REPORT OF
APOLLO 13 REVIEW BOARD

APPENDIX A
BASELINE DATA: APOLLO 13
FLIGHT SYSTEMS AND OPERATIONS

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

CONTENTS

BASELINE DATA: APOLLO 13 FLIGHT

SYSTEMS AND OPERATIONS

Appendix A is divided into five parts. Part A1 briefly describes the Apollo spacecraft configuration; Part A2 provides a systems description of the Apollo spacecraft configuration with special emphasis on the electrical power system (EPS); Part A3 describes the lunar module systems; Part A4 briefly describes the Mission Control Center at Houston, Texas, and its interface with the spacecraft during the mission; and Part A5 gives a detailed description of the fuel cells and cryogenic gas storage systems aboard the Apollo spacecraft. This baseline material may not always represent the precise Apollo 13 configuration in every case, since there is a continuous updating which is documented periodically. For example, Fuel Cell 2 on Apollo 13 was normally connected to bus A in the distribution system, rather that as described in Part A2.6.

The data were extracted from the following sources:

APPENDIX A

PART A1 and A2	Technical Manual SM2A-03-Block II-(1) Apollo Operations Handbook Block II Spacecraft, Volume 1, dated January 15, 1970.
PART A3	Technical Manual LMA790-3-LM, Apollo Operations Handbook, Lunar Module, Volume 1, dated February 1, 1970.
PART A4	Manned Spacecraft Center Flight Operations Plan - H Missions, dated August 31, 1969.
PART A5	Apollo Fuel Cell and Cryogenic Gas Storage System Flight Support Handbook, dated February 18, 1970, prepared by Propulsion and Power Division, Manned Spacecraft Center.

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PART A1

APOLLO SPACECRAFT CONFIGURATION

The Apollo spacecraft consists of a launch escape assembly (LEA), command module (CM), service module (SM), the spacecraft lunar module adapter (SLA), and the lunar module (LM). The reference system and stations are shown in figure A1-1.

LAUNCH ESCAPE ASSEMBLY

The LEA (fig. A1-2) provides the means for separating the CM from the launch vehicle during pad or first-stage booster operation. This assembly consists of a Q-ball instrumentation assembly (nose cone), ballast compartment, canard surfaces, pitch control motor, tower jettison motor, launch escape motor, a structural skirt, an open-frame tower, and a boost protective cover (BPC). The structural skirt at the base of the housing, which encloses the launch escape rocket motors, is secured to the forward portion of the tower. The BPC (fig. A1-3) is attached to the aft end of the tower to protect the CM from heat during boost, and from exhaust damage by the launch escape and tower jettison motors. Explosive nuts, one in each tower leg well, secure the tower to the CM structure.

COMMAND MODULE

The CM (fig. A1-4), the spacecraft control center, contains necessary automatic and manual equipment to control and monitor the spacecraft systems; it also contains the required equipment for safety and comfort of the flight crew. The module is an irregular-shaped, primary structure encompassed by three heat shields (coated with ablative material and joined or fastened to the primary structure) forming a truncated, conic structure. The CM consists of a forward compartment, a crew compartment, and an aft compartment for equipment. (See fig. A1-4.)

The command module is conical shaped, 11 feet 1.5 inches long, and 12 feet 6.5 inches in diameter without the ablative material. The ablative material is nonsymmetrical and adds approximately 4 inches to the height and 5 inches to the diameter.

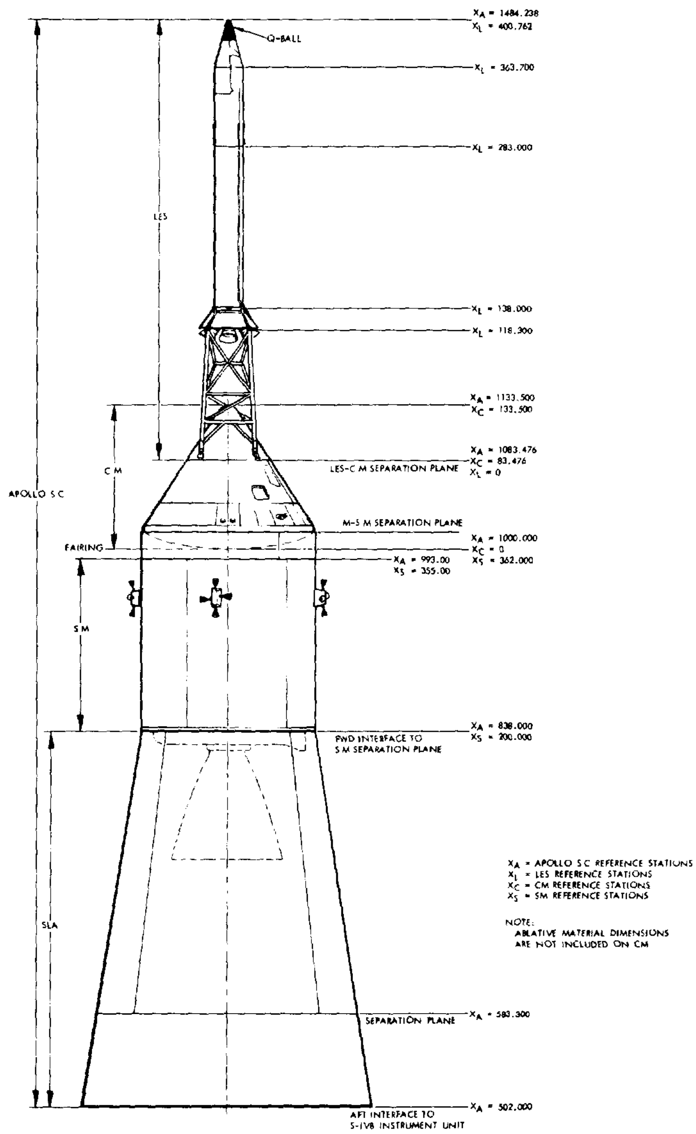


Figure A1-1.- Block II spacecraft reference stations.

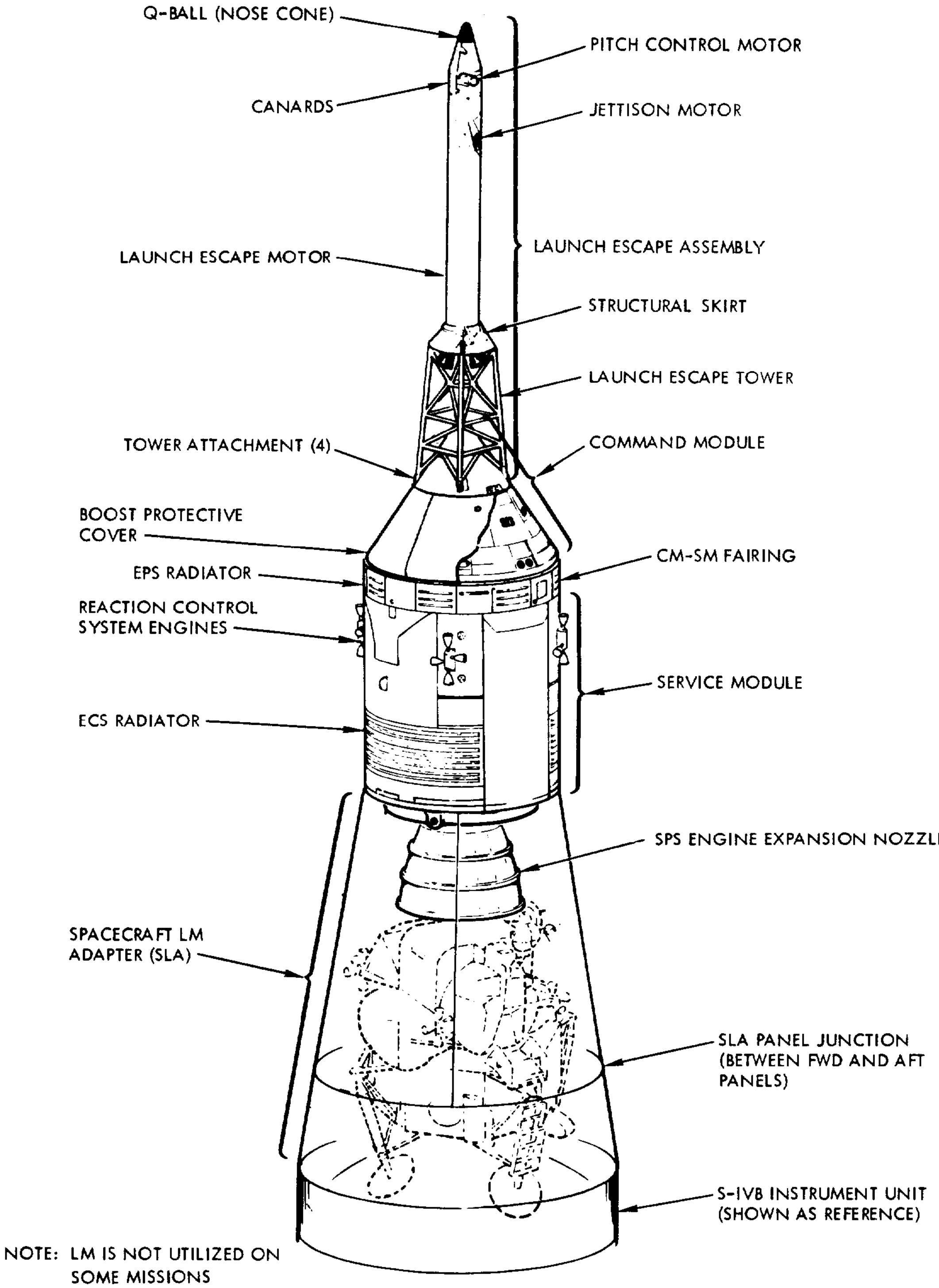


Figure A1-2.- Block II spacecraft configuration.

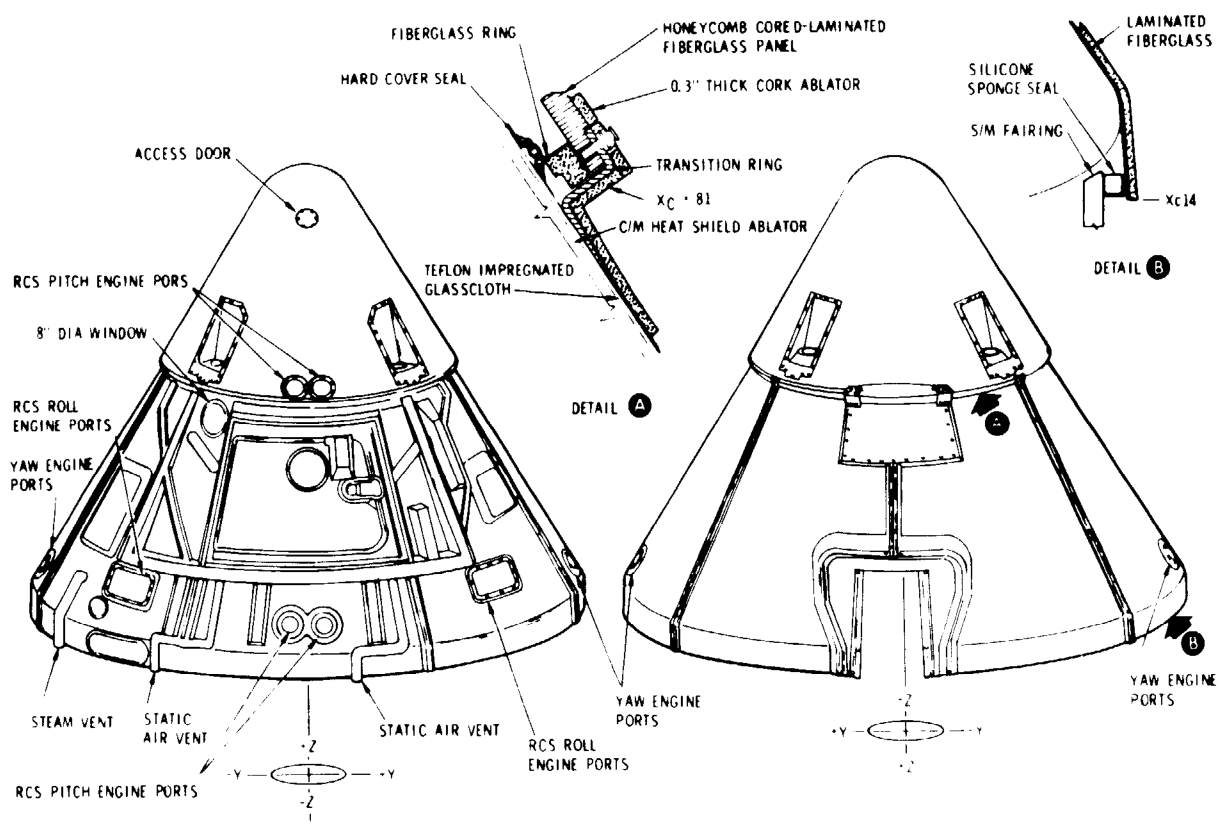


Figure A1-3.- Boost protective cover.

