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APOLLO EXPERIENCE REPORT — BATTERY SUBSYSTEM

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16. Abstract

Experience with the Apollo command-service module and lunar module batteries is discussed. Significant hardware development concepts and hardware test results are summarized, and the operational performance of batteries on the Apollo 7 to 13 missions is discussed in terms of performance data, mission constraints, and basic hardware design and capability. Also, the flight performance of the Apollo battery charger is discussed. Inflight data are presented.

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APOLLO EXPERIENCE REPORT

BATTERY SUBSYSTEM

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SUMMARY

The requirements established for the Apollo command-service module and lunar module batteries were well within the existing state of the art for batteries; hence, no unique problems were identified or experienced during battery development and qualification or during short-time unmanned flights.

The only significant problems resulted from the use of a relatively new type of nonabsorbent separator (Permion 307) in the command module entry and postlanding batteries and from failure to verify the effectiveness of the battery-charging system for those batteries. These two factors jointly resulted in severe undervoltage at the command module main buses at command module/service module separation during the Apollo 7 mission. The final solution of these problems for the flight of Apollo 11 was achieved by reverting to the original absorbent cellophane separator material and by raising the output voltage of the command module battery charger.

With the possible exception of auxiliary battery 2 in the unmanned Apollo 6 flight (insufficient data to prove a battery failure), no battery failure occurred in any flight through the Apollo 16 mission. This is proof that the design- and verification-test principles were valid and that procedures and criteria for acceptance testing at the battery vendor's facility and preparation for flight at the Kennedy Space Center were effective in culling batteries that might fail in flight.

INTRODUCTION

The batteries discussed herein are the operational flight batteries developed for use in the Apollo spacecraft. The report includes a discussion of the command module (CM) entry and postlanding batteries and battery charger, the lunar module (LM) main power batteries, and the CM and LM pyrotechnic batteries. This report does not include a discussion of the miscellaneous off-the-shelf batteries used in the various developmental boilerplate flights. All Apollo batteries were of the silver-oxide/zinc alkaline type. The LM main power batteries were primaries (single discharge), whereas the CM batteries and all pyrotechnic batteries were low-cycle life secondaries (rechargeable). The Apollo 7 to 13 missions are covered.

COMMAND-SERVICE MODULE BATTERIES

The unmanned command-service module (CSM) flights used from two to eight entry batteries in the command and service modules, two pyrotechnic batteries in the CM, and two pyrotechnic batteries in the SM to supply power to mission-sequencing control devices and service module (SM) jettison control devices. The manned vehicles used three entry batteries and two pyrotechnic batteries, all in the CM. The mounting locations of the various batteries used in manned missions are shown in figures 1 to 3.

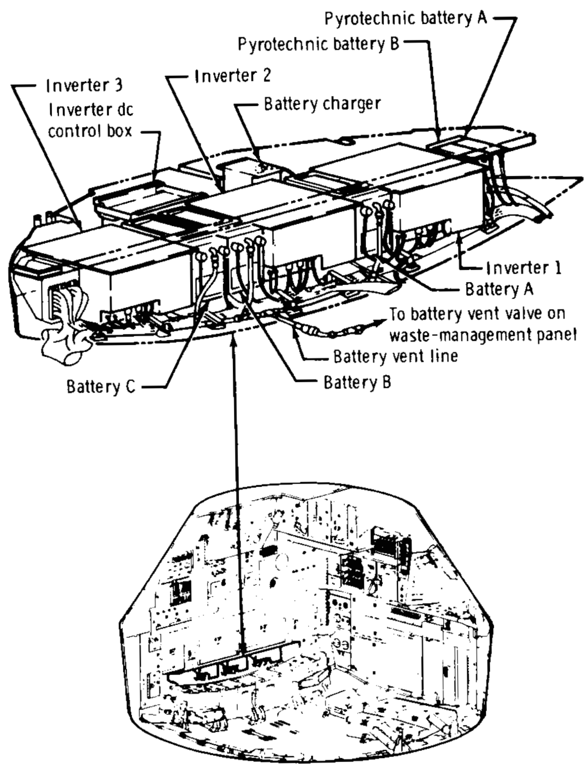
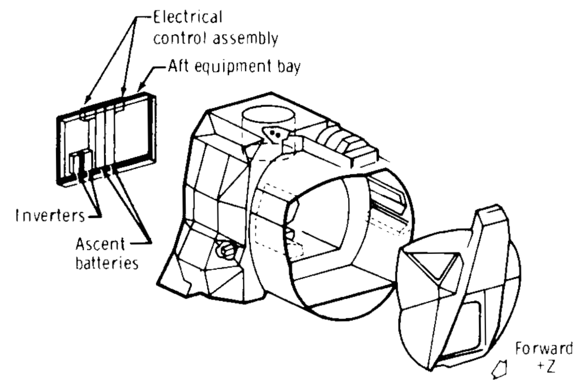
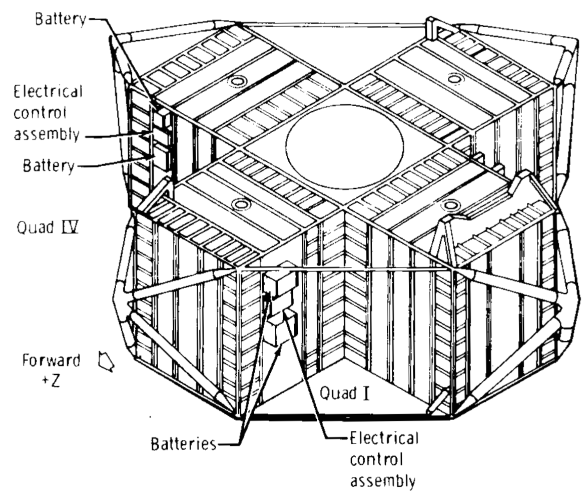


Figure 1. - Command module battery and charger locations in the lower equipment bay (block II).



(a) Ascent components.



(b) Descent components.

Figure 2. - Locations of LM main electrical power system components.

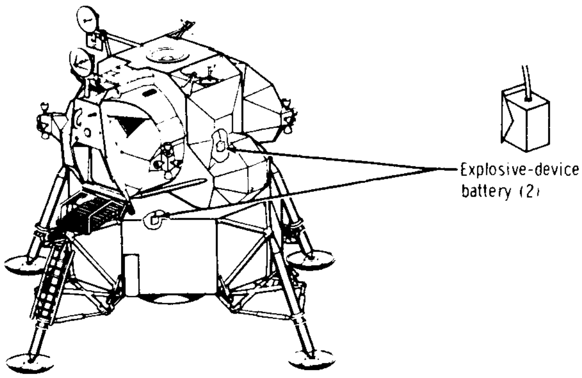


Figure 3. - Locations of LM explosive-device batteries.

Entry and Postlanding Batteries

The entry and postlanding batteries were used to supplement the main CSM power source (that is, the fuel cells) during peak loads (for example, service propulsion system gimbaling); to supply certain parasitic loads that had to be kept separate from the CM main direct-current (dc) buses; and, after CM/SM separation, to supply all CM electrical loads.

