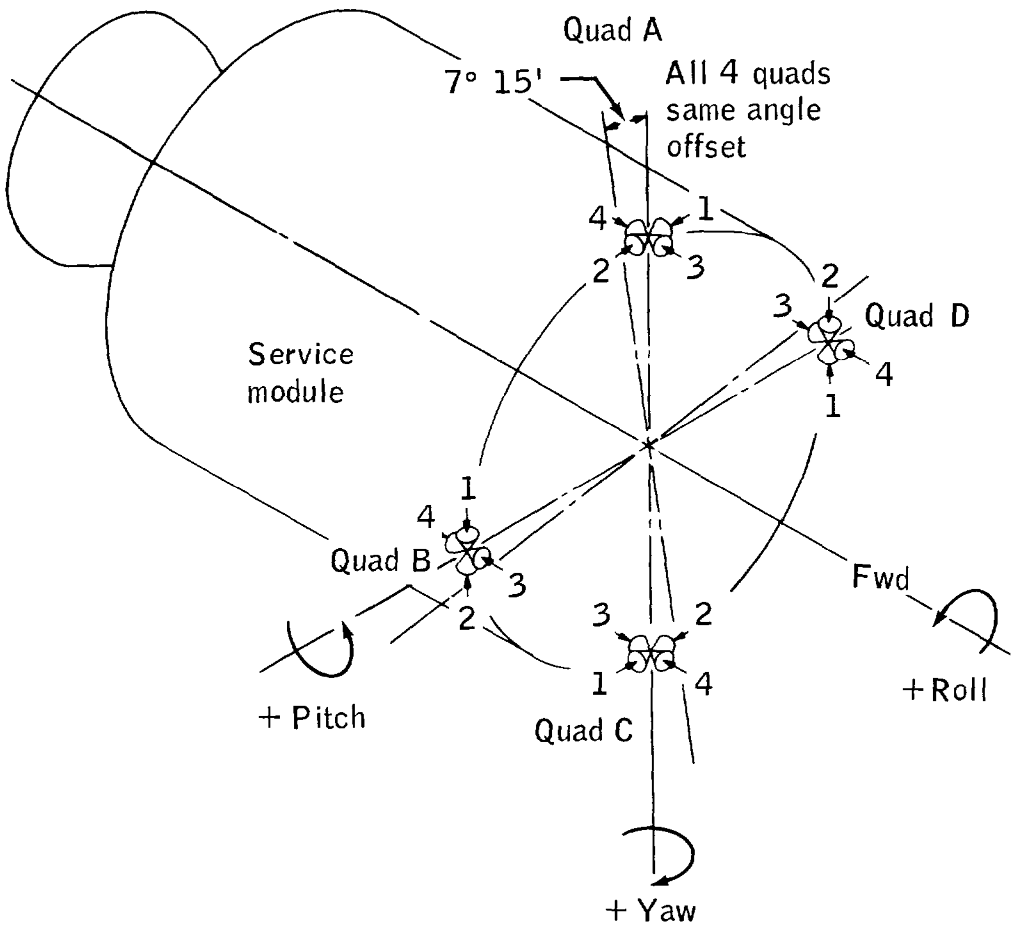
until some reconfiguration could be accomplished. Because power to the talkback indicators would also be lost, it would take some effort to determine the status of the control system.

At the time of the accident, the spacecraft was performing a computer-controlled roll maneuver and maintaining pitch and yaw attitude hold. The digital autopilot began firing RCS thrusters to counteract the attitude perturbations presumably caused by the oxygen tank no. 2 failure, and attitude was completely controlled until main bus B was lost. Soon after the loss of main bus B, Mission Control noted the spacecraft began to rotate about the pitch and yaw axes. It was also noted that the fuel and oxidizer pressures in quad D were decreasing and the crew was asked to verify that they had opened the helium isolation valves which had previously been reported as closed. Although the crew did not acknowledge this request, the pressures were observed to increase to the normal operating values shortly thereafter. The pressures had decreased in this quad because the helium pressurization valves had been jolted closed and subsequent firings of the thrusters had used some of the propellant. This increased the ullage volume and resulted in a noticeable decrease in tank pressures. The flight controllers correctly diagnosed the cause and were not mislead into thinking the tanks were leaking.

At 56:07 Mission Control noted that the crew had turned off all Auto RCS Select switches, because they were concerned that unwanted thruster firings were causing the continuing spacecraft attitude excursions. At about 56:13 the spacecraft was observed to be approaching gimbal lock of the inertial platform. Gimbal lock is a condition in which the inertial platform loses its reference alignment. To prevent a gimbal lock, the spacecraft attitude relative to the inertial platform must be kept out of certain regions. Mission Control advised the crew of this situation, and in an effort to achieve positive control about all axes of the spacecraft, the crew was directed to reconfigure the RCS Auto Select switches for thrusters 3 and 4 in quad B and all thrusters in quad C to be powered from main bus A. This would provide single-jet control authority about each axis (fig. B6-2). The other jets were not switched to main bus A power in order not to drag down the main bus A voltage any more than necessary. The LMP acknowledged and the drift toward gimbal lock was arrested, although all rotations were not stopped.

At 56:22 the CMP reported that the spacecraft was being subjected to pitch and yaw rates and that he had to use direct control with the rotational hand controller to stop them. The rates would start to increase again as soon as he stopped the direct control. He asked if the ground could see any spurious jet firings that might be causing the rates. Although the data available in Mission Control were not complete (the position of the propellant system valves in quads A and C was unknown and firing signals to the Direct coils are not on telemetry), it appeared to the flight controllers that the jet firings were not causing the



Axis	Direction	Thruster
Roll *	+ –	A-1, B-1, C-1, D-1 A-2, B-2, C-2, D-2
Pitch	+ –	A-3, C-3 A-4, C-4
Yaw	+ –	B-3, D-3 B-4, D-4

* Autopilot can be configured to use quads B and D for roll or A and C.

Figure B6-2.- SM RCS quad location and thruster numbering system.

spurious rates. It was observed that thruster 3 in quad C was receiving firing signals almost continuously, but was having no success in stopping the negative pitch rate. In an effort to gain control over the negative pitch rate, at 56:32 Mission Control requested the crew to put the Auto RCS Select switch of thruster 3 of quad A on main bus A. It was suspected that C-3 thruster was not really firing because there was no perceptable reduction in quad C propellant.

At about 56:35 the crew was requested to remove all power from the quad B thrusters auto coils and to power all quad D thrusters from main bus A. This request was made in an effort to determine if quad B thrusters were causing the unwanted pitch and yaw rates. Mission Control continued to monitor the RCS thruster firings and the spacecraft attitude response, trying to determine the status of the system. During the next 10 minutes, the crew pointed out that the quad temperature indications for A and B were out of the normal operating range, and Mission Control assured the crew that they were within acceptable operating limits. In this same time period the ground had noticed numerous firing signals of thruster C-1. Since the flight controllers could see no explanation for this, the crew was requested to remove all power from the C-1 auto coil at 56:53. About 10 minutes later, the CMP reported no negative pitch capability, and requested clearance to enable thruster A-4. Mission Control responded immediately to "bring A-4 on," and the pitch rate was stopped within a few seconds. At 57:20, Mission Control noted a discrepancy in the roll control jet configuration. The autopilot was configured to use quads A and C for roll control, but the auto coils for these jets were turned off. The crew was directed to configure the autopilot to use quads B and D for roll control.

Based on a close observation of firing signals to quad C and the resulting spacecraft response, the flight controllers thought that the quad C propellant isolation valves had been jolted closed by the incident that caused the loud bang. The computer was still sending firing signals to the auto coils, but they were apparently having no effect and propellant was not being used by this quad. Therefore, to save the small amount of electrical power that was being spent by sending firing signals to the coils, at 57:29 Mission Control directed the crew to turn off the auto coils to this quad.

Complete attitude control appeared to be established at this time and all further attitude control support to the CSM was directed toward transferring control to the LM. The overall LM activation support is described in more detail in the following section; however, establishment of the attitude control of the LM is briefly summarized as follows:

1. Mission Control referred the crew to specific pages in the LM Activation Checklist (part of the Flight Data File, ref. 7) for the

procedure to transfer the inertial platform alignment from the CSM to the LM.

2. The CMP was directed to power down all of the guidance, navigation, and control systems after the LM platform had been properly aligned.

3. Mission Control assisted the LM crew in getting attitude control established by pointing out specific circuit breakers that needed to be closed and switches that needed to be positioned.

It was approximagely 1-1/2 hours after the initial incident before complete automatic attitude control was established, although the crew had manual control capability at all times. The information on the ground was incomplete and was confused by the intermixing of automatic control and manual direct control. Furthermore, the major concern was the electrical and oxygen problems, and the only mandatory action in the control system area was to maintain a safe posture in the systems and avoid gimbal lock. These mandatory tasks were accomplished and in due time complete attitude control was established.

Lunar Module Activation

It was recognized at about 45 minutes after the accident that the LM might have to be used to provide the necessary life support, and the LM activation was started about 1-3/4 hours after the crew first reported the loud bang in the CSM. The first hour and 45 minutes were spent in regaining positive attitude control in the CSM, in troubleshooting the electrical problems in the CSM, and in attempting to halt the loss of oxygen from the service module. Since LM activation did not begin until the lifetime of the one functioning fuel cell was predicted to be about 15 minutes, there was a strong motivation to complete the LM activation and CSM powerdown as soon as possible.

The first order of business for LM activation was to get electrical power and the communications sytems operating. A specific procedure for this was read to the LMP at 57:37. Although three checklists for LM activation were available as part of the Flight Data File in the spacecraft, Mission Control did not direct the crew to follow any of them. These checklists were designed for three different situations at LM activation. The first, entitled "Apollo XIII LM-7 Activation Checklist" (contained in ref. 7), contains the nominal mission sequences from initial LM manning to undocking prior to the lunar landing. The other two activation checklists are in the "LM Contingency Checklist" (contained in ref. 7). They were written to cover the situations of having to use the LM to perform an Earth-return abort maneuver for the docked CSM/LM configuration. One checklist includes activation of the primary guidance and navigation system (inertial platform alignment, etc.) and is called the "2-Hour Activation List" because it was designed to be completed at a comfortable pace in time to execute a descent propulsion system maneuver in 2 hours elapsed time. The other contingency list is called the "30-Minute Activation List," and serves the same purpose, except that many steps, including the G&N activation, are omitted. There was no LM activation checklist available which was designed to cover the specific situation resulting from this incident. The features that were different are as follows:

1. The need to get the LM totally activated as soon as possible--including attitude control as well as supplying life support, communications, and electrical power.