COMMAND MODULE

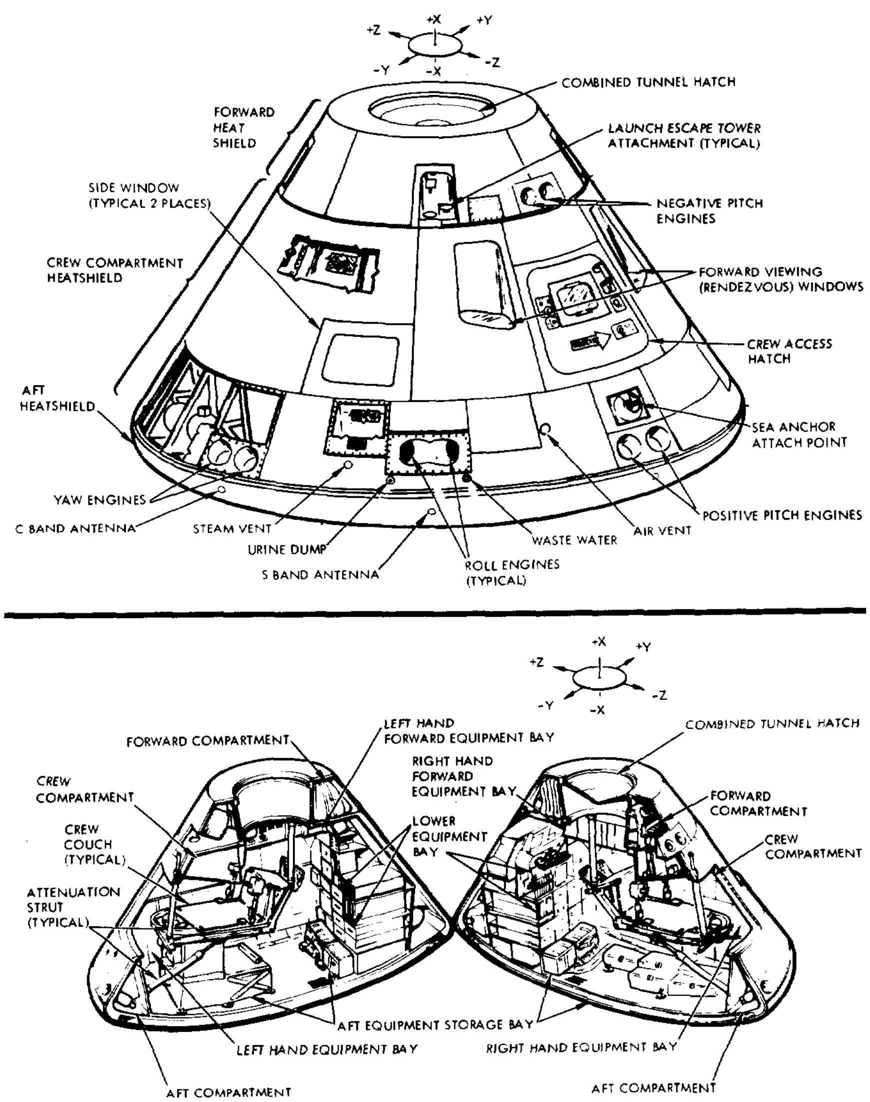
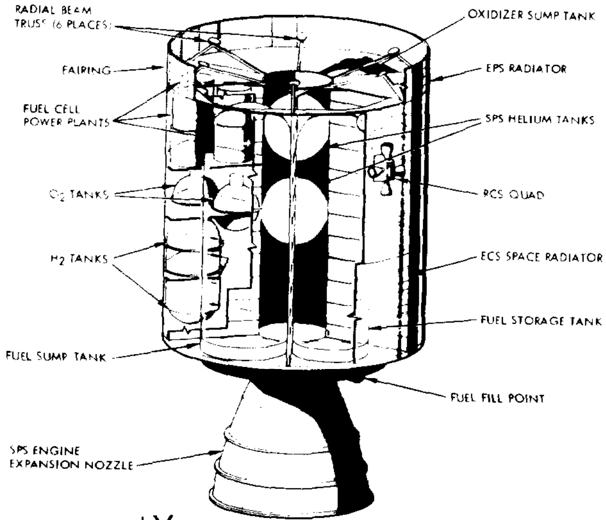
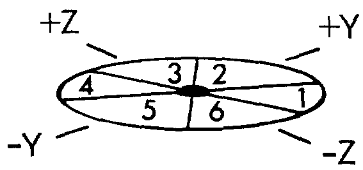


Figure A1-4.- Block II command module.

SERVICE MODULE

The service module (fig. A1-5) is a cylindrical structure formed by 1-inch-thick aluminum honeycomb panels. Radial beams, from milled aluminum alloy plates, separate the structure interior into six unequal sectors around a circular center section. Equipment contained within





1 and 4 are 50-degree sectors
2 and 5 are 70-degree sectors
3 and 6 are 60-degree sectors

Service module items

Sector 1
- Empty NASA equipment

Sector 2
- Environmental system space radiator
- Service propulsion system
- Reaction control system package (+Y -axis)
- Service propulsion system oxidizer sump tank

Sector 3
- Service propulsion system
- Reaction control system package (+Z -axis)
- Environmental system space radiator
- Service propulsion system oxidizer storage tank

Sector 4
- Fuel cell power plant (three)
- Helium servicing panel
- Super-critical oxygen tank (two)
- Super-critical hydrogen tank (two)
- Reaction control system control unit
- Electrical power system power control relay box
- Service module jettison controller sequencer (two)

Sector 5
- Environmental control system space radiator
- Service propulsion system fuel sump tank
- Reaction control system package (-Y axis)

Sector 6
- Environmental control system space radiator
- Reaction control system package (-Z axis)
- Service propulsion system fuel storage tank

Center Section
- Service propulsion system helium tank (two)
- Service propulsion system engine

Fairing
- Electrical power system space radiator's (eight)

Figure A1-5.- Service module.

the service module is accessible through maintenance doors located around the exterior surface of the module. Specific items, such as propulsion systems (SPS and RCS), fuel cells, and most of the SC onboard consumables (and storage tanks) contained in the SM compartments, are listed in figure A1-5. The service module is 12 feet 11 inches long (high) and 12 feet 10 inches in diameter.

Radial beam trusses on the forward portion of the SM structure provide a means for securing the CM to the SM. Alternate beams one, three, and five have compression pads for supporting the CM. Beams two, four, and six have shear-compression pads and tension ties. A flat center section in each tension tie incorporates redundant explosive charges for SM-CM separation. These beams and separation devices are enclosed within a fairing (26 inches high and 13 feet in diameter) between the CM and SM.

SPACECRAFT LM ADAPTER

The spacecraft LM adapter (SLA) (fig. A1-6) is a large truncated cone which connects the CSM and S-IVB on the launch vehicle. It houses the lunar module (LM), the nozzle of the service propulsion system, and the high-gain antenna in the stowed position. The adapter, constructed of eight 2-inch-thick aluminum panels, is 154 inches in diameter at the forward end (CM interface) and 260 inches at the aft end. Separation of the CSM from the SLA is accomplished by means of explosive charges which disengage the four SLA forward panels from the aft portion. The individual panels are restrained to the aft SLA by hinges and accelerated in rotation by pyrotechnic-actuated thrusters. When reaching an angle of 45 degrees measured from the vehicle's X-axis, spring thrusters (two per panel) jettison the panels. The panel jettison velocity and direction of travel is such as to minimize the possibility of recontact with the spacecraft or launch vehicle.

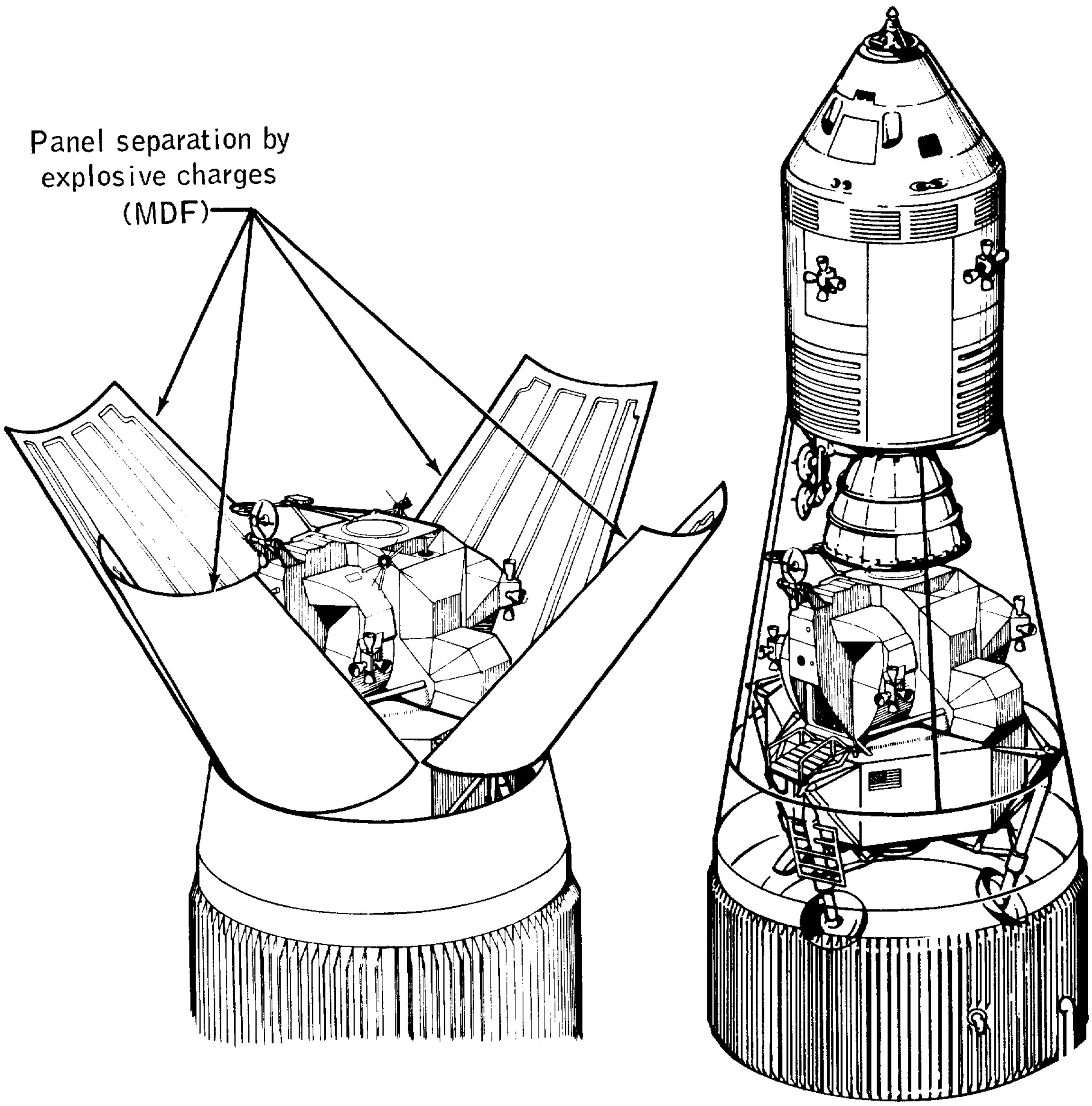


Figure A1-6.- Spacecraft LM adapter.