The original version of the entry and postlanding battery had a capacity of 25 A-h. The battery had a welded titanium case and a machined, anodized-magnesium cover. A coat of beryllium-aluminum paint was applied for increased emissivity and for corrosion resistance. The plate-separator material in the cells was Permion 307, an irradiated polyethylene film selected because of its thermal resistance. A thermal environment of 250° F in the CM lower equipment bay was expected for approximately 30 minutes because of entry heating. The available energy was 725 W-h per battery at 29 volts or 2175 W-h per shipset of three batteries. The battery had to be capable of six charges and discharges, the discharges consisting of 30 minutes at 35 amperes and then 225 minutes at 2.0 amperes (all at more than 27 volts).

The 25-A-h battery was qualified but was never flown. By means of better definition of the block II vehicle battery-energy requirements, it was shown in mid-1964 that the required entry and postlanding energy was 2234 W-h, which exceeded the energy available even without a battery failure. It was determined that the batteries should be capable of 40 A-h (1160 W-h each) to satisfy vehicle requirements with one allowable battery failure. In October 1964, a contractor was authorized to proceed with battery redesign.

The 40-A-h battery also contained Permion 307 plate-separator material and required additional active material to deliver the additional energy. The case material was changed to cemented Plexiglas, which was enveloped in glass-reinforced epoxy and covered with gray plastic paint. Also, this battery had to yield six complete charge/discharge cycles. The discharge profile was changed to 25 amperes for 1 hour and 2 amperes for 7.5 hours; a 25-ampere load was more representative of spacecraft battery loads. The weight and dimensions of the 40-A-h battery are shown in table I. The voltage-current characteristics of this battery are shown in figure 4. This battery was qualified and flown on all Apollo missions through Apollo 13.

TABLE I. - BATTERY DESCRIPTION

Item	Entry and postlanding battery, 25 A-h	Entry and postlanding battery, 40 A-h	Pyrotechnic battery	LM ascent-stage battery	LM descent-stage battery
Voltage, nominal, V	29	29	20 (minimum)	29	29
Number of cells	20	20	20	20	20
Capacity rating, A-h	25	40	.75	296	400
Dimensions, nominal					
Length, in.	10.15	11.75	6.25	35.75	16.94
Width,[a] in.	6.4	5.75	2.635	4.95	9.04
Height, in.	5.75	6.875	2.87	7.78	9.96
Weight, lb	22	28.5	3.5	123.7	132.7
Type of terminal	Stud and nut	Stud and nut	Stud and nut	Stud and nut	Stud and nut
Quantity used per vehicle	(b)	3	2	2	4
Location	CM	CM	CM and LM	LM	LM
Thermal-control method	Passive	Passive	CM: passive LM: coldplate[c]	Coldplate[c]	Coldplate[c]

[a]Excludes mounting rails where applicable.

[b]Never flown.

[c]32° F coolant.

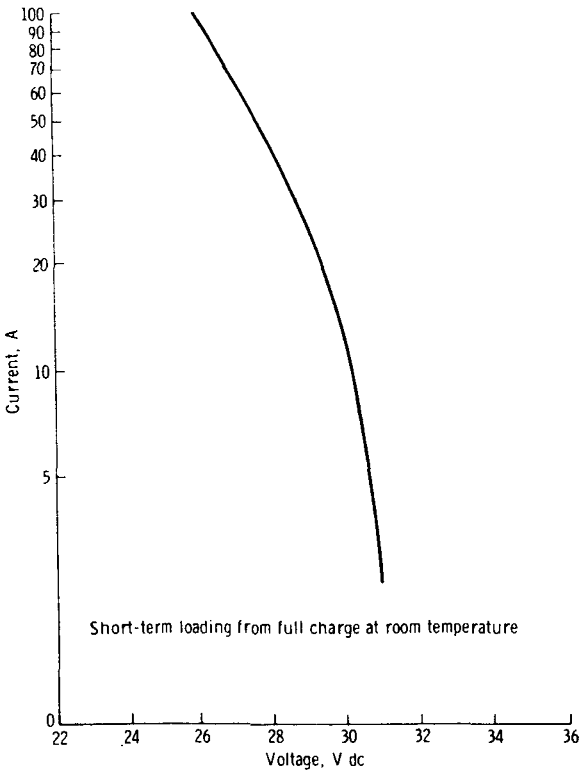


Figure 4. - Characteristics of the Apollo entry and postlanding battery.

Operational performance. - Performance was nominal on all unmanned missions. Each of these missions lasted less than 0.5 day; consequently, the spacecraft underwent very little zero-gravity time. However, in postflight tests of the eight entry batteries in the Apollo 6 spacecraft (spacecraft 020), internal cell shorting was discovered in auxiliary battery 2. Four of the remaining batteries developed cell shorts within one charge/discharge cycle. Soon afterward, two test batteries from spacecraft 007A at the Manned Spacecraft Center (MSC) failed in the same way, as did a battery being used to simulate the Apollo 7 mission performance at a contractor facility. The shorting was the result of zinc dendrites that had pierced the Permion 307 plate-separator material when cells were subjected either to slight overcharge or to slight overdischarge. Because this type of failure had not occurred before in these cells, it was concluded that the separator-material characteristic had changed; however, this conclusion could not be verified. In response to an urgent recommendation by the vendor and based on revised thermal requirements, a program was initiated to requalify the battery, using conventional cellulosic (cellophane) separator material that had an intermediate layer of plastic film.

A further indication that a change in separator material was required was observed in reduced battery-voltage characteristics after prolonged exposure of the battery to zero gravity (fig. 5). The very low voltage shown for the Apollo 7 batteries (relative to those of Apollo 8) was caused by the unexpected inability to recharge the batteries fully. Although the reduced voltage-current characteristic could not be reproduced in ground-based tests, it was concluded that hydrogen bubbles, formed by natural gassing at the negative plate, displaced electrolyte between the plates. This phenomenon is possible because the Permion separator is not absorptive and the hydrogen remains in place on the negative plates in zero gravity. Loss of electrolyte decreases conductivity, lowering the voltage characteristic. After qualifying the battery containing cellophane separators, one redesigned battery was flown on the Apollo 10 spacecraft; on subsequent flights, all three batteries were of this design. The improvement obtained from the redesign is shown in figure 6. The absorptivity of the cellophane eliminated the zero-gravity effect by retention of the electrolyte between the plates.

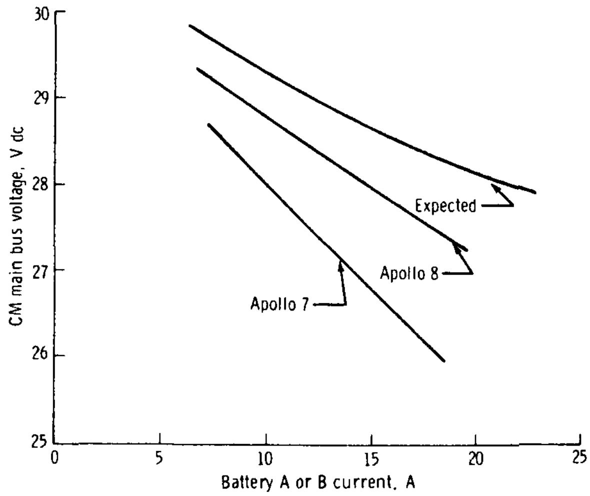


Figure 5. - Entry-battery flight-performance data at the time of CM/SM separation.

