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HUNTSVILLE, ALABAMA

SPENT STAGE EXPERIMENT SUPPORT MODULE PROPOSAL

JUNE 1966

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National Aeronautics and Space Administration

NASA

FOREWORD

This proposal is submitted to assure NASA management that MSFC possesses the technical and managerial capability to design, manufacture, and test the Spent Stage Experiment Support Module (SSESM) on a timely basis and within the framework of an austere program.

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SECTION I. INTRODUCTION

1.1 BACKGROUND AND PHILOSOPHY

Flight performance analysis of AAP near-Earth orbiting missions indicates that the S-IVB stage, firing in combination with the CSM to final orbit, can provide near optimum payload characteristics for a wide range of orbits. Since the spent S-IVB stage and the spacecraft would then be in proximity, the large volume of the LH_2 tank could be available for use as an enclosed workshop. A docking structure and airlock are needed on the spent stage for providing docking to the CM and access for astronauts into the large LH_2 tank.

The intent of this document is to propose a combined docking and airlock structure called (Spent Stage Experiment Support Module) (SSESM) which would be designed, manufactured, and tested in-house at MSFC. The approach covered in this proposal is applicable to experimental missions for evaluation of the orbital workshop beginning with flight AS-209, and to subsequent missions with operational workshops.

The facilities and engineering capabilities to accomplish this task are presently available at MSFC. In addition, this Center has complete knowledge and configuration control of the S-IVB stage and S-IVB Workshop. Accordingly, it is the intent of this proposal to delineate the management and close control of all production elements necessary to: (1) Meet the early launch dates; (2) provide fast reaction to program changes; and (3) restrict costs to present budget limitations.

1.2 SCOPE

This document proposes that MSFC design, build, and test the SSESM, which is a combined docking structure and airlock suitable for mating the CM to the spent S-IVB stage, thus, providing astronauts access to both the pressurized empty LH_2 tank and to the unpressurized exterior of the vehicle. The SSESM will occupy the space normally provided for the LEM and will attach to the descent stage attachment points. Volume allocation will be provided for mounting experiments on the exterior of this module. Where possible, equipment already designed for the Saturn/Apollo or Gemini subsystems will be used for subsystems of this module, such as environmental control, electrical power, instrumentation, and communications. All structural manufacture and equipment adaptation will be accomplished within MSFC Laboratories.

SECTION ONE. INTRODUCTION

1.1 BACKGROUND AND [illegible]

Flight performance analysis of [illegible] calculations indicates that the S-IVB stage, during operational [illegible] orbit, can provide near optimum payload characteristics [illegible] spent S-IVB stage and the spacecraft [illegible] simultaneously, the large volume of the LH_2 tank could be [illegible] A docking structure and airlock [illegible] attached to the spent stage, providing docking to the CM and access [illegible] into the spent LH_2 tank.

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SECTION II. PROGRAM SUMMARY

2.1 GENERAL

The S-IVB Spent Stage Experiment involves the establishment in Earth orbit of the spent S-IVB stage with a Spent Stage Support Module (SSESM) docked to the CSM for manned orbital missions. This system provides large volumes for 14 to 30 days on initial flights. Rendezvous and resupply techniques may be utilized on initial or subsequent flights to extend mission lifetime.

The spent S-IVB stage is utilized as a habitable workshop by including as a major element of the experiment a SSESM. The SSESM is basically a structural unit which includes an airlock, provides the dynamic and static structural interface between the CSM and spent stage, and provides attachment provisions for the desired supporting equipment and corollary experiments. It provides the passageway from the CSM to the S-IVB workshop and to the hatch for EVA. In this context, the SSESM will provide for either an unpressurized or pressurized workshop flight as required by mission assignments. Instrumentation and electrical power are provided integral to the SSESM as required to support the different degrees of sophistication desired in the assigned operational missions and corollary experiments. Capability for pressurizing the LH_2 tank can be provided for pressurized flights or combination flights (unpressurized X days, pressurized Y days) by including pressurization components as an experiment on the SSESM or as an operational subsystem on the SSESM.

The SSESM is designed within the framework of the mission and overall workshop design discussed in the subsequent paragraphs. A summary of the basic SSESM design is also discussed in a subsequent paragraph; a detailed technical description is presented in Section IV. Alternate designs and system flexibility are discussed in Section VIII, but the primary system proposed for the AS-209 mission is the system discussed as the basic SSESM design. MSFC has completed, as evidenced by the schedules and technical backup material, that portion of the final SSESM design necessary at this time to provide system readiness for flight AS-209. Efforts are continuing to maintain the requirements of this flight schedule.

2.2 MISSION DESCRIPTION

The Spent Stage Experiment mission considered for this program is a 20-day mission, 14 to 20 days depending on CSM lifetime, launched in early 1968 on flight vehicle AS-209. Longer duration flights are discussed as alternates. The S-IVB is burned for direct injection into an elliptical orbit, the CSM then transposes and docks with the SSESM, and the CSM is then ignited to circularize in a low inclination, 170 nautical mile Earth orbit. After establishment of 170 nautical mile orbit, the crew completes procedures to passivate the S-IVB stage and to activate the Workshop.

2.3 SPENT STAGE EXPERIMENT CONFIGURATION

The composite operational configuration in Earth orbit consists of the spent S-IVB stage, the IU, the SLA, and the SSESM which is docked to the Apollo Command/Service Module (refer to Figure 2.3-1). The SSESM is attached to the S-IVB forward interstage at the four LEM descent stage attach points and the SSESM airlock is attached to the upper LH_2 dome by a flexible bellows connection.

The S-IVB stage is passivated (made safe for occupancy) by disarming the command destruct receiver and venting the LH_2 tank, LO_2 tank, cold helium supply gases, engine pneumatic supply, engine start bottle, APS helium supply, and stage pneumatic supply gas. The S-IVB/Workshop is activated by deployment of the SLA panels, removal of the LH_2 tank hatch and connection of the SSESM airlock to the LH_2 tank, performing an SSESM subsystems monitor check, subsystem activation, equipment deployment as necessary, and pressurization of the LH_2 tank. The pressurization of the LH_2 tank may be delayed for the time period desired for performing experiments in an unpressurized container. Prior to pressurization the LH_2 tank interior is fitted with lights, blowers, and crew-assist ropes which will be stored on the SSESM. For the basic design, no major interface has been established with the CSM other than docking interface. This assembled operational configuration will allow one or two crew members to function in the SSESM airlock, the S-IVB tank, or outside the SSESM airlock, depending on the detailed scheduling of mission and experiment operations. Varying periods of allowable occupancy will be provided in each area which is discussed later in more detail. For passive thermal control, the assembled workship is rotated at approximately six revolutions per hour oriented at an attitude nominally broadside to the Sun or aligned with the velocity vector during periods of LH_2 tank occupancy. The workshop could assume a random attitude at other times.