PART A2

SYSTEMS DESCRIPTION DATA

INTRODUCTION

Systems description data include description of operations, component description and design data, and operational limitations and restrictions. Part 2.1 describes the overall spacecraft navigation, guidance, and control requirements and the resultant systems interface. Parts A2.2 through A2.10 present data grouped by spacecraft systems, arranged in the following order: guidance and navigation, stabilization and control, service propulsion, reaction control, electrical power, environmental control, telecommunications, sequential, and caution and warnings. Part A2.11 deals with miscellaneous systems data. Part A2.12 deals with crew personal equipment. Part A2.13 deals with docking and crew transfer.

These data were extracted from the technical manual SM2A-03-BLOCK II-(1), Apollo Operations Handbook, Block II Spacecraft, Volume 1, dated January 15, 1970.

PART A2.1

GUIDANCE AND CONTROL

Guidance and Control Systems Interface

The Apollo guidance and control functions are performed by the primary guidance, navigation, and control system (PGNCS), and stabilization and control system (SCS). The PGNCS and SCS systems contain rotational and translational attitude and rate sensors which provide discrete input information to control electronics which, in turn, integrate and condition the information into control commands to the spacecraft propulsion systems. Spacecraft attitude control is provided by commands to the reaction control system (RCS). Major velocity changes are provided by commands to the service propulsion system (SPS). Guidance and control provides the following basic functions:

a. Attitude reference

b. Attitude control

c. Thrust and thrust vector control.

The basic guidance and control functions may be performed automatically, with primary control furnished by the command module computer (CMC) or manually, with primary control furnished by the flight crew. The subsequent paragraphs provide a general description of the basic functions.

Attitude Reference

The attitude reference function (fig. A2.1-1) provides display of the spacecraft attitude with reference to an established inertial reference. The display is provided by two flight director attitude indicators (FDAI) located on the main display console, panels 1 and 2. The displayed information consists of total attitude, attitude errors, and angular rates. The total attitude is displayed by the FDAI ball. Attitude errors are displayed by three needles across scales on the top, right, and bottom of the apparent periphery of the ball. Angular rates are displayed by needles across the top right, and bottom of the FDAI face.

Total attitude information is derived from the IMU stable platform or the gyro display coupler (GDC). The IMU provides total attitude by maintaining a gimbaled, gyro-stabilized platform to an inertial reference orientation. The GDC provides total attitude by updating attitude information with angular rate inputs from gyro assembly 1 or 2.

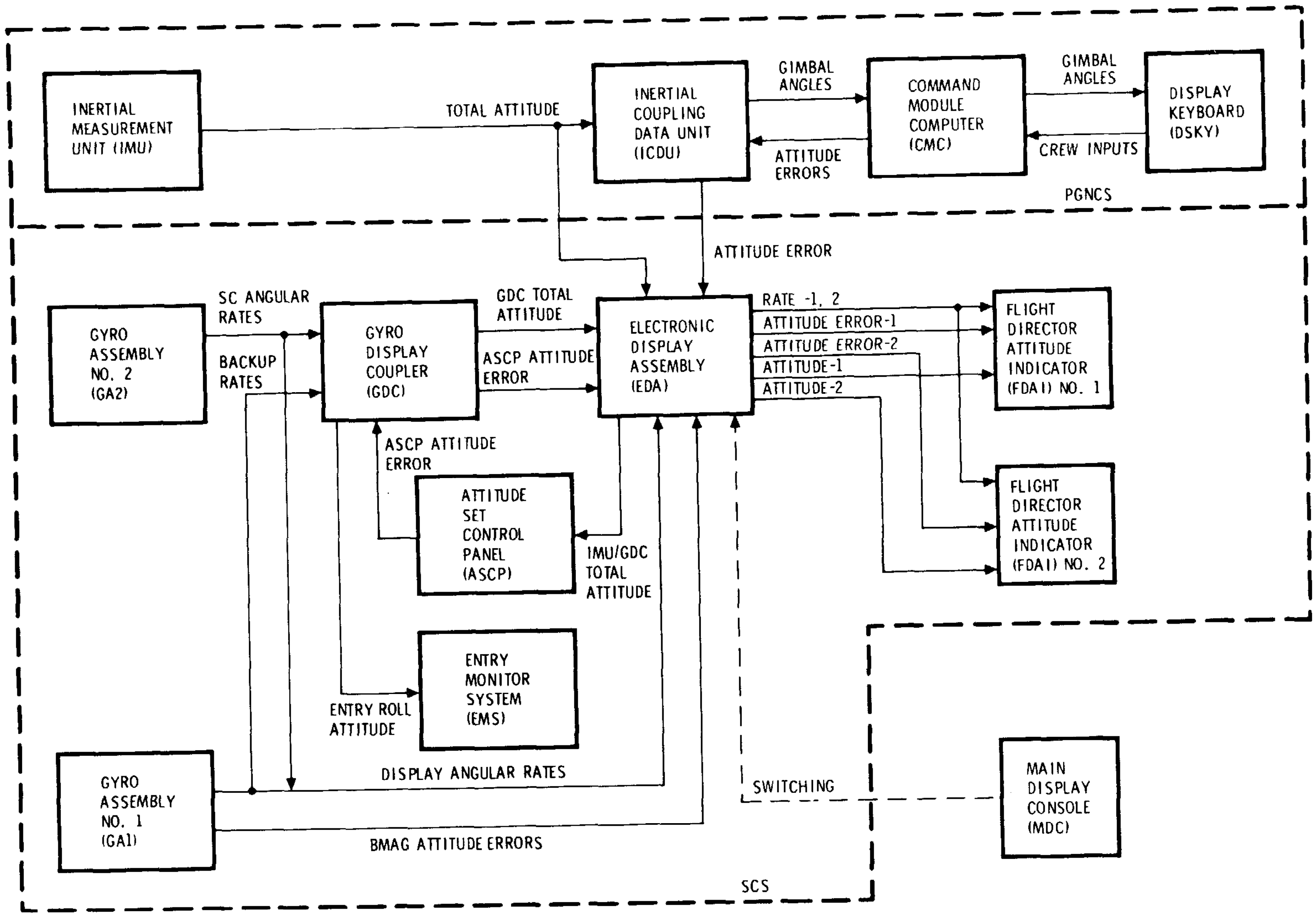


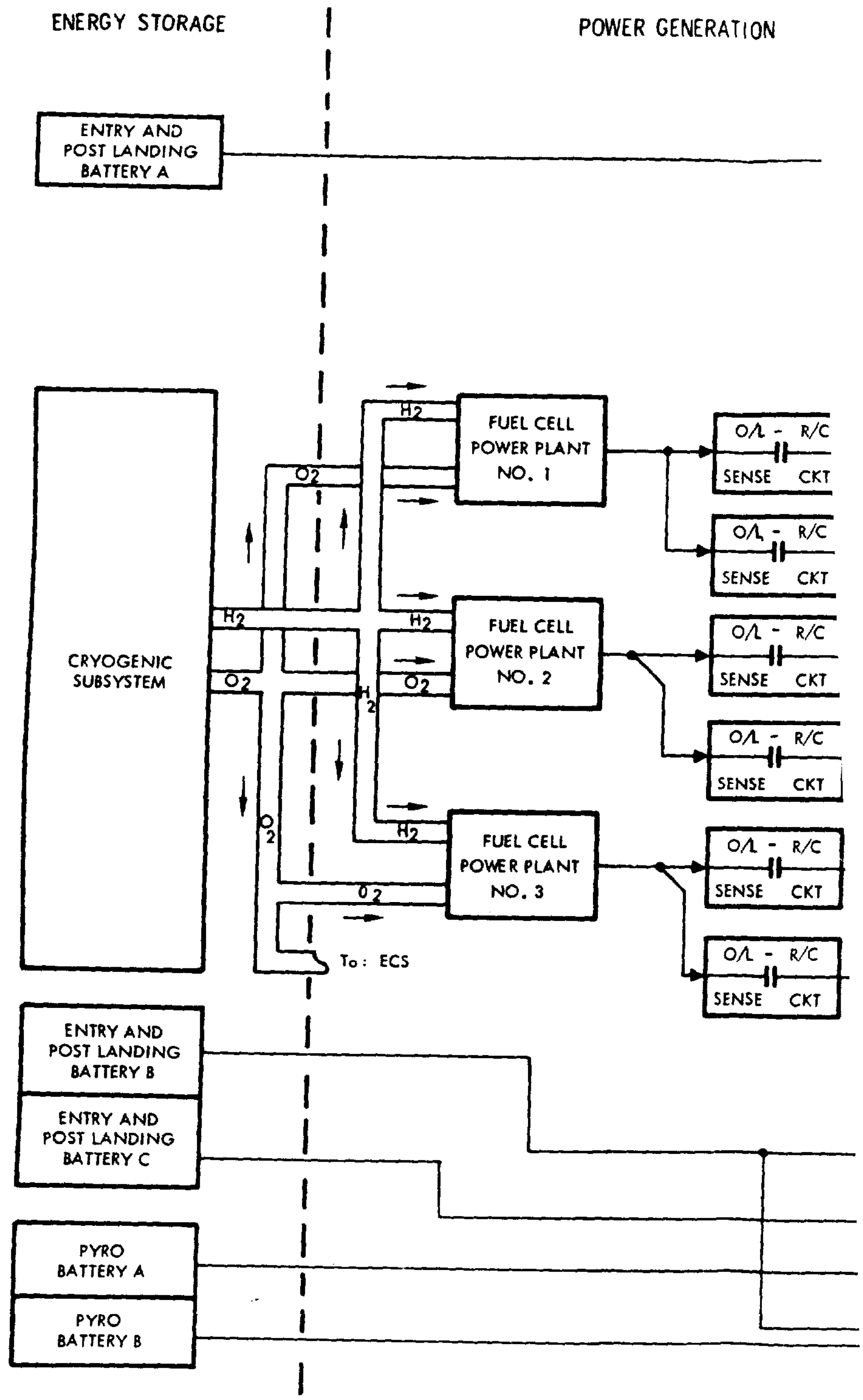
Figure A2.1-1.- Guidance and control.

Attitude error information is derived from three sources. The first source is from the IMU through the coupling data unit (CDU) which compares IMU gimbal angles with CMC commanded angles set into the CDU. Any angular difference between the IMU gimbals and the CDU angles is sent to the FDAI for display on the attitude error needles. The second source is from gyro assembly 1 which contains three (one for each of the X, Y, and Z axes) single-degree-of-freedom attitude gyros. Any spacecraft rotation about an axis will offset the case of a gyro from the float. This rotation is sensed as a displacement off null, and a signal is picked off which is representative of the magnitude and direction of rotation. This signal is sent to the FDAI for display on the attitude error needles. The third source is from the GDC which develops attitude errors by comparing angular rate inputs from gyro assembly 1 or 2 with an internally stored orientation. These data are sent to the FDAI for display on the attitude error needles.

