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APOLLO EXPERIENCE REPORT -
COMMAND AND SERVICE MODULE
ELECTRICAL POWER DISTRIBUTION SUBSYSTEM

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16. Abstract

A review of the design philosophy and development of the Apollo command and service modules electrical power distribution subsystem, a brief history of the evolution of the total system, and some of the more significant components within the system are discussed. The electrical power distribution primarily consisted of individual control units, interconnecting units, and associated protective devices. Because each unit within the system operated more or less independently of other units, the discussion of the subsystem proceeds generally in descending order of complexity; the discussion begins with the total system, progresses to the individual units of the system, and concludes with the components within the units.

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SUMMARY

The Apollo command and service module electrical power distribution subsystem was required to receive electrical power from any combination of seven direct-current sources (three fuel cells, three entry batteries, and one service module battery) and to condition, control, and deliver the power to the appropriate loads. To satisfy these requirements, the system was designed to perform the following functions.

1. Distribute power to the various locations through a system of interconnecting cables

2. Protect the wiring if a load malfunction occurred

3. Prevent additional damage to the vehicle if a wire became overloaded

4. Protect the various power sources from a malfunction in the loads or the wiring

5. Protect each power source from a malfunction in any other power source

6. Convert direct current to alternating current to satisfy load requirements

Major developmental effort was expended on the static inverter, the most complex electrical unit in the system; therefore, the history and design configurations of this device are discussed in detail in this report. Conversely, the motor-driven switch (which, electrically, is a relatively simple device) proved to be quite troublesome because of its complicated mechanical design and widespread use throughout the command and service module. It is discussed in this report, and special emphasis is placed on problems that developed during the Apollo Program. The developmental efforts and problems with other allied equipment are also reviewed.

Although numerous minor changes were made as a result of real or anticipated problems, the overall system performance was satisfactory throughout the program. Nevertheless, several recommendations worthy of consideration in the design of any new system are apparent from a review of the entire effort. Most of the recommendations were made as a result of many incidents rather than a particular problem and

are not specifically addressed in the text of the report, but are discussed more fully in the recommendations and concluding remarks.

INTRODUCTION

Throughout the active life of each Apollo vehicle, beginning with the initial power-up and continuing through checkout, flight, and postflight testing, the command and service module (CSM) electrical power distribution subsystem (EPDS) was used almost continually. Three fuel cells located in the service module (SM) provided the primary source of power for the major portion of a mission. The fuel cell output was supplemented by two of the three 40-ampere-hour command module (CM) entry batteries during high-load phases of the mission and was backed up by a 400-ampere-hour battery in the SM (added after the flight of Apollo 13), which could be used if a fuel cell failed. The electrical configurations of the spacecraft during various phases of a normal mission are described in the following paragraphs.

Near the end of the launch countdown period, the fuel cells were activated and remained active until CM/SM separation. A few minutes before lift-off, two entry batteries were connected to the main buses to provide power to the CM in the event of an abort. After the spacecraft was in orbit, the entry batteries were disconnected, recharged, and used only to supplement fuel cell power during service propulsion system (SPS) burns. Between burns, the load normally varied between 60 and 80 amperes (well within the capability of the fuel cells); however, when gimbals were operated during SPS burns, the load could reach 120 amperes, requiring the additional capacity of the two entry batteries.

All three entry batteries were switched to on-line status during preparation for entry and became, at CM/SM separation, the sole source of electrical power for the vehicle. During descent, the load was approximately 60 amperes and continued near that level until the vehicle was 800 feet above water and the main buses were deenergized. After this time, only the recovery aids and various small loads on the postlanding bus required electrical power.

DESIGN CONSIDERATIONS

The EPDS of the Apollo CSM was designed to control, condition, and deliver nominal 28 V dc and 115 V ac power. These tasks had to be accomplished with maximum reliability, flexibility, and safety yet remain within weight and volume constraints. To satisfy these goals, the system was designed around two main buses in the direct-current system and two buses in the alternating-current system.

The direct-current system (fig. 1) accepted power from three fuel cells and three entry batteries and distributed this power to two main direct-current buses. Loads on the buses were divided into the following three categories according to criticality and power source.

1. Nonredundant essential loads were fed from both buses through isolation diodes.

2. Redundant essential loads were arranged so that loads were fed from different buses.
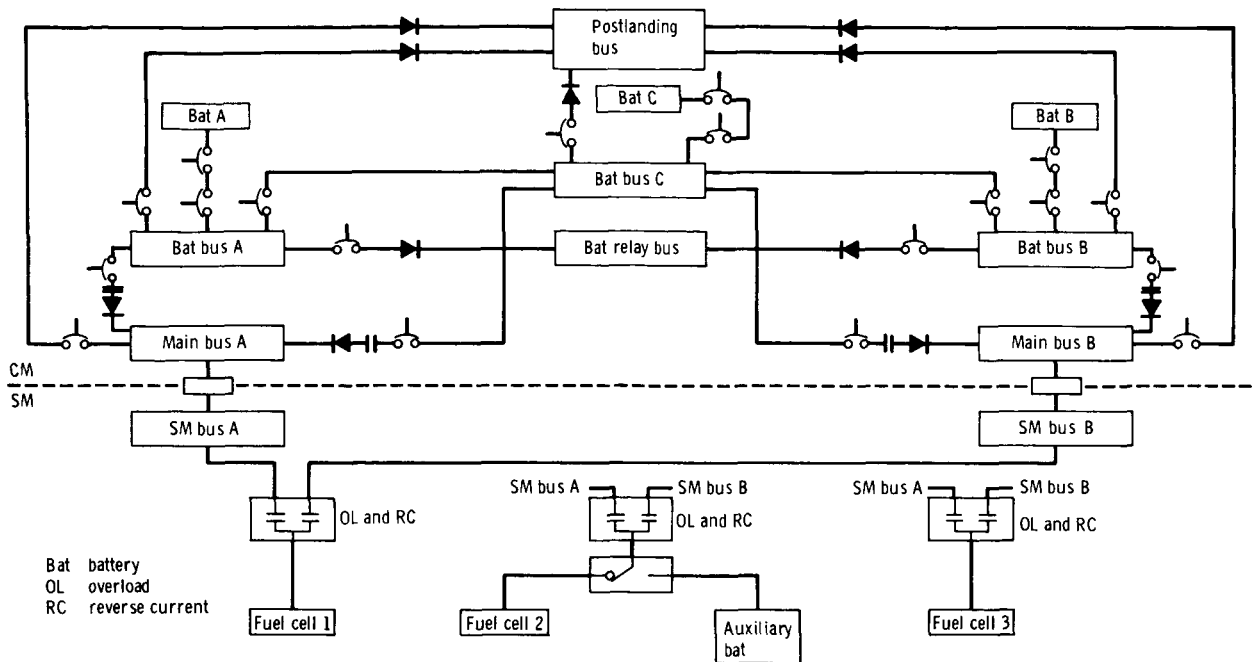
Figure 1. - Structure of direct-current buses.

3. Nonredundant nonessential loads were connected to the buses, as required, to equalize the loading on each bus.

In addition to the inherent reliability provided by this mechanical configuration, mission success was further ensured by rigid quality control and careful inspection by qualified inspectors during all phases of equipment fabrication. Each function was checked several times after the system was installed in the vehicle. During flight, the crew monitored the essential parameters by display and control read-outs, and ground controllers monitored telemetered data.

Typical production samples of each component or assembly were selected at random, subjected to qualification tests, and reevaluated. These tests, designed to simulate as nearly as possible the rigors of mission environments (plus an appropriate safety factor), were performed on each component or assembly. Results were then evaluated to determine the suitability of the equipment for flight. In addition, each component or assembly was subjected to acceptance testing.