2. The desire to power down the CSM as soon as possible in order to preserve all available battery power for reentry.

3. The LM was to serve as a "lifeboat" supplying oxygen, water, electrical power, and attitude control for 80 or 90 hours.

This presented a paradoxical situation in which almost total LM capability was required, but at the same time its consumables had to be conserved as much as possible. In responding to the situation, the flight controllers referred the crew to specific pages in the normal "LM Activation Checklist," augmented with additional instructions. The purpose was to bypass all steps that were not absolutely necessary for getting the LM power, communication, and environmental control system in operation. The total instructions given to the crew referred to only 4 pages of the 59 in the checklist. There were three single instruction additions to this shortly afterward which completed the LM configuration for supplying oxygen to the cabin. Although this particular contingency had never been simulated in the training exercises in preparation for the mission, similar cases had been considered, and Black Team personnel, including the Flight Director, Glynn Lunney, had prepared procedures and criteria for using the LM to augment the CSM. The simulations had been limited to cases where the LM ascent stage was to be retained following rendezvous in lunar orbit. These same personnel had participated in these simulations for the preceeding missions of Apollo 10, 11, and 12, and therefore were familiar with the problems.

The next procedure given to the crew was designed to get the LM guidance and navigation system operating and to get the LM inertial platform aligned to a known reference. Again, Mission Control referred the crew to specific pages in the "LM Activation Checklist," along with certain necessary circuit breaker closures which were not listed on those pages. Although the necessary circuit-breaker panel configuration for LM activation is shown on two pages in the checklist, the crew was not referred to those pages by Mission Control. In order to save time, only the necessary circuit breakers were given as part of each set of special instructions. The omission of a necessary circuit breaker closure later caused some delay in establishing LM attitude control.

Throughout this period of LM powerup, the CMP was given frequent instructions on the CM configuration to reduce power requirements. The crew completed an alignment of the LM IMU to the CSM IMU at 58:09. The platform gimbal angles for both spacecraft were read to the ground for computation of the fine-align torquing angles for the LM. As soon as the LM IMU was aligned, the CMP was directed to power down the CM computer and the IMU, including the IMU heaters.

At about 58:17 the temperature of the coolant loop in the LM began to rise and the LM crew was advised to activate the sublimator, referring to the appropriate page in the "LM Activation Checklist." During the next 2- to 3-minute period there was an unusually high density of conversation, both in the Mission Control Center and on the air-to-ground frequency between the CAPCOM and crewmen in both spacecraft modules. The CMP reported powering down the CM control system; the CDR reported he had no attitude reference system and requested permission to "close the FDAI circuit breakers so we could have a ball to see if we go to gimbal lock"; both the CMP and the LMP reported conditions and asked questions regarding configuration items; and on the ground the CSM flight controllers were trying to get their systems powered down as much as possible while the LM flight controllers were trying to "get through" to the LMP to pressurize the LM RCS and to turn the thruster heaters on.

At approximately 58:21, the CMP was told to continue his powerdown by turning off the power to the rotational hand controller almost simultaneously with the LM crew being directed to power up the FDAI and the RCS heaters, pressurize the RCS, and open the main shutoff valves. After about 5 minutes, when it became clear that neither spacecraft had control of the attitude, the CMP was directed to reactivate the CSM Direct attitude control capability. This was done and the LM crew then proceeded, following instructions from the ground, to pressurize the RCS and to perform the steps necessary to get the attitude reference system operating in the LM. Mission Control at 58:32 gave the LM crew the inputs for the onboard computer which set the proper system gains for the LM autopilot to control the docked spacecraft configuration. The LM achieved complete automatic attitude control capability at 58:34, when the crew received direction from Mission Control to close an essential circuit breaker that had been previously overlooked. The position of this circuit breaker is not indicated on telemetry, but the flight controller correctly diagnosed the problem when the crew stated they still did not have automatic control at 58:33.

After it was definitely established that the LM had attitude control, the CMP was given final instructions for completely powering down the CM, and work toward getting the LM configured for the long trip home proceeded. Mission Control gave the crew the LM IMU torquing angles to get the platform fine aligned to the reference orientation. Discussions were held between the ground and the spacecraft concerning the ability of the crew to use the stars as a reference for platform realignment. It was concluded that this would be difficult if not impossible to do, and the current alignment should be preserved until after the abort maneuver.

An abnormally high pressure reading was noted in one of the LM ascent stage oxygen tanks shortly after telemetry data were received in Mission Control, and the crew was directed to use oxygen from this tank instead of the descent tank. Later it was diagnosed that the shutoff valve leaked, allowing the higher pressure oxygen from the manifold to leak into this ascent tank. The condition in itself was not a problem; the net effect was that this ascent tank was raised to a slightly higher than normal pressure which was well within the tank limits. This degraded the system redundancy, however, and had a subsequent leak developed in this tank, the LM oxygen supply would have been depleted (fig. B6-3).

The next phase of activity was devoted to reducing the power drain from the LM batteries to as low a value as practical. This included turning off many of the displays in the LM and put Mission Control in the position of monitoring system parameters for the crew. The crew was also given all the information required to execute a return-to-Earth abort maneuver 2 hours after passing the point of closest approach to the Moon (pericynthion). Providing this data well in advance is a normal procedure which gives the crew the capability to perform the abort if communications are lost with the ground.

PLANS AND ACTIONS TAKEN TO RETURN THE CREW TO EARTH

After the crew had powered down the CM and activated the LM, the immediate situation had stabilized, and Mission Control could direct its full resources to the long-term problem of getting the crew safely home. The first item of concern was to determine an expected LM consumables lifetime and to develop a trajectory plan that would return the spacecraft to Earth within this lifetime. Also it was mandatory to reduce the expenditure of battery power and water as much as practical.

Subsequent efforts by Mission Control in support of the crew were varied and extensive. Much of this activity, however, is normally part of the routine functions of Mission Control. Such items as monitoring systems performance via telemetry parameters; keeping accurate records of consumables usage, and predicting future consumption rates; scheduling crew rest periods; and orbit determination are only some of the examples of this normal activity. However, only the special activities which were unique to this mission failure or which were of major importance to the

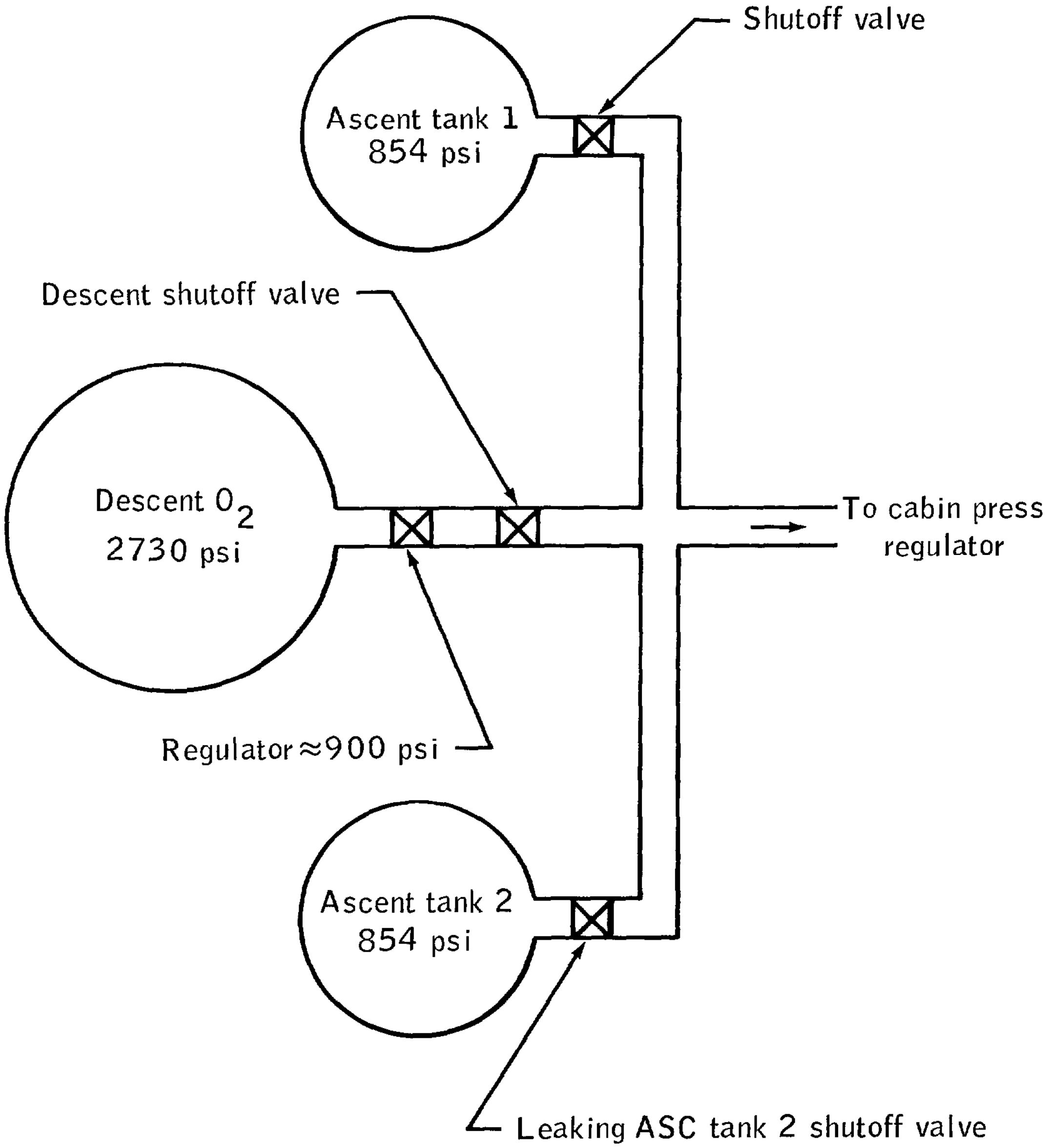


Figure B6-3.- Schematic of LM oxygen storage system.

successful return of the crew will be described. These activities are grouped in three categories in this report and described as independent subjects. These categories are consumables and system management, return-to-Earth trajectory control, and definition of procedures and checklists for reentry preparation. No attempt is made to describe the events chronologically. The Mission Operation Report (ref. 5) contains a comprehensive documentation of these events.