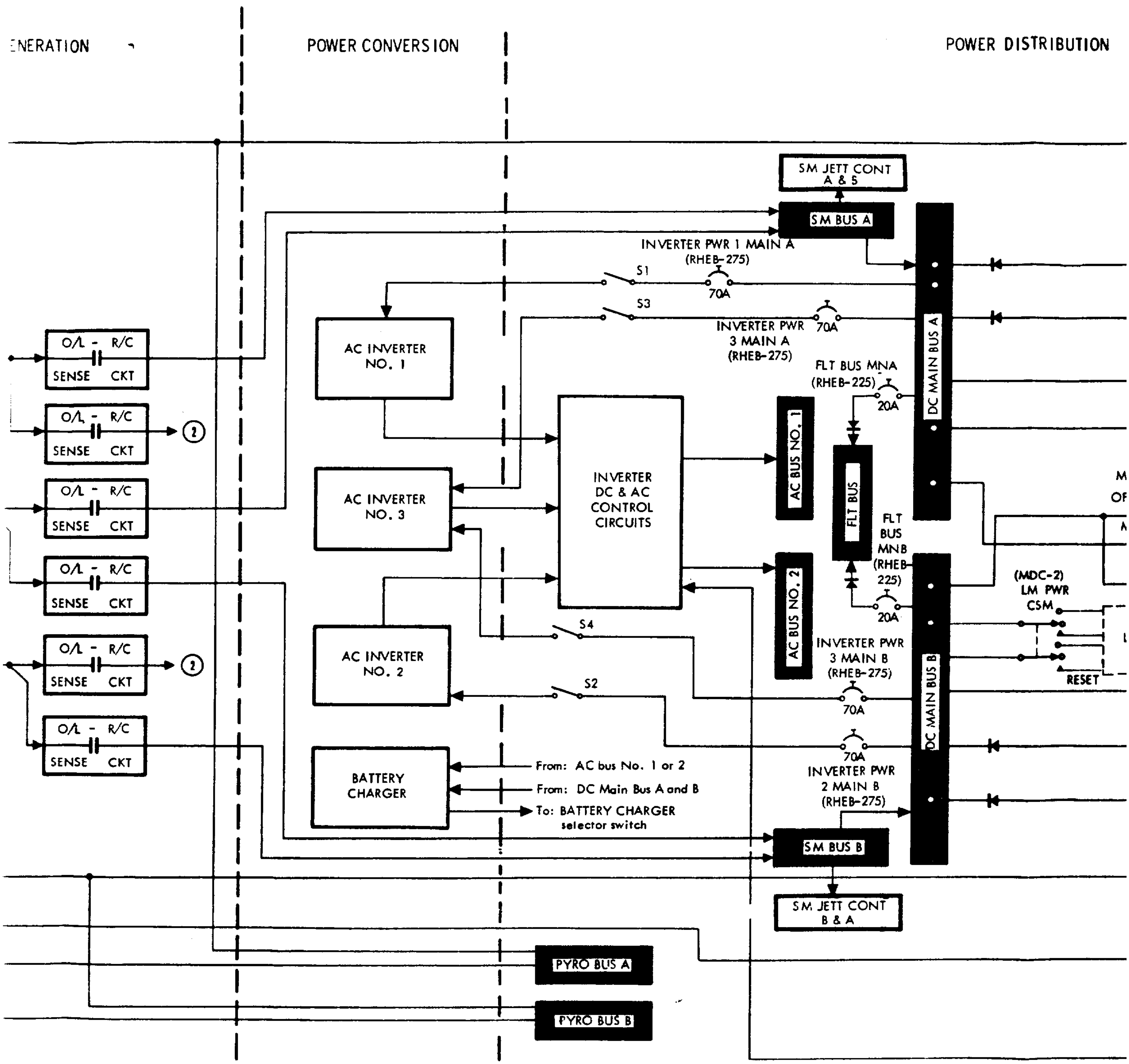
Angular rates are derived from either gyro assembly 1 or 2. Normally, the no. 2 assembly is used; however, gyro assembly 1 may be switched to a backup rate mode if desired. For developing rate information, the gyros are torqued to null when displaced; thus, they will produce an output only when the spacecraft is being rotated. The output signals are sent to the FDAI for display on the rate needles and to the GDC to enable updating of the spacecraft attitude.

Attitude Control

The attitude control function is illustrated in figure A2.1-2. The control may be to maintain a specific orientation, or to command small rotations or translations. To maintain a specific orientation, the attitude error signals, described in the preceding paragraph, are also routed to the control reaction jet on-off assembly. These signals are conditioned and applied to the proper reaction jet which fires in the direction necessary to return the spacecraft to the desired attitude. The attitude is maintained within specified deadband limits. The deadband is limited within both a rate and attitude limit to hold the spacecraft excursions from exceeding either an attitude limit or angular rate limit. To maneuver the spacecraft, the reaction jets are fired automatically under command of the CMC or manually by flight crew use of the rotation control. In either case, the attitude control function is inhibited until the maneuver is completed. Translations of small magnitude are performed along the +X axis for fuel settling of SPS propellants prior to burns, or for a backup deorbit by manual commands of the translation control. An additional control is afforded by enabling the minimum impulse control at the lower equipment bay. The minimum impulse control produces one directional pulse of small magnitude each time it is moved from detent. These small pulses are used to position the spacecraft for navigational sightings.



ENERGY STORAGE
POWER GENERATION
ENTRY AND POST LANDING BATTERY A
CRYOGENIC SUBSYSTEM
FUEL CELL POWER PLANT NO. 1
FUEL CELL POWER PLANT NO. 2
FUEL CELL POWER PLANT NO. 3
H2
O2
O/L - R/C
SENSE CKT
To: ECS
ENTRY AND POST LANDING BATTERY B
ENTRY AND POST LANDING BATTERY C
PYRO BATTERY A
PYRO BATTERY B



:NERATION
POWER CONVERSION
POWER DISTRIBUTION
SM JETT CONT A & B
SM BUS A
INVERTER PWR 1 MAIN A (RHEB-275)
S1
70A
S3
INVERTER PWR 3 MAIN A (RHEB-275)
70A
O/L - R/C SENSE CKT
AC INVERTER NO. 1
FLT BUS MNA (RHEB-225)
20A
DC MAIN BUS A
AC BUS NO. 1
FLT BUS
INVERTER DC & AC CONTROL CIRCUITS
AC INVERTER NO. 3
FLT BUS MNB (RHEB-225)
20A
AC BUS NO. 2
(MDC-2) LM PWR CSM
RESET
S4
AC INVERTER NO. 2
S2
INVERTER PWR 3 MAIN B (RHEB-275)
70A
DC MAIN BUS B
70A
INVERTER PWR 2 MAIN B (RHEB-275)
BATTERY CHARGER
From: AC bus No. 1 or 2
From: DC Main Bus A and B
To: BATTERY CHARGER selector switch
SM BUS B
SM JETT CONT B & A
PYRO BUS A
PYRO BUS B

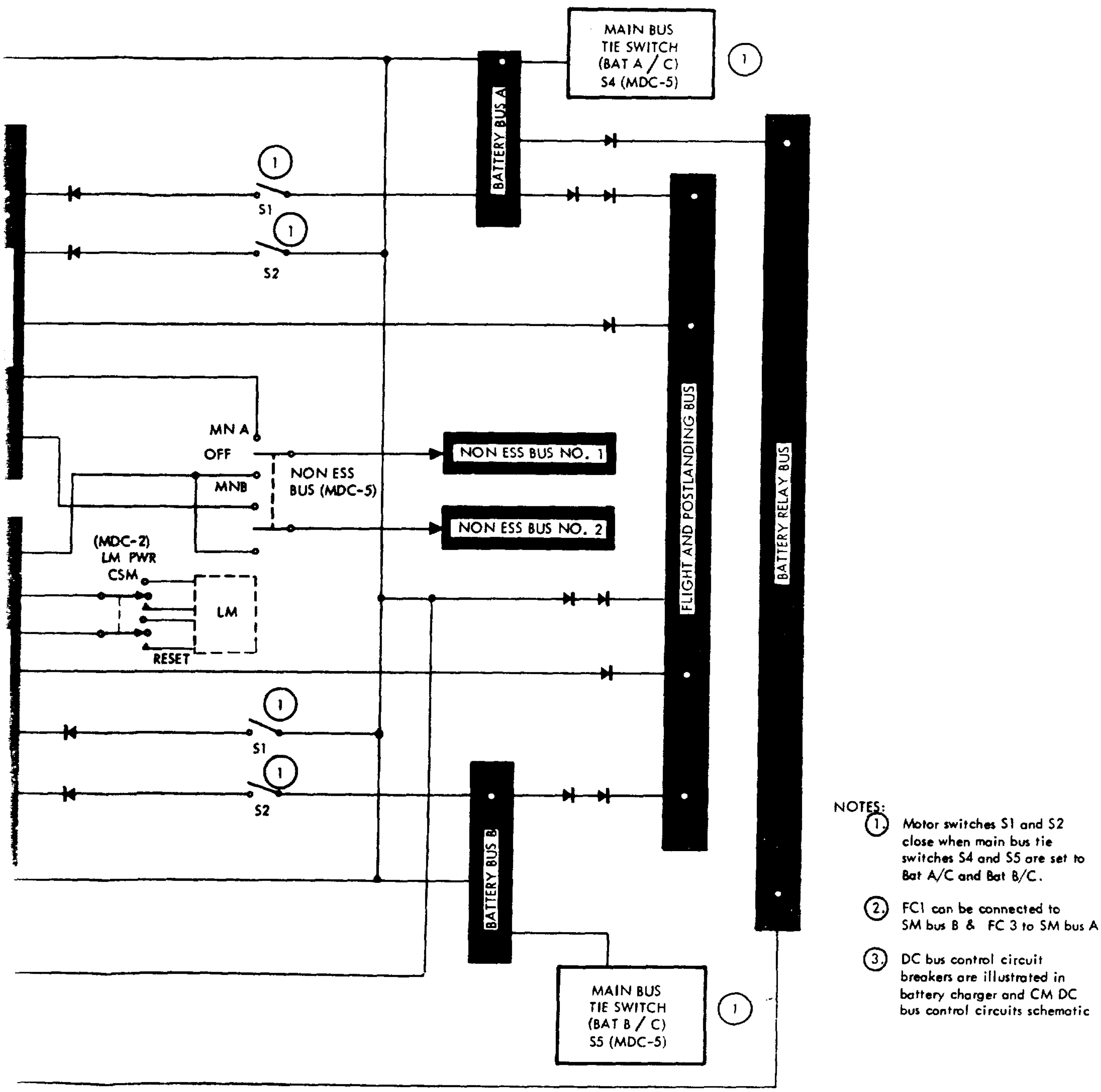


Figure A2.6-1.- Electrical power subsystem block diagram.

Functional Description

Energy storage.- The primary source of energy is the cryogenic gas storage system that provides fuel (H_2) and oxidizer (O_2) to the power generating system. Two hydrogen and two oxygen tanks, with the associated controls and plumbing, are located in the service module. Storage of reactants is accomplished under controlled cryogenic temperatures and pressures; automatic and manual pressure control is provided. Automatic heating of the reactants for repressurization is dependent on energy demand by the power generating and/or environmental control subsystems. Manual control can be used when required.