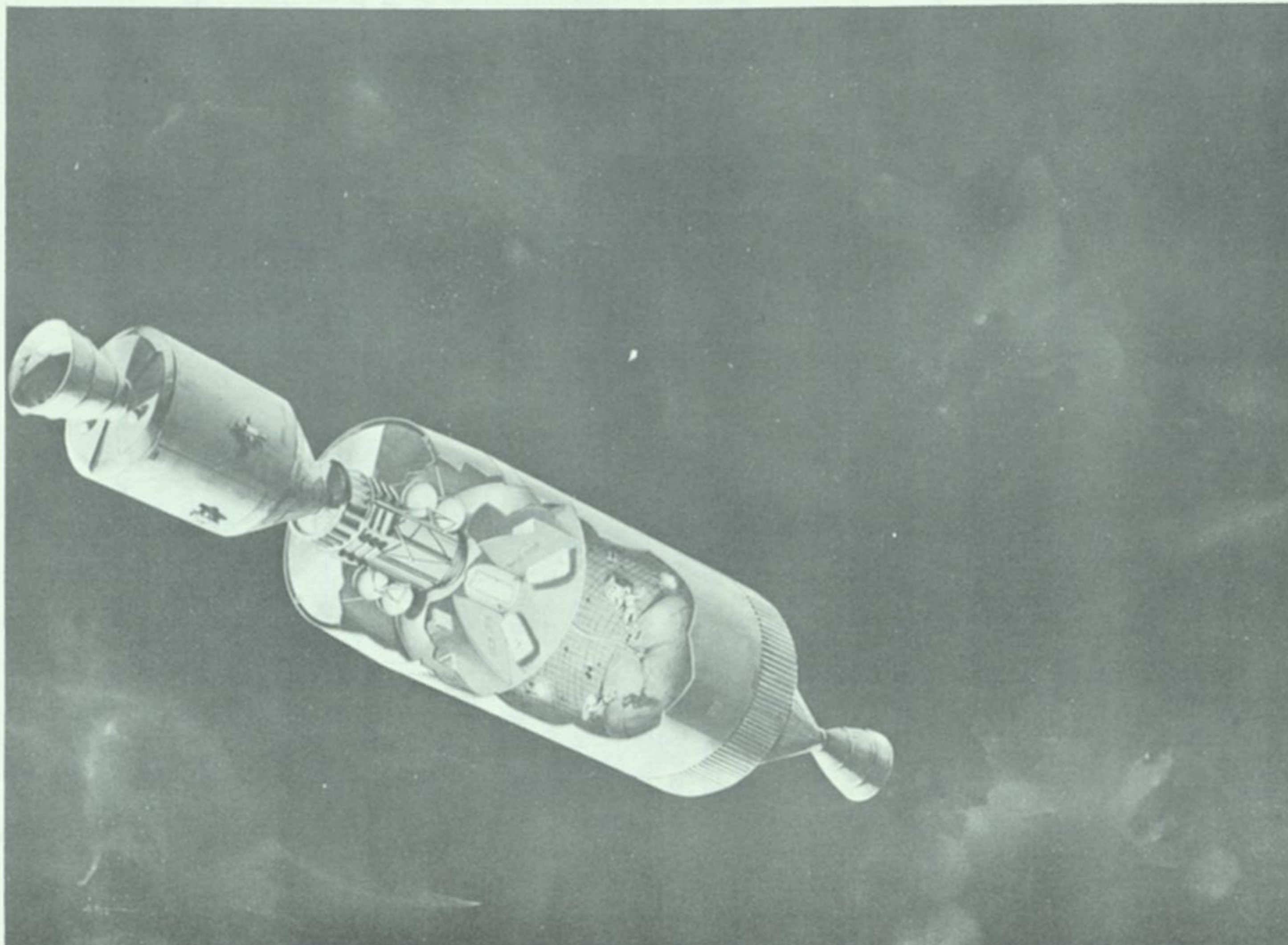

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FIGURE 2.3-1. ORBITAL CONFIGURATION

2.4 SPENT STAGE EXPERIMENT SUPPORT MODULE (SSESM)

The basic SSESM design being proposed is a straightforward design with an inherent 20-day capability. The module design provides electrical power, instrumentation, telemetry, environmental control, and experiment support with a composite weight of approximately 9, 200 pounds. The available weight for experiments is approximately 950 pounds.

The configuration (refer to Figure 2.4-1) contains a 200 ft^3 cylindrical airlock, 65 inches in diameter, and 15 feet long which will accommodate two crew members, required instrumentation, and storage for selected equipment. The system provides a 32-inch diameter upper ingress/egress hatch to the CSM, a 48-inch diameter lower ingress/egress hatch to the LH_2 tank, and a side located 36 by 55-inch rectangular sliding hatch for extra vehicular ingress/ egress. These hatch sizes, designs, and locations readily allow the transfer of men, equipment and experiments.

The SSESM, basically of aluminum construction, has structural truss members attaching the airlock to the S-IVB forward skirt at the four LEM attach points. The airlock is a skin/frame of welded construction with external ring frames and longerons for equipment attachment. A double skin meteoroid bumper provides a large protected equipment compartment around the lower half of the airlock exterior for selected experiments and electronics. These items are mounted to the airlock stiffeners and the truss support structure. The upper end of the airlock accommodates a LEM docking adapter for mating with the CSM and the lower end has an expandable non-metallic bellows which is provided for in orbit attachment to the upper LH_2 tank dome.

The electrical power system, utilizing 21 silver zinc batteries, provides approximately 300 KW hours of electrical power to the SSESM for housekeeping and experiments. The power profile can be varied substantially to accommodate periods of low and high load requirements. The batteries are mounted to hard points around the upper portion of the airlock. A power control panel is provided on the airlock and a portable display panel is provided for use in the LH_2 tank.

The instrumentation provides the capability of monitoring all required subsystem components and limited subsystem controls. Sufficient instrumentation is provided to monitor several experiments. A telemetry system using available hardware is provided for direct transmittal of data to ground via the IU antenna.

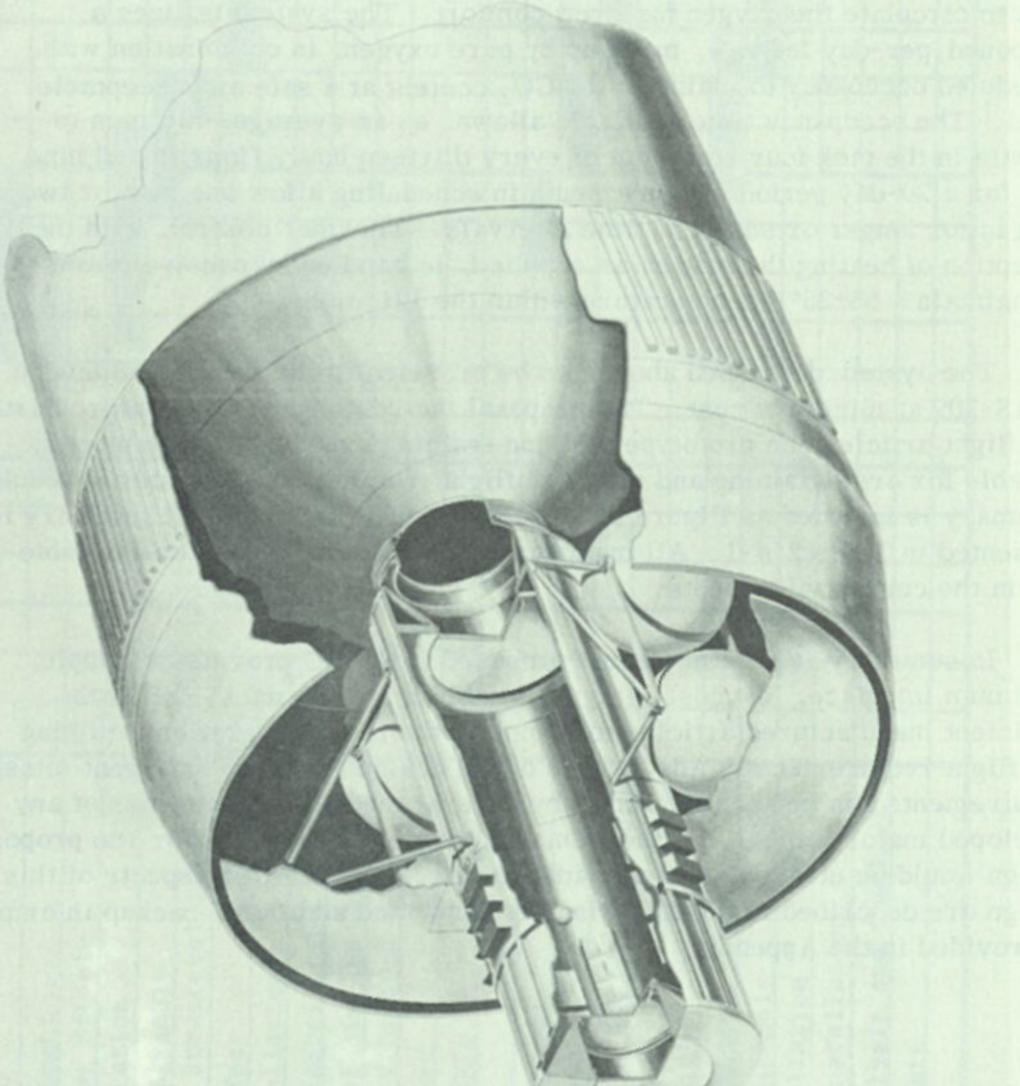

FIGURE 2.4-1 SPENT STAGE EXPERIMENT SUPPORT MODULE

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The environmental control system (ECS) has been designed with emphasis on simplicity and reliability. The system utilizes components which are largely available from existing manned space flight programs to provide pressurization capability for the LH_2 tank and airlock to 5 p.s.i.a. with 100 percent oxygen, and to provide PLSS recharge capability for extra vehicular activity. Four 20-ft^3 spheres are used for the ECS oxygen and one 3-ft^3 sphere for the PLSS oxygen. Four blowers are distributed within the LH_2 tank to circulate the oxygen for crew comfort. The system utilizes a 30-pound-per-day leakage, made up by pure oxygen, in combination with scheduled occupancy to maintain the CO_2 content at a safe and acceptable level. The occupancy time available allows, as an average, two men to remain in the tank four hours out of every thirteen hours (four in and nine out) for a 20-day period. Adjustments in scheduling allow one man or two men in for longer or shorter time intervals. Thermal control, with the exception of heating the oxygen as supplied, is handled by passive means to maintain a 65±25°F temperature within the LH_2 tank.