The EPDS was designed to increase reliability with maximum flexibility. Controls in the alternating-current system were provided to allow any inverter to be placed on either or both alternating-current buses; however, an interlock was provided to prevent connecting the inverter outputs in parallel. Similarly, any or all of the six normal direct-current sources (three entry batteries and three fuel cells) could be connected to

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either or both main buses. Although the pyrotechnics were not a part of the EPDS, provisions were made for using main entry battery power to fire the explosive devices if the pyrotechnic batteries failed.

Together with the normal precautions taken in any electrical installation to protect the user from shock and the hardware from physical damage, the primary concerns in the design of the EPDS were for mission safety and for the protection of the power sources from downstream faults and fault propagation if problems did arise. Diodes, circuit breakers, fuses, and reverse and overcurrent sensors were used to protect power sources.

Nonflammable, flame-retardant, and fire-resistant materials were used wherever possible. Both electrical and mechanical means were used to prevent the propagation of a fire. This was accomplished by coating the unprotected area behind the panels and exposed hardware with flame-retardant coatings, by using potting components, and by sealing enclosures. To prevent heat damage to adjacent wires if an overload occurred, each wire was selected to be compatible with its associated circuit breaker. Electrical cables were also routed through covered trays or behind closeout panels to protect them from physical damage.

Because components of the EPDS were distributed throughout the CSM, the system was exposed to a variety of environments including water, urine, salt, and dust. Atmospheric pressure varied from sea level to the vacuum of deep space, and temperatures varied widely. In addition, equipment in the CM pressure vessel had to be capable of maintaining mechanical integrity when exposed to a 78g impact shock if touchdown occurred on land. Verification that the equipment could withstand these environments was determined by qualification testing.

All the precautions mentioned previously were considered during the design, fabrication, and testing of the system to ensure crew safety and mission success. To improve the performance of the system further, the EPDS design was continuously reviewed, and each change or proposed change was evaluated in terms of reliability. The performance of the system during test and flight was reviewed to determine if any problems existed that might require design changes.

A formal, computer-aided program was devised to identify "sneak circuits." This program provided for the identification of current paths that were not readily apparent from a review of the electrical schematics but might cause undesirable results. Some of these evaluations resulted in procedural changes, component replacement, and system modifications.

SYSTEM DESCRIPTION

Direct-Current Distribution

The basic design of the direct-current distribution system (fig. 1) consisted of two buses (main bus A/SM bus A and main bus B/SM bus B) that were isolated from each other and energized by three fuel cells or three entry batteries. Redundant loads were connected to separate buses. Nonredundant critical loads were fed from both buses through diodes or double-throw switches to provide circuit isolation. Nonredundant noncritical loads were tied to either bus, as required, to equalize bus loading.

Entry batteries A and B supplied power to individual battery buses (battery buses A and B) that supplied logic and switching power to various controls. A third battery (entry battery C) supplemented A and B on the main buses during entry and after landing. Both battery buses supplied power to the battery relay bus, which distributed power to essential control functions.

Three fuel cells in the SM provided the bulk of the power for a normal mission. Any fuel cell could be connected to either or both of the main buses, but the normal configurations were with fuel cells 1 and 2 on main bus A and with fuel cell 3 on main bus B. After a cryogenic supply failure, which resulted in the loss of all three fuel cells on Apollo 13, an auxiliary battery was added in the SM to supplement the remaining power sources if one or more fuel cells failed.

A battery charger was provided to recharge the entry batteries from fuel cell power. The charger used both alternating current and direct current to apply a current-limited charge to one battery at a time. After being used, each battery was recharged as soon as practical to provide maximum power for entry or any contingency.

As time passed, the need for added flexibility of power sources became apparent, resulting in the addition of circuit breaker (CB) 15 and CB24 (fig. 2) to make it possible
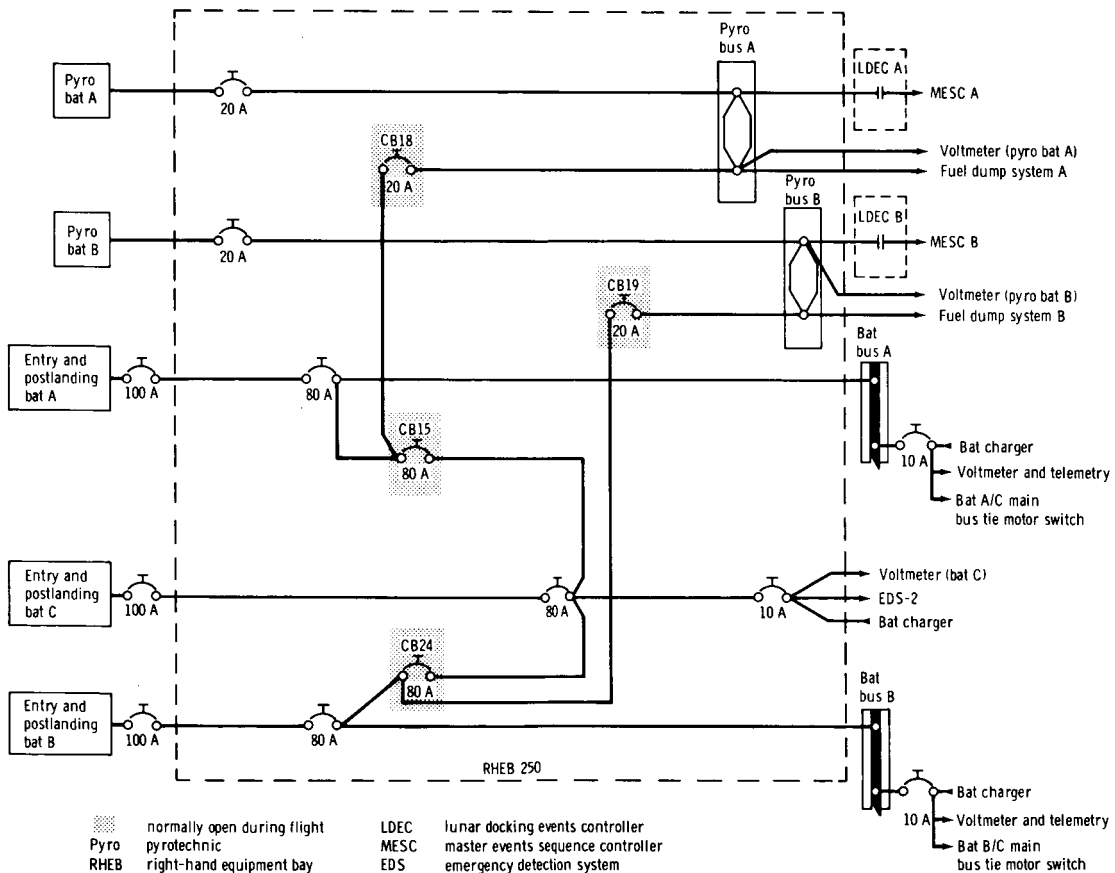

Figure 2. - Battery circuits.

to put any battery on any bus in the power system. In addition, CB18 and CB19 were added to permit activation of a pyrotechnic bus by an entry battery if a need occurred. Although many other minor changes were made in both design and procedures, the system and system components remained basically the same as those flown in 1966 on spacecraft 009, the first Block I vehicle launched from the NASA John F. Kennedy Space Center. The more significant of these changes are reviewed in the following discussion.