A secondary source of energy storage is provided by five silver oxide-zinc batteries located in the CM. Three rechargeable entry and postlanding batteries supply sequencer logic power at all times, supplemental dc power for peak loads, all operating power required for entry and postlanding, and can be connected to power either or both pyro circuits. Two pyro batteries provide energy for activation of pyro devices throughout all phases of a mission.

Power generation.- Three Bacon-type fuel cell power plants, generting power through electrochemical reaction of H_2 and O_2, supply primary dc power to spacecraft systems until CSM separation. Each power plant is capable of normally supplying from 400 to 1420 watts at 31 to 27 V dc (at fuel cell terminals) to the power distribution system. During normal operation all three power plants generate power, but two are adequate to complete the mission. Should two of the three malfunction, one power plant will insure successful mission termination; however, spacecraft loads must be reduced to operate within the limits of a single powerplant.

Normal fuel cell connection to the distribution system is: fuel cell 1 to main dc bus A; fuel cell 2 to main dc busses A and B; and fuel cell 3 to main dc bus B. Manual switch control is provided for power plant connection to the distribution system, and manual and/or automatic control for power plant isolation in case of a malfunction.

During the CSM separation maneuver, the power plants supply power through the SM buses to two SM jettison control sequencers. The sequencers sustain SM RCS retrofire during CSM separation and fire the SM positive roll RCS engines 2 seconds after separation to stabilize the SM during entry. Roll engine firing is terminated 7.5 seconds after separation. The power plants and SM buses are isolated from the umbilical through a SM deadface. The sequencers are connected to the SM buses when the CM/SM SEP switch (MDC-2) is activated; separation occurs 100 milliseconds after switch activation.

Power conversion.- Primary dc power is converted into ac by solid state static inverters that provide 115/200-volt 400-cps 3-phase ac power up to 1250 volt-amperes each. The ac power is connected by motor switch controls to two ac buses for distribution to the ac loads. One inverter has the capability of supplying all spacecraft primary ac power. One inverter can power both buses while the two remaining inverters act as redundant sources. However, throughout the flight, each bus is powered by a separate inverter. Provisions are made for inverter isolation in the event of malfunctions. Inverter outputs cannot be phase synchronized; therefore, interlocked motorized switching circuits are incorporated to prevent the connection of two inverters to the same bus.

A second conversion unit, the battery charger, assures keeping the three entry and postlanding batteries in a fully charged state. It is a solid state device utilizing dc from the fuel cells and ac from the inverter to develop charging voltage.

Power distribution.- Distribution of dc power is accomplished via two redundant dc buses in the service module which are connected to two redundant buses in the command module through a SM deadface, the CSM umbilical, and a CM deadface. Additional buses provided are: two dc buses for servicing nonessential loads; a flight bus for servicing inflight telecommunications equipment; two battery buses for distributing power to sequencers, gimbal motor controls, and servicing the battery relay bus for power distribution switching; and a flight and postlanding bus for servicing some communications equipment and the postlanding loads.

Three-phase ac is distributed via two redundant ac buses, providing bus selection through switches in the ac-operated component circuits.

Power to the lunar module is provided through two umbilicals which are manually connected after completion of transposition and docking. An average of 81 watts dc is provided to continuous heaters in the abort sensor assembly (ASA), and cycling heaters in the landing radar, rendezvous radar, S-band antenna, and inertial measurement unit (IMU). Power consumption with all heaters operating simultaneously is approximately 309 watts. LM floodlighting is also powered through the umbilical for use during manned lunar module operation while docked with the CSM.

A dc sensing circuit monitors voltage on each main dc bus, and an ac sensing circuit monitors voltage on each ac bus. The dc sensors provide an indication of an undervoltage by illuminating a warning light. The ac sensors illuminate a warning light when high- or low-voltage limits are exceeded. In addition, the ac sensors activate an automatic disconnect of the inverter from the ac bus during an overvoltage condition. The ac overload conditions are displayed by illumination of an overload warning light and are accompanied by a low voltage light. Additional sensors monitor fuel cell overload and reverse current conditions, providing an automatic disconnect, together with visual indications of the disconnect whenever either condition is exceeded.

Switches, meters, lights, and talk-back indicators are provided for controlling and monitoring all functions of the EPS.

Major Component/Subsystem Description

The subsequent paragraphs describe the cryogenic storage subsystem and each of the various EPS components.

Cryogenic storage.- The cryogenic storage subsystem (figs. A2.6-2 and A2.6-3) supplies hydrogen to the EPS, and oxygen to the EPS, ECS, and for initial LM pressurization. The two tanks in the hydrogen and oxygen systems are of sufficient size to provide a safe return from the furthest point of the mission on the fluid remaining in any one tank. The physical data of the cryogenic storage subsystem are as follows:

	Weight of usable cryogenics (lb/tank)	Design storage pressure (psia)	Minimum allowable operating pressure (psia)	Approximate flow rate at min dq/dm (+145° F environment) (lb/hr-2 tanks)	Approximate quantities at minimum heater and fan cycling (per tank) (min dq/dm)
O_2	320 (min)	900±35	150	1.71	45 to 25%
H_2	28 (min)	245 (+15, -20)	100	0.140	53 to 33%

Initial pressurization from fill to operating pressures is accomplished by GSE. After attaining operating pressures, the cryogenic fluids are in a single-phase condition, therefore, completely homogeneous. This avoids sloshing which could cause sudden pressure fluctuations, possible damage to internal components, and prevents positive mass quantity gauging. The single-phase expulsion process continues at nearly constant pressure and increasing temperature above the 2-phase region.

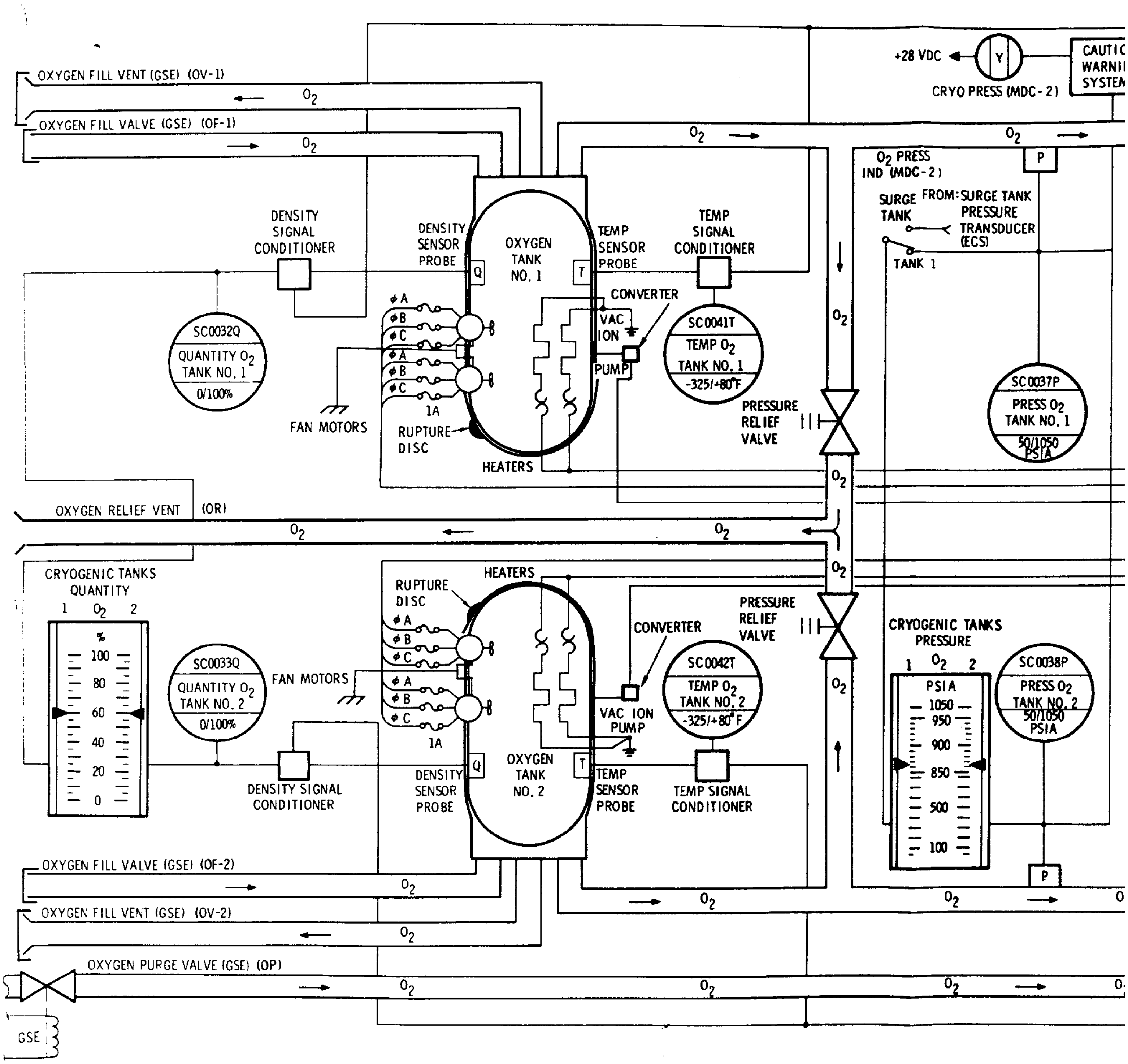
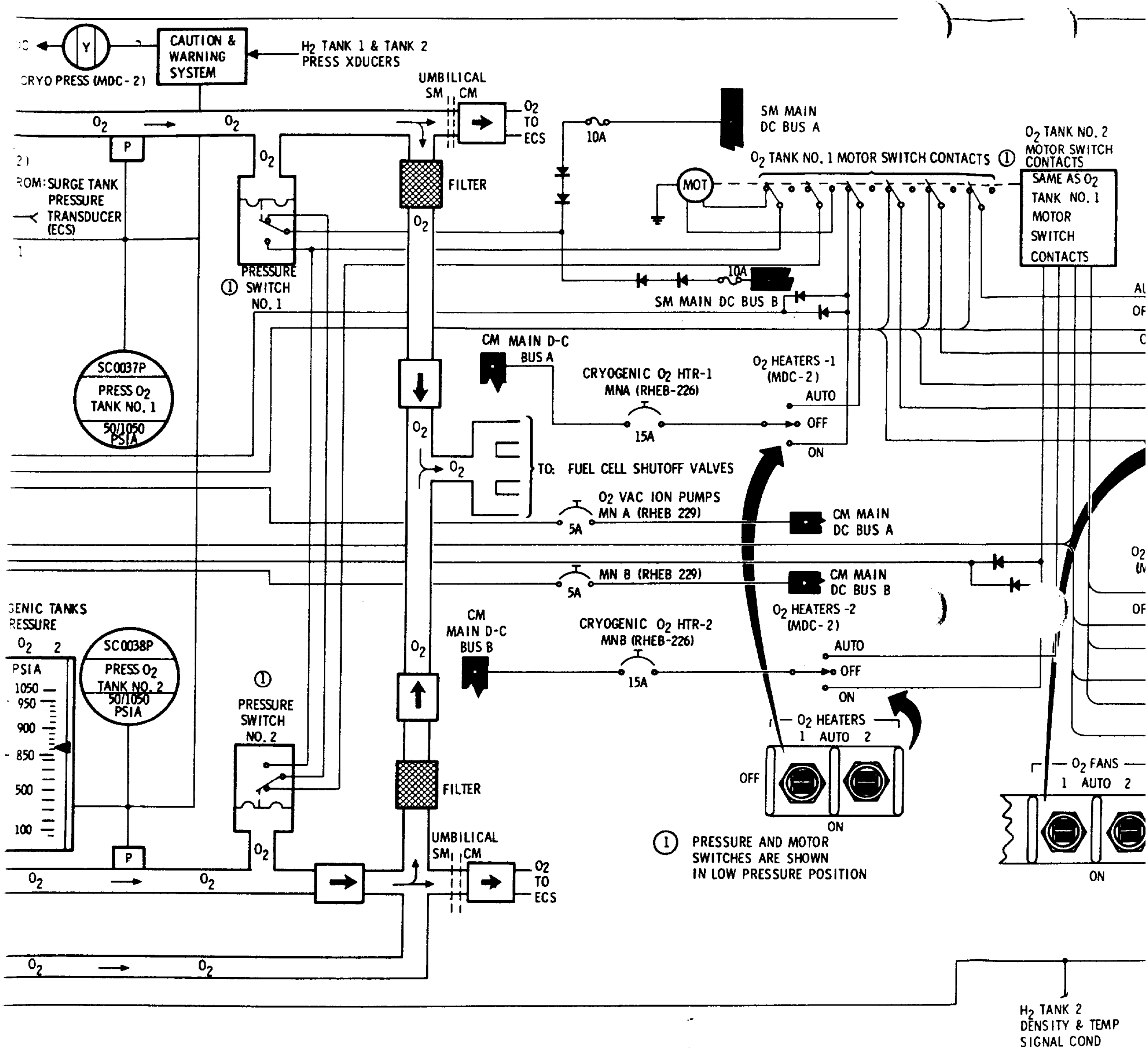
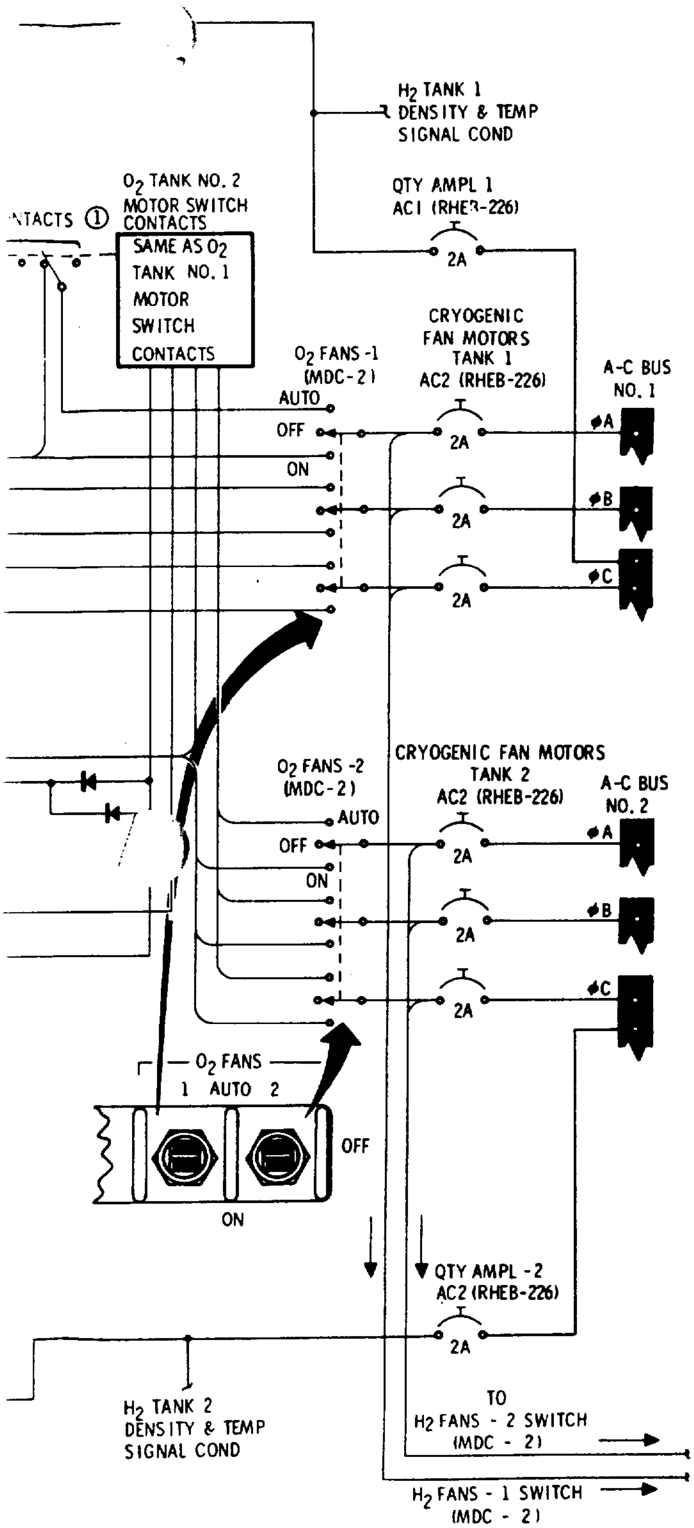