Consumables and Systems Management Actions

Consumables and systems management of both the LM and the CM were of vital importance and generated much activity in Mission Control.

Lunar module.-

Electrical power system: All LM electrical power is supplied by batteries. There are four in the descent stage with a total rated capacity of 1600 amp-hours and two in the ascent stage with a total rated capacity of 592 amp-hours. After the LM activation, analyses of power requirements and lifetime capability were completed. These analyses showed that after the abort maneuver at 61:30, the LM could be powered down to a total current requirement of about 27 amps and still keep the inertial platform aligned. This was extremely important because it made it possible to perform a guidance-controlled abort maneuver at 79:30 which could be used to reduce the return time back to Earth from 152 hours to 143 hours g.e.t. The analyses also indicated that if the guidance system was completely powered down after 79:30, the total power requirement could be reduced to about 17 amps, stretching the battery lifetime to approximately 165 hours g.e.t. This was a comfortable margin, even if the return time could not be reduced below 155 hours.

The flight controllers provided the crew with a list of specific switches to close and circuit breakers to open which would reduce the electrical load to the minimum possible consistent with safe operation. The fact that virtually all of the onboard displays were turned off is an indication of how extensively the spacecraft was powered down. Mission Control kept an accurate account of the switch and circuit breaker configuration, and was able to insure that the necessary equipment was powered up again when the subsequent trajectory maneuvers were made. The full powerdown configuration actually required only 12 amperes, instead of 17. The basis for this powerdown was contained in the LM Contingency Checklist (ref. 7). The Emergency Powerdown Checklist was developed for the case of the LM in lunar orbit awaiting rescue by the CSM. Some additions to this listing of turned-off equipment were made by Mission Control.

As soon as the electrical power system configuration was established and apparently performing well, Mission Control began planning for what actions to take if a LM battery failure were to occur. These plans included listing the few remaining items of equipment which could be taken off line in the powered-down condition. Since the current was already down to less than 17 amperes, there was not much left that could be removed except the communications equipment, but certain equipment could have been operated on a periodic basis rather than continuously. A schedule for this kind of operation was planned in case it became necessary.

At 97:14:26 the LMP called Mission Control to report an anomaly that he had observed in the LM. This anomaly was a "little thump" that was heard but not felt, and it seemed to come from the vicinity of the LM descent stage. The LMP also observed a "new shower of snowflakes come up that looked like they were emitted from down that way." The venting appeared to be going radially outward, perpendicular to the X-axis in the +Y, +Z quadrant, and it continued for approximately 2 minutes. Neither the flight controllers nor the LMP observed any anomalous behavior in the data. The LMP closed the essential display circuit breakers in order to scan his instruments. The flight controllers searched the various displays of telemetry data. Since no unusual readings were noted, the investigation of the "thump" incident was not pursued further at that time. A postflight review of the data indicates that at about the time of the "thump," a large, momentary increase in LM battery output occurred. The surge was of 2 to 3 seconds duration, and was experienced by all four descent batteries. The behavior of the four battery currents is summarized in the table:

Battery	Current output, amps		
	Before surge	Peak	After surge
1	3	37.5	3
2	2	Off-scale high 60 amps	6
3	3	36.8	1
4	3	30.5	1

The MSC investigation of this anomaly is still in progress, and the exact cause of the current increase, the "thump," and the venting is not known. It does appear that they were all related, but not connected with the previous service module failure.

At 99:51 g.e.t. a descent battery no. 2 malfunction warning light illuminated. Because the display system on board was powered down except for the caution and warning panels, the analysis of the problem was done in Mission Control where telemetry was available. There were three possible valid causes of the warning light: an overcurrent, a reverse current, or a battery overtemperature condition. The troubleshooting systematically eliminated all three, and Mission Control concluded the problem was a faulty temperature sensor. The crew was advised to reconnect the battery about an hour later. No problems with the battery ever developed, but the sensor indication later became erratic, causing several MC&W alarms. A plot of total usable amp-hours remaining in the LM batteries is contained in figure B6-4.

Coolant system: It was as essential to power down the LM as much as possible in order to reduce the cooling requirements as it was to reduce the battery amp-hours expended. The LM coolant loop uses the action of ice sublimination to take heat away from the spacecraft. Feed water for the sublimator is stored in tanks, and the rate of water usage to provide this cooling is proportional to the amount of electrical power expended because of the heat generated. The analysis showed that for the above-mentioned electrical power requirements, the LM water supply was most critical and would be depleted about 155 hours g.e.t. This analysis was based on data obtained several hours after the initial LM activation. Estimates based on the usage rate immediately after activation indicated the LM would be depleted of water by 94 hours g.e.t. As expected, the rate reduced drastically, however, after the initial cooling down was accomplished.

During the mission period before the postpericynthion abort, when the spacecraft was on a trajectory with a 155-hour g.e.t. landing time, efforts were made to find a method of increasing the LM water margin by means other than a further powerdown. Two procedures were developed as a result of this effort. The first allowed the crew to get drinking water from the CM potable water tank, and the second was a method of transferring water to the LM tanks for use in the LM coolant loop. The latter procedure involved the use of the portable life support systems (PLSS) water tanks as an intermediate container for transporting the water from the CM waste tank. Although it did not become necessary to use the second procedure, it was tested on the ground by engineering personnel at MSC, and was available in Mission Control. A plot of the usable water remaining in the LM is shown in figure B6-5.

Oxygen supply and carbon dioxide removal: The oxygen supply in the LM was adequate for more than 200 hours g.e.t., and was of no concern (fig. B6-6). This included a supply in the systems normally used for the lunar extravehicular activity (EVA). The initial problem with the ascent oxygen tank 2 had stabilized to the condition that the pressure in the tank was about 100 psi above the normal operating range. Engineering support personnel had advised Mission Control that this was no problem, and no further actions were taken in this area.

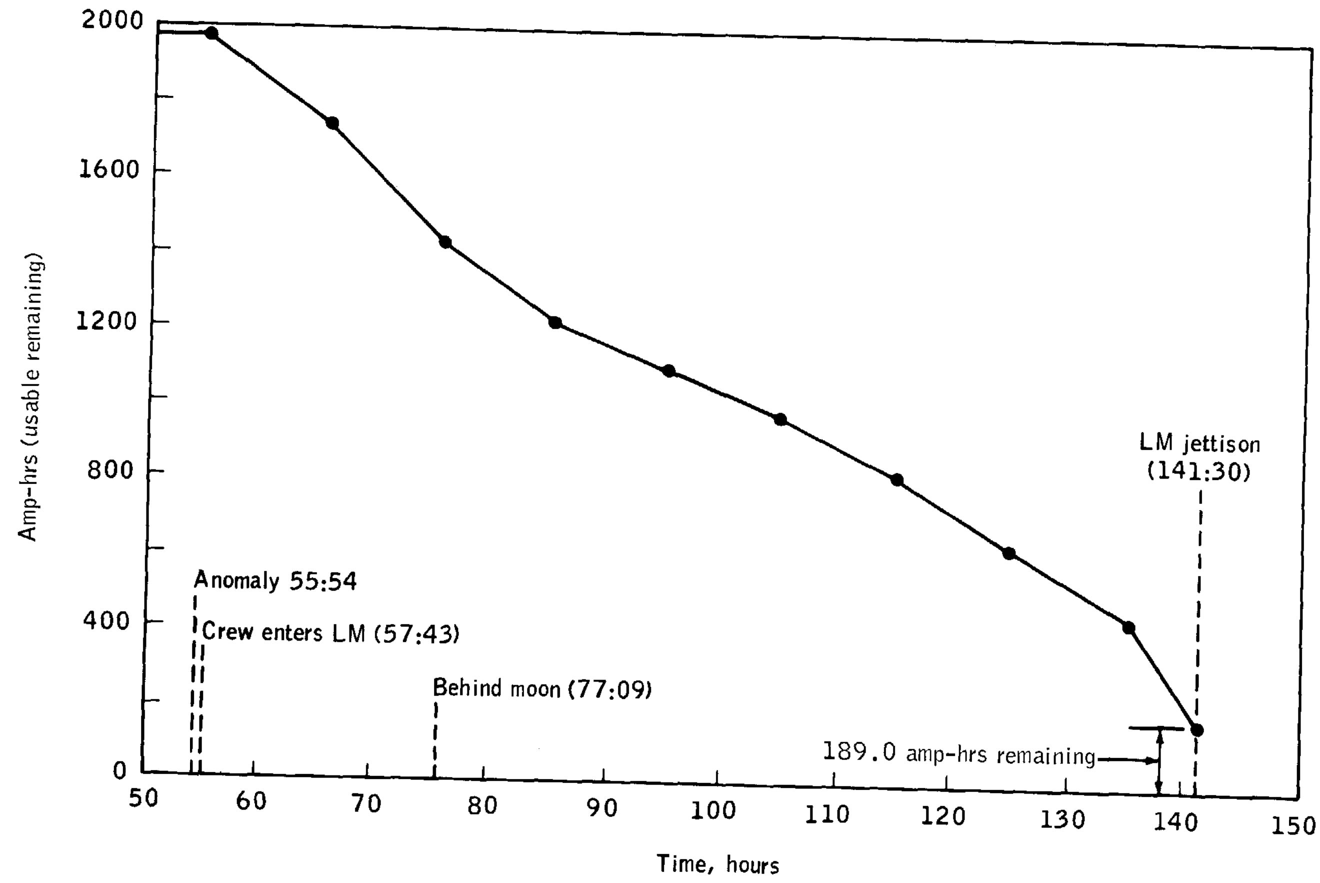


Figure B6-4.- Electrical power system consumables status.