As mentioned previously, the power distribution system was designed around a network of buses, each with loads and sources peculiar to the criticality of the application. The direct-current bus structure consisted of nine interconnected buses. Although the SM buses provided power to some SM loads, they were extensions of the main buses and served primarily to route power from the fuel cells to the CM.

The large circuit breakers on the output of the entry batteries were directly attached to the battery-mounting hardware to protect the wires leading to the first control panel. Overload and reverse-current sensors for each fuel cell (fig. 3) served a similar purpose in the SM.
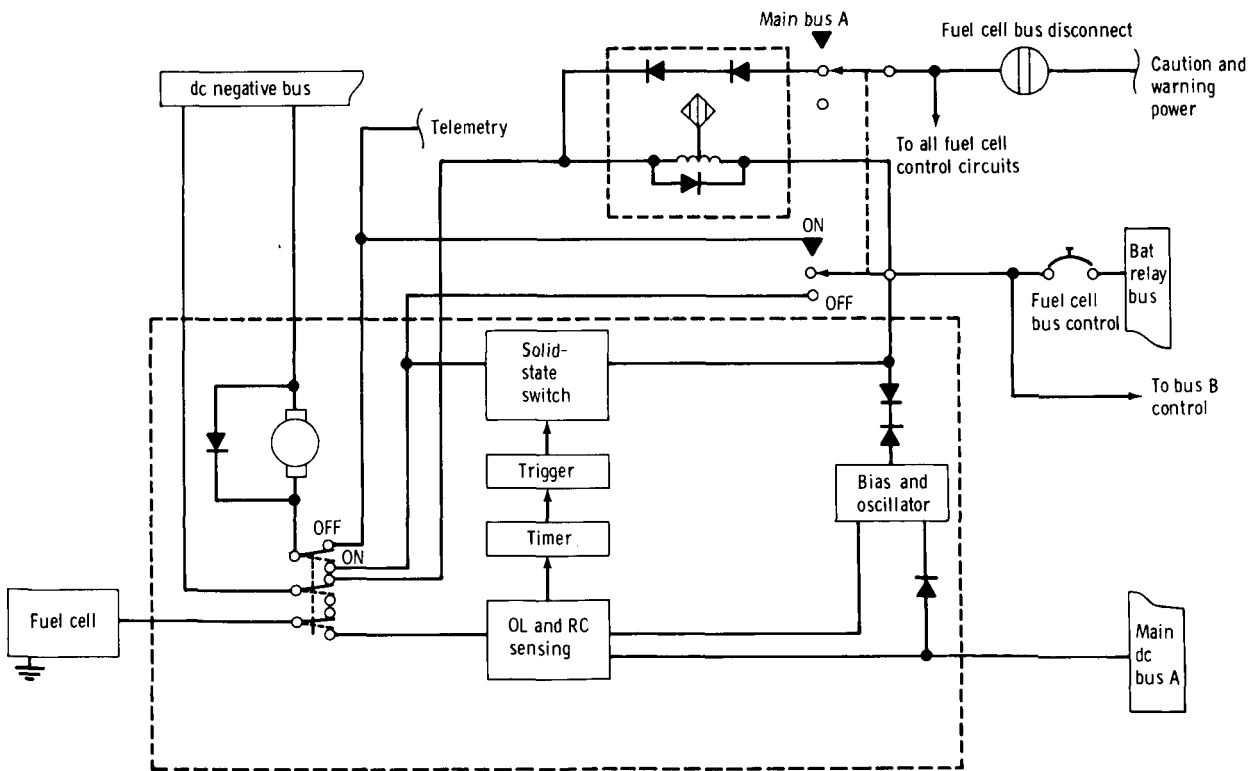
Figure 3.- Fuel cell control circuit.

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A timing circuit (fig. 4) in the reaction control system (RCS) provided automatic switching to connect the entry batteries to the main buses in the event of a pad abort. This feature allowed battery power to be conserved by operating the vehicle from ground power until just before lift-off. If an abort was necessary, crew action was not required to provide power to the CM after separation from the SM.

Grounding

To minimize electromagnetic interference (EMI) from structural current, a single-point ground was used for the direct-current return, alternating-current neutral, pyrotechnic-initiating circuit returns, and shield grounds. This design is shown in figure 5 with all grounds returning to the vehicle ground point (VGP). When the CM was docked with the lunar module (LM), all current flowing from the CSM to the LM was returned to the VGP through the CM/LM umbilical, thereby maintaining the single-point ground configuration. Individual buses were provided to separate the power and signal grounds from the LM and CM until the grounds were tied to the VGP.
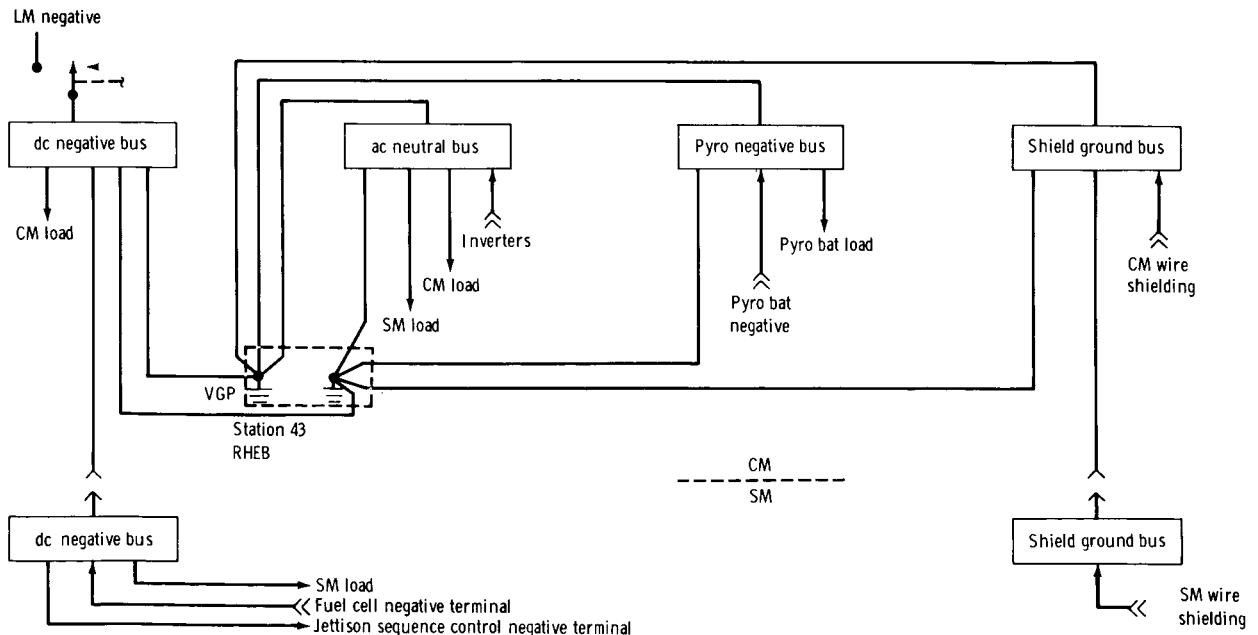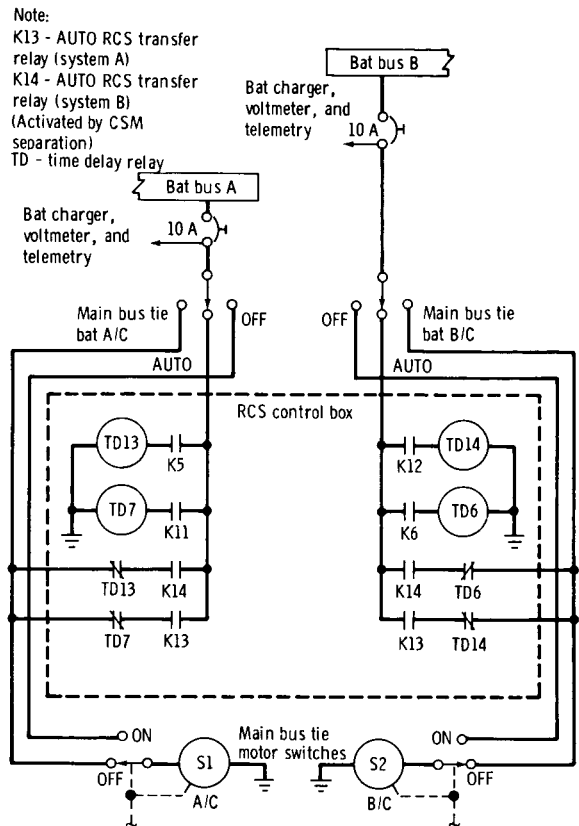

Figure 4. - Entry battery/main bus tie control.