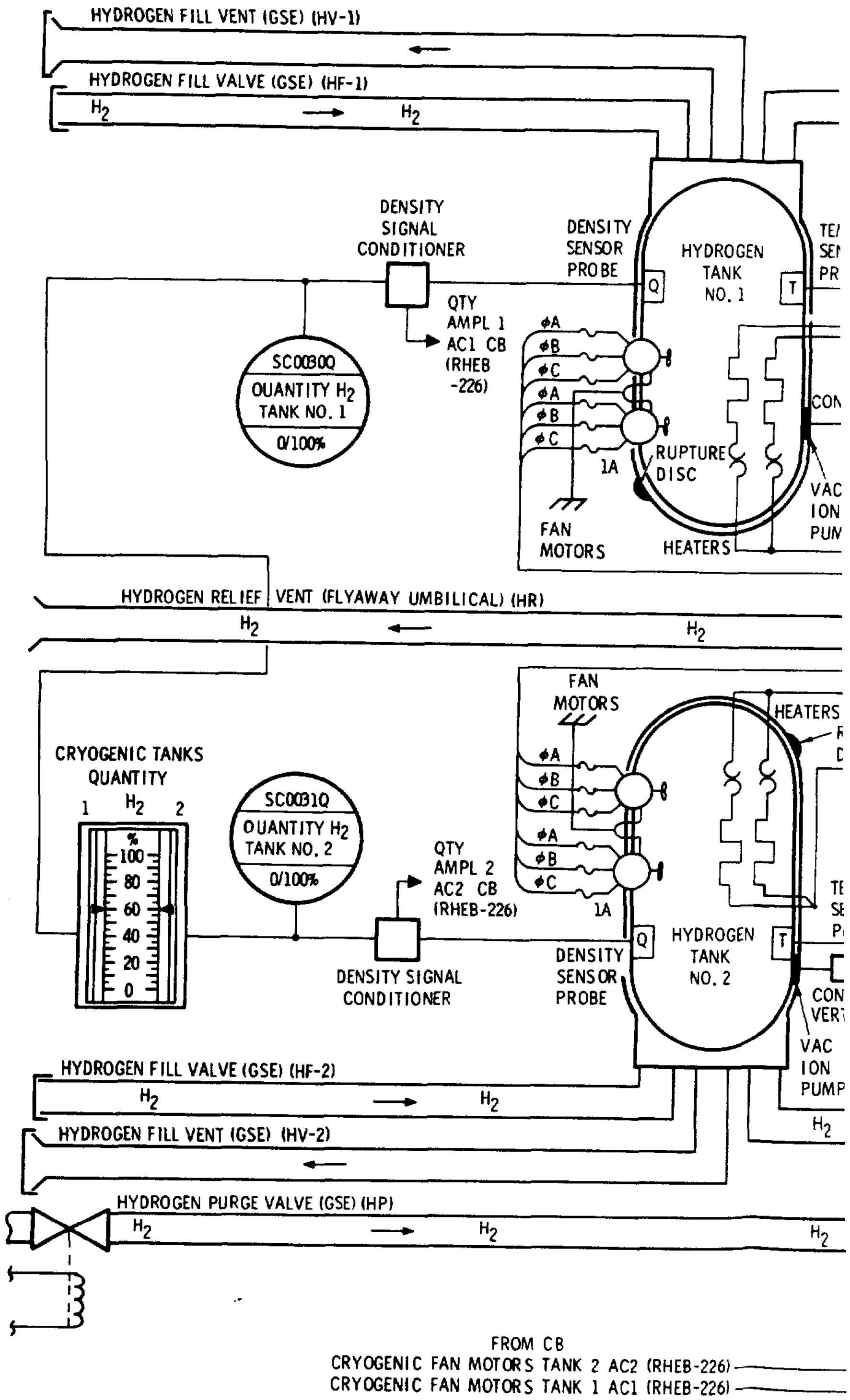
Figure A2.6-2.- Cryogenic storage subsystem (oxygen).



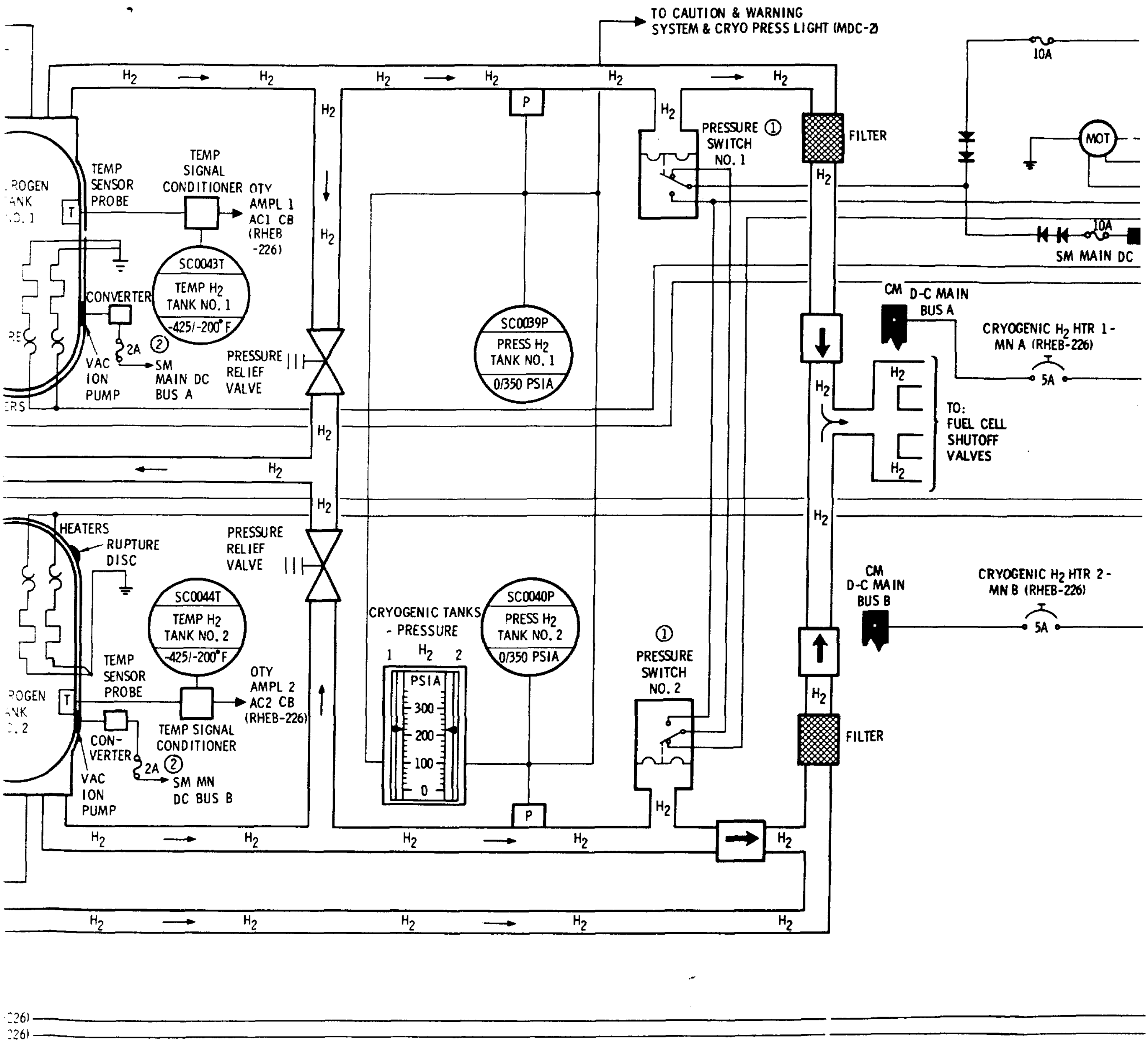
CAUTION & WARNING SYSTEM
CRYO PRESS (MDC-2)
H_2 TANK 1 & TANK 2 PRESS XDUCERS
UMBILICAL SM CM
O_2 TO ECS
SM MAIN DC BUS A
10A
O_2 TANK NO. 2 MOTOR SWITCH CONTACTS
SAME AS O_2 TANK NO. 1 MOTOR SWITCH CONTACTS
O_2 TANK NO. 1 MOTOR SWITCH CONTACTS ①
MOT
FROM: SURGE TANK PRESSURE TRANSDUCER (ECS)
FILTER
PRESSURE SWITCH NO. 1 ①
SM MAIN DC BUS B
SC0037P
PRESS O_2 TANK NO. 1
50/1050 PSIA
CM MAIN D-C BUS A
CRYOGENIC O_2 HTR-1 MNA (RHEB-226)
15A
O_2 HEATERS -1 (MDC-2)
AUTO
OFF
ON
TO: FUEL CELL SHUTOFF VALVES
O_2 VAC ION PUMPS MN A (RHEB 229)
5A
CM MAIN DC BUS A
MN B (RHEB 229)
CM MAIN DC BUS B
CRYOGENIC TANKS PRESSURE
O_2 2
PSIA
1050
950
900
850
500
100
SC0038P
PRESS O_2 TANK NO. 2
50/1050 PSIA
CM MAIN D-C BUS B
CRYOGENIC O_2 HTR-2 MNB (RHEB-226)
O_2 HEATERS -2 (MDC-2)
PRESSURE SWITCH NO. 2 ①
O_2 HEATERS
1 AUTO 2
O_2 FANS
1 AUTO 2
① PRESSURE AND MOTOR SWITCHES ARE SHOWN IN LOW PRESSURE POSITION
H_2 TANK 2 DENSITY & TEMP SIGNAL COND