The system described above can be provided for an early 1968 launch on AS-209 at minimum cost. The proposal includes three manufactured articles: one flight article, one prototype test and training article, and one mockup suitable for crew training and zero "g" flight training. The master schedule summary is included as Figure 2.4-2. A resource requirement summary is presented in Table 2.4-1. All manpower requirements would be available within the current allotments.

In summary, the basic design proposed by MSFC provides a simple, minimum interface, low cost design available for flight on AS-209 with sufficient manufactured articles to complete the test, integration, training and flight requirements. Adaptations of the design for more stringent mission requirements can be accommodated by several methods without loss of any developed major item, i.e., the items requiring development for the proposed design would be utilized on design adaptations. The detailed aspects of this design are described in the following sections, and additional backup information is provided in the Appendix.

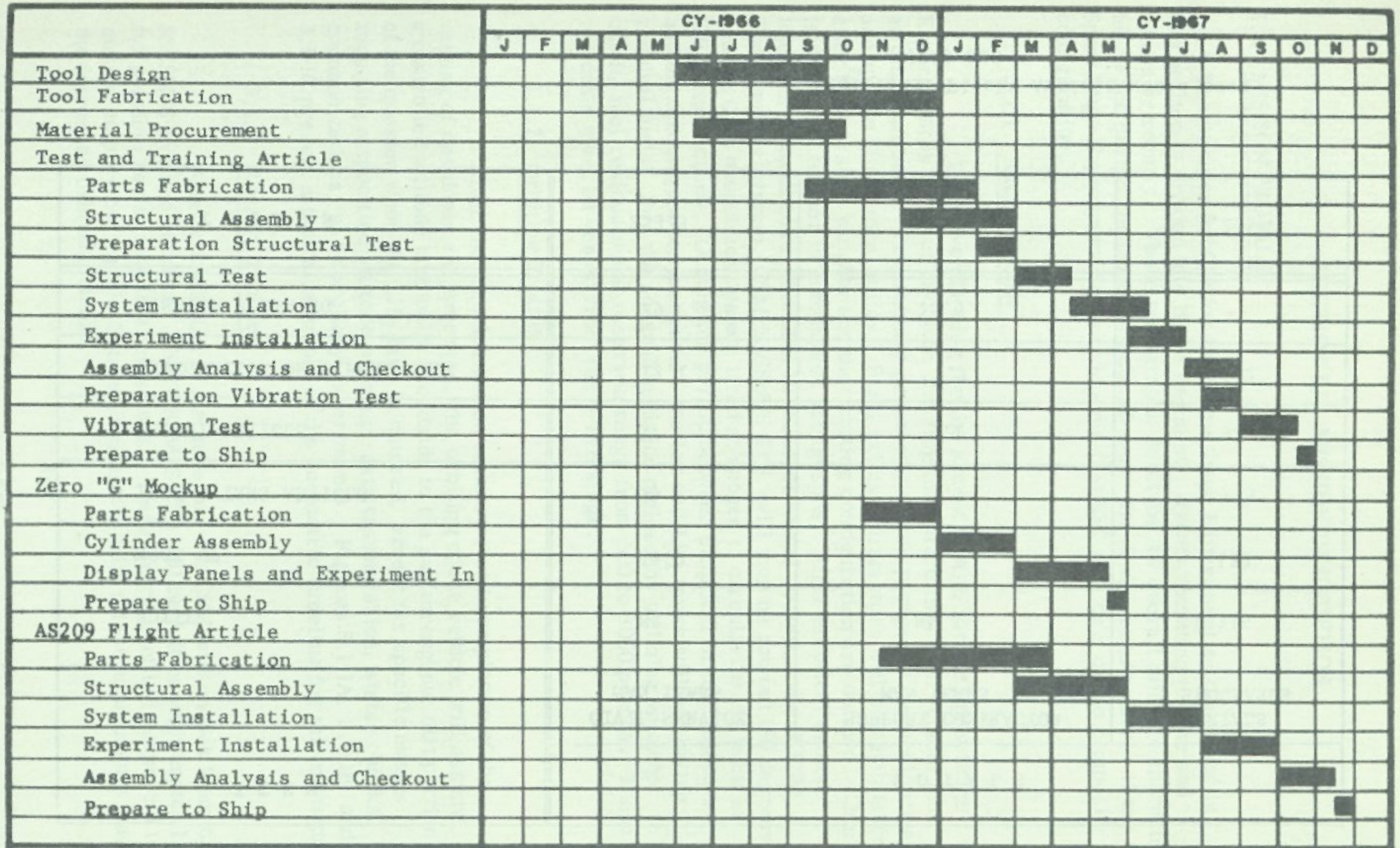


FIGURE 2.4-2 SSESM SCHEDULE SUMMARY

TABLE 2.4-1

RESOURCES REQUIREMENTS SUMMARY

ITEM	TOTALS		
	CIVIL SERVICE MAN YEARS	SUPPORT CONTRACTOR MAN YEARS	MATERIALS $ THOUSANDS
RDT&E	84	84.4	1,316
MOCK-UP	16.2	1.7	123
TEST & TRNG ARTICLE	55	9.9	795
FLIGHT ARTICLE	57	9.1	919
INTEGRATION OF EXPERIMENTS	12.1	7.7	122
MISSION PLANNING	29	16.6	--
TOTALS	253.3	129.4	3,275

Additional Flight Article would cost
$743,000

SECTION III. MISSION DESCRIPTION

3.1 MISSION PROFILE

The mission profile for the Spent Stage Experiment as discussed in this section is divided into three segments, ascent, orbiting profile and return or reentry. These paragraphs describe the overall mission characteristics and design conditions which influence the design requirements of the Spent Stage Experiment Support Module (SSESM) as a part of the composite configuration.

3.1.1 Ascent Profile

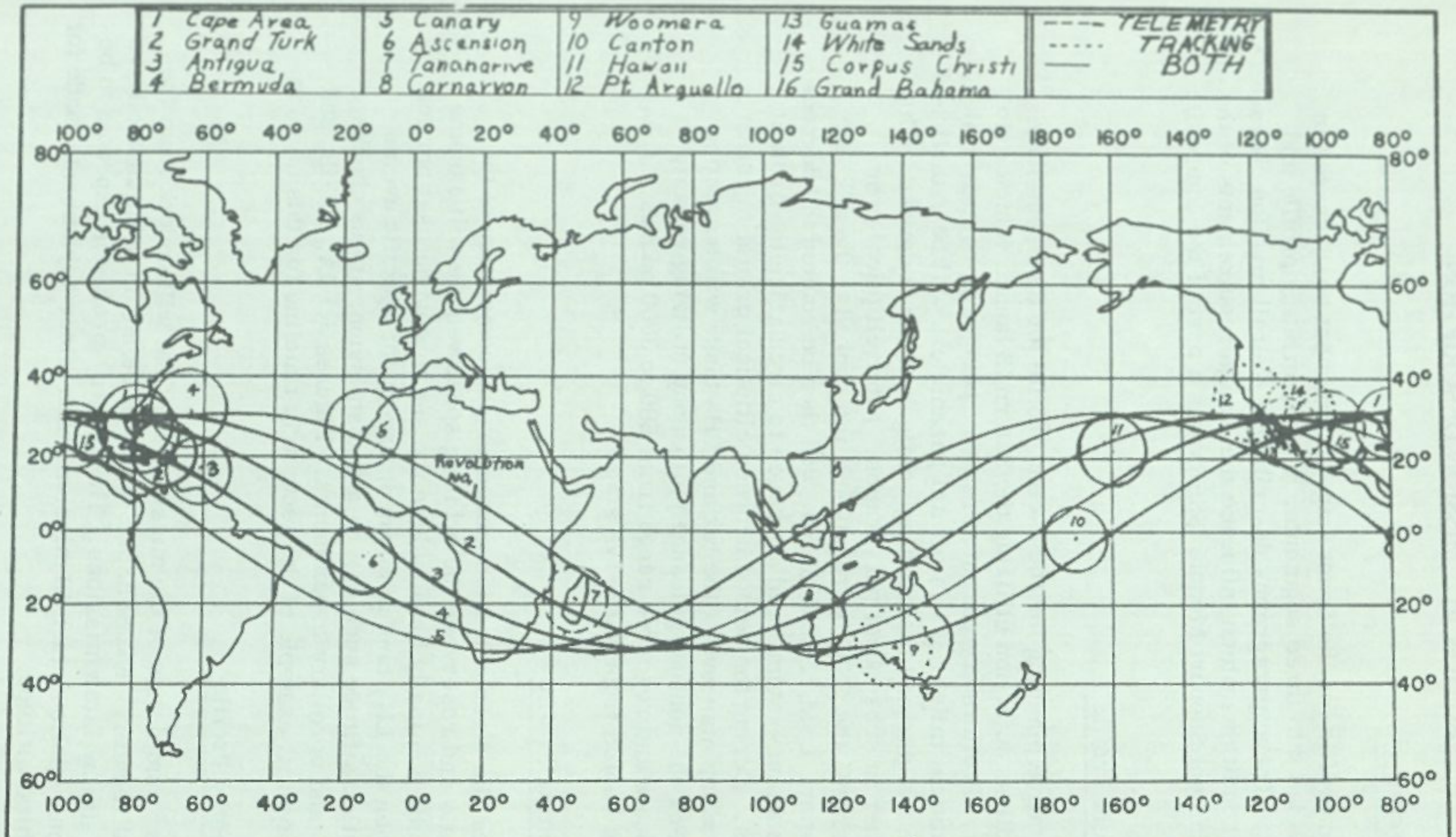
Tentative planning for the ascent profile for the Spent Stage Experiment is a direct injection utilizing the Saturn IB launch vehicle into an elliptical orbit. Preliminary parameters are: perigee 81 nautical miles, and apogee 170 nautical miles. Firm orbital parameters will be established after final system weights and the orbiting configuration are defined. This is discussed further in subsequent paragraphs. After elliptical orbital injection is confirmed, the CSM is transposed and docked to the SSESM. The complete system, CSM/SSESM/S-IVB, will then be rotated 180 degrees and the CSM propulsion system fired at apogee to circularize the orbit at 170 nautical miles. During the period between elliptical orbital injection and the circularization maneuver, the lox and LH_2 tanks will be venting residual fuels. For the nominal guidance philosophy utilizing velocity cutoff, fuel residuals and reserves range from 500 to 3000 pounds. These residuals must be vented from the S-IVB stage.