Figure 5. - Vehicle ground point.

Alternating-Current Distribution

Three-phase 400-hertz alternating current was delivered through two four-wire buses that could be energized by one or two of the three static inverters (fig. 6). Each inverter had sufficient capacity to supply all alternating-current power required by the vehicle. This could be accomplished by paralleling both alternating-current buses to a single inverter, or each bus could be energized by individual inverters; however, the inverters could not be paralleled without causing damage to the inverters.

Circuits were provided in the alternating-current power control box that prevented parallel operation of the inverter outputs if improper procedures were followed. However, this switching system had defects involving a race between motor switches that could cause the loss of all alternating current to the spacecraft. This problem was overcome by rewiring the control switches on a priority system so that no switch race was possible.

Figure 6. - Alternating-current power distribution.

During Block I checkout, it was discovered that a portion of the guidance and navigation (G&N) information was derived from a phase shift between two signals, each of which originated from a different alternating-current bus. For the information to be valid, each bus had to be in phase with the other. Consequently, a phase-synchronization box was designed for Block I manned spacecraft. The G&N system was modified for Block II to eliminate the need for buses to be in phase. Because phase synchronization was no longer a requirement, the phase-synchronization box was deleted from Block II spacecraft.

Sensors

Sensors monitored the main direct-current buses for low voltage (fig. 7) and signaled the caution and warning (C&W) system if the voltage level fell below 26.25 volts. Alternating-current buses were monitored by sensors (fig. 8) for low voltage, overload, and overvoltage. If either the load or voltage went out of limits, a C&W signal was initiated; however, of the three anomalous conditions, only an overvoltage condition required corrective action to be taken faster than a crewmember could react to the situation. Therefore, an overvoltage condition resulted in automatic disconnection of the faulty inverter.
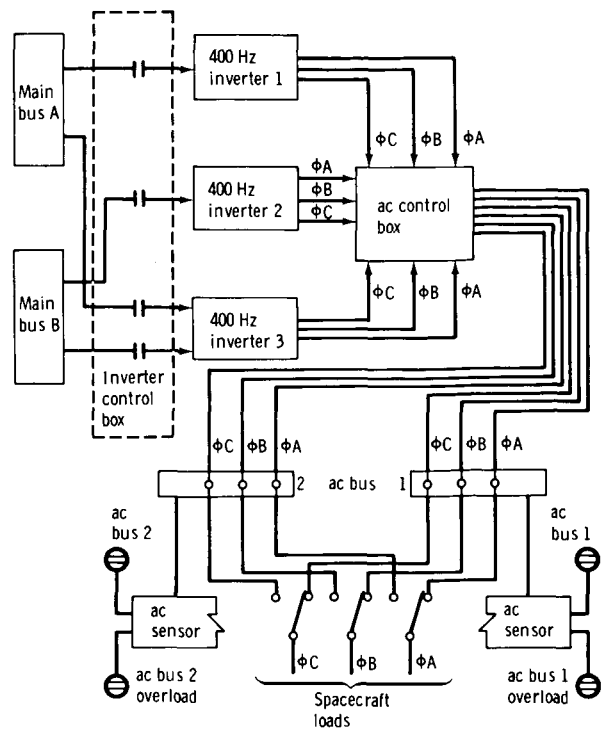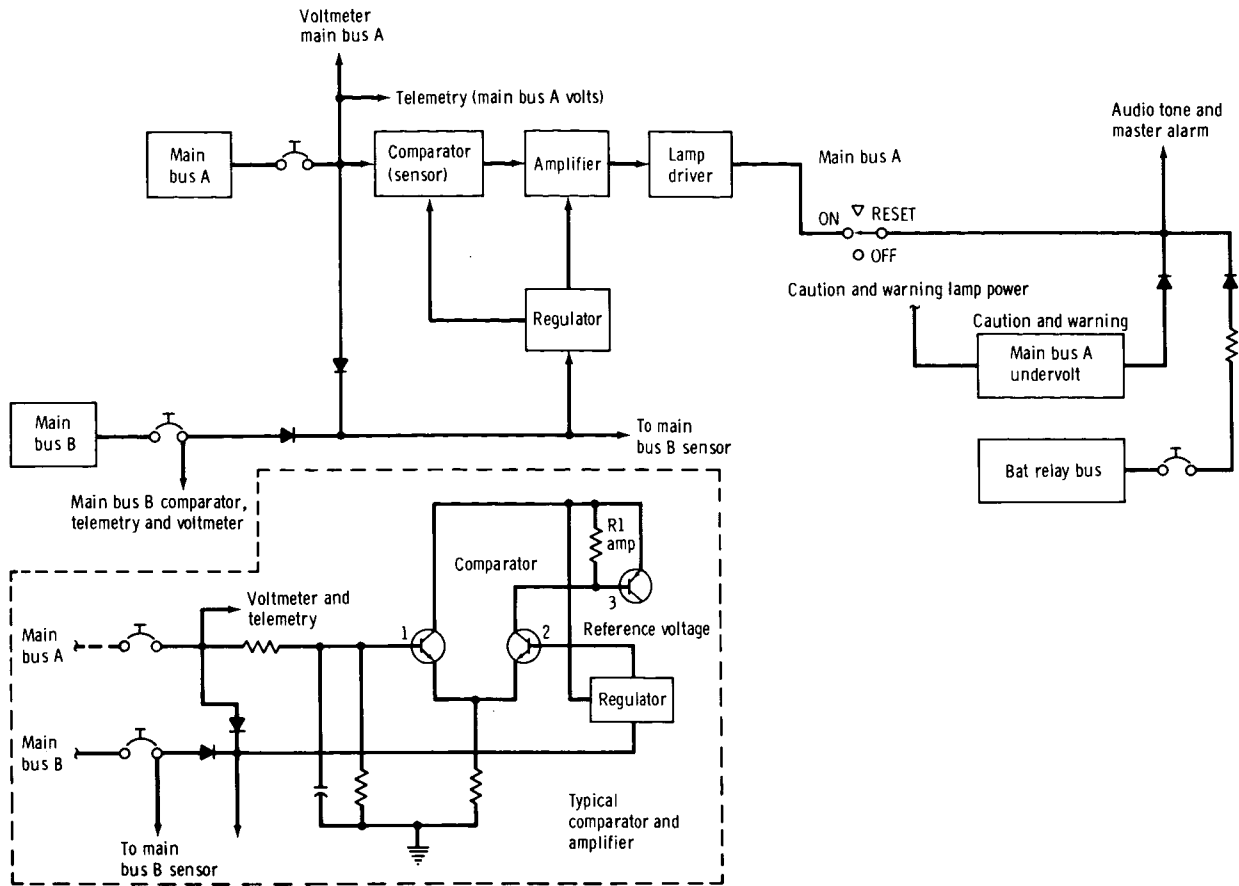
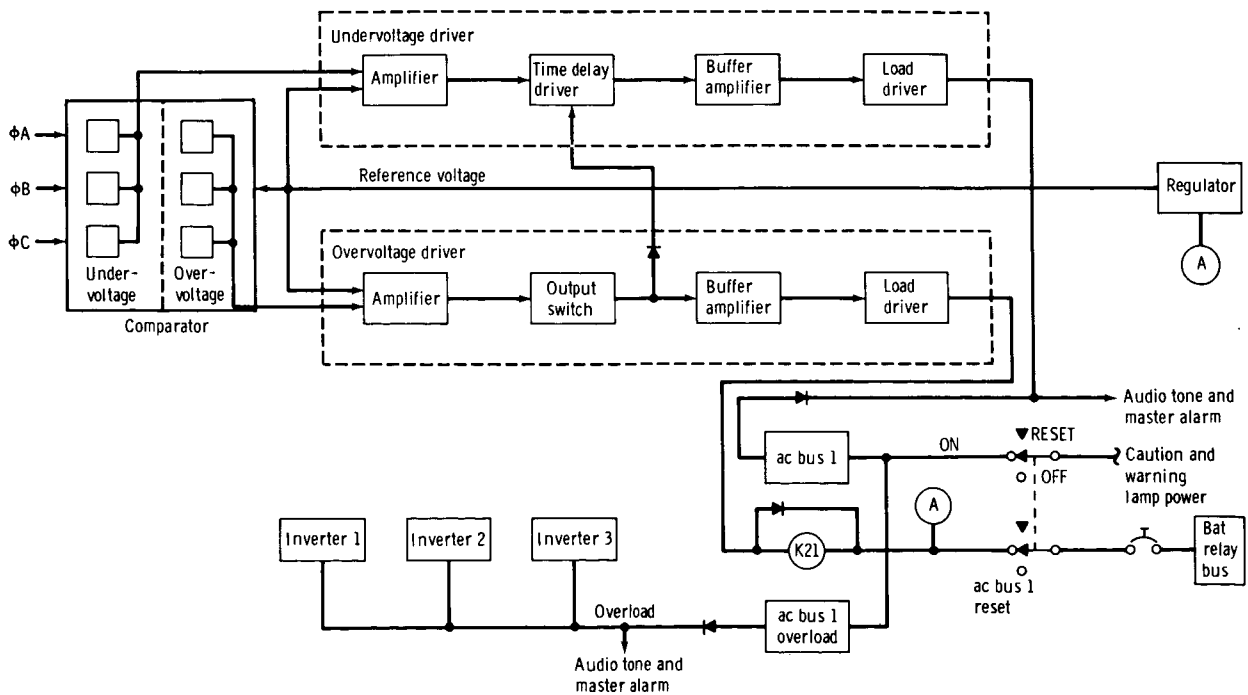Figure 7. - Direct-current undervoltage-sensing circuit.

