this problem. Revised processes and in-process checks eliminated this problem.

3. Inverted contact buttons, which reduce switch lifetime, were found on numerous switches. These buttons were assembled incorrectly by a subcontractor. Initially, the switch manufacturer did not inspect the subassemblies 100 percent before installing them into switches. Revised inspection procedures corrected this problem.

4. Short springs were found in momentary switches. These springs fell out of place and caused internal shorts. Analysis indicated that overheating of the spring under compression during case welding was causing the problem. An improved welding technique and proper fixture cooling corrected this problem.

5. The AT-type switches were found to be pressure sensitive. These switches would not release under ambient pressure greater than 1 atmosphere. The cause was traced to the use of modules with low release force. The force degraded during switch cycling to the point of failure. The problem was corrected by increasing the module release force and reducing the module actuating force. In addition to this corrective action, all critical AT-type switches were replaced with LS-type switches.

In all the corrective actions taken, X-ray of the switches has become an important and valuable tool. The impact of the problems would have been greatly reduced if adequate X-rays had been available. At present, all switches are X-rayed, and multiple views are taken of momentary switches.

Rotary switches. - Rotary switches are used to manually select various spacecraft system operating parameters for display purposes. Each spacecraft contains approximately 10 rotary switches which, depending on use, vary in the number of decks, poles, and positions. The basic switch mechanism is solder sealed within a cylindrical housing with a bellows assembly at one end and a header at the other. The basic mechanism consists of a stator and a rotor. Rotation of a cylindrical rotor shorting-rod shorts out the stator element, providing the switching. Positive detent is provided by a spring-held roller which rides in a star-shaped detent gear.

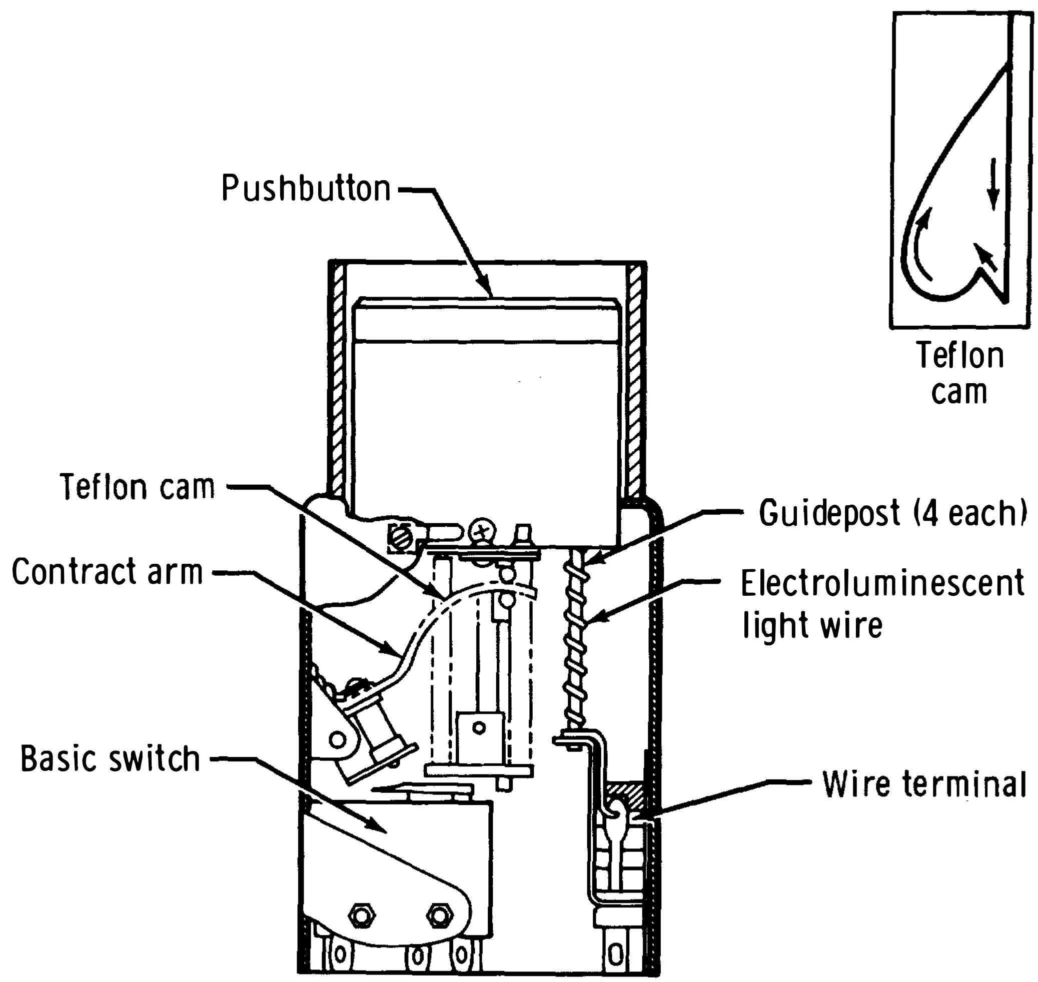
Design and testing of the rotary switch presented a few minor problems. Early in the design phase, problems were encountered with vibration on five deck switches and with the ability to meet lifetime requirements on high-current switches. The vibration problem was resolved by adding structural support to the switch. The lifetime problem was eliminated by reducing the lifetime requirements from 25 000 to 10 000 cycles. Although this reduction did not solve the problem, the switches worked well in the spacecraft because the 10 000-cycle lifetime was in excess of normal switch usage.

The use of these switches in the LM has been successful and trouble-free. There has never been a vehicle testing or confirmed flight problem. During the flight of Apollo 10 (LM-4), the probable cause of a nonfunctioning meter was determined by analysis to be a broken wire within a rotary switch. This type of failure has not been experienced in any phase of the switch program and seems to be an unlikely failure mode.

Pushbutton switches. - Pushbutton switches are used to manually override or alter automatic system operation in normal or emergency situations. Each spacecraft contains eight of these switches to perform the following six functions.

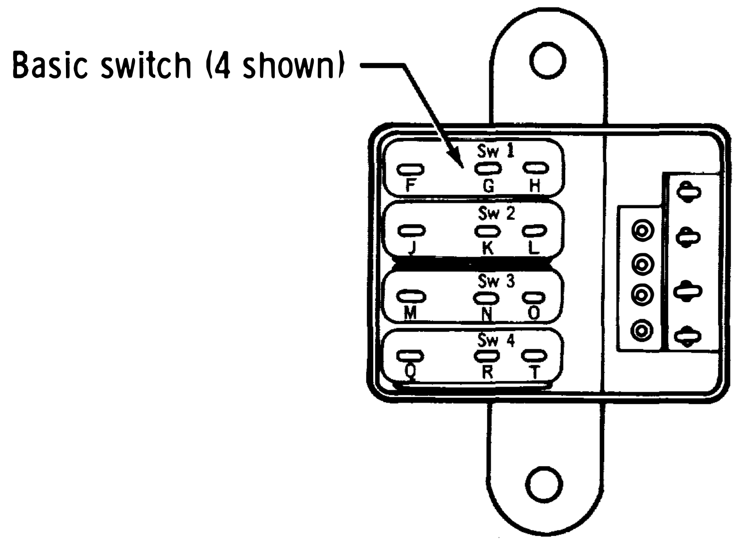
1. Abort
2. Abort stage
3. Master alarm
4. Engine stop
5. Engine start
6. X-translation

The switch mechanism actuates a lever to energize individual hermetically sealed snap-switch elements. The mechanism may also actuate a cam follower in a stationary cam to provide the actuation position required by the switch. Lighting is provided within the pushbutton to indicate status or action information to the astronauts. Switches were designed to perform specific electrical functions and therefore are not interchangeable. Each switch varies in the number of snap-switch elements, the amount of lighting, cam action, and shape. A typical pushbutton switch is presented in figure 2, and a comparison of the types of switches is shown in table II.



(a) Cutaway view.

Figure 2. - Schematic diagram of a typical pushbutton switch.



(b) Bottom view.

Figure 2. - Concluded.

TABLE II. - PUSHBUTTON SWITCH COMPARISON

Switch	Light	Number of basic switches	Movement	Function
115	EL[a] and I[b]	4	Lock and button return	Stop
116	EL	4	Lock	Abort stage
117	EL and I	2	Momentary	Start
118	I	2	Momentary	Master alarm
119	EL	4	Lock	Abort
120	EL	4	Momentary	X-translation

[a]Electroluminescent lamp.
[b]Incandescent lamp.

The pushbutton switches have been marked by numerous failures, including the following.

1. Problems were caused by incorrect manufacturing techniques and procedures easily corrected by revisions.

2. Contact bounce occurred during random vibration testing; corrective action included changing the size of a compression spring to increase actuating force from 13 to 20 ounces to 15 to 30 ounces and reduction of the vibration level. Vibration levels were reduced on the basis of more realistic levels determined during LM test article 3 testing.

3. Lighting problems normally required changing of incandescent lamps or the correction of color and brightness of electroluminescent lamps by traditional techniques.

In addition to these minor problem areas, more significant failures were experienced: contamination and stop-switch latching.

Contamination found in many of the switches prohibited proper switch actuation. Contamination included the following.