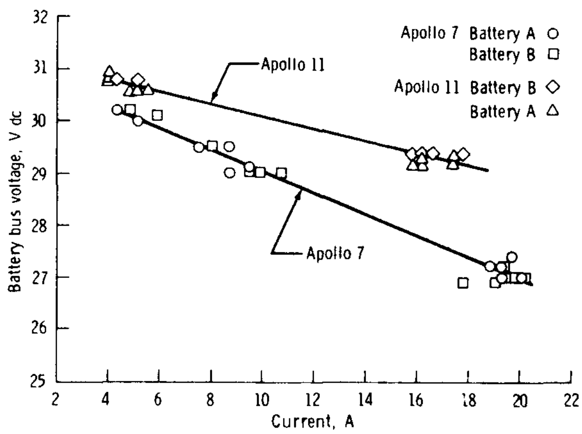


Figure 6. - Comparison of entry-battery characteristics at the time of CM/SM separation.

Mounting. - The entry and postlanding battery was hung from its top surface in the lower equipment bay (fig. 7). Active thermal control was not required because of the mild thermal environment of the CM cabin.

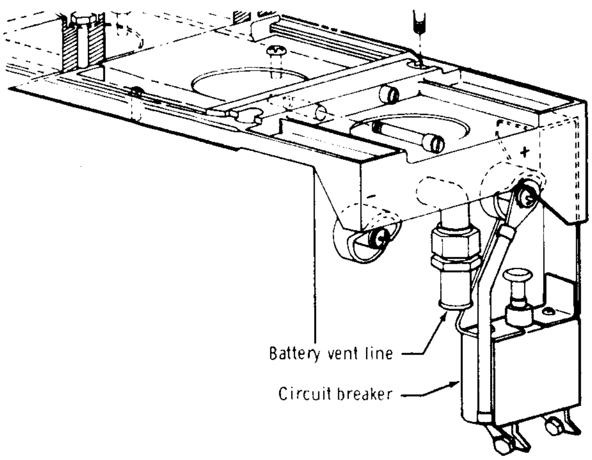


Figure 7. -The 100-ampere circuit breaker and entry-battery mounting adapter.

Pyrotechnic Battery

No significant problems were encountered in the development of the pyrotechnic battery. Qualification was completed in July 1965. The CSM pyrotechnic battery has been flown successfully on Apollo spacecraft from boilerplate 22 to Apollo 13.

Performance requirements. - The salient electrical requirements were as follows. The capacity was 75 amperes for 36 seconds at more than 20 volts at 60° to 143° F or for 15 seconds at more than 20 volts at 50° ± 3° F. Regarding cycle life, the battery had to deliver 6 cycles of 75 amperes for 36 seconds at any time within 36 days after activation. The battery also had to satisfy the preceding requirements when activated after storage for 1 year from the date of manufacture. The batteries were furnished in the dry, charged condition, and the electrolyte was supplied in separate containers. The weight and dimensions of this battery are given in table I. The separators were three layers of cellophane (only). The battery case was a hand-fabricated monobloc of G-10 circuit board, cemented together and lined with an epoxy cement. The cell relief valves, which had cracking pressures of 2 to 10 psig, were vented into a manifold space that was insulated from the intercell connectors; this space had a relief valve that had a cracking pressure of 30 ± 5 psig.

Mounting. - The pyrotechnic battery was hung from its top surface in the right-hand equipment bay by four bolts (two at each end) that threaded into the structure. Active thermal control was not required because of the mild thermal environment of the CM cabin.

Special Protective Devices

Before the spacecraft 012 fire, silicone-rubber potting was applied to all exposed terminal and cable-lug metallic surfaces after the installation of the CM flight batteries. (The potting was not used for test batteries.) As a result of the review after the fire, it was concluded that not only was the silicone rubber flammable but that the exposed terminals of the test batteries were a spark source if shorted. It was also concluded that additional overload protection was required for the long cable that connected each entry battery to its circuit breaker in the battery circuit-breaker panel (panel 250).

Entry battery. - To provide additional circuit protection, a 100-ampere circuit breaker was added at the battery. The battery mounting-adapter plate was extended downward to provide a bracket so that the 100-ampere circuit breaker could be connected to the battery by a very short cable. The use of silicone-rubber terminal potting was discontinued; instead, a polyimide shield was designed. The shield was attached to the mounting bracket and contained integrally molded baffles that blocked access of stray materials that might short circuit either the battery or the circuit-breaker terminals.

Pyrotechnic battery. - The use of silicone-rubber terminal potting was discontinued. Pod-shaped polyimide shields were designed. Each terminal had two shield halves: one shield half was mounted on the terminal behind the connecting cable lug; the other shield half, which enclosed the terminal and lug, snapped onto the first shield half. Because the pressurization-test plug on the terminal face of the battery physically interfered with the installation of the pods, the plug had to be removed.

Entry-Battery Charger

The salient characteristics of the entry-battery charger were as follows. The alternating-current (ac) input was 115 to 200 volts, 400 hertz, three phase; the dc input was 28 volts. The output is shown in figure 8. The entry-battery charger weighed 4.3 pounds (maximum), and the maximum input power was 55 watts ac and 84 watts dc. A simplified schematic of the charger electronics is shown in figure 9. No problems occurred during the development of this unit. Qualification was completed without complications in February 1966.

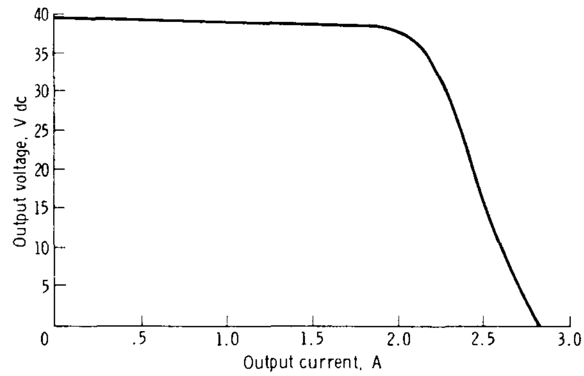


Figure 8. - Characteristics of entry-battery charger.

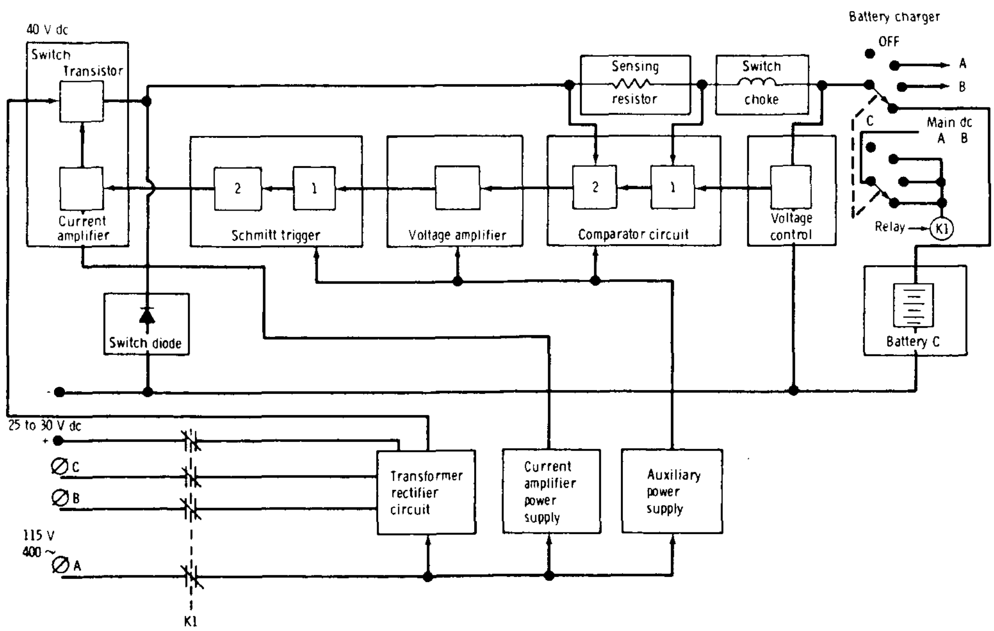


Figure 9. - Battery-charger functional diagram.