3.1.2 Orbiting Profile

After the circularization maneuver and completion of the venting of residuals and reserves, the orbiting space vehicle will assume a controlled attitude nominally broadside to the sun throughout that portion of the mission when the LH_2 tank is manned, except for specific events requiring a specified attitude such as: data transmission, status checks, communications, and corollary experiments. Figures 3.1-1A, 3.1-1B, and 3.1-1C present a typical example of the geometric timeline for this mission.

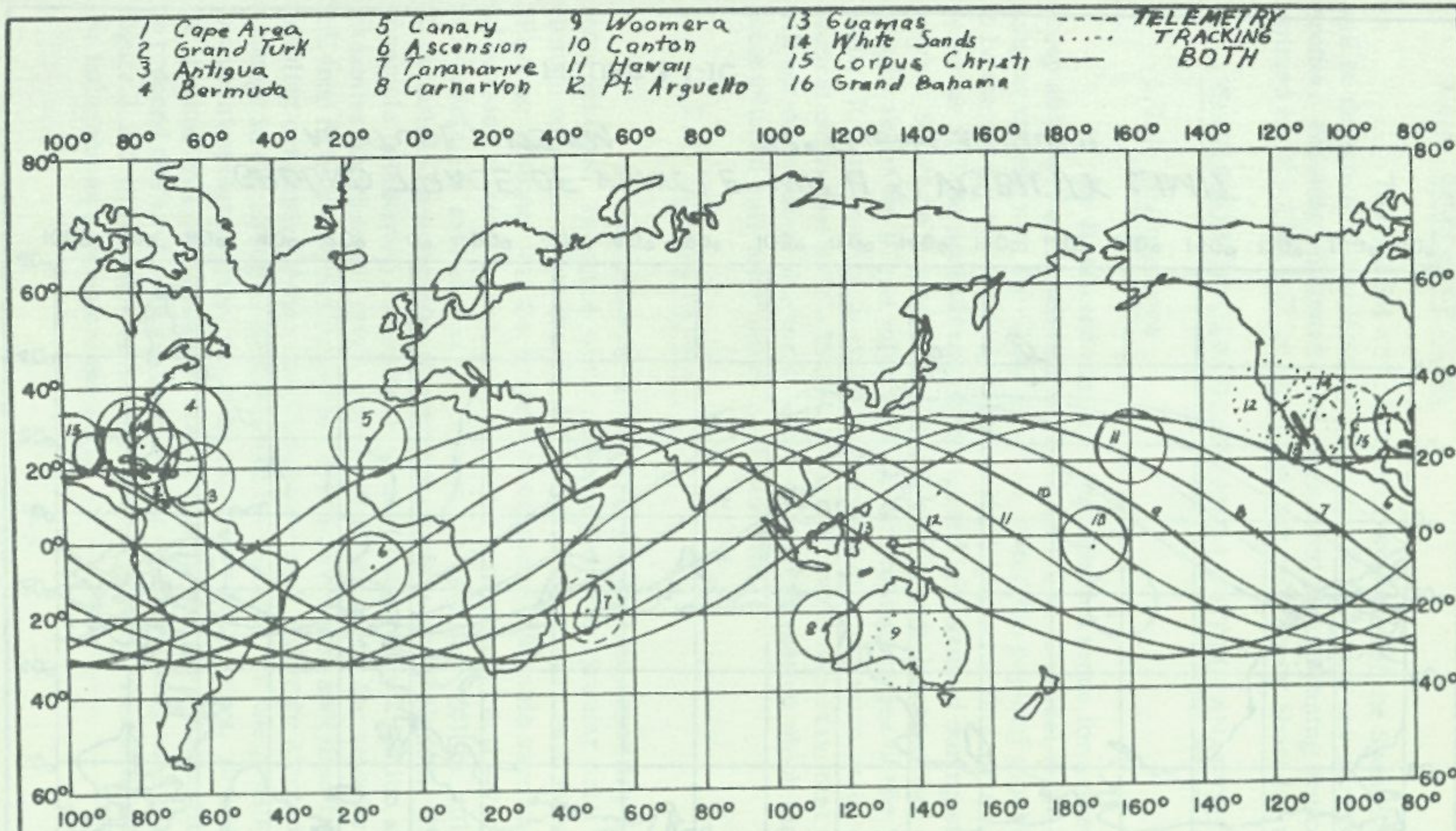
3.1.3 Reentry Profile

After completion of the mission, the CSM will undock from the SSESM/S-IVB and reenter. The SSESM/S-IVB will be left in orbit unstabilized for possible revisits by later missions. The reentry profile is expected to be similar to previous Apollo-Saturn IB missions with minor adjustments made for the increased orbital altitude.