1. Potting in the housing

2. Cement particles

3. A rubber stamp wedged into a switch

4. Dirt

Contamination can enter through various openings in the switch, including the unsealed rear of the switch. Some problems were associated with attempts to seal the rear of the switch with potting. Corrective action to preclude contamination problems included the following.

1. The X-ray techniques were developed to detect metallic and nonmetallic particles.

2. A new design rubber (Silastic) seal was added at the rear of the switch. This rubber seal provided better adhesion to the cement, therefore preventing a flow of the potting which could cause contamination.

3. Switches were fitted with dust covers before and after installation into panels. These covers remain on the switch until just before launch.

4. Layout changes were also made to check button-to-case alinement, and spacing was added to preclude binding.

Since the initiation of these procedures, contamination problems have been minimized during panel fabrication and eliminated once the switches are installed.

During vehicle testing, it was determined that when the engine stop switches were depressed, they could be inadvertently reset, which caused the engine to start again. To preclude this action, an external latch mechanism was added to this switch. The external latch mechanism was designed so that two actions were required to reset the stop switch, thereby eliminating the possibility of an engine restart. The design of this latch mechanism created many other problems before the latch could be properly installed in the vehicle. At present, this mechanism is working properly and has not caused any problems.

Pushbutton switches have performed well in flight. No failures have been attributed to these switches. The only recommendation as a result of this review is to make certain, during initial system design, that electronic circuits perform all complicated system switching and that mechanical switches be used to actuate this switching.

Variable Controls

Potentiometers. - Potentiometers are used within the spacecraft to provide continuous level adjustment, generally as volume controls. Each LM contains 14 of these devices in three basic configurations. The configurations include linear taper, nonlinear taper, clockwise rotation, and nonlinear taper counterclockwise rotation. The potentiometers are contained in a hermetically sealed housing that uses a bellows assembly to allow shaft rotation. Mechanical stops limit rotation to 300°. Basic resistance is provided by a carbon element that can vary between 2.5 and 25 000 ohms. Development, including qualification and acceptance testing, was satisfactory in that few failures occurred. Failures that did occur included leakage and dielectric absorption, which were easily detected and corrected.

Use of these potentiometers in the spacecraft presented numerous problems. During panel acceptance testing, many potentiometers were found to have been open or discontinuous. Analysis revealed that this problem was caused by inadequate contact pressure due to tolerance buildup between contact and contact cavity. Acceptance testing did not reveal this lack of buildup, because the potentiometers were checked only at four points. Corrective action included selective fitting of the contact into the contact housing and a screening test. The screening test consisted of monitoring the resistance over the dynamic range of the potentiometer. These potentiometers used a simple carbon element to provide resistance changes. Attempts have been made to use these devices as precision potentiometers, without success. Using these potentiometers as volume controls presented no problems during vehicle testing or flight.

Synchro. - The synchro is a rotary control used to actuate a synchro transmitter. This transmitter generates position commands to an antenna drive assembly in either the pitch or the yaw axis. The synchro is housed in a hermetically sealed case and uses a bellows assembly to provide continuous shaft rotation. Construction is that of a standard synchro with a single-phase rotor and a three-phase, delta-wound stator with carbon brushes. Development of this device was successful, resulting in only one problem. During qualification testing, the rotating (slip) torque increased after a few cycles of operation. Analysis indicated that this increase in torque was caused by high-friction areas developing between brake surfaces. This problem was corrected by the following.

1. Lapping and smoothing of the contact surfaces of the brake

2. Reduction of the spring constant of the spring that provides force between the brake contact surfaces

3. Installation of a glass-filled Teflon bushing on the output shaft

The effect of this corrective action was demonstrated by rotating a test model over 100 000 times, with negligible increase in torque and no appreciable wear.

Synchros have operated successfully on all vehicles during both testing and flight. However, a problem did occur during vehicle testing. Concern was expressed regarding the setting accuracy of the synchro. Normally, synchros of this type have accuracies in the $\pm 0.5^\circ$ range, while the accuracy for this particular synchro was $\pm 2.0^\circ$. Investigation indicated that, while the basic accuracy of the synchro is approximately $\pm 0.5^\circ$, hermetically sealing the device by using a bellows assembly decreased this accuracy to the specified $\pm 2.0^\circ$. In designing future devices of this type, consideration should be given to the need for hermetic sealing, because sealing causes increased cost and weight and reduces accuracy.

Moving Pointer

Servometric meters. - Servometric meters are used to display to the astronaut various operational parameters of the spacecraft. Included in each spacecraft are six round, two single-vertical, 10 dual-vertical, and two cross-pointer meters. The name "servometric" was coined to describe a device that is somewhat similar to a D'Arsonval meter movement. As shown in figure 3, a closed-loop rebalance principle is used to position the pointer in accord with the input signal. This concept provides fast response, high accuracy, stability, and mechanical ruggedness not found in other devices. Dial faces and legends, illuminated by electroluminescent patches, are imprinted to reflect the parameter displayed. Maximum accuracy is ensured by calibrating each device for full-scale operation.

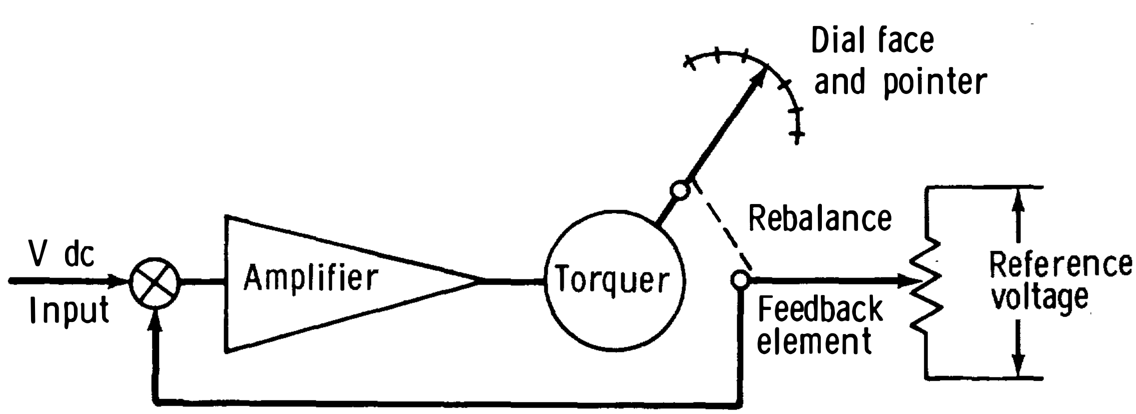


Figure 3. - Servomatic meter concept.

The meters are packaged within a hermetically sealed enclosure. The case is an aluminum investment casting with integral mounting flanges. The cover glass is cemented onto the case from the front, and a cast bezel is cemented over the glass to the case.

The back plate, with glass headers, is connected to the case by a solder-seal strip. The electronics, amplifier, and power supply are packaged in a conventional welded cordwood module that is solidly potted. Electrical adjustments, when required, are made by soldering discrete components to circuit boards mounted to the module.

During meter development through qualification testing, many problems were encountered. As they developed, these problems were reviewed, analyzed, and corrected. Some of the more significant problems and their corrective actions were as follows.

1. Vibration — The structure of the meter was not isolated, causing vibration problems. Corrective action included providing vibration isolation between the meter case and the internal components.

2. Uneven optical coating — Optical coating on the glass was not applied evenly. Corrective action included the fabrication of special jigs and fixtures to handle the odd-shaped glass and tight process control.

3. Cleanliness — Foreign materials of many types were found inside the meters. The materials included metal shims picked up by the torquer magnet. Corrective action included the use of brass shims and tighter control to keep work areas clean.

4. Sealing cases — The most significant problem during early testing of meters was in sealing the front glass within the case. Failures occurred during thermal vacuum testing. Specifically, breaks occurred in the bond between the glass and the case. During the thermal vacuum test, stresses developed which pulled the bond apart. Changing the potting to a stronger, more rigid material improved the bond, but caused the glass to crack. In analyzing the problem (fig. 4), the determination was made that a material had to be selected that had the proper shear strength, yet was flexible. The potting material between the meter case seal, the glass, and the case was changed to a flexible polyurethane, and the glass was made smaller to allow more potting. This change reduced the stress at the bond line so that seal breaking was prevented and the bond was flexible enough at a low temperature to prevent the glass from breaking.

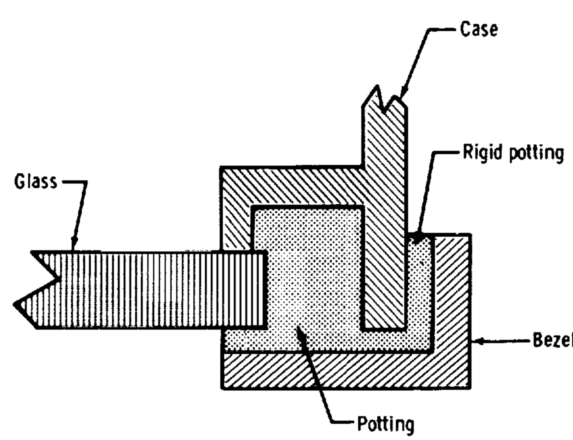


Figure 4. - Meter case seal.

Upon completion of this phase of the program, problems continued. In addition to numerous minor acceptance test problems such as lighting and case leaks, several major problems appeared. The problems were as follows.

1. Cleanliness — Contamination similar to that found prior to qualification testing was again found within the meters. A contamination review team visited the meter manufacturer and recommended improvements in many areas.