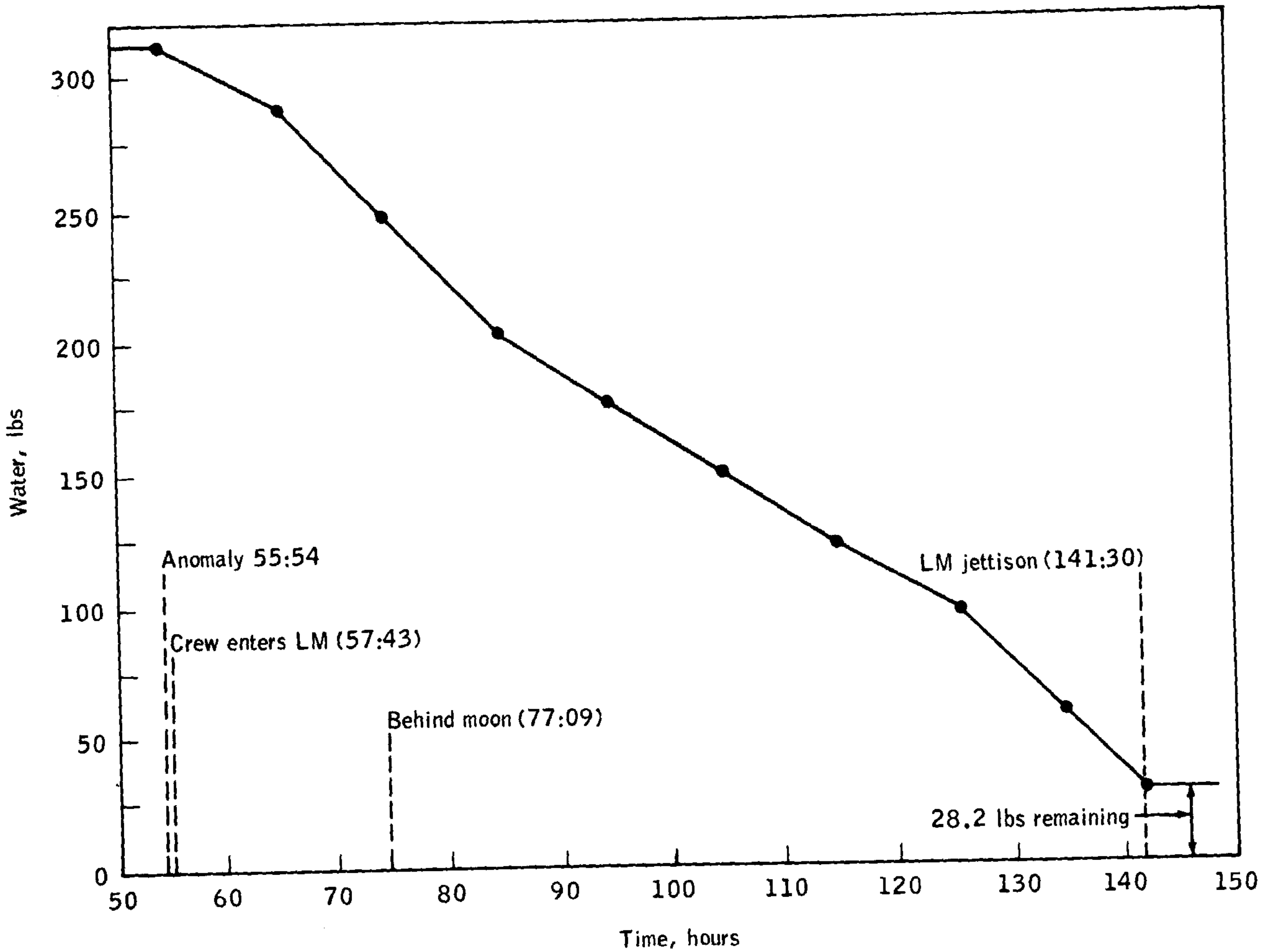


Figure B6-5.- Usable remaining water.

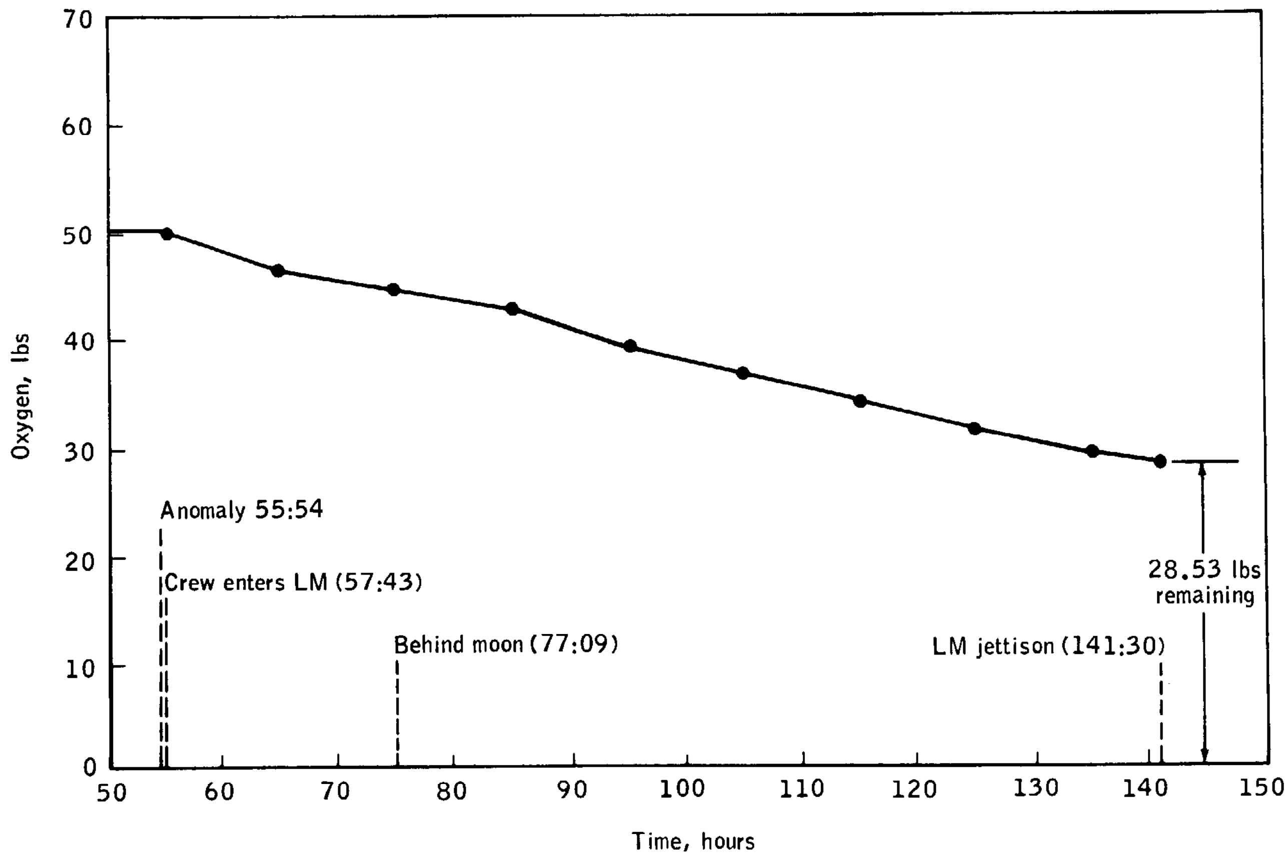


Figure B6-6.- Usable remaining oxygen.

The problem of removing carbon dioxide from the cabin oxygen was a serious one. The LM, like the CSM, uses lithium hydroxide (LiOH) cartridges to scrub the recirculated oxygen to remove odors and carbon dioxide. The LiOH cartridges are rated for a specified total man-hours capacity, and eventually must be replaced when they become saturated. The LM cartridges were not adequate for carbon dioxide removal for three men for the duration of the Earth-return trip. There were more than adequate cartridges in the CM, but they would not fit in the LM canisters. There were several methods suggested for solving the problem, including powering up the CM system to circulate cabin oxygen through its LiOH canisters. The method that was actually used was developed by Crew Systems Division personnel at MSC. It consisted of using tape, flight data file cards, and plastic bag material to connect the CM LiOH canisters to the LM oxygen circulation system. The crew implemented the modification and it worked very well. The partial pressure of carbon dioxide reading indicated by the onboard gage dropped rapidly from 8mm Hg to 0.1mm Hg soon after the rig was completed at 94 hours g.e.t. The modification was not tried until this time in order to get maximum use from the LM cartridges. About 20 hours later, the carbon dioxide partial pressure reading had increased to 1.8mm Hg, and a procedure for putting two additional cartridges in series to those in the CM canisters was given to the crew. This procedure was also developed by engineers at MSC (fig. B6-7). After this second modification was completed, the carbon dioxide partial pressure remained below 2mm Hg for the rest of the mission, without any further modifications necessary.

The modifications to the oxygen circulation systems were evaluated in the simulators at MSC before they were accepted by mission operations personnel. This included tests in the pressure chamber. As mentioned earlier, there were other methods that could have been adopted had this one proved to be unacceptable.

Reaction control system: The LM reaction control system (LMRCS) propellants were another consumable that had to be managed carefully. Maintaining attitude control of both the CSM and the LM, with a total weight in excess of 90,000 pounds, can be done by the LMRCS, but is a particularly taxing job. The LM control system was not designed to perform this task, and does not do it efficiently in terms of propellant expenditure. This was aggravated by the fact that there is some control moment loss and some cross coupling when the LM is in control due to thrust plume deflectors designed to protect the LM descent stage from extended thruster firings.

Shortly after the LM assumed attitude control, Mission Control gave the crew a procedure which increased the attitude excursion tolerance in the computer. This increased the attitude error tolerance and caused

Figure B6-7.- View of CM LiOH cannister modification as installed in the LM.

less thruster firings to be commanded by the computer which was maintaining automatic attitude control. The simulators at KSC and MSC were used to evaluate different techniques for maneuvering the spacecraft under manual control as well as automatic. Manual maneuvers became necessary after the LM inertial platform and computer were powered down after the post-pericynthion abort maneuver. Backup and support crews performed the evaluations and recommended certain techniques.

Mission Control kept a close watch on the RCS propellant consumption and was prepared to have the crew revert to an uncontrolled, drifting flight mode if necessary. This would have been requested if the RCS propellant decreased below the "red line" value. The flight controllers had computed a "red line" which provided enough propellant for meeting the midcourse correction maneuver requirements and the requirements to maneuver in preparation for the reentry sequence.