GROUND TRACE OF VEHICLE WITH 5° VISIBILITY LIMIT
ALTITUDE: 175 KM. 72° FLIGHT AZIMUTH

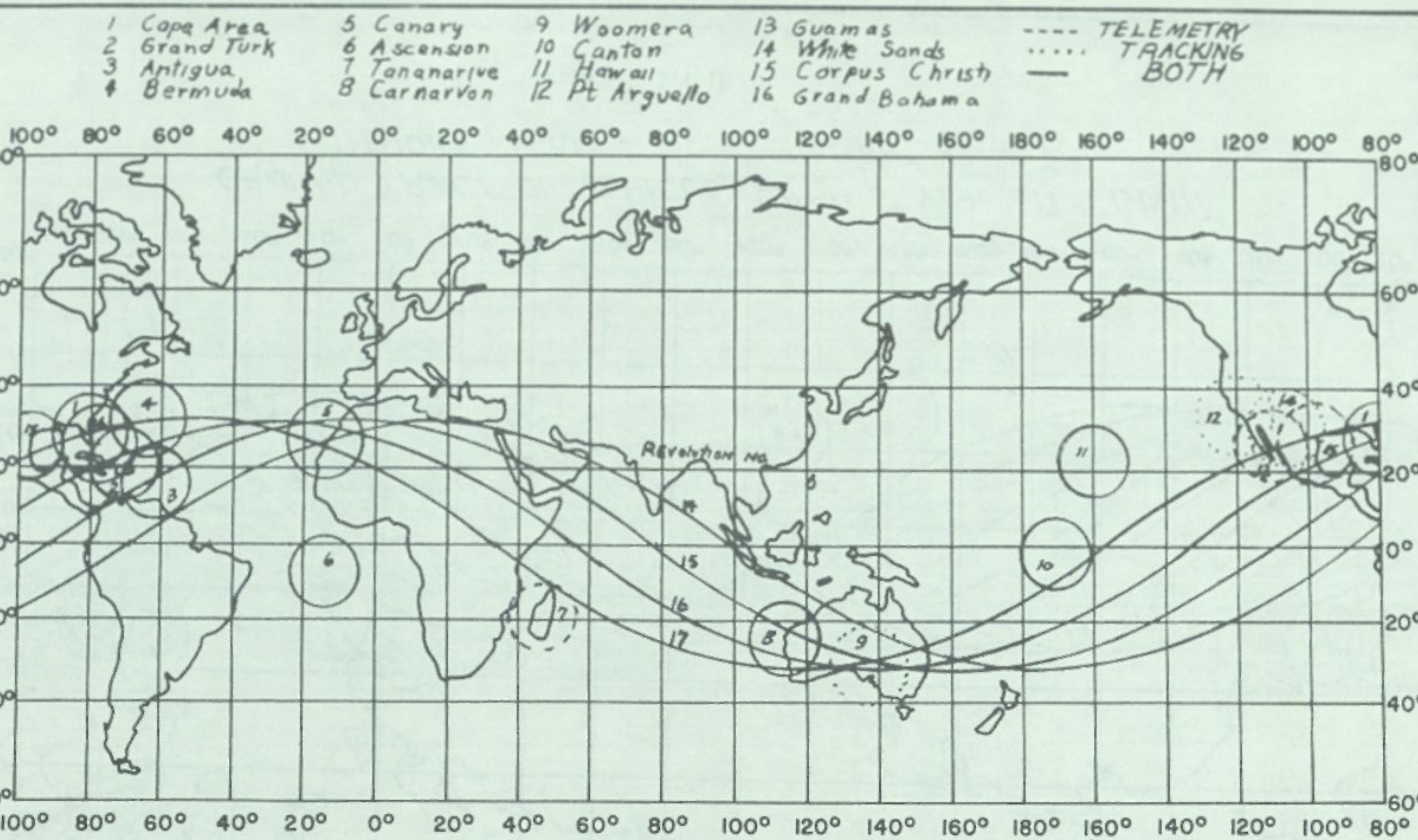
FIGURE 3.1-1A



GROUND TRACE OF VEHICLE WITH 5° VISIBILITY LIMIT

ALTITUDE: 175 KM. 72° FLIGHT AZIMUTH

FIGURE 3.1-1B



GROUND TRACE OF VEHICLE WITH 5° VISIBILITY LIMIT

ALTITUDE: 175 KM   72° FLIGHT AZIMUTH

FIGURE 3.1-1C

3.1.4 Emergency Abort

The SSESM and associated systems of the Spent Stage Experiment will be designed to facilitate emergency abort during any portion of the mission profile. Existing equipment and associated abort sensing devices presently utilized for the CSM will be employed for emergency abort for the mission.

3.2 PERFORMANCE CAPABILITY AND LIFETIME ANALYSIS

3.2.1 Performance

As discussed earlier, the selected mode for achieving the acceptable payload capability is via injection in elliptical orbit and circularizing at apogee with the CSM/SPS. As stated the payload capability is defined to be the gross injected weight above the IU less the fuel residuals and reserves and the pressurants to be vented and dumped during stage passivation. Shown on Figure 3.2-1 is a typical example of payload capability (weight above the IU) as influenced by increasing apogee of the "parking" orbit. It is necessary to maintain perigee at a minimum value of 80 nautical miles due to tracking constraints and to ascertain that the orbit will not decay unsatisfactorily prior to circularization which may be accomplished only after several revolutions.

3.2.2 Lifetime in Orbit

A major factor influencing the payload capability is the lifetime requirements. Selection of the 170 nautical mile circular orbit as the altitude required to yield a 20-day mission lifetime is established by the following criteria: (a) the system is assumed to tumble randomly; (b) mission occurs during a period of maximum solar activity; (c) the SLA panels are extended in the nominal 45° position; (d) the influence of 20 variations on principle lifetime par ameters are included. The effect of vehicle orientation during periods of ground communication and data transmission when the space vehicle is aligned with the velocity vector will tend to increase the sensible lifetime over that specified for the tumbling condition. Since the SLA panels may either be ejected or folded, a change in the effective lifetime will be realized. If the panels are folded to a position that reduces the configuration profile, thus, reducing drag, the lifetime will be increased. If the panels are ejected, the lifetime will be reduced due to the decrease in weight of the orbiting configuration. Figure 3.2-2 illustrates the variation of orbital lifetime with altitude and the influence on achievable payload.

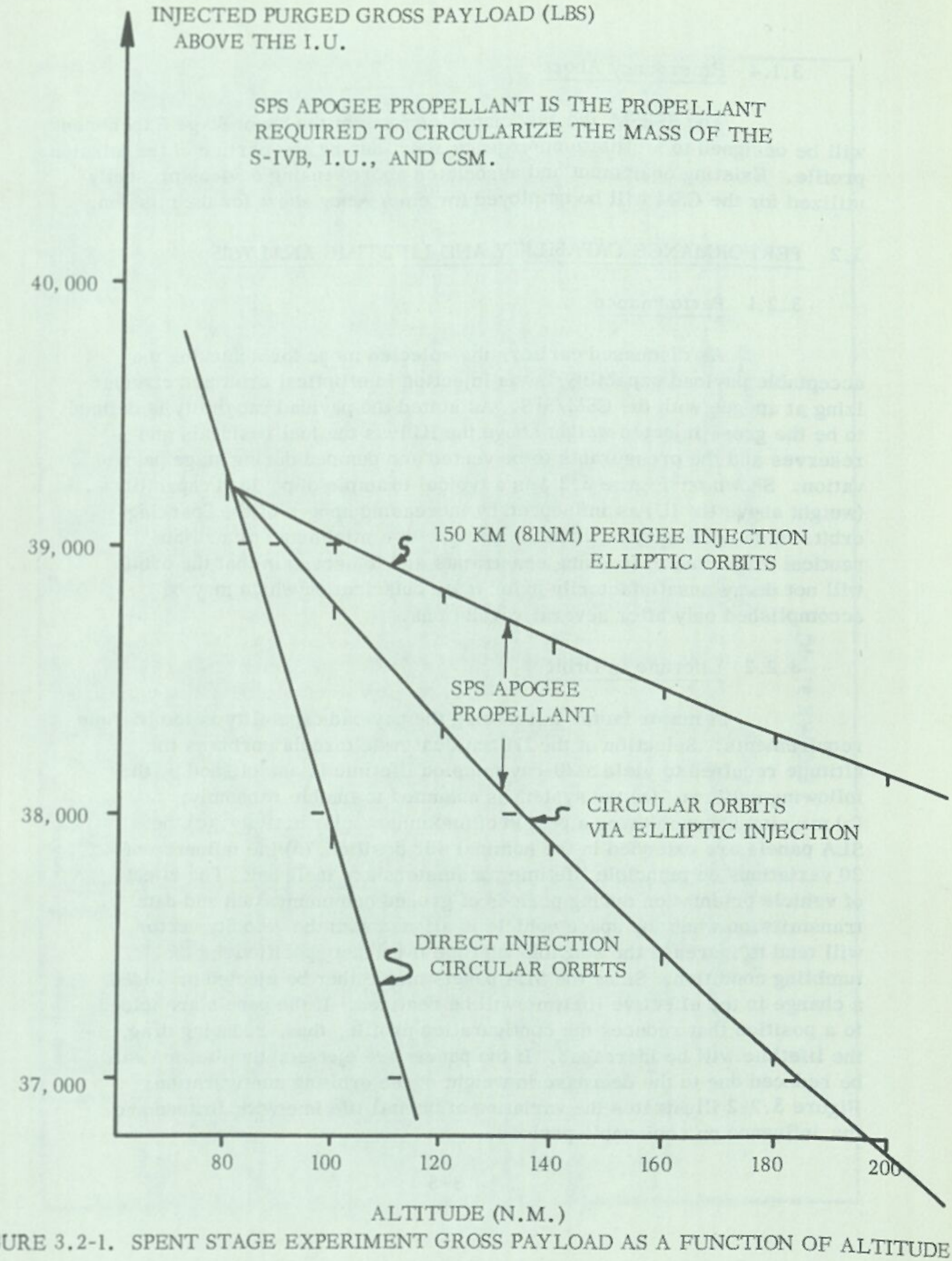


FIGURE 3.2-1. SPENT STAGE EXPERIMENT GROSS PAYLOAD AS A FUNCTION OF ALTITUDE

3.3 DYNAMIC ANALYSIS

During the ascent phase of the mission profile, the SSESM will experience dynamic responses, both longitudinal and lateral that will induce acceleration loads normally encountered by the LEM. With the maximum wind speeds, shears, and gusts occurring during the S-IB stage thrusting phase of flight, maximum lateral accelerations are expected to occur during the flight times where angle of attack and dynamic pressure are maximum. Maximum longitudinal acceleration normally occur just prior to first stage cutoff. The SSESM will be designed to withstand these induced loading conditions as described above.