2. Dial-holder adhesive — The technique used to bond the meter dial holder was changed to facilitate production. The change included the addition of a black dye to the adhesive holding the dial. The addition of this dye reduced the strength of the adhesive, thus allowing the dials to fall off. This change did not have program approval and, when detected, the manfacturer was directed to return to the original technique which did not include the black dye. In the interim, while the change was in effect, several meters were manufactured, shipped, and installed in a spacecraft. Additional testing of the meters was required, including X-ray analysis, before the meters could be certified for flight.

3. Sticking meter pointer — A sticking meter pointer is the most significant meter problem. After qualification and shipment of many meters, the pointers were found to be sticking. Sticking occurred at either end of the meter scale or in the middle. Many corrective actions were incorporated into the meters to correct this problem, but none have been completely successful. Corrective actions include the following.

a. Sticking in the center of the scale was caused by inadequate clearance between the dial and dial-holder tabs, causing the dial holder to raise in the center when expanded. Distortion of the dial was caused by unequal pressure between the dial-holder tabs and the bezel. Corrective action for these problems included changing the dial-holder and dial dimensions to provide adequate clearance for expansion. In addition, the center of the dial was cemented to prevent movement.

b. Pointer sticking against the stops was caused by cement on the stops. Several corrective actions were taken to correct this problem, including (1) production instructions to require cleaning the movable stops with alcohol; (2) removal of chromate protective coating on the stationary stops; (3) cleaning of the torquer assembly at the subassembly level; (4) revision of production procedures to ensure proper adjustment of the pointer arm counterweight; (5) examination for excess cement along the dial edge; and (6) the addition of a fluorescent dye to the cement and the use of black light to detect any residue.

c. Corrective action for sticky pointer stops was also instituted on the completed assembly after delivery. Each meter pointer was cycled stop-to-stop, 100 times, in an attempt to break through any chromate film. If the meter still stuck, the instrument was returned to the manufacturer for repair.

In summary, with all the problems, the meters have functioned satisfactorily during all Apollo flights. Future programs, however, should reconsider the need for such a complex meter. The meters used in the command module are less complex, less costly, and less troublesome than the LM servometric meters. Finally, when servometric meters lose power, they continue to provide some on-scale indication. On critical LM meters, small red lights were added to the system to indicate power loss. If the servometric meters are used, this detection system should be built into the meters to provide a more positive indication of power and signal loss.

Thurst-to-weight indicator. - The thrust-to-weight indicator — an indicating, acceleration-measuring device — is used to measure acceleration during lunar landing. Indication is provided by a moving pointer on a fixed vertical scale illuminated by an electroluminescent lamp. The scale is calibrated from 0 to 6 lunar g. The display is packaged within a hermetically sealed magnesium enclosure. The enclosure consists of a housing to which a cover and a plate glass window are epoxy sealed. The internal mechanism is basically a simple accelerometer consisting of a seismic mass, guide rods, a cord, a pulley, a calibration spring, a damper assembly, and a pointer assembly.

The initial development of this indicator required design changes to resolve two problems: sealing and corrosion. The indicator sealing problem, similar to the meter sealing problems, was solved in essentially the same manner. The initial solution was to use a heavier bezel and thinner glass. The heavier bezel provided more bonding area; however, the thin glass broke under test. The glass was made thicker, thus solving the problem. Corrosion problems during salt-fog testing were caused by the interaction of the salt and the magnesium case. This problem was resolved by treating the case with epoxy before final painting. With the solution of these problems, qualification and testing proceeded without incident.

After an interval of approximately 2 years, the manufacturer was required to fabricate additional indicators. During acceptance vibration testing of these indicators, a screw fell out. A review of the assembly procedure determined that there was no torque requirement for screw installation. At that time, a decision was made to return all indicators to the vendor for screw tightening. The manufacturers started to comply with this requirement and ran into many problems. A thorough review of assembly procedures indicated that they were incomplete and that the technicians were not familiar with them. Corrective action included calling back the original technician to comply with the update of all the process specification and assembly procedures. Since this action was taken, no problems have occurred. This required action indicated the importance of thorough procedures and the advantage of completing fabrication of all devices without interruption.

Fixed Pointer

The range indicator is a vertical-scale indicator displaying range/altitude and range-rate/altitude-rate by means of a moving scale indicating against a fixed pointer. Two 10-foot tapes supply the moving scales that range up to 400 nautical miles for range/altitude and ±700 ft/sec for range-rate/altitude-rate. The tape is part of a digital servofeedback loop with a gray code word on the back corresponding to numbers observed on the front. An electrostatic sensor, or read head, determines the position of the tape by this coded word for comparison to the commanded position. If these two positions disagree, digital logic commands a direct current (dc) stepper motor to drive the tape to the correct position.

The indicator, in addition to the stepper motor and tape circuitry section, has two other operating sections: the processor and the interface (fig. 5). The processor contains the basic time-generating circuits for the indicator and performs the comparison of input against feedback to operate the motors. The interface section allows for the selection of the desired information source, stores needed input information, and provides digital conversion. Digital conversion is required because information enters the system as either a frequency word or as 15- to 18-bit digital words.

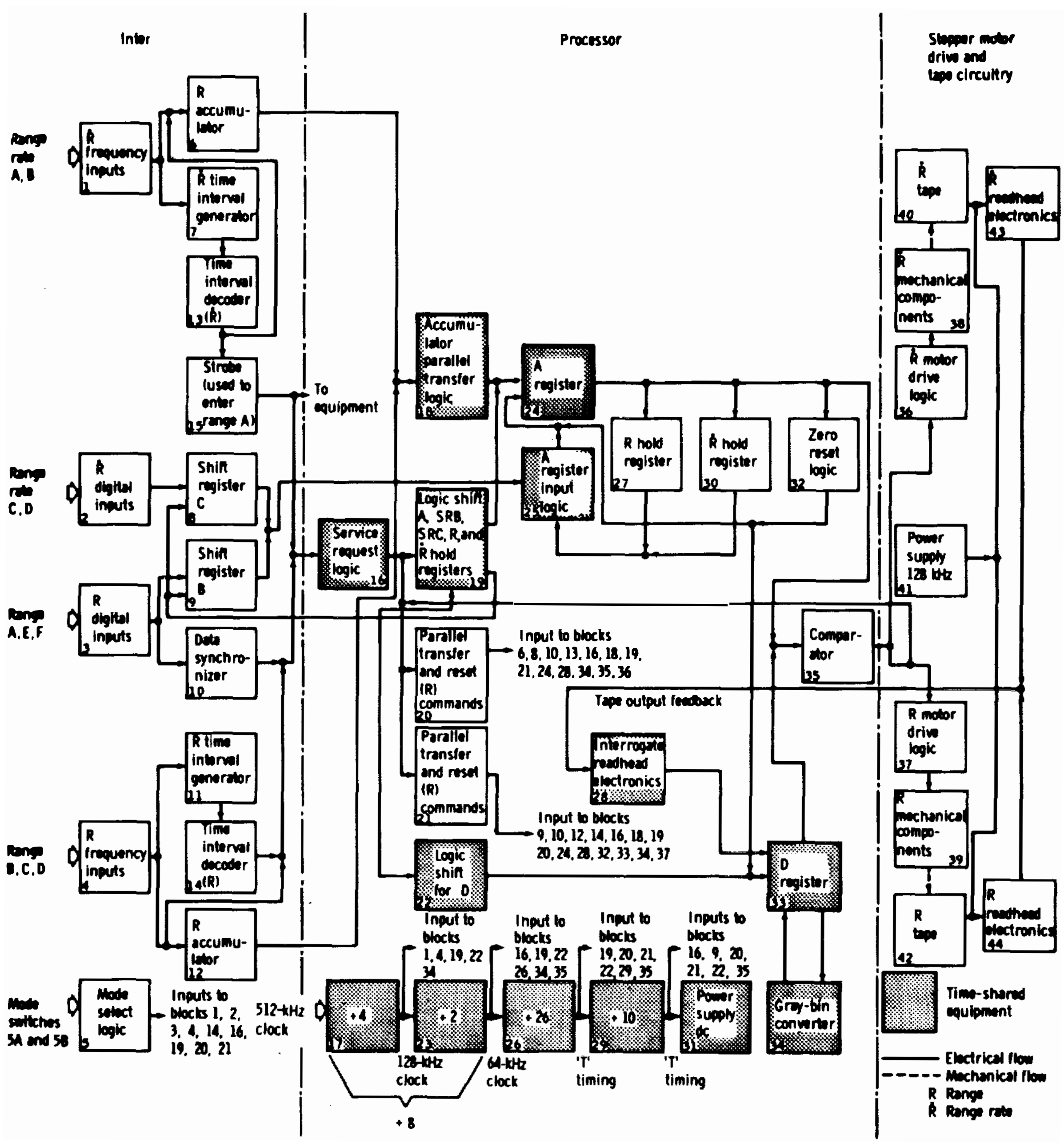


Figure 5. - Simplified signal flow diagram, lunar module range/range-rate indicator.