H_2 TANK 1
DENSITY & TEMP
SIGNAL COND
QTY AMPL 1
AC1 (RHEB-226)
2A
O_2 TANK NO. 2
MOTOR SWITCH
CONTACTS
SAME AS O_2
TANK NO. 1
MOTOR
SWITCH
CONTACTS
CRYOGENIC
FAN MOTORS
TANK 1
AC2 (RHEB-226)
O_2 FANS -1
(MDC-2)
AUTO
OFF
ON
A-C BUS
NO. 1
ϕA
ϕB
ϕC
CRYOGENIC FAN MOTORS
TANK 2
AC2 (RHEB-226)
O_2 FANS -2
(MDC-2)
A-C BUS
NO. 2
— O_2 FANS —
1 AUTO 2
OFF
ON
QTY AMPL -2
AC2 (RHEB-226)
H_2 TANK 2
DENSITY & TEMP
SIGNAL COND
TO
H_2 FANS - 2 SWITCH
(MDC - 2)
H_2 FANS - 1 SWITCH
(MDC - 2)



HYDROGEN FILL VENT (GSE) (HV-1)
HYDROGEN FILL VALVE (GSE) (HF-1)
H2
H2
DENSITY
SIGNAL
CONDITIONER
DENSITY
SENSOR
PROBE
HYDROGEN
TANK
NO. 1
QTY
AMPL 1
AC1 CB
(RHEB
-226)
SC0030Q
QUANTITY H2
TANK NO. 1
0/100%
RUPTURE
DISC
1A
FAN
MOTORS
HEATERS
HYDROGEN RELIEF VENT (FLYAWAY UMBILICAL) (HR)
H2
H2
FAN
MOTORS
HEATERS
CRYOGENIC TANKS
QUANTITY
1 H2 2
%
100
80
60
40
20
0
SC0031Q
QUANTITY H2
TANK NO. 2
0/100%
QTY
AMPL 2
AC2 CB
(RHEB-226)
1A
HYDROGEN
TANK
NO. 2
DENSITY SIGNAL
CONDITIONER
DENSITY
SENSOR
PROBE
VAC
ION
PUMP
HYDROGEN FILL VALVE (GSE) (HF-2)
H2
H2
HYDROGEN FILL VENT (GSE) (HV-2)
H2
HYDROGEN PURGE VALVE (GSE) (HP)
H2
H2
H2
FROM CB
CRYOGENIC FAN MOTORS TANK 2 AC2 (RHEB-226)
CRYOGENIC FAN MOTORS TANK 1 AC1 (RHEB-226)



TO CAUTION & WARNING
SYSTEM & CRYO PRESS LIGHT (MDC-2)
10A
MOT
PRESSURE ① SWITCH NO. 1
FILTER
TEMP SENSOR PROBE
TEMP SIGNAL CONDITIONER
QTY AMPL 1 AC1 CB (RHEB -226)
SC0043T
TEMP H_2 TANK NO. 1
-425/-200°F
CONVERTER
2A ②
SM MAIN DC BUS A
VAC ION PUMP
PRESSURE RELIEF VALVE
SC0039P
PRESS H_2 TANK NO. 1
0/350 PSIA
SM MAIN DC
CM D-C MAIN BUS A
CRYOGENIC H_2 HTR 1 - MN A (RHEB-226)
5A
TO: FUEL CELL SHUTOFF VALVES
HEATERS
RUPTURE DISC
PRESSURE RELIEF VALVE
SC0044T
TEMP H_2 TANK NO. 2
-425/-200°F
CRYOGENIC TANKS - PRESSURE
PSIA
300
200
100
0
SC0040P
PRESS H_2 TANK NO. 2
0/350 PSIA
① PRESSURE SWITCH NO. 2
CM D-C MAIN BUS B
CRYOGENIC H_2 HTR 2 - MN B (RHEB-226)
TEMP SENSOR PROBE
QTY AMPL 2 AC2 CB (RHEB-226)
TEMP SIGNAL CONDITIONER
CON-VERTER
SM MN DC BUS B
VAC ION PUMP
FILTER

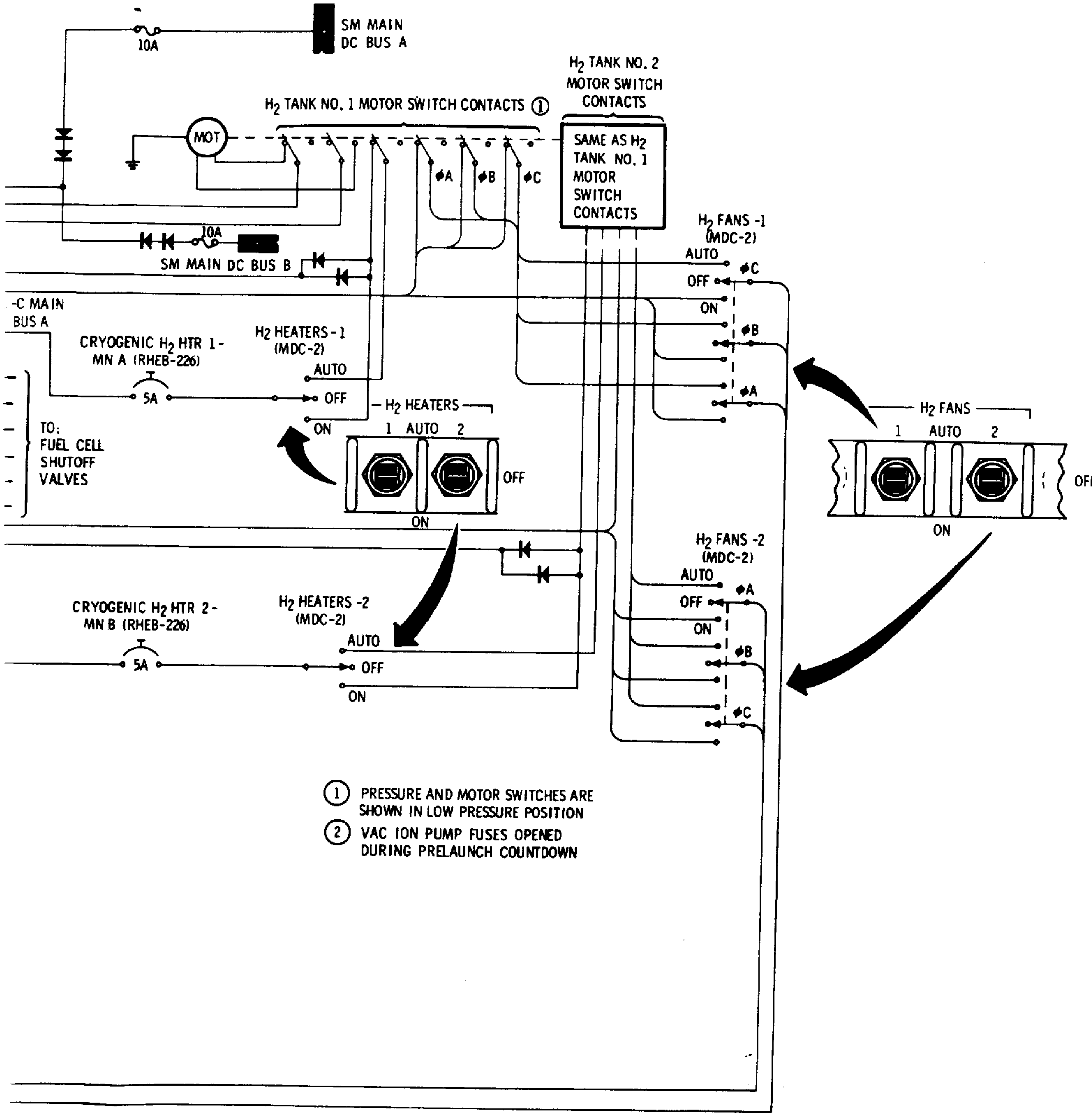


Figure A2.6-3.- Cryogenic storage subsystem (hydrogen)

Two parallel dc heaters in each tank supply the heat necessary to maintain design pressures. Two parallel 3-phase ac circulating fans circulate the fluid over the heating elements to maintain a uniform density and decrease the probability of stratification. A typical heater and fan installation is shown in figure A2.6-4. Relief valves provide overpressure relief, check valves provide tank isolation, and individual fuel cell shutoff valves provide isolation of malfunctioning power plants. Filters extract particles from the flowing fluid to protect the ECS and EPS components. The pressure transducers and temperature probes indicate the thermodynamic state of the fluid. A capacitive quantity probe indicates quantity of fluid remaining in the tanks.