Operation. - The charger was operated manually by the crewmen. When a battery was to be charged, the charger was turned on, the battery was removed from the battery relay bus, and charging power was applied to the battery through a selecting switch and the applicable battery bus (fig. 10).

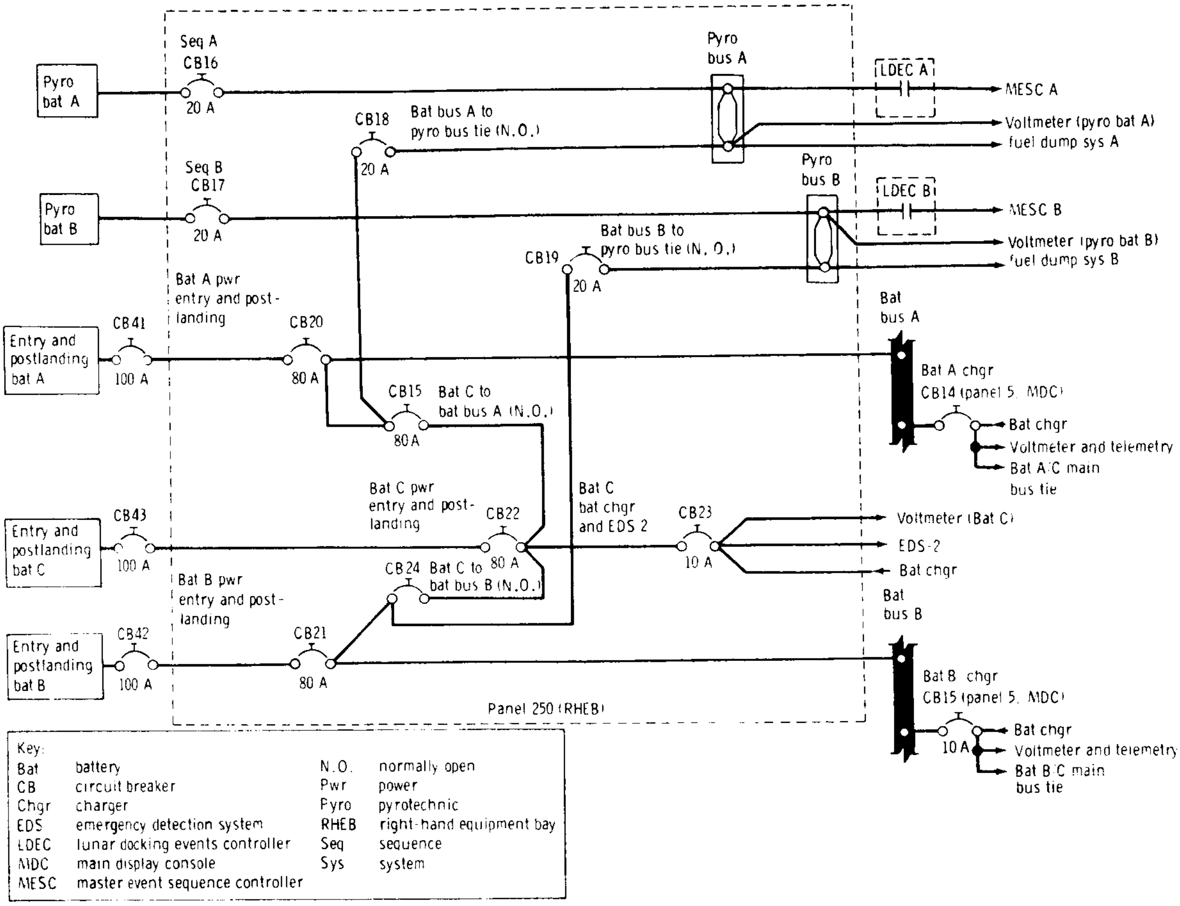


Figure 10. - Battery circuits.

Flight use. - Extensive testing and evaluation of the charger and entry battery were performed at a contractor facility and at the MSC to define the interdependent characteristics and rules for inflight charging timing and termination. These laboratory tests showed that the charger could recharge the batteries fully in a time approximately equal to $T = (A\text{-}h_o/1A) + 0.5$, where T is the time required for charging, in hours, and $A\text{-}h_o$ is the ampere-hours removed from the battery. This relationship was based on terminating charges when the charger current decreased to 0.4 ampere. Typical charging curves from these tests are shown in figure 11.

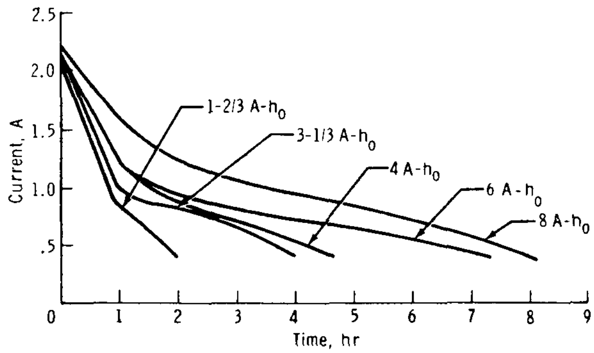


Figure 11. - Entry-battery charging current as a function of prior ampere-hours output.

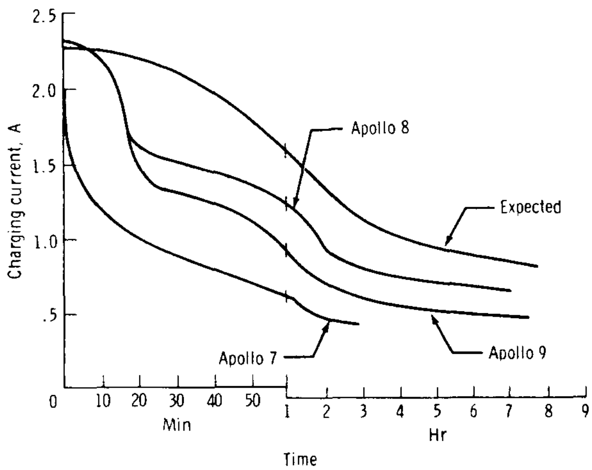


Figure 12. - Charging experience for approximately 10 A-h output.

The first formal test of the CM battery-charging system occurred during the flight of Apollo 7 (spacecraft 101). The batteries could not be recharged. A comparison of the expected and the actual performance of the Apollo 7 charging system is shown in figure 12. Postflight analysis showed that the line impedances between the charger and the battery (previously unevaluated) were sufficient to reduce significantly the charge voltage applied at the battery. The charger output voltage, although within specification limits, was on the low side of the allowable tolerances, aggravating the low-voltage condition. It was further postulated that the battery had a higher-than-expected internal resistance because of the gassing effects mentioned previously.