The indicator is packaged within a hermetically sealed enclosure. For the most part, the system electronics consist of micrologic (flat packs) integrated circuits that are mounted on 10 multilayer circuit boards. Each tape is made of 1-inch-wide, 3-mil-thick H film. One side of the tape has the range and range-rate display scales painted in black and white, while the other side has the gray code word with an exposed copper coating signifying a "1" and the lack of copper signifying a "0." A variable-polarity stepper motor is used to drive each tape. These 600-rpm motors operate through an 86:1 gear ratio for range tape drive and a 55:1 gear ratio for range rate tape drive. The gear train is connected to a spring that is attached to separate rollers to maintain constant tension on the tape.

The indicator, actually a complex digital computer, was developed under a two-phase program. The first phase produced a standard version for use on early spacecraft, and the second phase produced a high-reliability version for use on manned spacecraft. The high-reliability version was basically identical to the standard version, with the following improvements added.

1. Power signal monitor — The power signal monitor provides an alarm light in the event of (a) alternating current voltage below 85 volts, (b) direct current voltage below 20 volts, or (c) input data loss.

2. High-reliability parts — The read head and all electronic components are subjected to a 168-hour burn-in, with a 100-hour burn-in at 160° F for the 10 circuit cards. In addition, each card is tested at -30°, +25°, and +80° F, and each assembly is subjected to a 32-hour temperature cycling.

3. Mechanical improvements — Mechanical improvements included the use of a 7-watt permanent-magnet motor to replace the 4-watt four-pole variable-reluctance motor; a 100-percent-helium filling medium for improved heat transfer; enlarged tape spools; and a heat sink for the motors.

The high-reliability indicator has been the source of many problems throughout the program. The most significant problem occurred on the Apollo 9 mission (LM-3) just before launch. Test personnel observed erratic range tape operation while conducting various tests. The tape dropped below zero off-scale, oscillated below zero, or moved slightly below zero and then continued normal operation. Extensive testing on the vehicle could not duplicate the failure. A decision was made to replace the indicator and return the malfunctioning equipment to the manufacturer for failure analysis. Extensive testing by the manufacturer, including vibration and thermal environment, reproduced the failure; however, failure occurrence was random. Analysis finally identified two causes for the problems:

1. Oscillation below zero was caused by the serial number on the tape. This failure occurred only if the tape went below zero, not a normal condition.

2. A dual inverter gate micrologic circuit was found to have a poor weld on the common emitter connection. Visual inspection showed the lead surface to be mottled. This failure caused incorrect information to be shifted into the comparator, and this shift resulted in the tape slewing below zero.

A thorough analysis of the micrologic circuit indicated that the emitter weld joint, a thermal compression bond, did not adhere to the pad. Tests were conducted on similar circuits, and in all cases, it was impossible to break the weld joint. The failure was concluded to be random and within the reliability of the parts; therefore, no corrective action was taken. Other problems and corrective actions taken were as follows.

1. Mechanical installation — Cracked mounting tabs were found on indicators after installation into a vehicle panel. The problem was traced to improper installation into the panel. Modifications to the vehicle panels and installation procedures corrected the problem.

2. Motor drive circuit — Numerous indicators were rejected during preinstallation tests. An investigation indicated that transient voltages in both the panel fabrication shop and the test area were causing the destruction of motor drive circuits. Corrective action included proper grounding and the installation of isolation transformers.

3. Calibration — Indicator problems, such as oscillations, inability to read below certain points on a tape, and erratic tape operation, were traced to improper calibration techniques. To ensure proper calibration, procedure changes, including mandatory inspection points, were added, and the acceptance test plan was reviewed. The test plan revisions provided a supercalibration which added many additional test points.

Actual flight experience with the range indicator has been successful. The only criticism regarding this indicator by a flight crew was that the indicator does not operate like the simulator. Steps have been taken to make the simulators more like the actual flight vehicle to correct this situation.

Numeric Displays

Helium pressure/temperature indicator. - The helium pressure/temperature indicator is a digital voltmeter that accepts a 0- to 5-volt dc input and provides a four-decimal-digit electroluminescent read-out. An external switch selects the read-out parameter and also sets the proper scale factor ratio. In selecting temperature, the read-out displays three digits and a sign (plus or minus). When pressure is selected, four digits are displayed; however, the last digit is always zero.

The display is packaged within a hermetically sealed enclosure consisting of a housing with end cap and glass front face. Sealing is accomplished by a solder seal band attaching the end cap and by epoxy sealing the glass front face. Components (integrated circuits) are mounted to multilayer printed circuit boards by welding. The boards are mounted to an aluminum frame and encapsulated. The potting used for encapsulation provides both protection during shock and vibration, and proper heat transfer.

Development of the indicator was plagued by case sealing problems and the unavailability of electroluminescent lamps. In addition to problems associated with solder seal bands (corrected by applying improved procedures and techniques), sealing of the glass front face was a difficult problem to resolve. The initial design employed glass to obtain a glass-to-metal seal. Because of problems encountered by the glass manufacturer in obtaining these seals, a design change was made. An epoxy seal, similar to that used on meters, was developed. This seal has worked well throughout the program.

The problems associated with electroluminescent lamps were more basic than the seal problem. At the time the lamps were required, electroluminescence was in the early stages of development. Other procurements including meters, thrust-to-weight indicators, propellant quantity indicators, range indicators, timers, flags, and electroluminescent lamp panels were experiencing similar difficulties. The problems were especially acute in other digital-type electroluminescent lamps. Basically, problems were associated with inability to seal the lamps, removal of moisture from the basic material used, and voltage breakdown.

After completion of development and qualification testing, relatively few problems ensued. A series of failures occurring after fabrication of all indicators caused concern. The problem was associated with initiation of the thermal cycle part of acceptance testing. During this testing, approximately eight indicators were found to have shorts to the case. These shorts were found between the cases and signal grounds. Analysis indicated that the problem was not detrimental to indicator operation, but that the problem was undesirable. Failure analysis of the indicators determined that the problems were related to integrated circuits, and were found to be caused as follows.

1. Cement used to bond and electrically isolate the chip from the case was contaminated.

2. The chip was set into the case at an angle so that clearance between the chips and case was less than minimum specifications.

3. Conductive heat-sink epoxy under the body of the integrated circuit was under two leads.

Corrective action for all causes was similar; closer quality control was conducted during fabrication. In addition, vendor acceptance tests were revised to include an insulation resistance check from all pins to ground.

Propellant quantity indicator. - The propellant quantity indicator is a dual digital display that accepts an eight-bit binary-coded decimal word and provides a two-decimal-digit electroluminescent read-out. The read-out simultaneously indicates the percentage of fuel and oxidizer quantities. Input is an 8-4-2-1 format with the least significant bit first. The display is packaged within a hermetically sealed enclosure, consisting of a housing with an end cap attached by a solder sealing band at one end and by epoxy-sealed glass at the other end. Components, generally integrated circuits, are mounted to multilayer printed circuit boards by welding. The boards are mounted to an aluminum frame and encapsulated. Potting used for encapsulation was selected both to protect during shock and to provide heat transfer.

The propellant quantity indicator was manufactured by the same vendor as the helium pressure and temperature indicator. Throughout the program development, testing, and use, the same problems associated with this indicator were exhibited by the propellant quantity indicator. These problems included the failures associated with electroluminescent lamps, sealing, and, later in the program, integrated circuits.

Mission timer. - The mission timer is a seven-digit electroluminescent display showing mission elapsed time from 0 hours 0 minutes 0 seconds to 999 hours 59 minutes 59 seconds. The display counts real time from either an external 10-pps input or an internal time source. The internal time source is actuated automatically when the external source fails and is indicated by an illuminated tuning fork appearing on the display face. External controls to reset, stop, start, and slew are provided to facilitate timer operation.

The mission timer program is notable for numerous problems that eventually required complete redesign and repackaging by a different manufacturer. The basic design of the original mission timer was of modular cordwood construction. This type of construction has electrical components (resistors, capacitors, diodes, etc.) soldered between two circuit boards. The void between the boards is filled with a potting compound (fig. 6). Differential expansion between the potting compound and the circuit boards caused the solder joints to crack, thus breaking electrical contact. Initiation of thermal acceptance testing accelerated the interaction, causing failures to occur. Corrective action, whereby the timers were disassembled and additional solder was added to the connections, was attempted. This work created many new problems and did not solve the original problem. After many additional ground failures and a flight failure during the Apollo 11 mission (LM-5), a decision was reached to procure newly manufactured timers. The new timers were mechanically and electrically interchangeable with the old timers. The new timers, in addition to correcting the solder problem, included design changes to correct other timer problems.

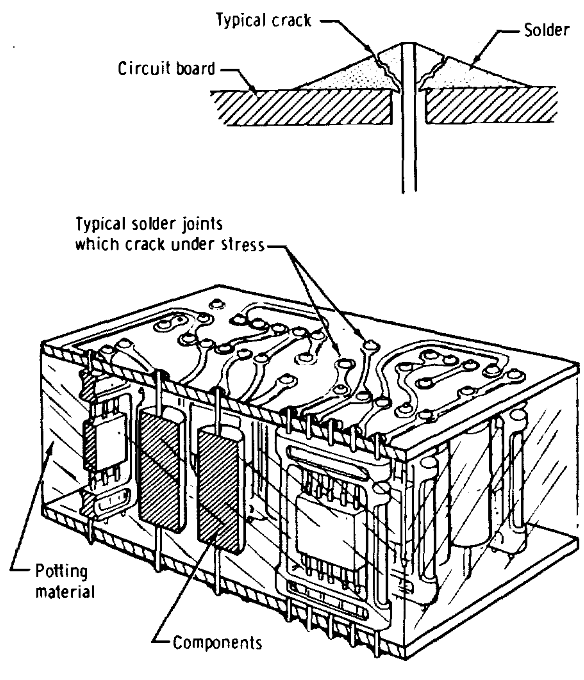


Figure 6. - Typical modular construction.