Figure 8. - Alternating-current sensing circuit:

Originally, the alternating-current sensing circuitry was designed to automatically disconnect an overloaded bus; however, because silicone-controlled rectifiers (SCR) in the inverter overload sensor circuit proved to be overly sensitive to current surges, several unnecessary losses of alternating-current power occurred. As a result, the disconnect feature of the sensor was disconnected, but the C&W alarm was retained. Mission success was not impacted because a crewman was always present in the CM to take the action required for an overload-bus condition.

Control Units

The greatest concentration of control functions in any single unit was in the SM power distribution box. This box, located on the upper deck of the SM, contained both automatic and remotely actuated motor switches to control direct current in the SM. Fuel cell outputs could be connected to either CM bus by motor switches that provided reverse current and overload protection. Provisions also were included for switching ground support equipment power to the vehicle and for protecting the buses if a heater short occurred in the environmental control subsystem.

The CM direct-current power control box used motor-driven switches and stud-mounted diodes to switch power output from the CM batteries to the main buses. To achieve maximum heat transfer, the diodes were mounted directly on plated heat sinks; the plating was the only electrical insulation between the diode mounting base and the heat sink. Early in the test program, several failures occurred when the insulation was damaged and the diode shorted to ground. This problem was overcome by using plated washers between the diode and the heat sink; as a result, at least two failures in the plating were required before a short would occur.

The battery charger (fig. 9) also was modified after fabrication because the output was originally calibrated for a particular current and wire routing. However, subsequent reconfiguration of the wiring and the addition of circuit breakers resulted in a buildup of tolerances which, when combined, caused the charger output to be inadequate. The unit had to be partly depotted to gain access to the potentiometer that controlled the output voltage level. Setting the potentiometer control at the proper level solved the problem.

A major change was made in the configuration of the direct-current distribution system after the Apollo 13 mission. To enhance the probability of a safe return if all three fuel cells were lost, a 400-ampere-hour auxiliary battery was added in the SM. Two control boxes were added to the SM to provide switching and protection for the additional battery.
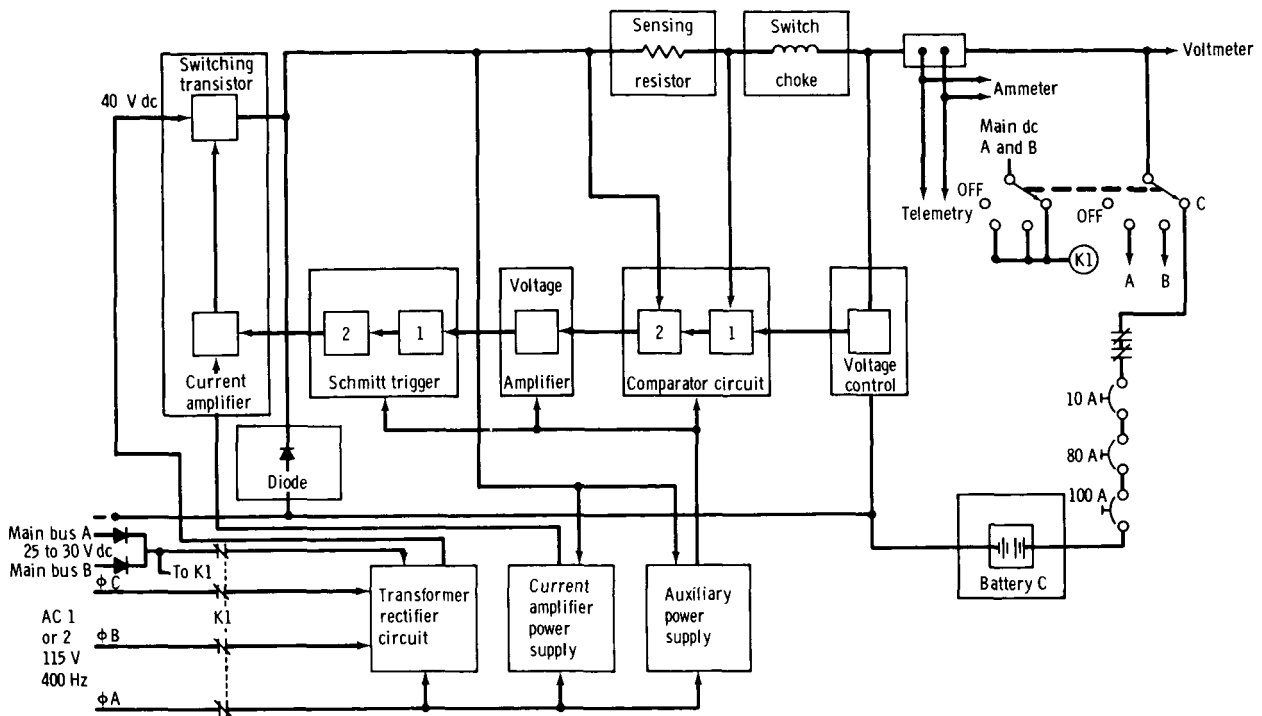
Figure 9. - Functional diagram of battery charger.