Reduction of line impedances was not feasible for subsequent flights because it involved breaking into already fabricated, complex wiring harnesses. Reduction of battery impedance by using absorbent cellophane separators was not feasible until the Apollo 10 mission. Although it was proven possible to drill a hole in the charger down to the adjustment screw of an output potentiometer (raising the charger output voltage) and then reseal the charger, the reliability of this adjustment could not be verified until after the Apollo 10 mission. Hence, for the Apollo 8, 9, and 10 missions, chargers were selected that had outputs on the high side of the specified voltage tolerances. Wherever necessary, chargers were removed from other command modules and installed in the next flight vehicle. This procedure permitted full recharging but required excessively long times (fig. 12). In the Apollo 11 and subsequent vehicles, chargers were used that had adjusted outputs permitting charging at the rates expected before the Apollo 7 mission. The small increase in voltage required to achieve this improvement is shown in figure 13.

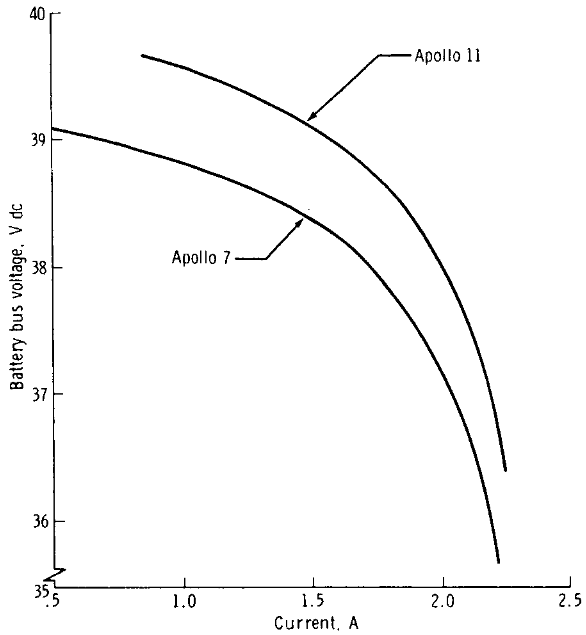


Figure 13. - Characteristics of entry-battery chargers on the Apollo 7 and 11 missions.

LUNAR MODULE BATTERIES

The LM battery complement consisted of two 29-volt, 296-A-h batteries in the ascent stage and four 29-volt, 400-A-h batteries in the descent stage. The LM originally was configured for fuel-cell power generation. Development of the LM fuel cell was begun with the contractor who developed the CSM fuel-cell system. The LM fuel cell was to have the following general characteristics.

1. Each of three fuel cells was to produce 32 kWh at 900 watts within the band of 27 to 36 volts.

2. The minimum load was to be 100 watts.

3. The peak load was to be 1125 watts for 1 hour.

4. The fuel cell was to accept transient load changes of 450 watts within voltage regulation.

5. Each fuel cell was to weigh 76 pounds (maximum).

6. The fuel cells were to be the open-cycle-operation type.

Regarding open-cycle operation, hydrogen was to be used both as a fuel and as a coolant, without recycling fuel. By this means, each fuel cell was not to dissipate more than 75 Btu/hr at 160° F to the structure. By mid-1964, significant technical difficulties were being experienced in both the CSM and LM fuel- cell developments. Additionally, the specified time from lunar lift-off to docking with the CSM was reduced more than 20 hours. The resulting decrease in energy requirements made a battery system feasible. As a consequence, a feasibility study was initiated by a contractor to evaluate the effect of converting to a silver-oxide/zinc alkaline primary-battery power source. This type of battery was considered to be reliable, and was available off the shelf relative to fuel cells, and had been flight proven during Project Mercury and the Gemini Program. In the study, it was shown that a battery system could be developed that could provide electrical power for a 35-hour lunar stay but that would result in approximately 100 pounds more of power-system weight than would the fuel-cell system. A decision was made to accept this penalty as being favorable over the developmental and reliability risks assessed against the fuel cells. Therefore, the LM fuel-cell program was reduced in April 1965 and terminated on June 30, 1965 (concurrent with the awarding of the LM battery subcontract). The battery-system configuration that was selected was composed of two 8.7-kWh batteries in the LM ascent stage and four 11.6-kWh batteries in the descent stage (a total of 63.8 kWh of installed energy).

Battery development began in June 1965 and culminated in battery qualification by March 1968. In addition to 20 monthly progress reports, special reports were issued regarding ascent- and descent-battery efforts on the following subjects.

1. High-temperature performance tests
2. Electrolyte volume and cell position
3. Thermal-runaway test
4. Short-circuit test (descent battery)
5. Battery-container weight-reduction study
6. Thermal- and ac-impedance tests
7. Specific-heat determination
8. Regression analysis of process and usage factors

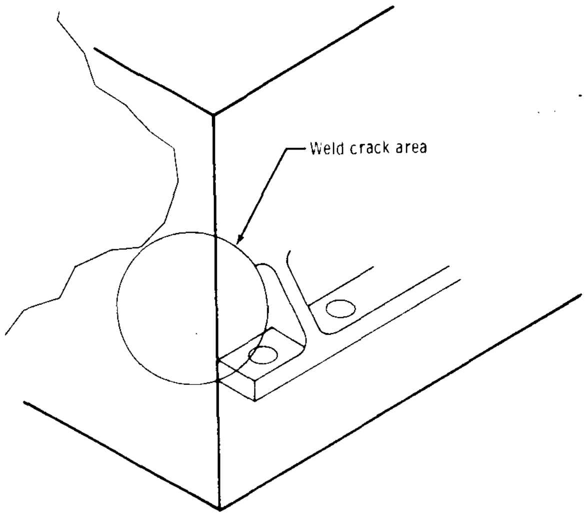
Capacity-prediction reports were issued on the following subjects.

1. Transient-voltage performance
2. Separator-material selection and evaluation tests
3. Systems simulation tests
4. Ascent- and descent-battery load sharing
5. Parametric testing of electrical characteristics
6. Qualification test
7. Postqualification test
8. Supplemental qualification test

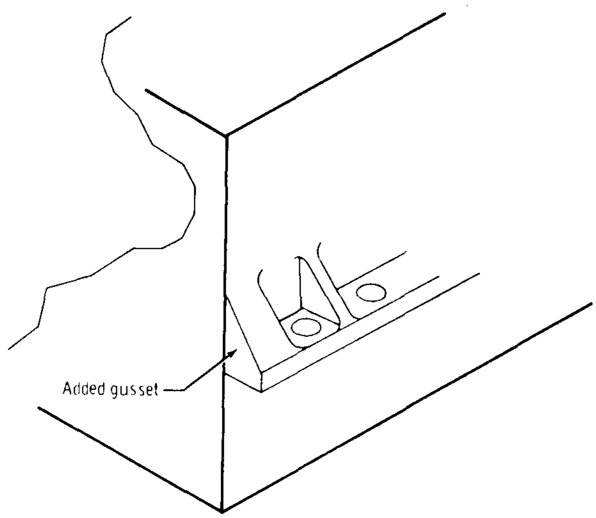
An integrated thermal/electrical computer model of the ascent and descent batteries (as if mounted on the cold rails and surrounded by the LM structure in space vacuum) was generated and was verified by the use of instrumentation on the LM-1 and LM-2 flights.