Command service module.- After the CM powerdown at 58:40 there was very little system management that could be or needed to be done. The electrical power system, however, did require some attention. The first action was to get the CM into a known configuration. So much had happened so quickly during the period following the accident, that neither the crew nor Mission Control had a complete knowledge of the switch configuration in the CM. Therefore, a checklist was developed which listed the desired position of every switch, circuit breaker, and actuator handle in the spacecraft. The lift-off configuration in the CSM launch checklist portion of the Flight Data File served as the baseline for this list, and the modifications were read to the crew. The crew then configured the CM as defined by this list.

The next task was to determine the status of main dc bus B. Because power had not been applied to the bus since the failure of fuel cell 3 at 55:58, it was not certain that a major short did not exist on it. Mission Control defined a procedure which used entry battery B to apply power to the bus. The procedure contained 12 steps, and the displays the crew should monitor were defined, along with the expected indications. The baseline configuration described in the preceding paragraph insured that all loads were isolated from the bus. The procedure was implemented at 94:21 hours and verified that there were no shorts on the bus.

After the CM had been powered down for about 24 hours, it began to cool down to a temperature well below the minimum expected operating temperatures. Engineering support personnel became concerned about the motor switches which are normally used to connect the battery busses to the main dc busses. When it was realized that the CM was going to get unusually cold before the initiation of the entry sequence, the ability of the batteries to provide sufficient potential to drive these switches was questioned. The analysis of the situation was difficult because of the uncertainty as to how cold the battery compartment would get, and it could not be proven that a problem would exist. However, to circumvent the situation, it was decided to close the bus tie motor switches after the main bus B checkout. Subsequently, the appropriate circuit breakers would have to be used as switches to connect and disconnect the batteries from the busses (fig. B6-8). A step-by-step procedure was defined and read to the crew and the bus tie switches were closed at 94:21 g.e.t.

A procedure was also developed for charging the CM entry batteries with the LM electrical power system. Approximately 20 amp-hours of the 40 amp-hours capacity had been used from entry battery A during the period immediately following the accident; a much smaller amount had been taken from battery B since that time. Since the LM battery capacity provided a comfortable power margin for the return to Earth, Mission Control decided to invest some of that power in charging the CM batteries. Preliminary examinations of an entry preparation sequence indicated that in order to not rush the crew, the CM powerup should be initiated about 6 hours before entry. To do this demanded that all three CM batteries be fully charged. The procedure to charge the CM batteries was defined in complete detail by Mission Control. In its most basic terms, it was simply a procedure that used the LM/CM electrical umbilical to get power to the CM main bus B. Then the CM battery charger was tied to this bus and the battery to be charged. The procedure as read to the crew consisted of four typewritten pages of notes and a step-by-step switch position definition. The battery charging was initiated at about 112 hours g.e.t. to demonstrate that it could be done and was completed at 128 hours after 18 of the 20 amp-hours had been replaced. This was done well before the reentry preparation, to allow the entry planning to proceed with the assurance that all batteries would be fully charged at the beginning of the entry preparations.

Return to Earth Trajectory Control

All trajectory determination and maneuver targeting for getting the crew back to Earth was performed by the Mission Control Center. This is the normal procedure, but usually the crew also has the capability to do this. This serves as a backup in case communications are lost with the ground. However, with the command module G&N system completely shut down, the crew was totally dependent on Mission Control for navigation, and abort and midcourse correction maneuver targeting. There was no backup.

There were four trajectory change maneuvers performed to return the spacecraft to the recovery area in the mid-Pacific Ocean following the command module powerdown (fig. B6-9). The first, performed at 61:30 g.e.t., placed the spacecraft on a safe reentry trajectory. The second, performed at 79:28 g.e.t., adjusted the Earth landing point to

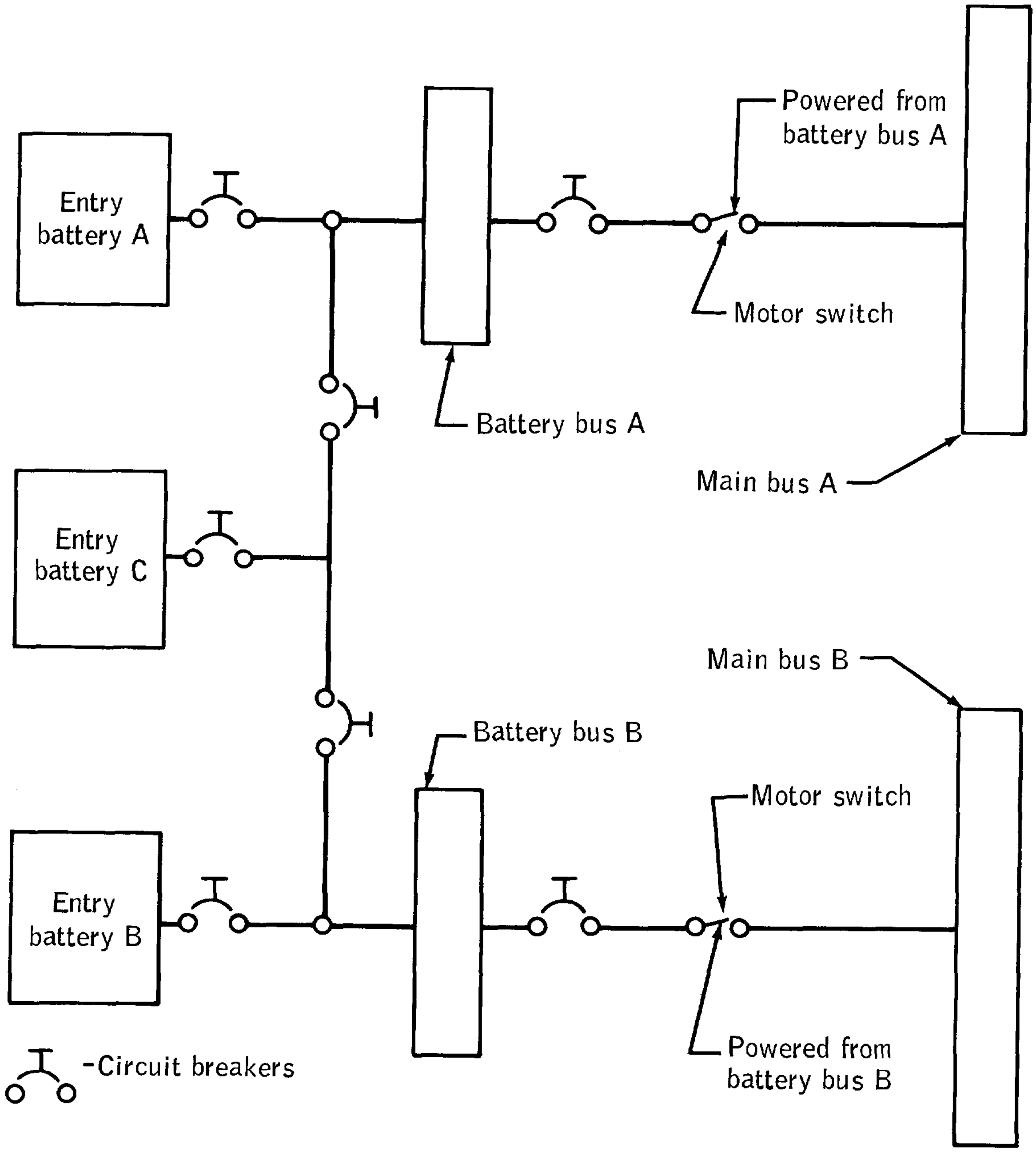


Figure B6-8.- Simplified wiring diagram showing battery power to main busses.

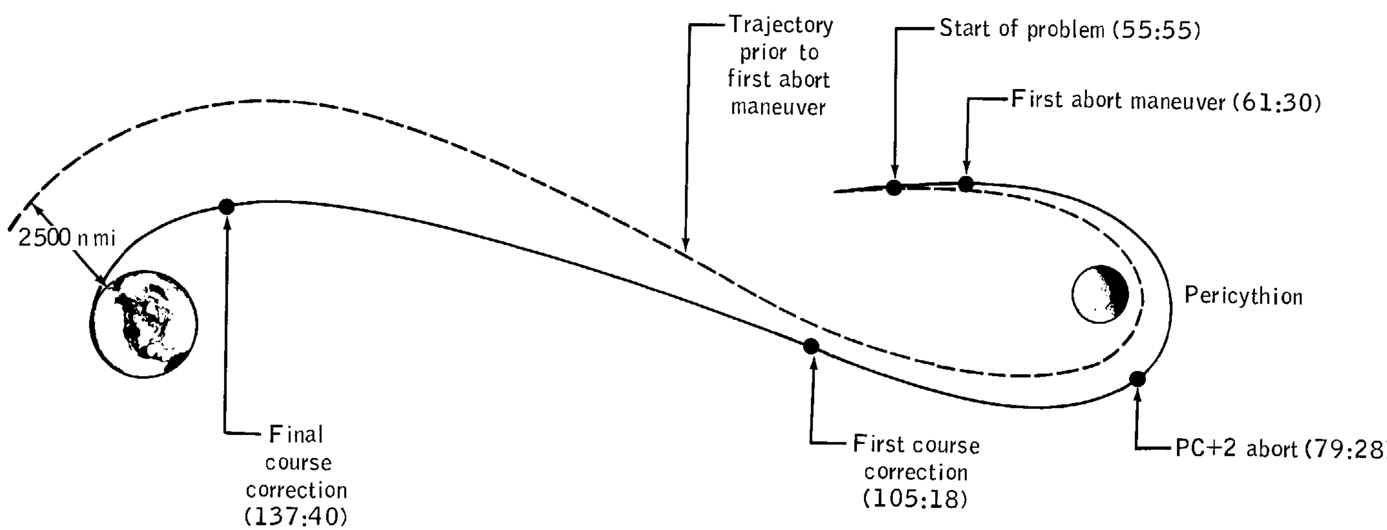


Figure B6-9.- Return-to-Earth trajectory control.

the mid-Pacific recovery area. The last two maneuvers, performed at 105:18 and at 137:40 g.e.t., were course corrections which adjusted the entry conditions to be in the middle of the safe entry corridor. These maneuvers and the decisions related to the choice of specific course changes are described in the following paragraphs.