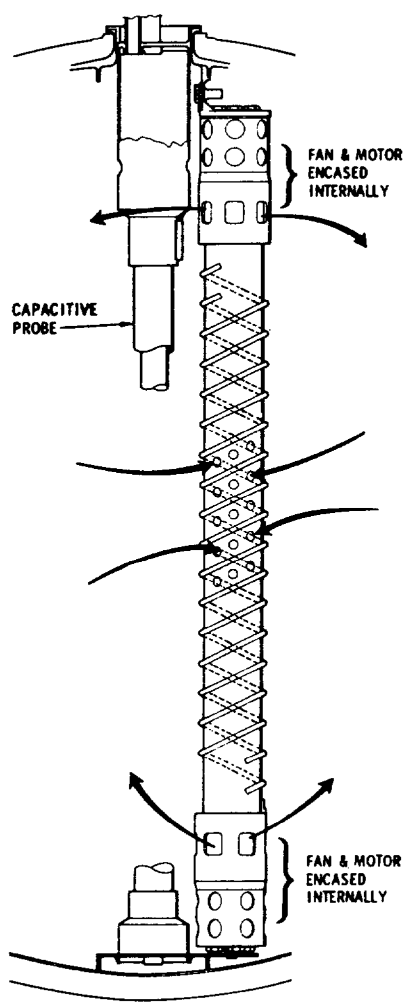


Figure A2.6-4.- Cryogenic pressurization and quantity measurement devices.

Repressurization of the systems can be automatically or manually controlled by switch selection. The automatic mode is designed to give a single-phase reactant flow into the feed lines at design pressures. The heaters and fans are automatically controlled through a pressure switch-motor switch arrangement. As pressure in the tanks decreases, the pressure switch in each tank closes to energize the motor switch, closing contacts in the heater and fan circuits. Both tanks have to decrease in pressure before heater and fan circuits are energized. When either tank reaches the upper operating pressure limit, that respective pressure switch opens to again energize the motor switch, thus opening the heater and fan circuits to both tanks. The O_2 circuits are energized at 865 psia minimum and de-energized at 935 psia maximum. The H_2 circuits energize at 225 psia minimum and de-energize at 260 psia maximum. The most accurate quantity readout will be acquired shortly after the fans have stopped. During all other periods partial stratification may degrade quantity readout accuracy.

When the systems reach the point where heater and fan cycling is at a minimum (due to a reduced heat requirement), heat leak of the tank is sufficient to maintain design pressures, provided flow is within the min dq/dm values shown in the preceding tabulation. This realm of operation is referred to as the min dq/dm region. The minimum heat requirement region for oxygen starts at approximately 45-percent quantity and terminates at approximately 25-percent quantity. Between these tank quantities, minimum heater and fan cycling will occur under normal usage. The amount of heat required for repressurization at quantities below 25-percent starts to increase until below the 3-percent level practically continuous heater and fan operation is required. In the hydrogen system, the quantity levels for minimum heater and fan cycling are between approximately 53 and 33 percent, with continuous operation occurring at approximately the 5 percent-level.

Assuming a constant level flow from each tank (O_2 - 1 lb/hr, H_2 - 0.09 lb/hr) each successive repressurization period is of longer duration. The periods between repressurizations lengthen as quantity decreases from full to the minimum dq/dm level, and become shorter as quantity decreases from the minimum dq/dm level to the residual level. Approximate repressurization periods are shown in table A2.6-I, which also shows the maximum flow rate in pounds per hour from a single tank with the repressurization circuits maintaining minimum design pressure.

The maximum continuous flow that each cryogenic tank can provide at minimum design pressure is dependent on the quantity level and the heat required to maintain that pressure. The heat required to maintain a constant pressure decreases as quantity decreases from full to the minimum dq/dm point. As quantity decreases beyond the minimum dq/dm region, the heat required to maintain a constant pressure increases. As fluid is withdrawn, a specific amount of heat is withdrawn. When the withdrawal rate exceeds the heat that can be supplied by the heaters, fan motors, and heat leak, there is a resultant pressure decrease below the minimum design operating level.

The ability to sustain pressure and flow is a factor of the amount of heat required versus the heat provided by heaters, fan motors, and heat leak. Since heat leak characteristics of each tank vary slightly, the flow each tank can provide will also vary to a small degree. Heat input from heaters, fan motors, and heat leak into an O_2 tank is 595.87 Btu/hour (113.88-watt heaters supply 389.67 Btu, 52.8-watt fan motors supply 180.2 Btu, and heat leak supplies 26 Btu). Heat input from similar sources into a H_2 tank is 94.6 Btu/hr (18.6-watt heaters supply 63.48 Btu, 7-watt fan motors supply 23.89 Btu, and heat leak supplies 7.24 Btu). These figures take into consideration the line loss between the power source and the operating component.

TABLE A2.6-I.- OXYGEN AND HYDROGEN REPRESSURIZATION AND FLOW.

Quantity (percent)	Oxygen		Hydrogen	
	Repressurization time, minutes (865 to 935 psia)	Flow at 865 psia	Repressurization time, minutes (225 to 260 psia)	Flow at 225 psia
100	4.0	3.56	20.0	0.38
95	4.3	3.97	21.0	0.42
90	4.6	4.55	22.0	0.46
85	5.0	5.27	23.0	0.49
80	5.4	6.02	24.5	0.52
75	5.7	7.01	26.5	0.65
70	6.5	7.94	28.5	0.76
65	7.4	9.01	31.0	0.80
60	8.7	10.80	33.5	0.87
55	9.6	12.54	36.0	0.93
50	10.8	14.19	39.0	0.97
45	11.5	15.69	41.0	0.98
40	12.4	17.01	41.0	0.97
35	12.6	17.56	41.0	0.94
30	13.0	17.56	40.5	0.91
25	13.1	16.55	40.5	0.83
20	13.2	15.48	42.0	0.71
15	14.5	12.28	47.0	0.54
10	17.8	8.76	58.0	0.37
7.5	21.4	7.09	71.0	0.23
5	24.0	5.37	Continuous	0.16

To avoid excessive temperatures, which could be realized during continuous heater and fan operation at extremely low quantity levels, a thermal sensitive interlock device is in series with each heater element. The device automatically opens the heater circuits when internal tank shell temperatures reach +90° F., and closes the circuits at +70° F. Assuming normal consumption, oxygen temperature will be approximately -157° F., at mission termination, while hydrogen temperature will be approximately -385° F.

The manual mode of operation bypasses the pressure switches, and supplies power directly to the heaters and/or fans through the individual control switches. It can be used in case of automatic control failure, heater failure, or fan failure.

Tank pressures and quantities are monitored on meters located on MDC-2. The caution and warning system (CRYO PRESS) will alarm when oxygen pressure in either tank exceeds 950 psia or falls below 800 psia. The hydrogen system alarms above 270 psia and below 220 psia. Since a common lamp is provided, reference must be made to the individual pressure and quantity meters (MDC-2) to determine the malfunctioning tank. Tank pressures, quantities, and reactant temperatures of each tank are telemetered to MSFN.

Oxygen relief valves vent at a pressure between 983 and 1010 psig and reseat at 965 psig minimum. Hydrogen relief valves vent at a pressure between 273 and 285 psig, and reseat at 268 psig minimum. Full flow venting occurs approximately 2 pounds above relief valve opening pressure.

All the reactant tanks have vac-ion pumps to maintain the integrity of the vacuum between the inner and outer shell, thus maintaining heat leak at or below the design level. SM main dc bus A distributes power to the H_2 tank 1 pump and bus B to the H_2 tank 2 pump. Fuses provide power source protection. These fuses are removed during prelaunch to disable the circuit for flight. Circuit breakers, O_2 VAC ION PUMPS - MNA - MNB (RHEB-229), provide power source protection for the CM main buses, which distribute power to the O_2 vac-ion pumps. The circuit breakers allow use of the O_2 vac-ion pump circuits throughout flight, and provide a means of disabling circuit if necessary. The O_2 circuit breakers are opened on the launch pad, and closed at 90 percent tank quantity.

The most likely period of overpressurization in the cryogenic system will occur during operation in the minimum dq/dm region. The possibility of overpressurization is predicted on the assumption of a vacuum breakdown, resulting in an increase in heat leak. Also, under certain conditions, that is, extremely low power levels and/or a depressurized cabin, demand may be lower than the minimum dq/dm flow necessary. Any of the preceding conditions would result in an increase of pressure within a tank.

In the case of hydrogen tank overpressurization, prior to reaching relief valve cracking pressure, tank pressure can be decreased by performing an unscheduled fuel cell hydrogen purge. A second method for relieving overpressure is to increase electrical loads, thus increasing fuel cell demand. However, in using this method, consideration must be given to the fact that there will be an increase in oxygen consumption, which may not be desirable.

Several procedures can be used to correct an overpressure condition in the oxygen system. One is to perform an unscheduled fuel cell purge. A second is to increase oxygen flow into the command module by opening the ECS DIRECT O_2 valve. The third is to increase electrical loads, which may not be desirable because this method will also increase hydrogen consumption.

A requirement for an overpressure correction in both reactant systems simultaneously is remote, since both reactant systems do not reach the minimum dq/dm region in parallel.

During all missions, to retain a single tank return capability, there is a requirement to maintain a balance between the two tanks in each of the reactant systems. When a 2- to 4-percent difference is indicated on the oxygen quantity meters (MDC-2), the O_2 HEATERS switch (MDC-2) of the lesser tank is positioned to OFF until tank quantities equalize. A 3-percent difference in the hydrogen quantity meters (MDC-2) will require positioning the H_2 HEATERS switch (MDC-2) of the lesser tank to OFF until tank quantities equalize. This procedure retains the automatic operation of the repressurization circuits, and provides for operation of the fan motors during repressurization to retain an accurate quantity readout in all tanks. The necessity for balancing should be determined shortly after a repressurization cycle, since quantity readouts will be most accurate at this time.

Batteries.- Five silver oxide-zinc storage batteries are incorporated in the EPS. These batteries are located in the CM lower equipment bay.

Three rechargeable entry and postlanding batteries (A, B, and C) power the CM systems after CSM separation and during postlanding. Prior to CSM separation, the batteries provide a secondary source of power while the fuel cells are the primary source. The entry batteries are used for the following purposes:

a. Provide CM power after CSM separation

b. Supplement fuel cell power during peak load periods (Delta V maneuvers)

c. Provide power during emergency operations (failure of two fuel cells)

d. Provide power for EPS control circuitry (relays, indicators, etc.)

e. Provide sequencer logic power

f. Provide power for recovery aids during postlanding

g. Batteries A, B, or C can power pyro circuits by selection.

Each entry and postlanding battery is mounted in a vented plastic case and consists of 20 silver oxide-zinc cells connected in series. The cells are individually encased in plastic containers which contain relief valves that open at 35 ± 5 psig, venting during an overpressure into the battery case. The three cases can be vented overboard through a common manifold, the BATTERY VENT valve (RHEB-252), and the ECS waste water dump line.

Since the BATTERY VENT is closed prior to lift-off, the interior of the battery cases is at a pressure of one atmosphere. The pressure is relieved after earth orbit insertion and completion of cabin purge by positioning the control to VENT for 5 seconds. After completion the control is closed, and pressure as read out on position 4A of the System Test Meter (LEB-101) should remain at zero unless there is battery outgassing. Outgassing can be caused by an internal battery failure, an abnormal high-rate discharge, or by overcharging. If a pressure increase is noted on the system test meter, the BATTERY VENT is positioned to VENT for 5 seconds, and reclosed. Normal battery charging procedures require a check of the battery manifold after completion of each recharge.

Since the battery vent line is connected to the waste water dump line, it provides a means of monitoring waste water dump line plugging, which would be indicated by a pressure rise in the battery manifold line when the BATTERY VENT control is positioned to VENT.

Each battery is rated at 40-ampere hours (AH) minimum and will deliver this at a current output of 35 amps for 30 minutes and a subsequent output of 2 amps for the remainder of the rating.