The basic design of the new timers used integrated circuits welded on printed circuit boards instead of cordwood construction. Other timer problems and their resolution were as follows.

1. Electromagnetic interference (EMI) susceptibility — The original timers were extremely susceptible to electrical noise generated during spacecraft testing. The installation of filters, wire shielding, changing switches, et cetera did not correct the problems. The following design and test changes for the new timers completely eliminated all noise problems.

a. Input commands (stop, start, slew, etc.) use +28 volts dc for activation instead of the +3 volts used for the original timers.

b. Logic circuitry in the new timers is isolated from the input command lines by a buffer stage. The activation signal must be present for a specific time period to initiate the logic circuitry. The old timers were activated by the leading edge of a pulse. Therefore, a high-voltage short-duration transient of the type normally found on the spacecraft will not affect the logic of the new timer.

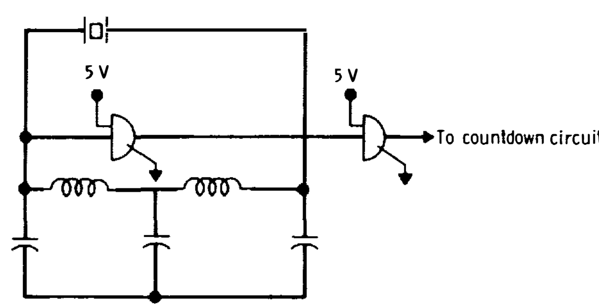
c. Extensive EMI testing included transients of -50 and +100 volts.

2. Cracked glass — The glass display panels of the timers cracked after installation into the spacecraft. The exact nature of the crack was not determined; however, cracks were thought to result from excessive stress built into the glass during fabrication. The new timer uses a glass seal in front of the display panel and an epoxy-sealed display panel, void of stresses.

Unfortunately, the new timer design was not problem-free. Because of the short procurement schedule, these problems were especially critical. The use of a new electroluminescent display panel and an internal oscillator circuit, were troublesome.

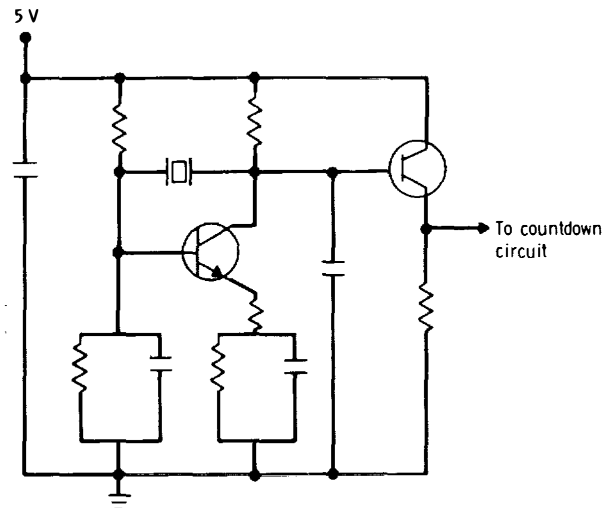
The new manufacturer decided to use electroluminescent lamp panels fabricated in his own facility. A four-digit display was fabricated and passed extensive qualification testing. While fabricating the mission timers (seven-digit display) many problems were encountered, primarily the inability of the electroluminescent lamp to withstand high voltage. After many attempts to resolve the problem, a new supplier was sought. The lead time required by the new supplier resulted in the procurement of an intermediate supply of electroluminescent lamp panels. The intermediate panels were obtained by cannibalizing the old timers and extracting the electroluminescent lamp panels. The initial qualification and use of the new timers included the old electroluminescent lamp panel and the new electronics.

The new internal oscillator circuit used, as did all other circuits, an integrated circuit. During panel testing, a discovery was made that at certain narrow temperature bands, the oscillator would not start once it had been stopped. During continuous operation over the required temperature range, no difficulties were encountered. After extensive analysis of the circuit (fig. 7(a)), a determination was made that some combination of resistance, inductance, capacitance, and temperature reduced the circuit gain. This reduction in gain would not allow the circuit to start reliably. The final solution of this problem was a design change using transistors (fig. 7(b)) instead of logic gates.



(a) Original concept.

Figure 7. - Oscillator circuit.



(b) Final concept.

Figure 7. - Concluded.

Other problems, including the procurement of multilayer printed circuit boards, the availability of integrated circuits, and an insulation resistance problem with inductors, caused some concern. These difficulties were solved while resolution of the major problems was underway.

A few of the new timers have been installed in a spacecraft and tested. The problems experienced with the old timers, especially noise susceptibility, have not recurred. The only problems to date have been related to use of old electroluminescent lamp panels. Because these panels were used in both the event and the mission timers, an apparent brightness difference was noticed. To correct this problem, dropping resistors were added to the electroluminescent lamp input power lines of the brighter of the two timers; thus adjusting the brightness of the displays to the same apparent level.

Event timer. - The event timer is a four-digit electroluminescent display that counts from 0 minutes 0 seconds to 59 minutes 59 seconds. The display counts real time either up or down, starting or stopping from external control switches. An external switch allows the presetting of any time value by slewing any digit. This program experienced the problems associated with the mission timer, and the solutions were the same. The only difference was that the new event timer did not have problems with the oscillator circuit. The old timers included an external oscillator to provide the 10-pps counting input. The use of an internal oscillator in the new event timer provided required timing accuracy and eliminated a source of system problems.

To date, with limited use, no serious event timer problems have occurred. As indicated for the mission timer, the only difficulty has been light intensity, but with the new electroluminescent lamp panels, this problem no longer exists.

Status Display

Flag indicators. - Flag indicators are electromechanical devices used within the spacecraft to present to the astronaut an indication of system operating status, mainly value position indicators. The LM uses 32 of these two- or three-position indicators. Flags are packaged within hermetically sealed enclosures, each consisting of a housing and end cap. The end cap includes an epoxy-sealed glass window. Mechanical (fig. 8) and electrical (fig. 9) designs for both the two- and three-position flags are essentially the same. The principal difference is the addition of another input and of discrete components to the three-position flag. Mechanically, the flags follow a basic leaf spring positioning concept. The center flag, appearing in the viewing aperture with no signal, is held in position by a cantilevered leaf spring acting against a rectangular positioning block. A 22- to 32-volt dc signal sends current through the electrical circuit and the torquer coil, moving the dial against a stop, thus exposing another flag. Removal of the signal will return the flag to the original position.

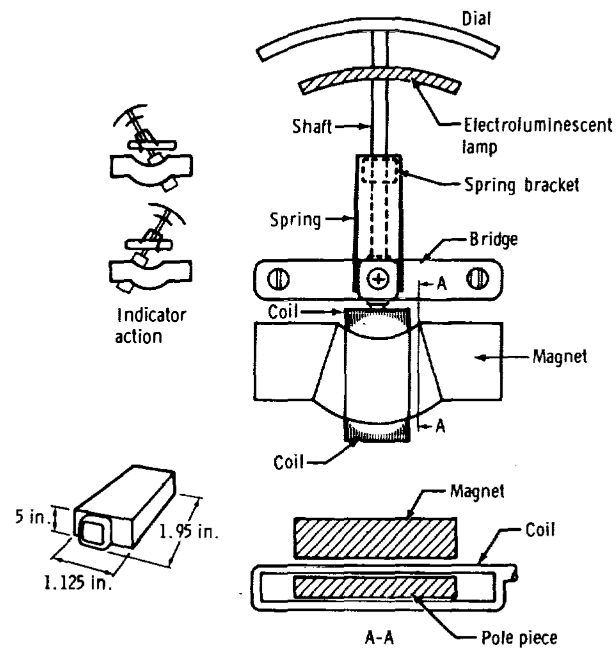
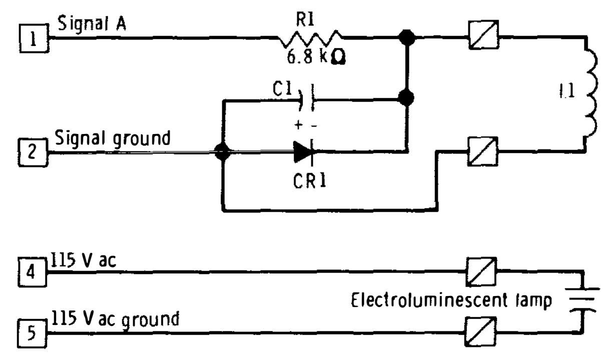
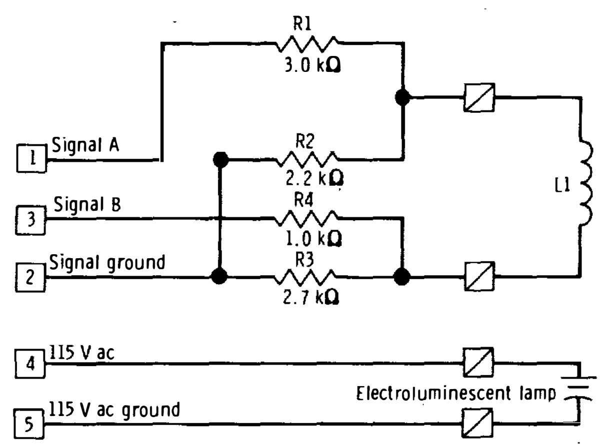


Figure 8. - Flag indicator.