Abort maneuver at 61:30 hours.- Soon after the failure in the CSM it became obvious that the lunar landing mission could not be achieved and that all effort would have to be focused on getting the crew back to Earth as soon as possible. At the time, the spacecraft was not on a trajectory that would return to a safe reentry of the Earth's atmosphere --so a trajectory change was mandatory. The following questions needed to be answered: What path should be followed back to Earth? When should the trajectory-changing maneuver be executed?

Because the spacecraft was on its way to the Moon, there were two basic types of abort paths that could have been followed: (1) a direct abort in which the trajectory would be turned around and the spacecraft returned to Earth without circumnavigating the Moon; and (2) a circumlunar abort in which the spacecraft would follow a path around the Moon before it returned to Earth. The disadvantage of the circumlunar abort path is that the flight back to Earth takes a longer time than for direct aborts. However, circumlunar aborts require much less velocity change and consequently much less propellant to perform, and part of the flight time can be made up by executing an additional "speedup" maneuver after the spacecraft has passed the Moon.

The direct abort was ruled out for Apollo 13 because the propellant requirements were so large. It would have been necessary to jettison the LM in order to reduce the spacecraft weight so that the service propulsion system (SPS) engine could make the necessary velocity change. The LM was essential to the crew's survival, and must not be jettisoned. Therefore, the choice was narrowed to the circumlunar abort which could be executed with the LM descent propulsion system (DPS), but there were still some decisions to be made. The options were as follows:

1. Do nothing until after the spacecraft passed the Moon; then execute a maneuver to place it on an Earth-return trajectory.

2. Execute a maneuver as soon as practical to place the spacecraft on an Earth-return trajectory and power down the LM immediately thereafter.

3. The combination of both the above: Get on an Earth-return trajectory as soon as practical, and after the spacecraft passed the Moon, perform a maneuver to speed up the return to Earth.

Option 2 was selected. The principal reason was that the LM systems necessary for executing the maneuver were working at the time, and they might not be working 20 hours from then when the spacecraft was in position to do option 1. Another consideration was the fact that the velocity requirement to get on an Earth-return trajectory would increase from 40 fps to 160 fps, making it impossible to perform with the RCS system if this became necessary. So even though option 1 would have allowed an immediate partial LM powerdown, saving some electrical power and water, it was decided that the risk was not worth the savings. Also, option 2 left option 3 available if the guidance and navigation system could be powered up to perform the second maneuver.

The decision having been made to perform a circumlunar abort, and to perform as soon as possible the maneuver to place the spacecraft on a safe reentry trajectory, the only question remaining open was what Earth landing point to target for. Because of the LM consumables status, getting back to Earth as soon as possible was the overriding factor. The quickest return resulted in a landing in the Indian Ocean at 152 hours g.e.t. This meant giving up the ability to bring the spacecraft down in the vicinity of the prime recovery force in the Pacific, although at least a water landing was provided. This was considered to be acceptable because the abort maneuver after passing the Moon probably could be used to decrease the flight time and to land in the prime recovery area.

Post-pericynthion abort maneuver.- Although the spacecraft was placed on a reentry trajectory by the abort maneuver at 61:30 with a landing at 152 hours g.e.t. in the Indian Ocean, it was decided that a post-pericynthion abort maneuver (PC + 2) should be performed. There were two reasons: (1) to reduce the return time to increase the LM consumable margin (the prediction at the time indicated only a 3-hour margin); and (2) to change the landing point to the mid-Pacific where the recovery force could be on station.

During the first few hours after LM activation, detailed analysis of LM consumable usage had shown that the guidance and navigation system could be kept powered up until after the PC + 2 abort maneuver at 79:30 g.e.t. It was predicted that all consumables would last at least until 155 hours g.e.t. even if the LM powerdown to 15 amperes total current were delayed until after 80 hours g.e.t.

There were several options available for decreasing the flight time, but only the three listed in the following table provided a landing in the mid-Pacific.

Option	Delta V, fps	Engine used	Landing time, hours g.e.t.
1	850	DPS	142
2	4000	DPS	118
3	4000	SPS	118

Option 1 was selected even though it resulted in the longest flight time, because of some very undesirable characteristics of options 2 and 3. The problem with option 2 was that it would be necessary to jettison the service module in order to be able to get a 4000 fps velocity change with the LM descent propulsion system. Such a maneuver would almost deplete the descent propellant, leaving a very limited capability should subsequent maneuvers be necessary. There was a high probability that a large course correction would have to be made later. Option 2 was seriously considered, but eventually rejected because it left the CM heat shield exposed to the space environment for such a long period of time, and the possible thermal degradation that might result from this was an unknown risk. The heatshield capability to withstand reentry might be compromised by the prolonged period of cold temperature it would experience. Option 3 was rejected because of the unknown status of the SPS; it was thought that the SPS or the SM might have been damaged by whatever had caused the "bang" and that the SPS should not be used unless absolutely necessary.

Since option 1 provided a comfortable consumables margin and allowed retention of the service module, it was selected. Option 1 also allowed a descent propulsion system delta V capability of approximately 1000 fps to be retained after the abort maneuver.

Part of the preparation for each mission is the establishment of "ground rules" and maneuver monitoring criteria for each planned maneuver. The "ground rules" are general statements which define what should be done if certain events occur. The maneuver monitoring criteria define explicitly the conditions under which the crew will deliberately terminate the maneuver early. The criteria are not the same for all maneuvers because there is a wide variation in the seriousness of the effect of dispersions, and in the seriousness of the effects of early or late engine shutdown. The trajectory and mission situations for the post-pericynthion abort burn were different from any of those for which criteria had been defined; therefore, it was necessary to establish these "rules."

The pertinent characteristics that would affect the rules were as follows:

(a) The spacecraft was on a safe reentry trajectory, although small course corrections probably would be required before reentry.

(b) The primary purpose of the maneuver was to place the landing point in the vicinity of the recovery force.

(c) The secondary purpose of decreasing the flight time was of major importance.

(d) The LM inertial platform had not been fine aligned for approximately 20 hours.

(e) The maneuver could be delayed for 2 hours with an increase in delta V of only 24 fps.

(f) The LM descent propulsion system was to be used.

The following ground rules based on these characteristics were established by the Mission Control team and were given to the crew:

(a) If the engine does not light, do not attempt any emergency start procedures.

(b) If the primary guidance and navigation system (PGNS) has failed, do not perform the maneuver.

(c) Do not attempt to null the indicated velocity errors after engine shutdown.

(d) If an engine shutdown occurred, a subsequent midcourse correction would be performed no sooner than 2 hours later.

The criteria for early termination of the maneuver were defined as follows:

1. Propulsion System Parameters

(a)	Engine chamber pressure	≤85 psi (TM)
		≤77 percent thrust (on board)
(b)	Inlet pressure	≤150 psi (TM)
		≤160 psi (on board)
(c)	Delta P fuel/oxidizer	>25 psi (ground monitored)

2. Guidance and Control System Parameters

(a) Attitude rate >10 deg/sec (except during start transient)

(b) Attitude error >10 degrees

(c) Engine gimbal light

(d) Inertial platform failure with a program alarm

(e) Computer warning light

(f) Control electronics system dc fail light

A final rule that was defined stated that if an early engine shutdown was experienced not due to any of the above, a relight should be attempted, using the engine-start pushbutton and the Descent Engine Command Override switch.

A contingency LM activation checklist had been defined prior to the mission and was part of the crew's Flight Data File. This checklist was designed to prepare the LM for a docked descent propulsion system burn from a completely dormant state. The majority of this checklist had been accomplished with the initial LM powerup at 58 hours g.e.t. The flight controllers reviewed the list in detail and defined a modified list of steps necessary to prepare the LM for the abort maneuver. The modification was basically a deletion of steps already accomplished or not necessary; however, there was one change which revised the time at which the helium regulator shutoff valve was to be closed. This was done to preclude the possibility of a shift in the regulator operating pressure causing a freezing of the propellant lines after this burn. Such an event would prevent further use of the descent engine and it was mandatory to maintain this engine for probable subsequent trajectory changes.

Midcourse correction maneuver.- Postmaneuver tracking data indicated that the second abort maneuver had placed the spacecraft grossly on the right path. However, because the LM inertial platform could not be fine aligned prior to the maneuver, the execution errors were larger than normal and the spacecraft was not on a safe reentry trajectory. This was expected and subsequent corrections were planned for in the LM consumables budget. The correction delta V magnitude was projected to be about 7 fps if executed at 104 hours g.e.t. Unlike the abort maneuver, the course correction maneuvers are not extremely sensitive to pointing accuracy, and with the delta V of only 7 fps it could probably be executed with sufficient accuracy without the inertial guidance system. A special team, composed of off-duty flight controllers and members of the backup flightcrew, was formed to define the maneuver ground rules and procedures to be followed for the course correction maneuver. A detailed crew checklist was to be developed also. None of the procedures or checklists in the Flight Data File were applicable because of the unique situation that existed for this case.

The major issues addressed by this team were as follows:

1. How to get the spacecraft aligned in the proper direction for the maneuver? Was it necessary to power up the inertial platform?

2. Which engine should be used, descent propulsion system or LM RCS?

3. What burn monitoring criteria should be used?

4. What attitude control modes should be used?

The team determined that it was unnecessary to use the inertial platform for the maneuver. The spacecraft could be oriented in the proper pitch direction by sighting on the center of the Earth with the Crew Optical Alignment System (COAS) fixed along the LM +Z axis. The approximately correct azimuth could be achieved by aligning the sunset terminator parallel to the LM Y-axis. This procedure had been developed in the preparation for Apollo 8 when it was discovered that course correction maneuvers could best be made in a local horizontal attitude (that is, perpendicular to a vector from the center of the Earth to the spacecraft). It could easily be applied to the LM-active maneuver, and would give adequate thrust pointing accuracy, so it was not necessary to power up the LM G&N system and try to align its inertial platform.