The only significant problem encountered in the development and qualification of the ascent batteries was undervoltage. This problem was encountered during large load transients related to single-battery aborts and abort-stage maneuvers. The problem was resolved by better definition of permissible maneuvers and by requiring "conditioning" discharges of 2.5 to 20 A-h at low loads immediately before mission stages involving possible large load increases on the ascent batteries. This procedure prevented undervoltage transients by a mechanism that is not fully understood at this time.

The only significant problem encountered in the development and qualification of the descent batteries was random-vibration-test failures of the magnesium-alloy battery container. Cracks occurred in the area where the ends and sides of the container were welded (just above the mounting rail). The failure mode was eliminated by adding a second gusset at the end of the mounting rail (fig. 14).



(a) Failure configuration.



(b) Corrected configuration.

Figure 14. - Failure and corrected configurations of the battery container.

The explosive-devices (ED) battery was identical to the CSM pyrotechnic battery, except that the pressurization port was retained in the ED battery. The ED battery was subjected to qualification tests (in addition to those required for CSM use) to verify the ability of the battery to withstand LM-vibration spectra and thermal vacuum in the cold-plated, thermally wrapped configuration. No difficulties relative to design or operation were encountered.

Ascent-Stage Batteries

The requirements for the ascent-stage batteries were as follows. The specified nominal capacity was 296 A-h within 28.0- to 32.5-volt limits and at rates equivalent to 350 to 1600 watts. The specified single-battery abort capacity was 268 A-h within 27.5- to 32.5-volt limits and at rates equivalent to 2200 to 3000 watts. The acceptance-test capacity (sample cells only) was 296 A-h at 50 amperes to a minimum of 1.41 volts per cell. It was required that the specified nominal and abort capacities be deliverable after idle stand for 30 days at 80° F. The required capacity on the acceptance test was determined after a 10-day charged stand at 95° F. The specified weight was a maximum of 131.7 pounds (activated), and the qualified weight was 123.7 pounds (activated). Dimensions and other pertinent data are summarized in table I. The battery was mounted in the vehicle in such a way that the plane of the plates was perpendicular to the natural gravity vector (earth or moon). Originally, it was required that the battery yield one 100-percent-capacity discharge and (with recharges) two 85-percent-capacity discharges. The requirement for the last two cycles was deleted because no charging is required in flight.

Descent-Stage Batteries

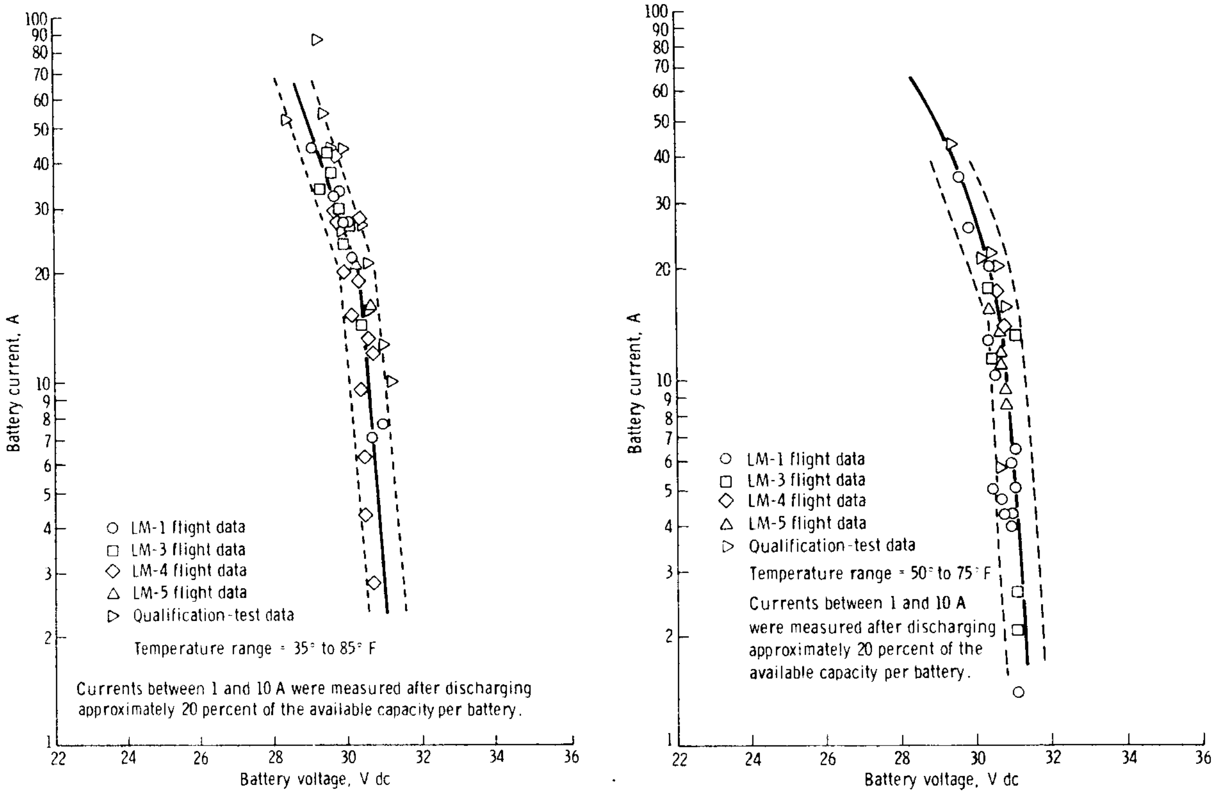
The requirements for the descent-stage battery were as follows. The specified nominal capacity was 400 A-h within 28. 0- to 32. 5-volt limits at a 25-ampere discharge rate. A tapped connection was provided from the 17th cell of the battery to prevent application of overvoltage to the LM systems at low loads when the battery was at near-full capacity and could deliver the high silver dioxide voltage on discharge at low currents. In a typical mission, the 17th-cell tap was used to supply small loads for heaters in the LM only from prelaunch until after CSM transposition and docking. After this, these loads were supplied from the CSM through the CSM/LM umbilical until LM powerup, at which time the LM loads were high enough to depress the voltage to acceptable levels. Use of the full battery or the 17th-cell tap was controlled manually by the LM crewmen. The specified contingency (one of four batteries failed) capacity was 389 A-h within 28. 0 to 32. 5 volts at rates equivalent to 15 to 1330 watts. The contingency mission requirements resulted in higher individual battery loadings and a shortened lunar staytime. The acceptance-test (sample cells only) capacity was 400 A-h at 25 amperes to a minimum of 1. 41 volts per cell. The specified capacities were required to be delivered after the same wet-stand time periods and temperatures as were used for the ascent batteries. The specified weight was 139. 6 pounds and the qualified weight was 132. 7 pounds. Dimensions and other pertinent data are summarized in table I. The mounting orientation and the cycle-life information were the same as for the ascent-stage batteries.