(a) Two-position indicator.



(b) Three-position indicator.

Figure 9. - Flag indicator circuit diagram.

The flag program, from initial development to present usage, has had many problems. The most annoying problem has been flag hangup, or hesitation. Although many corrective actions have been taken, this problem has never been solved. An analysis of acceptance and qualification failures identified the following failure modes.

1. Bearing bridge resonance — The bearing bridge resonated under random vibration, causing the flag to hang up. Design changes were made to the bridge to increase the resonant frequency above the critical level.

2. Coil clearance — The clearance between the coil and the pole piece or magnet (or both) was not sufficient; under random vibration, the coil would bind. Corrective action included increasing clearance and more accurate part alinement.

3. Spring system — Many flag hangups were caused by improperly formed springs. Procedures were implemented for proper spring formation and for spring tension measurement.

4. Flex lead drag — The flex lead which carries current to the coil was binding on the shaft. Corrective action included wire layout revision.

5. Dial clearance — The dial stuck to the case and stops because of lack of clearance. This problem was corrected by providing proper tolerances and by adding a rubber bumper to the stop.

Solution of the previously mentioned undesirable flag characteristics did not eliminate the hangup problems. Additional analyses determined that flag hangup was also attributable to the following.

1. Cement residue or paint on the stops

2. Flex lead drag because of slight misalinement

3. Random peak of spring-block friction from rough spots on the spring or actuator block

4. Nonuniform loading of spring leaves against the actuating block

5. Particulate contamination

A review of these problems and earlier flag problems indicates that the best corrective action would be flag redesign. Redesign would provide enough dial restoring torque to overcome any slight increase in friction or sticking. Steps were taken to improve the yield without redesign. A quality control audit was conducted to eliminate contamination; acceptance thermal cycling was added; manufacturing procedures were refined; and additional screening was added to acceptance testing. These actions, while reducing the number of field failures, have not eliminated them. Field failures that do occur are not hard hangups. The flags recover after a few seconds or require tapping to cause transfer. Because of low mission criticality of the flags, this solution to the problem was considered acceptable. Failures of this type (hesitation) have occurred on almost every flight of the LM.

Component caution indicator. - Component caution indicators are small incandescent lamp assemblies used to indicate system malfunction (red and yellow lamps) and lunar touchdown (blue lamps). The lamps are mounted in aluminum housings secured to the panel by a two-bolt flange (red and blue lamps) or integral screw threads (yellow lamps). Clear silicone supports the lamps in the housing. The electrical terminals are epoxy sealed. The legend is provided by a filter with a white diffuser cemented to the housing.

Design, development testing, and use of these indicators for the LM have been successful. The only problem has been an occasional broken filament. The failures that did occur happened early in the indicator life and did not affect the program. One shortcoming of the lamps is that the panels cannot be removed and replaced easily. The assemblies are back mounted and hard wired into the panels, requiring panel removal for lamp change. The lamps also are molded into the assembly and cannot be removed without being destroyed. Redesign to make lamp replacement easier would be an advantage over the present design.

Caution and warning display. - The caution and warning display is an incandescent lamp assembly used in conjunction with the caution and warning electronics to indicate and isolate system malfunctions. Each spacecraft has four display matrices, two red matrices located on panel 1 and two yellow matrices located on panel 2. The individual legends are illuminated when a 5-volt dc signal is commanded from the caution and warning electronics.

Each display assembly consists of 10 identical lamp modules located in a flange-mounted aluminum housing. Filters, legends, and color diffusers form separate subassemblies that are completely interchangeable. Electrical connections among lamp modules and external power are provided in printed circuit boards. These boards also include blocking diodes and inrush current-limiting resistors. Each lamp module contains two subminiature bulbs potted into place. Five lamp modules and a circuit board assembly are mounted together and potted into a subassembly. Two of these subassemblies and 10 legend subassemblies are mounted to a bezel by eight screws to form a complete assembly.

The caution and warning display, which is similar to the component caution indicator, has a history of success. Similar to the caution indicator, the only problem to date has been an occasional broken filament. The impact of a broken filament could be minimized by redesigning the display to include removable bulbs. Another design improvement would be to include the ability to make legend changes without completely removing the assembly from the panel.

Interior Lighting

Floodlights. - Floodlights, used to illuminate the display panels within the spacecraft, consist of the following four types of incandescent lamp assemblies.

1. Ceiling floodlight — The ceiling floodlight uses four miniature lamps electrically and mechanically affixed to a printed circuit board. Each lamp uses a clear polycarbonate lens for magnification and redirection of light for optimum lighting. The housing is made of an aluminum sheet metal cover fastened to tapered baffles arranged to produce sharp light cut-off. A hood is provided to direct the light within the spacecraft.

2. Forward floodlight — The forward floodlight uses one miniature lamp mechanically retained by a nylon insert. Electrical connection and mechanical support are provided by a spring-loaded contact. The assembly contains a clear polycarbonate lens designed for magnification and direction of the direct and reflected light to the proper area.

3. Post floodlight — The post floodlight uses one miniature lamp with integral wire leads for electrical connection. The lamp is soldered directly into the unit lead wires and potted into a polycarbonate lens. The housing is made from an aluminum spinning and employs baffles to direct the light to the proper area.

4. Side-panel floodlight — The side-panel floodlight uses two miniature lamps with integral wire leads for electrical connection. The lamps are soldered to terminals

within the housing and potted into place. The housing is made from molded diallyl phthalate. The light produced is directed through a sandblasted nylon light filter secured to the housing with a lens retainer.

The history of floodlights throughout the LM program has been similar to that of other incandescent lamps. Failures have been few in number and have been limited to filament failures. During early development, problems were encountered in meeting vibration requirements. These problems were resolved by simple modifications to floodlight assemblies and, in the case of ceiling lights, by providing a special vibration-isolation mounting in the spacecraft. The only other problem occurred during installation of the post floodlights into the spacecraft panels. Unless these lights were handled with extreme care, wires were broken. To correct this problem, the circuit boards were modified to lock into the housing. Here again, as with other incandescent light assemblies, floodlights should be made to permit easier replacement of defective lamps. As previously indicated, changing of lamps requires removal from the spacecraft and disassembly of the light.

Utility floodlight. - Utility floodlights are portable lights used for additional panel lighting, search of remote locations in the spacecraft, and reading. Utility floodlights use two miniature incandescent lamps in a machined Teflon housing which acts as a thermal and electrical insulator. The light assembly includes an 8-foot flexible cable and a universal lamp at the base. This arrangement allows the light to be either handheld or clamped to the spacecraft structure. The assembly has two separate lamp beams, provided by clear glass lenses designed for magnification and light direction.

The utility floodlight program was similar to all other incandescent light programs. No significant problems occurred during development or qualification testing, and except for a few broken filaments, no acceptance testing or use problems were found.

Electroluminescent lamp panels. - Electroluminescent lamp panels are solid-state lamps of various shapes and sizes used to illuminate panel nomenclature. Light is emitted from a thin layer of phosphor when the layer is electrically excited. Lamp construction is similar to that of a capacitor, as shown in figure 10. A phosphor/ dielectric is placed between two closely spaced electrodes and is hermetically sealed into a plastic envelope. The light produced is dependent on voltage for intensity and on frequency for color.

The procurement of electroluminescent lamp panels has been relatively free of problems since the start of the program. Initially, electroluminescence was state of the art, which led to problems in manufacturing techniques. When these initial problems were corrected, only two additional difficulties developed. The first concern was the development of dark spots in the electroluminescent lamps after extensive periods at high temperature. This problem was brought to the attention of the manufacturer, and corrective action was taken. Because fabrication of electroluminescent lamps is proprietary information, corrective action details could not be obtained. The only other concern was high-resistance shorts that developed around the leads. This problem was caused by the phosphor bronze mesh lead floating to the top of the plastic covering during processing. Corrective action included process specification revision and changing of the mesh material. Testing has proved this and all other corrective actions acceptable.

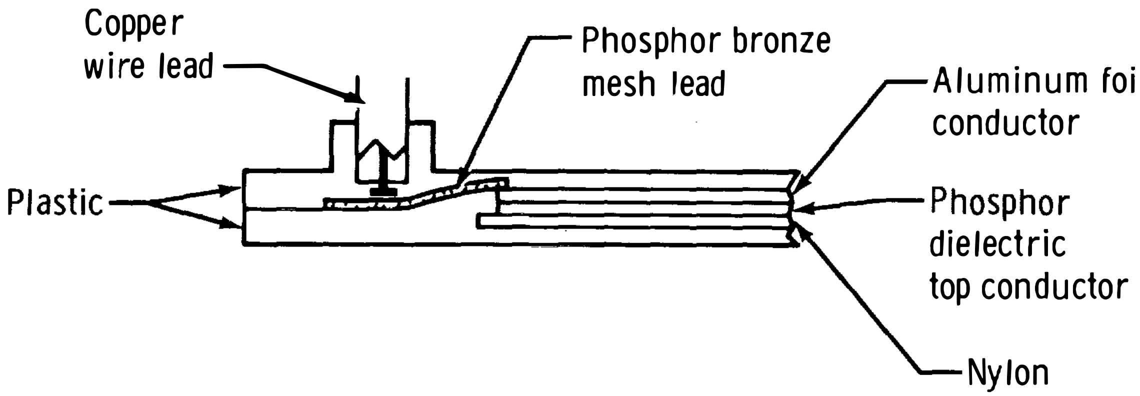


Figure 10. - Electroluminescent panel.