It was decided to use the descent propulsion system for the maneuver instead of the RCS engines, because the engine-on time for an RCS maneuver would exceed a constraint which protects the LM RCS plume deflectors. The engine was to be left at the low throttle point (about 12.6 percent of full thrust) to give the crew more time to monitor the burn and the lower acceleration should increase the shutdown accuracy. The engine shutdown criteria were the same as for the previous burn. It was decided to monitor the delta V with the backup guidance system accelerometers, but to shut the engine down at a fixed delta time specified by Mission Control. Studies had shown that the burn time computed by Mission Control was very accurate. Since the accelerometers had not been maintained at their proper temperature (heaters had been turned off to reduce consumables expenditure), their status was questionable and the team decided to not use the backup guidance system as an engine shutdown cue. However, if this system appeared to perform nominally during the maneuver, it would be used to null the velocity residuals in the X direction. Velocity errors in either Y or Z direction had an insignificant effect on the entry conditions and were not to be nulled.

Attitude control of the docked vehicle with the backup system required both the CDR and the LMP to actively participate, and Y- and Z-axis translation thrusters had to be used to get adequate control torque. The team defined the modes and procedures to be used in getting the spacecraft in the correct attitude and in controlling the attitude during the engine burn. A procedure to return the spacecraft to the passive thermal control condition was also defined.

All plans were completed after two lengthy sessions. A subgroup from the team defined a detailed crew checklist to be followed in preparing for the maneuver and in preparing for the coasting flight following the maneuver. The checklist was evaluated by members of the backup crew in mission simulators at MSC and some minor modifications were made as a result. The checklist and the procedures were reviewed by the on-duty Mission Control team and then read to the crew approximately 5 hours prior to the scheduled course correction. This allowed the crew ample time to study them and to rehearse their roles.

Entry Procedures and Checklist Definition

After the situation in the spacecraft was stabilized, one of the several parallel activities that was initiated was the definition of procedures for the pre-reentry phase. The total loss of electrical power in the service module forced some major revisions to the activities and the crew procedures for this part of the mission. The most significant consequences of this loss were the following: (1) SM RCS engines would not continue to fire to separate it from the CM after jettison; and (2) LM electrical power and RCS should be used to conserve the CM batteries and RCS propellant as much as possible. This meant that the LM should be retained through as much of the pre-entry sequence as possible, and that a plan for jettisoning the SM and the LM had to be worked out.

A first iteration plan for the pre-entry phase was available as early as 12 hours after the LM activation. This plan called for CM powerup 2 hours before arrival at the entry interface (EI - 2 hours), and required the total remaining capacity from the CM entry batteries, 98 amp-hours. After the plan was thoroughly reviewed by all elements of the operations team, including mission planning and flight crew support personnel, several modifications and additions were considered necessary. The principal difficulty was that the crew would probably be rushed, and there was little or no extra time allowed for contingencies. It was evident that the timeline needed to be extended and the CM batteries would have to be recharged to at least 115 amp-hours. The recharging was accomplished and the procedure is described in Part A2 of Appendix A.

The White Team, one of the four flight control teams assigned to the mission, was taken off its normal rotation of duty in order to devote full attention to developing the reentry preparation sequence of events, crew procedures, and checklists. With this flight control team as the lead element, all MSC organizations normally involved in this type of premission activity were enlisted in this effort. In the course of defining the procedures, extensive use was made of the spacecraft simulators at MSC and KSC. These simulations, performed by members of the backup crews, served two essential purposes. The first was simply to evaluate them--to determine if they were practical, safe, efficient, and adequate. The second purpose was to determine the time required to complete certain parts of the procedures. The latter was important because a completely defined timeline had to be given to the crew in order to insure that everything was accomplished on time. It was essential that this timeline be realistic because the crew could not afford to get behind and fail to complete it, but neither could they start too early and use too much power from the CM batteries.

Another source of data used to develop the procedures was a series of contingency separation studies that was performed prior to the flight by mission planning personnel. These studies had examined the trajectory-related considerations for several different methods of jettisoning the SM and the LM. They had defined the effects of different attitudes, time, and velocity of jettison on the subsequent separation distances. It was only necessary to verify that these studies were valid for the Apollo 13 conditions, and then select the one with the most optimum characteristics.

The planning and evaluation of the pre-entry activities continued for approximately 2 days. At the end of this time, a complete plan had been defined and thoroughly reviewed. It was read to the crew at about 120 hours g.e.t., which gave them about a day to study and rehearse their procedures.

The pre-entry sequence plan (fig. B6-10) called for initiating the powerup at EI - $6\frac{1}{2}$ hours, with the LM supplying power to main bus B in the CM and entry battery C supplying power to main bus A. A total of 115 amp-hours was required of the CM entry batteries, including a 23 amp-hour allowance for contingency after splashdown. A detailed expected battery current profile was plotted and used during the actual preparation to verify that a safe power margin was maintained throughout the reentry preparations. Battery utilization was planned so that all three entry batteries would be available throughout the entry phase. It

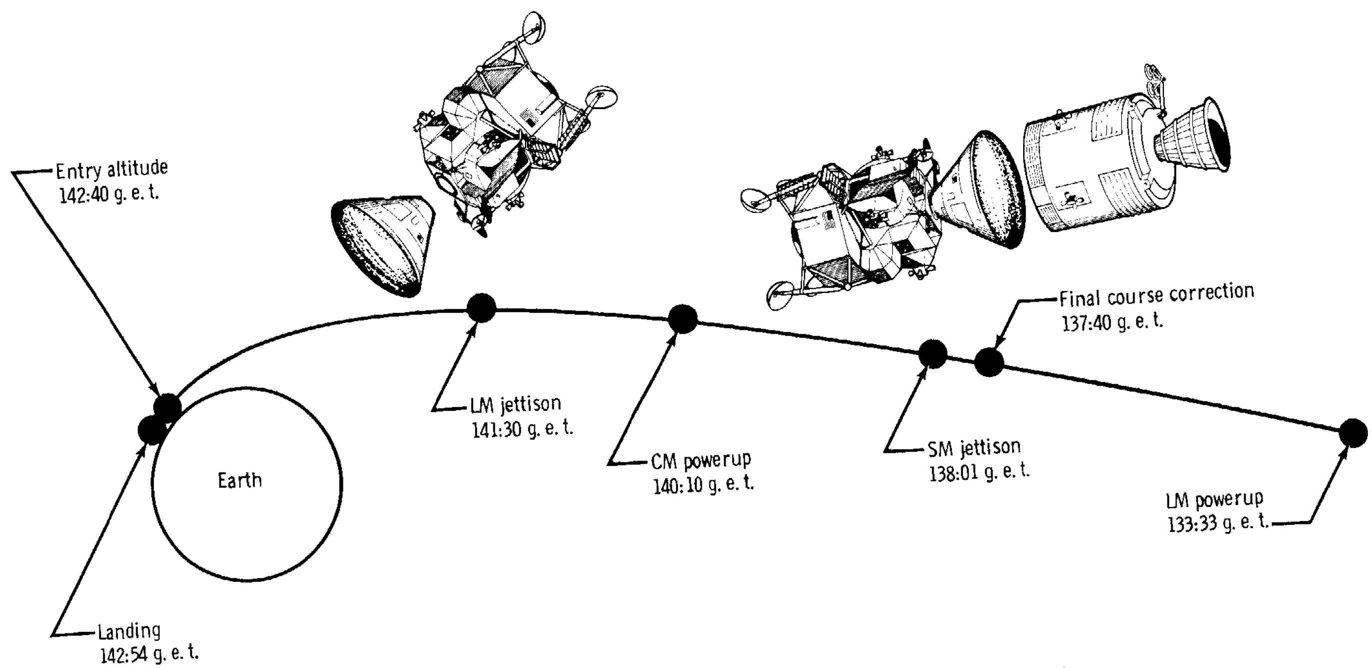


Figure B6-10.- Pre-reentry sequence milestones.

was predicted that battery C would be depleted after deployment of the main chutes, and in fact it was. This left the redundant capability of two batteries available to inflate the uprighting bags after splashdown.

The initial part of the reentry preparations, LM powerup, was performed about 3 hours earlier than planned. The crew was not resting comfortably due to the cold environment, and since there was ample margin in the LM batteries and water tanks, it was decided to turn on some equipment to try to warm up the spacecraft.

After activating the LM guidance and control system, the first major milestone in the entry sequence was to execute the final course correction to place the spacecraft on a trajectory that was in the center of the safe entry corridor. Prior to the final course correction, the trajectory had an entry angle error of about +0.5 degree, which is a safe condition, but slightly shallow (fig. B6-11). It is a standard practice to perform a final trim maneuver a few hours prior to entry to try to remove any entry angle error greater than ±0.1 degree, and this course correction was incorporated in the timeline before it was known whether or not it could be required.

The planned procedures for the final course correction were the same as for the earlier one performed at about 104 hours g.e.t., including the alignment procedure which only required sighting the Earth through the COAS. Manual control of the actual delta V maneuver was also planned. However, since the LM powerup was started 3 hours earlier than originally expected, it was decided to use part of this time to align the LM inertial platform. This was done with the crew sighting on the Moon and the Sun for orientation determination. A further modification to the planned procedures of using the primary guidance system to perform the course correction had to be abandoned, because the attitude error indications did not behave properly. It was suspected that there might be something wrong with the guidance computer, so the crew performed the maneuver manually, following the original plan. Subsequent analysis has shown that the attitude error indications were not indicative of a system problem, but were a result of the guidance system activation procedures. These same indications did not show up in the simulator evaluations performed before the crew was given the procedures because of the limitations of simulator initialization.