Preliminary indications are that dynamic response and acceleration loads will be essentially those prescribed for previous Apollo - Saturn launch vehicles (ex: AS-207) having similar ascent profiles. During the transposition and docking maneuver, the S-IVB stage auxiliary propulsion system will be required to maintain attitude stabilization of the spent S-IVB stage and SSESM while the CSM is rotating and docking with the forward end of the SSESM. This requirement dictates that this maneuver must be completed prior to depletion of the stage electrical power and instrument unit. To minimize disturbances during the transposition and docking maneuver, the S-IVB lox and LH_2 tank venting will have to be interrupted. Nominal transposition and docking times for CSM/LEM docking are essentially 30 minutes. It is expected that similar docking times and communication requirements will also be imposed on the operation.

After the transposition and docking maneuver is completed, the CSM/RCS is responsible for maintaining attitude control for the complete system as required. Although the S-IVB stage is to be passivated soon after the final circularization maneuver, it appears advantageous to utilize the S-IVB APS to assist in attitude control prior to the passivation process, thus, potentially increasing the useful lifetime of the CSM attitude control system. The final circularization maneuver which utilizes the CSM propulsion system will impose dynamic loading conditions on the SSESM since the entire system will be accelerated and controlled by the CSM. After the circularization is completed, attitude control will also induce bending and shear loads through the SSESM.

Analysis is now being accomplished to assess the structural stiffness requirements of the SSESM for the loading conditions that will be experienced during the Spent Stage Experiment. Preliminary analysis indicates that the present SSESM structural design affords sufficient stiffness to allow controlling the system.

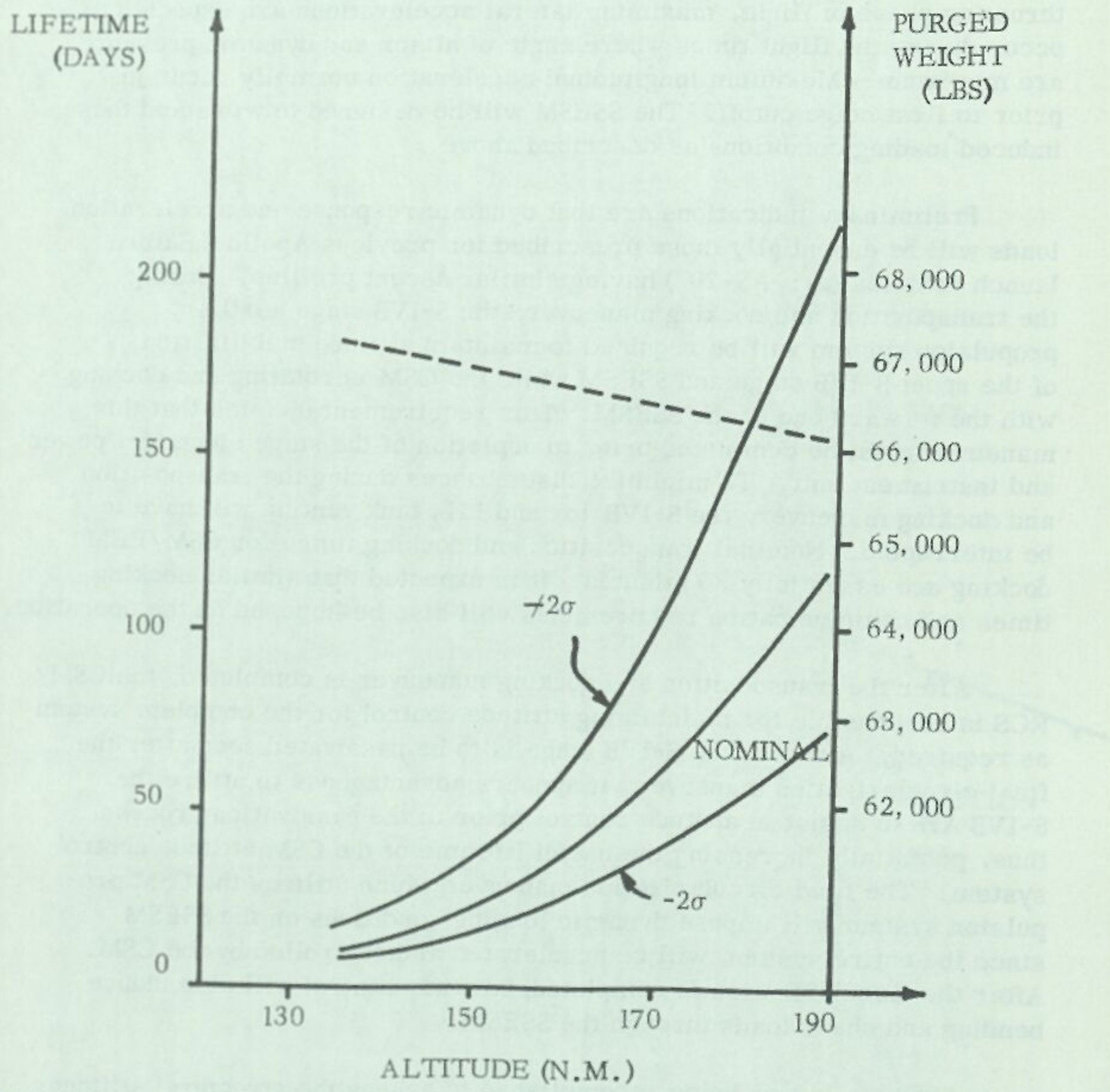


FIGURE 3.2-2. LIFETIME AND PURGED WEIGHT VS. ALTITUDE FOR THE DOCKED CONFIGURATION, TUMBLING, FOR DEC. 1, 1968

3.4 MISSION SEQUENCE AND ANALYSIS

The Spent Stage Experiment has as its primary mission the passivation and activation of the spent S-IVB stage making it suitable for habitation. The secondary mission of the Spent Stage Experiment is the performing of corollary experiments either within the LH_2 tank or EVA; the experiments are essentially self contained and independent of the S-IVB stage and SSESM. A preliminary functional analysis of the early mission functions is included in Figure 3.4-1. This diagram includes events and functions from pre-launch to pressurization of the LH_2 tank as a workshop.

A preliminary time line analysis and considerable detail on specific mission events and their sequence are included in the Appendix. The event sequence is for the initial and terminal portions of the mission.