Inverter

During the initial phases of the Apollo Program (1962 and 1963), studies were performed to determine the most feasible power system for motor applications. Because of these studies, a central inverter system was selected instead of individual inverters for each application. This system was chosen because it had the lowest weight and greatest reliability. Inverters were designed with an output capability of 1250 volt-amperes, making it possible for a single unit to supply all the alternating current required by the vehicle (fig. 10). Three such inverters were needed on each vehicle to meet the reliability requirements for mission success and crew safety. After the essential requirements were established, the inverter was designed to meet the following criteria.

1. Phases: three-phase, $120° ± 2°$ displacement

2. Voltage: $115 ± 2$ V ac (three-phase average), steady state

3. Frequency: 400 hertz with 6400 hertz external timing; $400 ± 7$ hertz when free running

The original inverters, designated as the -0001 model, were modified several times after the flight of spacecraft 009 in early 1966. Each time a major modification was made in the design of the inverter, the model was designated by a new dash number to control usage. One such modification, which resulted in the -0004 model, was made because alternating-current modulation adversely affected the performance of the S-band transponder.
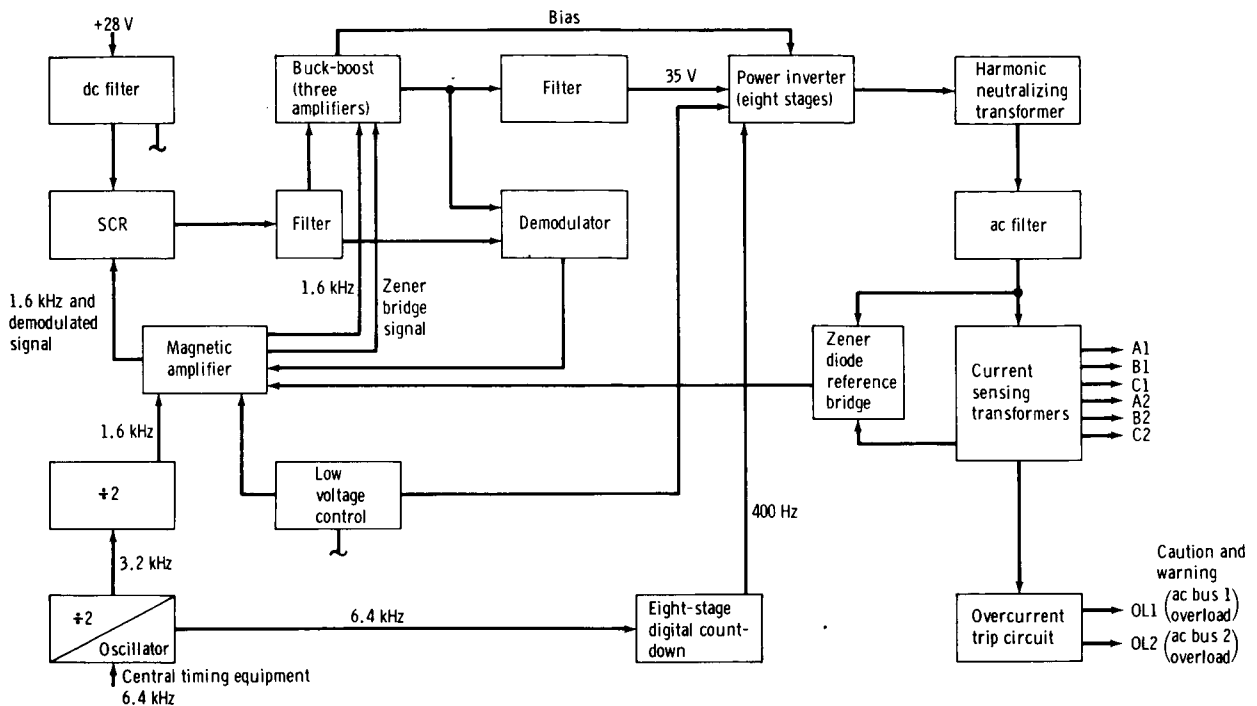
Figure 10. - Inverter block diagram.

The inverter housing was redesigned for Block II spacecraft inverters to facilitate easy removal for repair purposes. At the same time, output connectors were installed in place of the terminals that were used on earlier models. Another change to the inverter was made after the Apollo 204 fire. The inverter housing was sealed to prevent propagation of fire if any component within the inverter ignited. This inverter became the -0008 model.

The -0008 model was installed on the first series of Block II spacecraft and performed satisfactorily; however, the manufacturer of the power-switching transistors used in the inverters experienced difficulties in meeting delivery schedules. Consequently, a second source was selected to supply transistors with similar characteristics. The transistors from the second source had a slightly faster switching speed than the original ones; therefore, minor circuit modifications had to be incorporated, resulting in the -0010 inverter configuration. The -0010 model was first flown on the Apollo 10 mission and performance was normal. Although the design was not altered after this configuration, thermal-acceptance testing of all black boxes became mandatory later in the Apollo Program. As a result, the -0008 and -0010 units became, after testing, the -0012 and -0014 units, respectively.

During factory checkout of Apollo 12 hardware, occasionally the inverters tripped early during overload testing, thus failing to meet specification limits. The cause of these anomalies was determined to be electromagnetic interference, which was caused partly by the faster switching of the new transistors. A reassessment of mission constraints and failure modes indicated that only a remote possibility existed of a

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failure being caused by a fast timeout or premature tripping of the inverter. This condition was tolerable; therefore, no further changes were made.

Power Factor Correction Box

Most of the alternating current generated was used to power fuel cell pump motors; consequently, the load was highly inductive. To compensate for the undesirable power factor resulting from this load, a capacitor bank was added to the motor circuits (fig. 11). These capacitors, together with protective fuses, were installed in the power factor correction box, which was originally located in the CM. The box was not changed for the Block II configuration, but the fuel cell pump motors were redesigned. Because the new pump motors had a much better power factor than the old motors, it was found that, when the correction unit was used, the power factor was overcompensated and effectively nullified all gains from the correction unit. To rectify this problem without impacting schedules, some of the capacitance was removed by altering the panel wiring rather than by redesigning the box. This modification was made on early Block II vehicles.
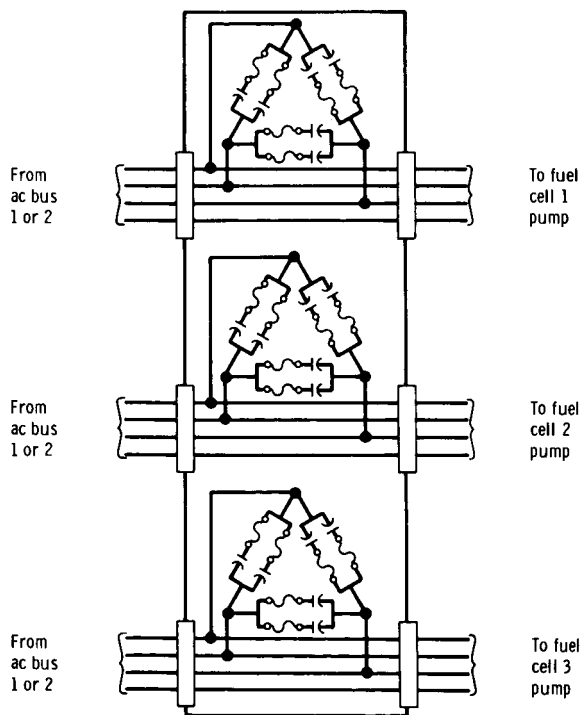
Figure 11.- Power factor correction box.