The service module jettison was the next major milestone in the pre-entry sequence. It was performed at about 4-1/2 hours prior to reentry. The techniques used and the attitude and delta V requirements for it were obtained from premission studies. Basically, the technique was very similar to that used by a railroad switch engine to get rid of the end boxcar. The spacecraft was given an impulse with the LM RCS that caused a velocity change in the desired direction of about 0.5 fps; the CM/SM separation pyrotechnics were fired, physically disconnecting

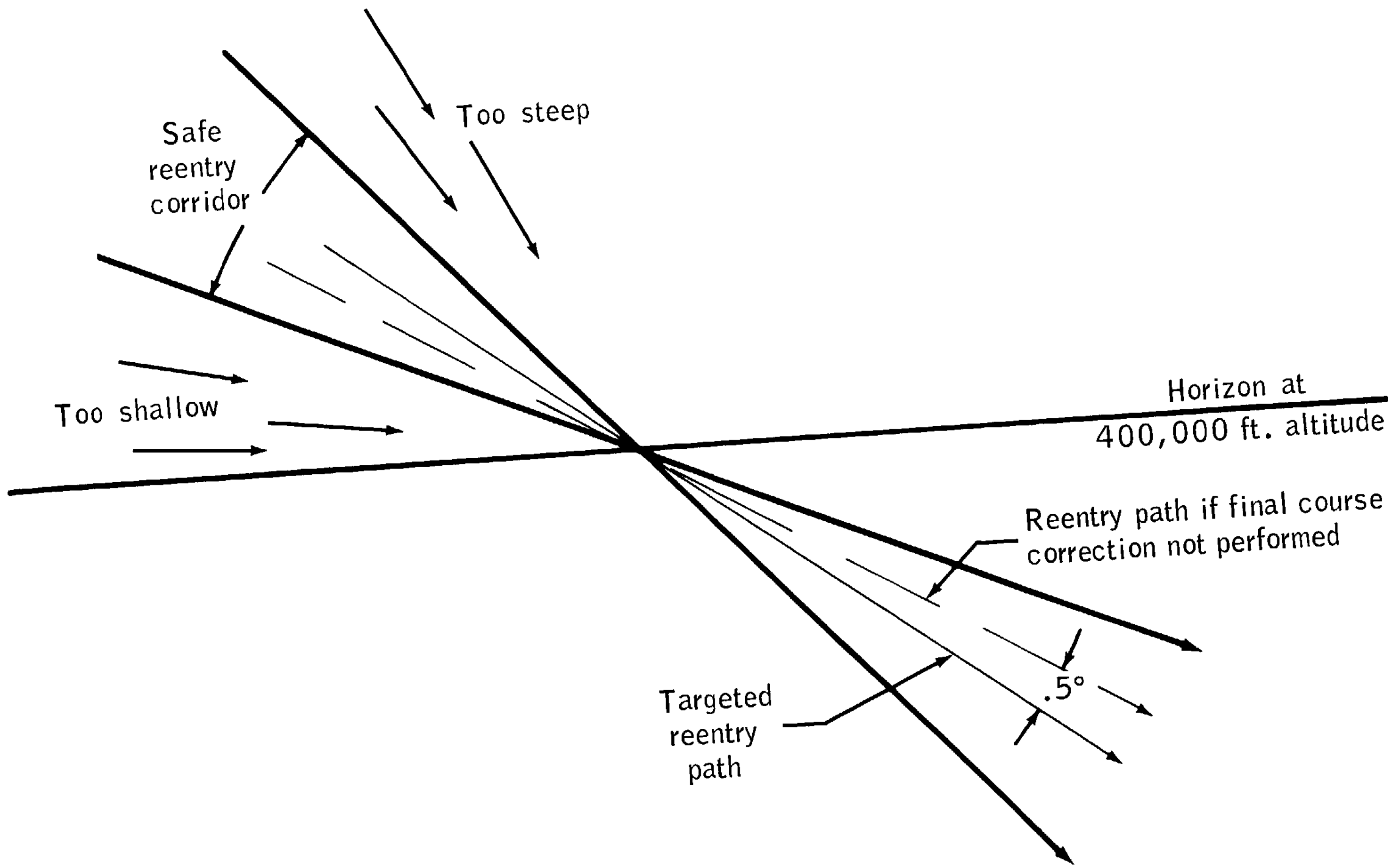


Figure B6-11.- Flight path angle and the safe reentry corridor.

the two modules; and a velocity change of the LM and CM was accomplished by reverse thrust from the LM RCS. The service module continued to translate relative to the manned modules, and separated from them at a rate of 0.5 fps. The normal method of using SM RCS jets to drive the SM away would not work because there was no way to get electrical power to keep the jets firing after CM/SM separation. The fuel cells which normally perform this function were inoperative.

The next major step was to get the CM inertial platform aligned. An automatic guidance controlled reentry was planned, which meant that the platform needed to be aligned to a known reference direction. There were several methods that could have been used to accomplish this, and a considerable amount of time was spent by the White Team in determining the best one. The selected plan used the docked align transfer procedure to get the CM platform coarsely aligned to the LM platform. The CM platform was then very accurately aligned to the desired direction by optical sightings with the CM sextant. Mission Control was standing by with an alternate procedure in case stars could not be seen through the CM optics; however, this was not necessary.

There was much interference on the voice and telemetry communication signals during this time period, which was later diagnosed to be due to the spacecraft attitude. Apparently the spacecraft was oriented so that the LM structure was blocking the signal from all of the omni antennas arrayed around the CM, and the received signal strength was very low. The antenna blockage problem was not recognized and several reconfigurations of the communication equipment were made to try to correct the problem, none of which were successful. In order to maintain adequate signal strength, it was necessary to receive data at the low bit rate only. This was not a major handicap, but it did cause some delay in completing the preparation of the CM guidance system for reentry.

The LM jettison from the CM was accomplished at about 1 hour prior to reentry. The attitude was based on premission studies, but no technique had been defined for achieving the actual separation with LM jettison from the CM only (no service module). The technique was defined by the White Team and consisted of using pressure in the LM/CM tunnel to impart a relative velocity to the two modules when the final separation pyrotechnics were fired. This method of separation had inadvertantly occurred at the LM final jettison on Apollo 10 and was known to give sufficient separation velocity.

It was planned to jettison the LM in a direction 45 degrees south of the spacecraft plane of motion; however, the crew maneuvered the spacecraft to an attitude 65 degrees north of this plane. Mission Control was monitoring the spacecraft attitude, but did not realize the mistake until the crew was in the process of final closeout of the LM. Flight controllers quickly analyzed the situation and determined that, although the 65 degrees north attitude did not give as much separation, it was acceptable. The major problem in being in error by 110 degrees was that it placed the CM in an attitude much closer to gimbal lock than is normally done. The crew had to be especially alert during the jettison and to use manual control of the CM to avoid gimbal lock.

The remainder of the sequence, from LM jettison to splashdown, followed normal procedures. The only difference was that the CM was completely independent of other spacecraft components at 1 hour prior to reentry instead of the usual 15 minutes.

PART B7

INSTRUMENT SYSTEM CHARACTERISTICS

Part 7 provides additional technical information of systems design and characteristics which are pertinent to interpretation of data presented in earlier parts of this Appendix. The following systems are discussed:

Oxygen Tank Temperature Instrumentation

Oxygen Tank Quantity Instrumentation

Oxygen Tank Pressure Instrumentation

Apollo PCM Telemetry System

Mission Control

OXYGEN TANK TEMPERATURE MEASUREMENT

The temperature measurement is made with a platinum resistance thermometer (R/T) encased in an Inconel sheath attached to the Teflon insulator part of the quantity probe (fig. B7-1). The resistance of the R/T and the transducer output voltage increase with temperature. The signal conditioner which serves as a reference voltage generator and amplifier is located on the oxygen tank shelf. An electrical schematic of the transducer is shown in figure B7-2.

The system electrical and performance parameters can be summarized as follows:

Data sample rate	one per second
Range	-320° F to $+80^{\circ}$ F
Corresponding R/T values	71 to 553 ohms
Output voltage	0 to 5 V dc
Accuracy	± 2.68 percent or $\pm 11^{\circ}$ F
Output impedance	5000 ohms

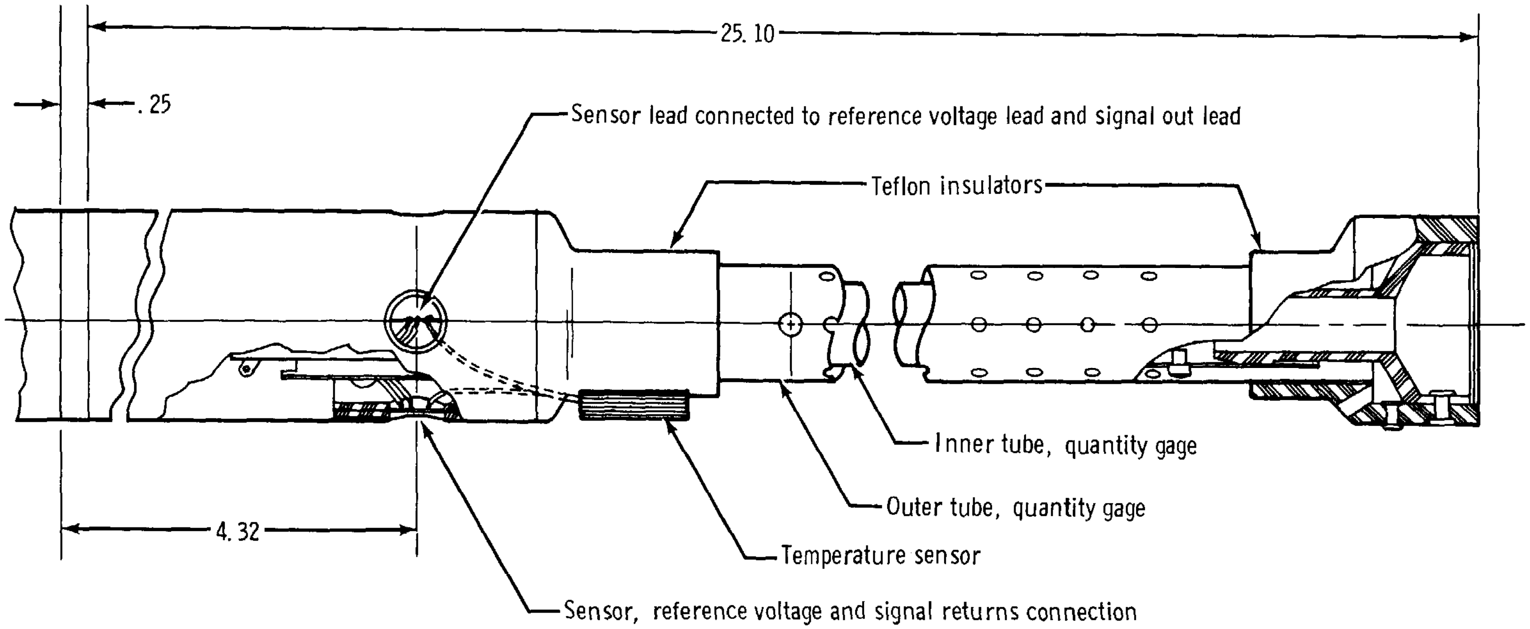


Figure B7–1. – Oxygen quantity gage and temperature sensor location.