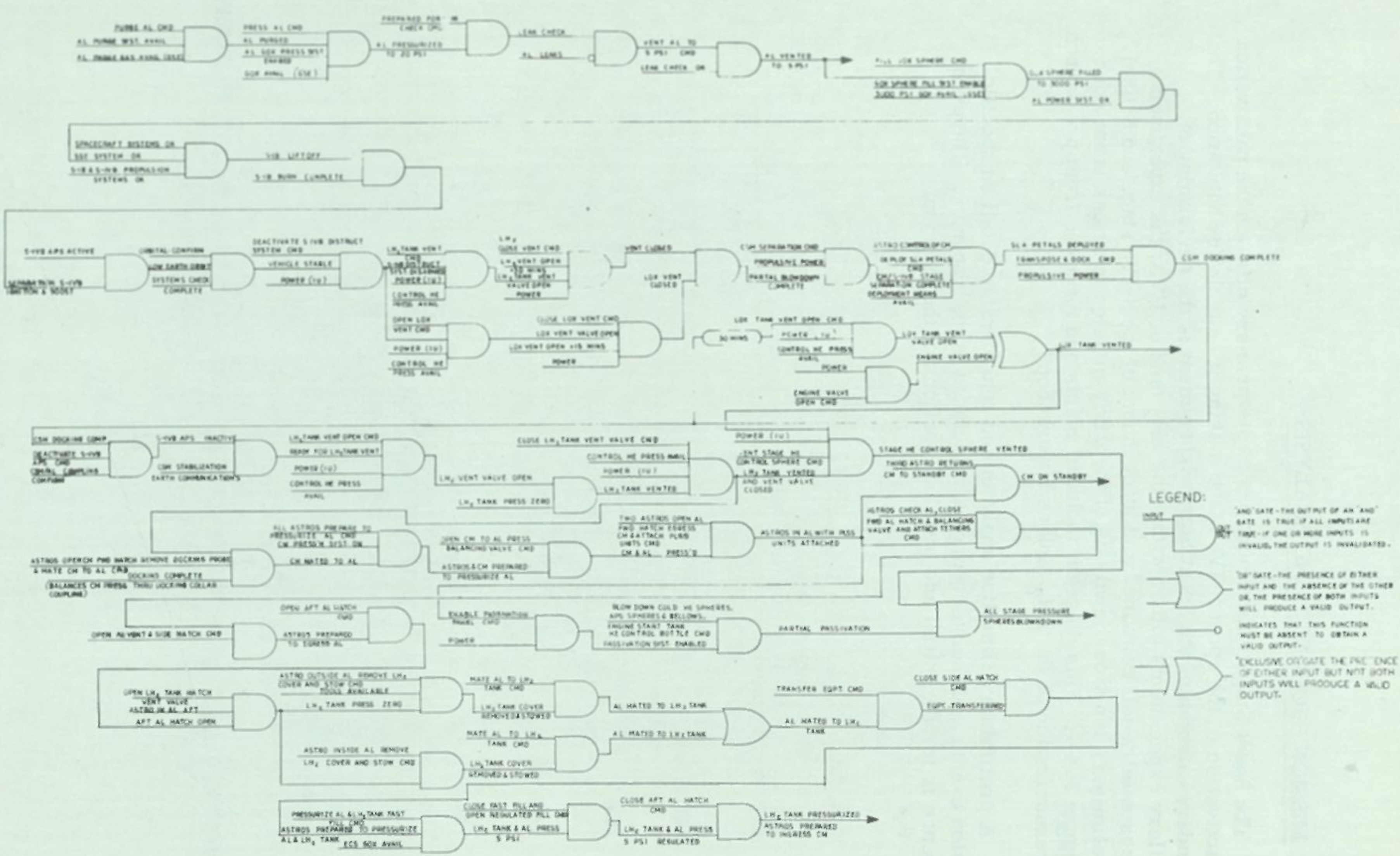


FIGURE 3.4-1. FUNCTIONAL LOGIC DIAGRAM

SECTION IV. TECHNICAL DESCRIPTION

4.1 DESIGN DESCRIPTION

4.1.1 Introduction

This section provides an overall systems description of the Spent Stage Experiment Support Module (SSESM) system proposed to be designed, manufactured, and tested by MSFC. Pertinent design guidelines are summarized, system characteristics and the integrated configuration are described, weight summaries are presented, and major systems interfaces are defined. Brief descriptions are also presented on the non-flight articles, the composite documentation, and the test philosophy. Subsystem descriptions are presented in substantial detail in the subsequent sections and the Appendix (under separate cover) contains additional data on many items, including design ground rules, SSESM handling sequence, crew familiarization requirements, maintenance concept, mission sequencing, and subsystem data. Alternate designs and system flexibility are discussed in Section VIII of this document.

4.1.2 SSESM Design Ground Rules

The detailed design ground rules used in the definition of the MSFC proposal for a minimum cost 20-day mission duration Spent Stage Experiment Support Module are contained in the Appendix (under separate cover).

The design ground rules are based upon a mission objective of providing, on flight AS-209, a pressurized S-IVB stage LH_2 tank in which astronauts can operate in a shirt-sleeve environment for 20 days.

The following design objectives and design approach considerations were made which are reflected by the design ground rules contained in the Appendix:

1. Maximum design simplicity.

2. Minimum program cost.

3. Passivation and activation of the S-IVB stage LH_2 tank into habitable shirt-sleeve environment volume is the primary experiment of the mission.

4. No major interfaces with the CSM

5. Minimum modifications to Saturn/Apollo hardware.

6. Retain AS-209 capability for Apollo backup.

7. Maximum utilization of existing, and available Saturn/ Apollo subsystem components.

8. SSESM design to provide inherent flexibility and growth potential for extended mission durations in a follow-on program, without major modification to the basic systems design.

4.1.3 Description

General - The SSESM is defined as an independent airlock unit which interconnects the CSM and the S-IVB LH_2 tank and is mounted at the attach points in the LEM adapter. The SSESM will include an airlock, docking structure, environmental control system, electrical power system, instrumentation and communication system, and support equipment as defined below. It will also include the support structure for these systems, expendables, and experiment stowage. The airlock will have the capability for independent and integrated operation with the CSM and S-IVB Workshop. A schematic of the SSESM system is given in Figure 4.1-1 outlining all major elements and systems of this module.

Airlock - The airlock is used as a connecting link between the CM, the LH_2 tank, and the extra vehicular area. It provides a meteoroid protected, environment controlled area for the crew, controls, and selected equipments. The system is 65 inches in diameter and approximately 200 inches in length containing a 32-inch-diameter hatch at top, a 48-inch-diameter hatch at bottom, and a 36-by-55-inch rectangular side hatch. These hatch sizes and locations are designed to permit the crew with equipment to readily move between zones of the Workshop. The airlock, 200 cubic feet, is sized to be capable of accommodating two suited astronauts allowing for suit donning and doffing. Provision is made for a pressure-tight connection to the LH_2 tank forward dome mounting surface after which removal of the dome cover provides a pressure environment for passage from the airlock to the tank.

Structure - The SSESM structure consists of the airlock cylinder, the lower flexible boot, the support structure, the forward meteoroid protection, the pressure spheres support structure, and the electrical batteries support