Switch tips. - Switch tips are small self-illuminated plastic caps used as handles for all LM toggle switches. The illumination is required to indicate switch position in a darkened spacecraft. The following four types of switch tips were used.

1. Maintain
2. Momentary
3. Lever lock
4. Slew (four-way momentary)

The tips are fabricated from transparent plastic (Kel-F) and illuminated by means of self-luminous paint deposited in holes. During fabrication, the self-luminous paint, activated with promethium-147 in the form of microspheres, is sealed into glass tubes. The tubes are placed in holes in the switch tip and are sealed by an epoxy plug. The exact number of tubes in each tip and their locations are determined by the switch function.

The problems associated with switch tips would have been insignificant if the tips did not contain promethium-147. This material emits beta particles and has a half-life of 2.6 years. Being radioactive, the material comes under control of the Atomic Energy Commission and must be handled according to their procedures. These procedures include (1) a periodic smear test to check for leakage and (2) tight control in the event of leakage. Problems associated with this material were vividly demonstrated just before the flight of Apollo 11 (LM-5). Mechanical damage to several tips caused leakage of the radioactive material during panel rework and required extensive cleaning of the spacecraft and additional medical tests for the crew after completion of the flight. As a result of this Apollo 11 problem, tighter control was initiated by NASA to ensure that similar situations did not recur. In addition to these handling problems, other significant problems appeared.

As a result of flammability investigations, the original switch tip material was changed to Kel-F. As a part of this modification, the method of implanting the luminous material was revised. The depositing of the luminous material into the plastic was causing a reaction and eventual radioactive leakage. To eliminate this difficulty, glass tubes were used. This method provides both glass and plastic seals, thereby eliminating leakage. The glass tubes were used in all switch tips, except for a large dot in the center of the lever-lock tip. This tip was not originally changed because of the size of the dot. Later, this dot was changed to two glass tubes, required because of excessive leakage caused by material reaction around the dot.

Use of these switch tips under actual flight conditions does not appear necessary. Debriefing of flight crews has indicated that the crew did not use the switch tips and would not operate a switch unless they could see what they were doing. Based on these discussions and potential problems with radioactive devices, a hard look should be directed to the use of switch tips in future programs.

Exterior Lighting

Self-luminous disks. - Self-luminous disks are small illuminated disks used to illuminate the docking target. The target is attached to the exterior LM structure as an aid in final docking maneuvers. Additional disks are located on the spacecraft external surface to aid an astronaut during extravehicular activity. Each disk is made of a fused silica outer housing which contains a luminous compound. The luminous compound consists of promethium-147 microspheres as a physical mixture with a suitable phosphor and inorganic binders. After curing at elevated temperatures, a ceramic plug is cast behind the disk to seal the luminous material in place.

Problems related to switch tips also appeared in these disks. The disks, being more rugged and mounted externally to the spacecraft, have presented fewer handling problems. The only problem associated with the disks occurred early in the program during temperature testing. The problem, cracking of the disks, was resolved by modifying the mounting used to clamp the disk to the target. Since then, no problems have occurred. An important consideration is the limited lifetime of the target because of radioactive half-life. To ensure correct brightness during flight, disk fabrication must take place as close to launch time as possible. To accomplish this, the target is shipped from the manufacturer to NASA Kennedy Space Center for installation as close to launch time as practical.

Docking lights. - Docking lights are running lights mounted on the LM exterior to provide visual orientation and to permit gross attitude determination during rendezvous and docking. The lights are visible and their color recognizable at a maximum distance of 1000 feet. The colors of the lights and their locations are as follows.

Color	Location
Red	Port light
Green	Starboard light

Color	Location
White	Aft light
Yellow	Forward light (negative Y-axis)
White	Forward light (positive Y-axis)

The lights are fabricated by using up to six miniature lamps, wired with two lamps in series to a terminal board. A silicone rubber color filter is placed over each lamp before it is mounted to the terminal board. The terminal board lamp assembly is mounted on an aluminum housing, wires are potted in place, and an aluminum cover is installed. A glass window is joined to a support which is then secured to a stainless steel housing, forming a window subassembly. This assembly is fastened to the housing assembly and secured; thus completing the light as shown in figure 11.

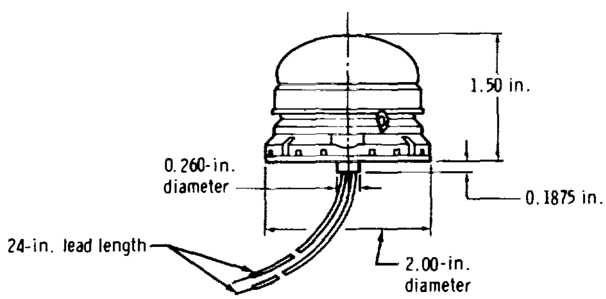


Figure 11. - Exterior docking light.

This program has been quite successful. Three failures that occurred during qualification tests were attributed to an easily corrected manufacturing defect. There have been no failures on vehicles during testing or in flight. The initial contract for docking lights was terminated after many serious design problems occurred. The history outlined relates only to the second contractor.

Tracking light. - The tracking light is a high-intensity light that permits, at distances of up to 400 nautical miles, visual tracking of the LM vehicle by the command and service module. The light has a 60° beam spread and flashes at a rate of 50 flashes per minute. The light consists of two separate assemblies (electronics and flash head) joined by a high-voltage cable. The electronics section consists of a filter network, an oscillator, a magnetic amplifier, high-voltage output bridges, a pulse-forming network (PFN), and a trigger circuit. The flash head section consists of a special design reflector and a xenon flash tube. Successful operation of the light is dependent upon the design and assembly of the high-voltage section depicted in figure 12.

Tracking light history has been marked by numerous failures that have occurred in testing, acceptance and qualification, and flight during the Apollo 9 and 12 missions. As a result of design problems, the contract with the original light manufacturer was terminated before any manned Apollo missions were flown, and a new manufacturer was obtained. The new manufacturer, along with the prime LM contractor, was responsible for design and fabrication of the light used during all LM flights. Failures

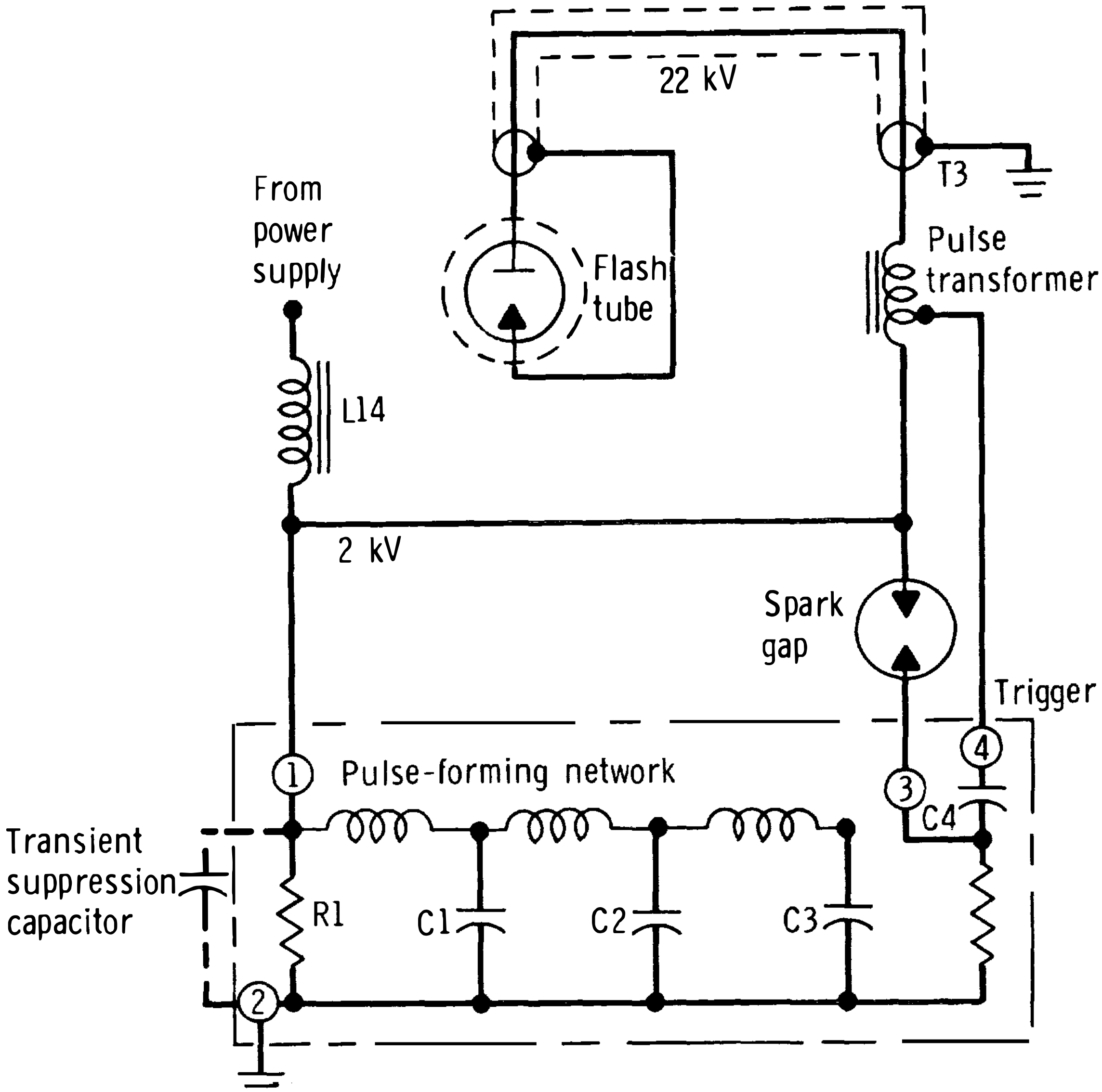


Figure 12. - Pulse-forming network, trigger, and lamp circuits.

that occurred were attributed to high-voltage section problems. These failures, and the corrective action listed as follows were encountered after the Apollo 9 failure.