At a subsequent design review, it was determined that the correction unit contained a single failure point because current from both buses was routed through a single connector. At approximately the same time, an effort was underway to remove weight from the CM to lessen the load on the parachutes during entry. Because the power factor correction box functioned only when the fuel cell pump motors were in use, there was no requirement for the unit after the SM was jettisoned. Consequently, after the box was redesigned to eliminate the single failure point and to optimize the power factor, it was installed in the SM.

COMPONENTS

Motor Switches

The most complex piece of mechanical gear in the electrical power system was the motor-driven switch (fig. 12). Thirty-eight of these switches, in various configurations, were installed in each Block II vehicle; therefore, a problem with any one switch created a potential impact on total design reliability.
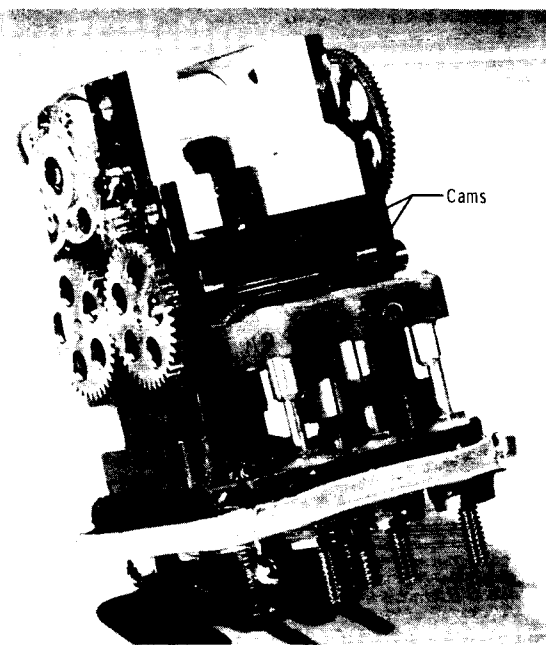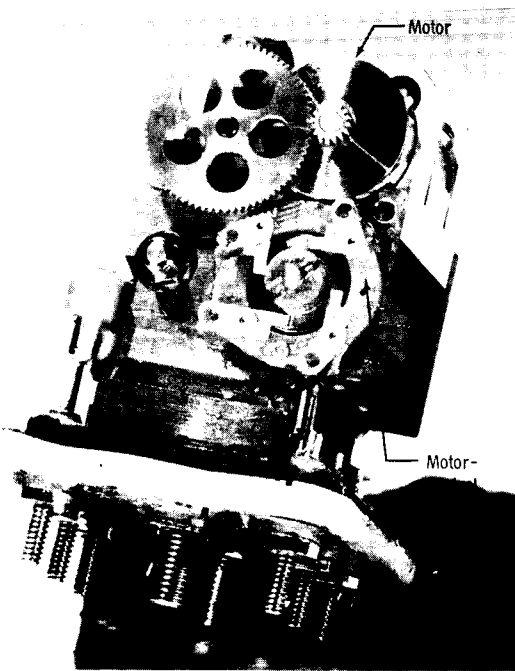
Figure 12. - Motor switch assembly.

Although some motor-switch problems occurred during flight, the majority of the problems resulted from ground tests and operation. The motor switches proved to be extremely sensitive to mishandling, improper operation, or incorrect control circuitry. Most of the failures resulted from the application of a low control voltage or the simultaneous application of "on" and "off" commands. Both misapplications resulted in excessive buildup of heat and subsequent destruction of the motor. These problems were overcome, in part, by using special test consoles to test any equipment that contained motor switches and by providing strict procedures for operating the motor switches.

During the first flight of a Block II vehicle, a problem appeared in a motor switch while it was removing three-phase power from fan motors. It was determined that the switch, which normally was back-filled with an inert gas and sealed with an environmental seal (metal to epoxy), had leaked to an altitude favorable to arcing. As a result, when the switch interrupted the highly inductive load, arcing occurred in the switch and sufficient EMI was created to trip the alternating-current sensor. The problem was overcome procedurally in real time; nevertheless, motor switches that are not hermetically sealed (metal to metal or glass to metal only) are no longer considered suitable for alternating-current circuits in vacuum applications.

A more serious problem became apparent after the Apollo 14 mission. As a result of an investigation, a new failure mode of the motor switch was discovered. A switch in the direct-current power control box failed in midtravel. When the switch was opened for inspection, it was found that the motor armature had undergone excessive heating and the armature windings had opened. Additional investigation and disassembly of the motor showed that the commutator bars were separated from the rotor and presented evidence of extreme heat in the area of the brush/commutator interface. Furthermore, the buildup of residue on the commutator was far greater than normal.

During a thorough investigation, many switches were tested and then disassembled to facilitate a microscopic examination of the mechanism. No abnormal conditions were found in the gear train or other mechanical parts of the switch, and, in all anomalies of this type, it was determined that the motor was at fault. Small silicone grommets, used as seals in the motor housing, were not cured under uniform conditions; those not cured completely tended to revert to their precured condition. This reversion process was present at all temperatures; however, heat aggravated the condition. As the reversion progressed, the catalyst that was used for curing the silicone was released and migrated to the motor commutator where it combined with bearing oil and brush residue to form a high-resistance deposit between the brushes and commutator. This deposit resulted in excessive heat being generated at the interface, a low armature current, and, consequently, low motor torque. Loss of torque caused the motor to stall until the heat generated within the motor was sufficient to cause a wire in the armature to open.

Motor-current traces, which depict the time required for the switch to transfer, and the amplitude of the current were taken before each operable switch was disassembled. These traces were compared with similar traces taken by the vendor during the fabrication of the switches. As a result of numerous trace evaluations, the transfer time, which proved to be a reliable indicator of the condition of a motor commutator, was used to identify switches with potential motor problems. The evaluation also established a maximum time in which a switch with a clean commutator should transfer; consequently, current traces were made for all switches in several phases of the checkout to determine the transfer time of each. Any switch with excessive transfer time was replaced.

Another anomaly was discovered during the evaluation of current traces. Two switches did not begin transferring current until 50 to 100 milliseconds after power was applied. This condition was caused by a sticking brake on the motor. The brakes stuck because of cold flow of the brake lining after long periods of inactivity. After the switches were operated once in the vehicle checkout, they no longer had a tendency to stick, and that particular failure mode was of no further concern.

Circuit Breakers

Several difficulties were experienced during the qualification testing of circuit breakers. Most of the difficulties were caused by faulty test equipment and unrealistic performance requirements rather than the breaker design. When the tests were begun, the breakers consistently failed to pass the operational life test because of actuator knob breakage. Reappraisal of the actual breaker use, however, indicated that the requirements were unrealistically severe. The breaker requirements were reviewed and then redefined to establish realistic requirements, after which the breakers performed satisfactorily.

After a salt-spray test, a group of breakers was set aside for several hours before being checked functionally. When the test was attempted, salt had built up between the actuator collar and the mounting sleeve to such an extent that the actuator could not be moved. This requirement was also determined to be unrealistic because it was not representative of actual conditions. Therefore, the breaker was not considered to be unsatisfactory.