Explosive-Devices Battery

The ED battery was identical to the CM pyrotechnic battery except that the pressurization port was retained in the ED battery. The use environment in the LM differed from that in the CSM; thus, the LM ED battery was mounted on cold rails outside the pressurized cabin and was wrapped in a thermal blanket to limit temperature extremes. The ED battery supplied power to the explosive devices that enabled such functions as staging, landing-gear extension, helium- and fuel-tank pressurization, and so forth.

Flight Performance

The LM ascent, descent, and ED batteries performed in accordance with specifications and satisfied the electrical-energy requirements for all LM flights. The voltage-current characteristics of these batteries are shown in figures 15 and 16. The performance of these batteries was noteworthy during the rescue phase of the aborted Apollo 13 mission. The batteries supplied total vehicle power for 83 hours under conditions for which they had not been qualified or tested; that is, low electrical loads (350 watts compared with a nominal 1000 watts), continuous zero gravity while loaded, and continuous low temperature (approximately 37° F). Telemetry measurements indicated that two of the descent batteries were used 5 to 6 percent above the specified capacity. A CM entry battery was charged from LM battery power in preparation for CM/SM separation and CM entry.



(a) Ascent battery.

(b) Descent battery.

Figure 15. - Lunar module battery characteristics.

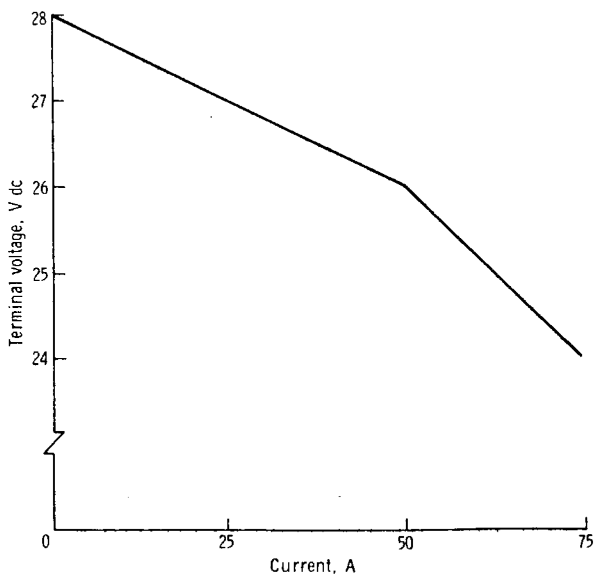


Figure 16. - Characteristics of the pyrotechnic and ED battery.

COMMAND MODULE BATTERY-FAILURE EXPERIENCE

Entry and Postlanding Battery

With the possible exception of the Apollo 6 auxiliary battery 2, for which no telemetry data are available for verification of an inflight malfunction ("Operational performance," p. 4), no inflight failure of an entry and postlanding battery has occurred. Ground-based failures include the Permion-separator failures (previously discussed) and the following problems.

Low voltage occurred after vibration tests during qualification testing. The voltage decrease was caused by groundpaths that resulted from electrolyte leakage from cracked cell cases. This problem was solved by reducing the cell-relief-valve pressure from 50 to 30 psig and increasing the internal corner radii in the molded cell cases (that is, making the corners thicker and stronger).

An incremental qualification test was performed on the entry and postlanding battery to determine if it could withstand the increased vibration levels of the SM for use without shock mounting in spacecraft 009. The battery failed because of cell-case cracks; thus, it was shock mounted for flight.

The two terminal-end cells of a battery in a laboratory became grounded by electrolyte leakage through cell-case cracks located directly behind the battery vent elbow. This failure was the result of mishandling the elbow, which caused it to be jammed backward into the cell cases.

Pyrotechnic Battery

No inflight failure of a CM pyrotechnic battery has occurred. None of the ground failures was significant to battery design or performance capabilities.

Battery Charger

No recorded failures of the CM battery charger have occurred. Two chargers were damaged during laboratory testing because of incorrect connection of ac input power.

LUNAR MODULE BATTERY-FAILURE EXPERIENCE

Many LM battery failures have occurred; therefore, the failures are summarized in broad groupings.

Qualification-Test Failures

Qualification-test failures are summarized as follows. Descent-battery weld cracks occurred during the random vibration testing; the solutions to this problem included the following.

1. On the basis of data obtained from vibration testing of an LM test vehicle, the vibration levels were reduced.

2. The inspection procedures were revised.

3. A gusset was added on the mounting rail.

Descent-battery cell-case cracks occurred after a -20° to 160° F thermal shock (unactivated) because of differential thermal expansion of the cell case and thermal fins in the battery case to which the cells were potted. The following changes were made.

1. The lower temperature was changed from -20° to 20° F, based on the ability to control shipment and storage temperatures easily within this limit.

2. The cell-case walls were lubricated to prevent adhesion to the thermal fins.

An ascent-battery voltage transient occurred below 28.0 volts (27.6 volts) when the thermal-vacuum discharge load was changed from 50 to 80 amperes. The specified voltage minimum was reduced to 27.5 volts, based on a review of the capabilities of the voltage-limited equipment operated from the battery. The ED battery was internally grounded because of electrolyte leakage (caused by improper addition of electrolyte during activation). The activation procedure and equipment were improved.

Acceptance-Test Failures

No ED-battery failures have occurred. However, two ascent-battery failures occurred: one because of low insulation resistance between the battery terminal and the case, caused by inadequate drying of potting material, and the second because of broken cases on sample cells, caused by vibration overtest. As of December 1969, 26 descent-battery failures had occurred. All failures except one were caused by improper processing, improper handling during fabrication, or improper testing. The one exceptional failure (low capacity) is still unexplained.

Failures of Flight Batteries Used in Vehicle Testing

All but one of the 15 reported ED-battery failures were caused by improper usage (that is, use beyond specified life or improper handling). In the one exception, the battery had a groundpath from terminal to case that could not be analyzed because the battery was discharged completely through an external load after discovery of the ground. No significant ascent-battery problems occurred. Six descent-battery failures occurred because of cells that leaked electrolyte when the batteries were stood on end during activation or while the batteries were in a vehicle at the Kennedy Space Center.

The leakage was attributed to excess free electrolyte caused by a reduction of plate porosity in long-term storage at temperatures between 50° and 90° F. Requirements for refrigerated storage were imposed. Also, a battery used in LM-4 tests had low voltage in one cell after discharging 320 A-h in the laboratory before being returned to the vendor for use as scrap. A lump of material found on a plate lug had punctured a separator, causing loss of cell capacity by means of a short between the positive and negative plates. Closer inspection by the vendor and hand cleaning of plate tabs were instituted to eliminate this type of failure.

Failures of Batteries Prepared for Flight

No ED-battery failures occurred. The one ascent-battery failure was caused by an apparent crack in one plastic cell cover at the flow mark near the fill screw. The flow mark was made to leak air slightly after extreme pressure cycling during the failure investigation. There had been no leakage before cycling. An inspection point was added to the test and checkout procedure to ensure that possible "leakers" would not get past the battery shop. One descent-battery failure occurred because a cell was installed in the reversed-polarity position in a battery, lowering the battery voltage by the equivalent of two cells. Three failures occurred because activated batteries had open-circuit voltages lower than required (37.0 volts) by 0.005 volt or less. The open-circuit-voltage criterion was reduced to 36.98 volts. No battery failures occurred in flight. The incident involving battery 2 on the Apollo 13 flight (following section) did not diminish the ability of that battery to continue to deliver power in a normal manner.