1. Pulse-forming-network failure — Because of qualification failures, design changes were implemented on the PFN. Analysis indicated that the most probable cause of failure was a low-pressure void within the PFN, and this void allowed voltage breakdown. To ensure that a void could not occur and to correct other design problems, the following changes were implemented:

a. Terminals were changed to a more reliable type.

b. Terminals were brazed into place.

c. Manufacturing procedures for cans were changed to heliarc welding.

d. A controlled-pressure nitrogen bubble was created within cans.

e. Inductors were potted.

After these changes were completed, the PFN underwent extensive testing as a component and as part of an assembled light. No failures were attributed to the PFN during any of the testing.

2. Flash head arcing — Another possible cause of failure was arcing within the flash head. A review of the flash head design indicated changes could be made to increase reliability. These changes included the following:

a. Development of new potting procedures

b. Relocation of connections to reduce electric field

c. Procurement of a new high-voltage cable with higher breakdown voltage and better bonding capability

d. Use of a wavy-type spring washer to prevent cracking of the ceramic insulator

e. Elimination of the void area at the connection of the high-voltage cable

3. High-voltage arcing — To eliminate the possibility of arcing within the electronic high-voltage section, various changes were made, including the following:

a. Use of new potting procedures, including sandblasting of certain components

b. Addition of an arc-suppression capacitor to reduce transient voltages

c. Relocation of connections to reduce electric field and provide longer path

These changes and improvements, along with an extensive thermal vacuum acceptance screening, were implemented in an attempt to preclude further light failures. However, these actions were not successful, as evidenced by the Apollo 12 failure.

As a result of the Apollo 12 failure, another thorough analysis of the light design and its operation was undertaken. The analysis did not pinpoint a design deficiency, but did indicate the probability of high-voltage failures. A determination was also made that voltage breakdown at the flash head could occur as a result of potting deterioration. The deterioration could be caused in flight when the light is directed at the sun. Because positive corrective action could not be taken, the following changes were made.

1. Reduction of the operating voltage of the light — A change in operating voltage increased the safety margin of all high-voltage components (power supply 2000 volts dc as opposed to 4000 volts dc; trigger pulse 16 kilovolts as opposed to 22 kilovolts; power 145 watts as opposed to 175 watts).

2. Reduction of flash rate — The change in flash rate decreased the duty cycle.

3. Acceptance testing — The time the light is on during critical pressures of acceptance thermal vacuum testing was increased.

4. Daylight restriction — The use of the light was restricted to night operation only to preclude heating by the sun. Thermal reflection was added to the paint to dissipate heat if the light is turned on in sunlight.

These changes have been incorporated into tracking lights now being installed into the spacecraft. At the present time, no flight data exist to ascertain if the changes are adequate. Should the present design light fail during flight, only one corrective action remains. The latest analyses, including a review of other devices, indicate that the ultimate solution to high-voltage breakdown problems is pressurization. Therefore, despite the possibility of flying all remaining flights with the present configuration even if it fails, the light should be redesigned to include pressurization of all high-voltage areas. This solution to the problem would be expensive and time consuming and, in addition to deletion of the light, should be considered if another flight failure occurs.

Miscellaneous Components

Time delay. - The time delay is an electronic circuit and relay assembly used to delay supercritical helium flow to the descent engine fuel tanks. The delay eliminates the possibility of fuel freezing as a result of helium flow. The time delay is packaged within a hermetically sealed enclosure. Electronic components, including resistors, capacitors, transistors, diodes, and relays, are soldered to printed circuit boards mounted within the enclosure. The components are potted in place to ensure thermal transfer and provide vibration isolation.

A review of the history of this device indicates a successful system. The initial design phases experienced some minor problems, easily corrected before completion of qualification testing. Process control and acceptance testing has been a reliable screen. There have been no problems during vehicle testing or flight.

Control assembly. - The control assemblies consist of three black boxes used to provide signal processing and switching between the various LM subsystems. Two of the boxes are located in the cabin behind panel 3, and the third box is located in the descent stage. Components (resistors, diodes, relays, transistors, capacitors, etc.) are mounted on printed circuit cards secured within the boxes. Electrical connections are brought out of the boxes on pigtail connections sized to mate with vehicle connections. The three hermetically sealed boxes were standardized to facilitate design and minimize testing. The original LM design included a stabilization and control panel with a few electronic components mounted on the back. These components grew in number until it became necessary to find another location. Control assemblies 1 and 2 were developed first, while control assembly 3 was developed later.

The circuitry within control assemblies 1 and 2 is generally associated with the following functions.

1. Steering error signals

2. Guidance control

3. Ascent and descent engine control circuitry including the following:

    a. Automatic engine ON/OFF

    b. Abort

    c. Abort stage

    d. Time delays

    e. Fire override

    f. Ascent engine latching device logic

4. Translation

5. Despiking filters

6. Dead-face simulator

7. Caution and warning tone generator

Control assembly 3, located in the descent stage, includes functions associated with lunar surface contact lights and descent engine shutoff.

Problems associated with control assemblies have been mainly in the area of reliability rather than actual hardware failures. After the design and qualification of assemblies 1 and 2, a determination was made that many single-point failures could occur. To preclude these failures, design changes were necessary. Because assembly 3 was in the early stages of fabrication, redundant paths were included within the box. Control assemblies 1 and 2, being in later stages of fabrication, could not be changed easily. Analysis indicated that external wiring changes could be made to improve system reliability. These wiring changes were made, and the system has operated successfully throughout all missions. The only hardware problem to date has been with relays. Several relays failed to close after a temperature test at 20° F. Failure analysis indicated that the failure was caused by a polyethylene glycol contaminant which had a freezing temperature in the 34° F range. Because the disposition and quantity of the polyethylene glycol in the can is critical, no foolproof corrective action was available. Thermal cycling of the control assemblies was thought to be an effective screen and was included as part of acceptance testing.

Other than the temperature problem, the control assemblies have been successful. There have been no known flight failures or vehicle test failures. The use of these boxes should be considered for other programs; however, consideration should be given to make design changes more practicable.

Overlays. - Overlays are plastic (polycarbonate) panels of various shapes and sizes used to provide identification of all vehicle displays and controls. The overlays are placed over the electroluminescent panels, with the resulting assembly secured to the panel structure. Overlay panels are processed to include clear areas, yellow and black stripping, white indicia, and gray background. Markings (indicia) are produced by a photographic etching process. Overlays of various shapes and sizes are required to accommodate display panels.

The overlay program has been successful, with few failures. However, two areas of concern should be reviewed. The first area is the potential fire hazard of the plastic material. Design changes were initiated to install fire barriers along all overlay edges. These barriers of aluminum channel, along with the panel-mounted components, act as heat sinks in case a fire is started. Tests have indicated that even in an oxygen atmosphere, when a direct external flame is removed from a burning panel, the overlay will not continue to burn. The other concern pertains to panel changes. With the present overlay system, nomenclature changes are costly in time and money. Simple changes can be made by overpainting; however, these changes will not be illuminated. A marking system should be developed that is more easily changed.

CONCLUSIONS

The lunar module display and control subsystem has successfully completed the required mission. Numerous failures of all devices have occurred during all phases of testing and during a few flight anomalies. The initiation of acceptance vibration and thermal testing has been instrumental in reducing the number of vehicle test and flight failures. Flight anomalies including tracking light failure on the Apollo 9 and 12 missions and mission timer failure on the Apollo 11 flight have had positive corrective action taken to preclude their recurrence. Components including (1) flag indicators, (2) servometric meters, (3) toggle switches, and (4) tracking lights continue to cause concern because of their high failure rate. Corrective actions taken to date, in addition to extensive acceptance testing, have minimized the probability of a flight failure, but have not eliminated the possibility.

The most significant recommendation of this review concerns the spacecraft panels rather than the components mounted therein. Apollo design philosophy required components to be designed so as to be rear mounted and hard wired into panels which were then mounted into the spacecraft. This philosophy necessitated panel removal and electrical and mechanical demating for replacement of components, as well as correction of minor problems. In designing future spacecraft, consideration should be given to providing more accessible and replaceable components. This action will minimize vehicle downtime, improve reliability, and simplify retesting.

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