Trip limits for the breakers were selected without access to performance data. When tests were performed, many breakers failed to meet specified values. A review of trip limit values indicated that the specified limits were unrealistic and that breaker performance was satisfactory under the redefined limit values.

A problem peculiar only to large breakers developed when wires were attached to the terminals. When the bolts were tightened to specified torque values, the seal between the breaker body and the terminal lugs was cracked. Although this crack did not affect the strength of the breaker body, it violated the environmental seal. A room-temperature-vulcanizing (RTV) silicone compound was applied to the area to reseal the breaker.

Because the three-phase circuit breakers were large and heavy, they were mounted by screws rather than by stem mounting. This condition resulted in excessive excursions of the internal mechanism during vibration testing. The excursions, in turn, caused the breakers to trip when the unit was vibrated. The problem was solved by adding metal pads to the front of the case to provide a more solid mounting.

The direct-current breakers contained two cams in a mechanism for holding the actuator in the closed position that could, under selected conditions, cause erroneous performance. If only one cam dropped in place when the breaker was reset, it would appear to be in its normal closed configuration but would no longer operate within normal vibration limits. This potential problem was overcome procedurally, however, by cautioning the crewmen to press firmly on the actuator knob when closing a breaker.
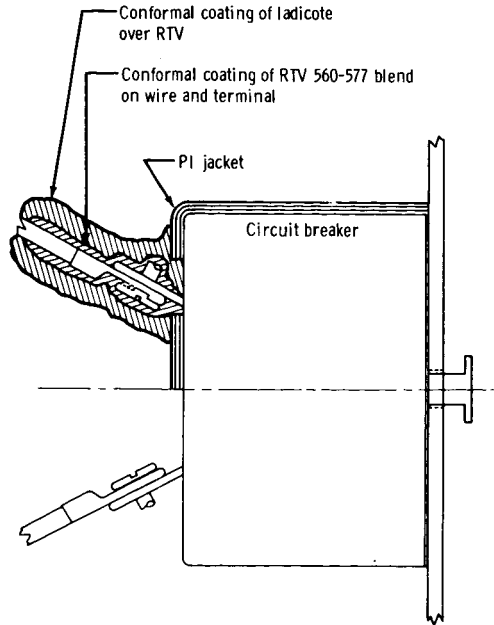
During the Apollo 14 mission, a breaker failed to make contact when it was mechanically closed. Postflight analysis disclosed a contaminant between the breaker contacts. The contaminant was lost before it could be analyzed; however, the substance on the contacts was probably bits of glass from a small fiberglass board inside the breaker. Because this was the only known instance of this problem, corrective action was not considered necessary.

Relays

Block I vehicles included an essential alternating-current bus that was automatically switched to the alternate source if low voltage was experienced. Switching was accomplished by using relays that were qualified for single-phase operation. However, three-phase alternating current was being switched, which resulted in arcing between the contacts and destruction of the relays. The possibility of the aforementioned condition was not considered during the original parts selection; as a result, a relay capable of switching three-phase power was selected.

Relays were used in a cryogenic control box that was added to the vehicle during the modification resulting from the Apollo 13 inflight problems. The original control box was designed to have motor switches because they had no contact bounce, had high current and voltage ratings, and were immune to vibration; however, as previously noted, the motor switches were unsuitable for switching alternating current in a vacuum. Consequently, when the additional box was designed, relays were used for all switching functions.

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Conformal coating of ladicote over RTV

Conformal coating of RTV 560-577 blend on wire and terminal

PI jacket

Circuit breaker

PI jacket: Premolded polyimide glass jacket. Consists of three layers, each of which is 8-mil glass cloth impregnated with polyimide resin and molded to a snug fit. Minimum thickness = 0.025 in.; used in the CM crew compartment as a fire retardant

RTV 560-577 blend : Room-temperature-vulcanizing silicone rubber; used for electrical insulation, moisture, and physical protection around the wire and circuit breaker terminal

Ladicote: A fire-retardant material for protection against burning in pure oxygen atmospheres.

Figure 13. - Component coating.

SAFETY MEASURES

A major consideration in the design of the EPDS was the prevention of fault propagation. Loads were placed on individual circuits and isolated by circuit breakers, fuses, or overload sensors. Buses were isolated with diodes and reverse-current sensors. Maximum physical protection was provided for boxes, components, and wires to prevent damage during manufacture, checkout, and flight. Closeout panels and cable covers were used to shield the equipment. Conformal coating was used behind the panels and in other inaccessible areas. A typical example of the coating applied to exposed electrical connections is shown in figure 13. The coating not only protected the system from shorts caused by floating debris but also sealed the component against moisture and provided an inhibitor to prevent flame propagation.

Wires were especially vulnerable to damage from overheating of adjacent wires because most of the wiring in the vehicle is bound tightly in bundles. To assess the reliability of the wiring and to establish proper ground rules for harness design, extensive testing was performed. Wires were exposed to a variety of temperatures and pressures, some typical of those that were anticipated and others in excess of what the wires were expected to undergo. Tests also were performed to establish the pertinent characteristics of the insulation at different temperatures. Data from these tests were used to establish time limits for various combinations of ambient temperature and pressure for single wires and for those in bundles.

Trip characteristics of each type circuit breaker were determined at various altitudes and pressures in a series of tests that established the derating necessary for operation in mission environments. The breakers also were subjected to several different combinations of ambient conditions, during which the trip times were recorded for different degrees of overload. Results from these tests then were used to establish wire-size/circuit-breaker compatibility criteria. Design was kept within these limits to ensure the integrity of adjacent wires if an overload condition occurred.

RECOMMENDATIONS AND CONCLUDING REMARKS

The final overall design of the Apollo electrical power distribution system was adequate. All the original concepts, however, were not satisfactory, as can be seen in retrospect. Examples of some of the designs that were either modified or discontinued are the automatic main bus disconnect, the automatic alternating-current overload disconnect, and the inverter switching arrangement.

Based on a review of experiences described in this report, several recommendations are listed that might be beneficial to new programs.

1. All equipment should be designed, fabricated, located, and mounted for easy removal, repair, and replacement.

2. Inverters or other alternating-current sources should be in phase and should have the capability of being paralleled.

3. Motor switches, if used, should be hermetically sealed (metal to metal or metal to glass), and the effects of long-term exposure to environments normally considered inert should be investigated thoroughly.

4. Solid-state power controllers should be considered in any application for which remote switching is required.

5. All stud-mounted diodes used for power-source isolation should be anode mounted. If a diode mounted in this manner shorts to ground, only one power source will be affected.

6. An active program for identifying and correcting single failure points and "sneak circuits" should begin early in the design and continue until the end of the program.

Additional recommendations, not obvious from this report but based on an evaluation of the final design of the command and service module electrical power system and its overall performance throughout the Apollo Program, are also listed.

1. Arc-suppression diodes should be the fail-open type.

2. Diodes used in steering functions should be the fail-short type.

3. Some type of laboratory device (e.g., breadboard, test jig, functional mockup, or prototype) should be available for evaluation of the electrical power distribution subsystem design early in the program and for use in providing real-time support during missions. Several circumstances arose during flight in which a dedicated device for electrical power distribution subsystem simulation would have been helpful in recreating problems and evaluating workarounds. The device, regardless of nomenclature, should be primarily for electrical power distribution subsystem support with quick-change capabilities to keep the configuration current.

A realistic qualification program with rigid quality controls is necessary in the selection of effective equipment. Comprehensive testing of each component and of higher assemblies is required. The experience gained from the performance of the Apollo electrical power distribution subsystem should be used in the formulation of guidelines for the design of electrical power systems for future space vehicles.

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