APOLLO 13 BATTERY EXPERIENCE

During the Apollo 13 cryogenic-oxygen-system failure and the subsequent inability to operate the CSM fuel cells, the CSM first obtained power (approximately 400 watts) from one entry and postlanding battery. Subsequently, the CSM/LM assembly was switched onto LM-battery power. The spacecraft was powered down to 350 watts. Intermittent spikes of an additional 70 watts occurred as heaters cycled on and off. Approximately 97 hours into the mission, the LM pilot reported "a little thump" in the LM descent stage. Within a few minutes, he reported snowflakes in the area of descent-stage quad IV, in which descent batteries 1 and 2 were mounted. From a review of the telemetry data, it was concluded that, at the time of the reported thump, an apparent short circuit of approximately 100 to 150 amperes occurred between the battery 1 and 2 electrical control assembly and the battery 2 ground. The battery 2 malfunction light subsequently cycled on and off for approximately 1 day, even though all batteries performed nominally for the remainder of the mission. This malfunction-light cycling was attributed to a malfunction of the battery 2 overtemperature-sensing circuitry. The battery temperature was approximately 37° F, and the overtemperature sensors do not activate at temperatures less than 140° F.

From the foregoing information, it was inferred (but never proved) that electrolyte escaped from some cells in battery 2, was displaced to the terminal end of the battery by the transearth descent-engine burn, and shorted a positive terminal to the battery case (mounted to a grounding structure). This short could have ignited the hydrogen and oxygen mixture under the battery cover, causing the thump and an efflux of electrolyte from the battery, which produced snowflakes. The capability of such a short to carry the estimated fault currents was verified by a battery test performed by a contractor.

On the premise that the postulated failure sequence might be valid and because the quantity of electrolyte in the descent-battery cells was considered to be higher than was necessary to obtain full battery capacity on discharge, a program of corrective redesign and investigation was initiated. The significant findings and design improvements are summarized in the following sections.

Findings

Twenty cubic centimeters (5.6 percent of 360 cubic centimeters) of the electrolyte in a descent-battery cell could be removed from a cell without affecting its ability to deliver capacity (420 to 450 A-h using 360 cubic centimeters compared with the specified 400 A-h). This reduction of electrolyte volume became a design change.

Cells that had plates more than 280 days old (from completion of the plate-making processes until the day of activation), whether stored refrigerated or unrefrigerated, absorbed significantly less electrolyte than did younger cells and tended to leak. Cells that had plates younger than 280 days caused no problems. No work was done or is planned to identify and define the mechanism of this plate behavior.

While on discharge, silver-oxide/zinc primary cells can evolve significant amounts of oxygen. Volumes greater than 50 percent of the effluent cell gas were measured. Work is planned to identify and define the basis for this behavior, which is believed to be related to the plate-formation process, formation-bath temperature and composition, drying conditions, and plate storage time and temperature before activation.

Design Improvements

The inside surfaces of the battery container and lid were coated with two layers of insulating, alkali-resistant paint. Individual cell vents were manifolded together and were vented external to the battery through a relief valve. This design provided a path overboard of the battery for any electrolyte that leaked from the cell vents and substantially reduced the volume of any detonatable gas mixtures that came from inside the cells. An analog temperature sensor was removed from the battery because it was no longer used in flight and because it complicated potting of the inside of the battery. To have both ascent-battery terminals at the same end of the battery (which was a single row of 20 cells), the terminal lead from the cell at the far end of the battery ran the full length of the inside of the battery and emerged at the terminal end. In lieu of this arrangement, the design was revised to have a terminal at each end of the battery. This redesign eliminated both an internal potting complexity and a possible source of internal shorts. The internal battery potting, which had previously covered all current-conducting parts but had not adhered reliably to some parts, was revised so that it adhered properly and extended over the entire top surfaces of all cells. This potting reduced the possibility of internal grounds caused by escaped electrolyte.

As an improvement to the overall Apollo spacecraft, an LM descent battery was added to subsequent service modules (fig. 17). This battery could have provided 12 kWh of emergency energy and was designated the SM auxiliary battery. It could be connected to the CM main buses through the distribution system for fuel cell number 2.

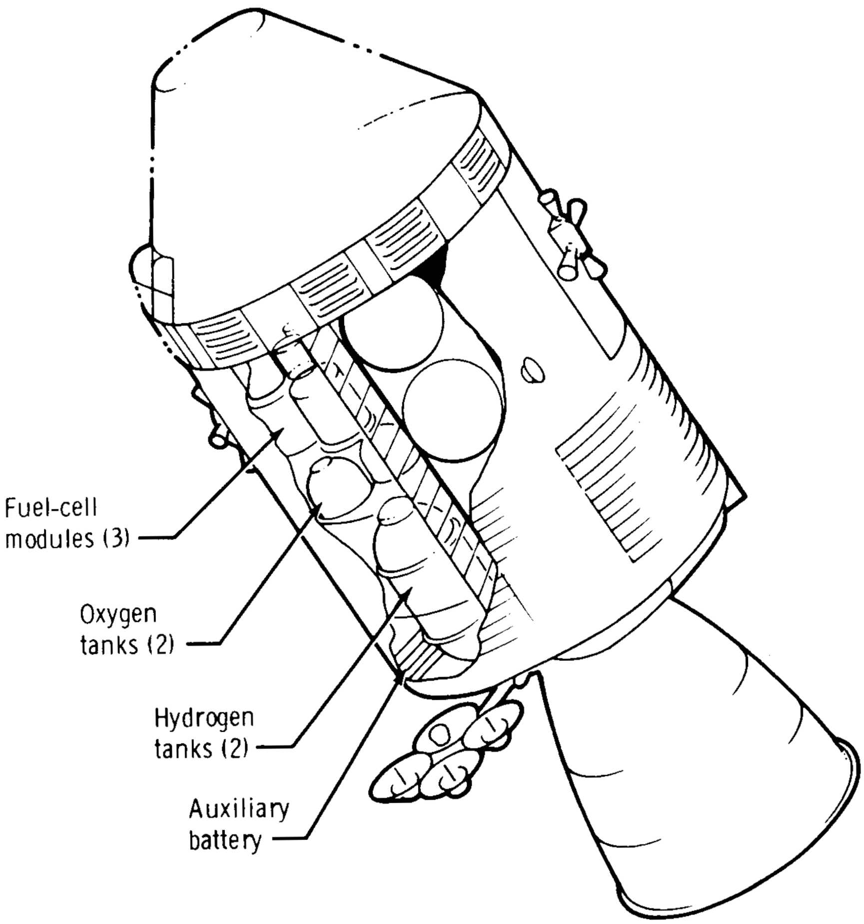


Figure 17. - Location of auxiliary battery in sector IV of the SM.

CONCLUDING REMARKS

The design features and methods of development, qualification, acceptance testing, activation, and preinstallation checkout of all the Apollo command-service module and lunar module batteries have prevented inflight failures, with the possible unproved exception of auxiliary battery 2 on the Apollo 6 spacecraft.

Two major conclusions, both based on the Apollo 7 problems, are made. First, untried and novel components (such as new separator materials) should not be used in batteries unless no other feasible, standard alternative exists. Second, each spacecraft system design (such as charging circuits) must be verified quantitatively, either in a flight vehicle or in an accurate simulation, to avoid unexpected developments in flight.

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