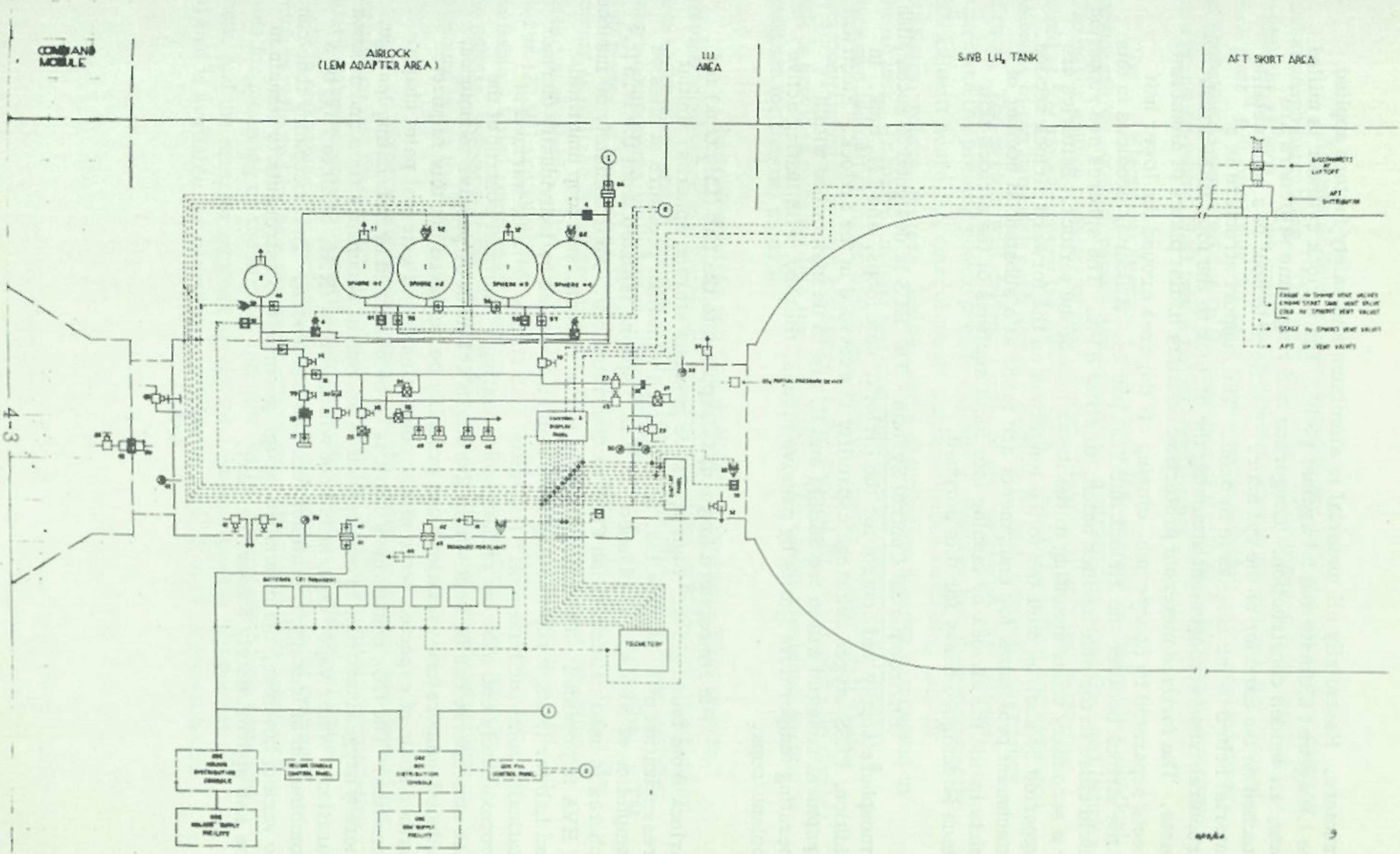


FIGURE 4.1-1. SYSTEM SCHEMATIC DIAGRAM

structure. Essentially all material is aluminum and safety factors applied are 1.4 against ultimate and 1.1 against yield. The airlock cylinder is milled plate, all welded construction. Longerons and ringframe stiffeners are attached on the exterior of the cylinder. The flexible boot is a non-metallic material bolted to the LH_2 dome in orbit. The support structure is a structural truss arrangement attaching the airlock to the four LEM attach points. The forward meteroid protection consists of flat plates of aluminum sheets separated by honeycomb, closing off the area around the lower half of the airlock between the airlock and SLA panels. A door is provided in this shield close to the side airlock hatch, to allow EVA. The spheres are supported by a secondary truss mounting to the lower half of the airlock. Batteries are supported by shelves mounted to the longerons on the forward airlock section. Attachment provisions for equipments are made at a substantial number of points inside the airlock by tapping into local sections of the airlock skin which is designed heavy for this purpose.

Environmental Control System - The ECS is designed to accomplish atmosphere supply and control for the airlock, suit loops, and LH_2 tank. In addition, PLSS oxygen recharge capability is provided in the airlock. Electrical equipment mounted within the SSESM and IU are to be maintained within operating temperature limits by passive means, thereby, eliminating active coolant loops.

The atmosphere (oxygen) is supplied to the Lab (LH_2 tank) and airlock from four 19.5 ft.^{3} high pressure gaseous storage spheres. Initial pressurization of 3,000 p.s.i.a. is permitted with final pressure at mission completion of 50 p.s.i.a. planned. A 3 ft.^{3} sphere isolated from the 19.5 ft.^{3} spheres is used to accomplish PLSS oxygen recharge. Approximately 60 manhours of EVA is provided. The oxygen supply to the airlock, suit loop umbilicals and Lab for leakage replinishing is heated as required by individually thermostatically controlled heaters. The initial Lab charge is to be warmed in approximately one orbit by radiant solar heating. Thermal control of the Lab atmosphere can be kept within the 65±25°F limits by passive techniques; reduced temperature variations can probably be achieved but this requires further study of expected retro motor contamination of external paint characteristics. The CO_2 contaminent limit is to be observed by scheduling leakage, thereby oxygen make-up, in accordance to astronaut occupancy time and mission duration. Water vapor content is kept within limits (R.H. of 30 to 70%) by a combination of leakage/replinishments and adsorbers. The occupancy time can be varied; however, the described design permits two astronauts to remain in the Lab approximately 8 hours per day.

Adjustable fans are provided in the Lab to afford crew comfort and atmosphere mixing. The ECS controls and displays will be located within the airlock. Two umbilicals are provided in the airlock for "closed face plate" suit operation during ingress/egress cycles. This suited mode of operation is required due to the inability of the PLSS sublimator to operate in a pressure (above H_2O triple point) environment.

Electrical System - The electrical system includes the electrical power source, control panels and circuitry, distribution networks, and lighting. The power source consists of twenty-one, 28 volt, silver zinc batteries rated at 500 ampere hours (14 Kw-hr) each. Twenty batteries are arranged in banks of ten batteries each and one battery is on a separate emergency circuit. The system will deliver a total of approximately 300 kilowatt hours and the present power profile defines a requirement of approximately 200 kilowatt hours. A control panel which provides switches, circuit breakers, relays, meters, lights, and a distribution system is mounted in the airlock. Displays include warning lights, ammeters, a voltmeter, a pressure indicating meter and a temperature meter. Power distribution is accomplished by circuit breakers on the control panel. A portable display unit is provided to carry into the LH_2 tank. The entire electrical system is designed for manual control.

Instrumentation and Communication System - This system provides equipment to acquire and present and relay system and experiment data to the crew and to ground. The system is comprised of flight qualified Saturn components including an FM/FM Telemeter, and RF Assembly, a TM Multiplexer, a Measuring Rack, Telemeter Calibrator, and displays. The antenna is utilized for telemetry to ground. Fifty-five signal conditioning slots are available. One hundred seventy telemetry channels at 12 SPS and four at 120 SPS are available. Fifteen FM/FM continuous channels are available. Voice communication is provided by a Gemini voice system for the SSESM, SSESM to CSM, and SSESM to LH_2 tank. Voice transmission to ground is via the CSM.

Experiment Provisions - Specific experiments are not defined for integration into this design; however, to provide flexibility for accommodating varied experiments many provisions have been made. Substantial space exists in and around the airlock system for mounting experiment packages which is reflected in the subsequent drawings. Structural provisions are made for mounting on the airlock interior walls, the exterior longerons and rings, and the exterior support structure. Approximately 950 pounds of payload is available for carrying experiments and 75 to 100 kilowatt hours of electrical power are currently available. Monitoring instrumentation and data telemetry capability are also provided for a significant amount of experimentation.

Crew Equipment and GFAE - Tools and equipment will be furnished as determined by specific task analysis. Typical items are handholds, racks, reactionless wrenches, lights, and tethers. During launch operations this equipment will be stowed in equipment storage assembly boxes. Three portable life support systems will be furnished, one will be stowed in the CM and two in the SSESM airlock. The PLSS is a portable back pack life support system used in conjunction with a space suit. Basically, it is a closed-loop gas/liquid environmental control system designed to maintain temperature and breathing oxygen to tolerable limits. Three extra pressure suits will be furnished to insure astronaut safety and comfort. GFAE of the following types will be used and stored on the SSESM:

1. Extravehicular activity hardware.

2. Camera equipment.

3. Others as required by mission.

4.1.4 Configuration

Configuration drawings are included as Figures 4.1-2 through 4.1-6 presenting detailed information on the SSESM, the SSESM launch arrangement, operational arrangement, overall layout and equipment placement, structure, and docking provisions. A digest of the five drawings, all titled Spent Stage Experiment Support Module (20-day mission) Inboard Profile and numbered sheet 1 through 5 of SK10-7284, is given below.

Figure 4.1-2 - This drawing defines the launch configuration of the SSESM and CSM with the bellows detached from the bulkhead, typical experiments attached, major external equipments, and other details.

Figure 4.1-3 - This drawing defines the SSESM in its orbital operational position and defines the arrangement of external equipment including batteries, gox spheres, etc.

Figures 4.1-4A and 4.1-4B - These drawings define the interior of SSESM airlock, the location of interior and exterior equipments such as control panels and fans, and the position of the SSESM instrumentation on the IU cold plates.

Figures 4.1-5A and 4.1-5B - These drawings define the internal configuration and characteristics of the S-IVB LH_2 tank.

Figures 4.1-6A and 4.1-6B - These drawings define the SSESM docking adapter provisions and the mated configuration of the CM/SSESM docking tunnel.

4.1.5 Weight Summaries

The overall payload, items above the IU, weight summary is presented in Table 4.1-1 reflecting a weight available for the SSESM and associated equipment and experiments of 10,132 pounds. Table 4.1-2 provides a weight summary of the SSESM reflecting a total weight of 9,190 pounds. Table 4.1-3 presents a detailed weight breakdown for the SSESM including all subsystems and supporting equipment.

4.1.6 Non-Flight Articles

In addition to the flight article (5) proposed herein, MSFC will furnish two additional hardware articles. A prototype (test and training article) and a mockup (zero "g" test article) will be provided. The characteristics of these articles are defined below.

Prototype (test and training article) - This system will be a prototype unit of the flight article which has been described in detail in the above sections. The system will be utilized for structural system tests, high altitude mission simulation tests and detailed crew familiarization. Selected dummy components will be utilized during the structural tests and the unit will later be completely equipped with prototype components.

Mockup (zero "g" test article) - This system will be a mockup of the SSESM sophisticated to the degree required to: be suitable for aircraft flight; simulate internal configuration of flight article; interconnect with the CSM; provide hatches, docking assembly and other mechanisms representative of SSESM torques, forces and configurations; have operating connections for permit crew training in pressure suits; and mock-up portions of the instrument panels not required to be operational.

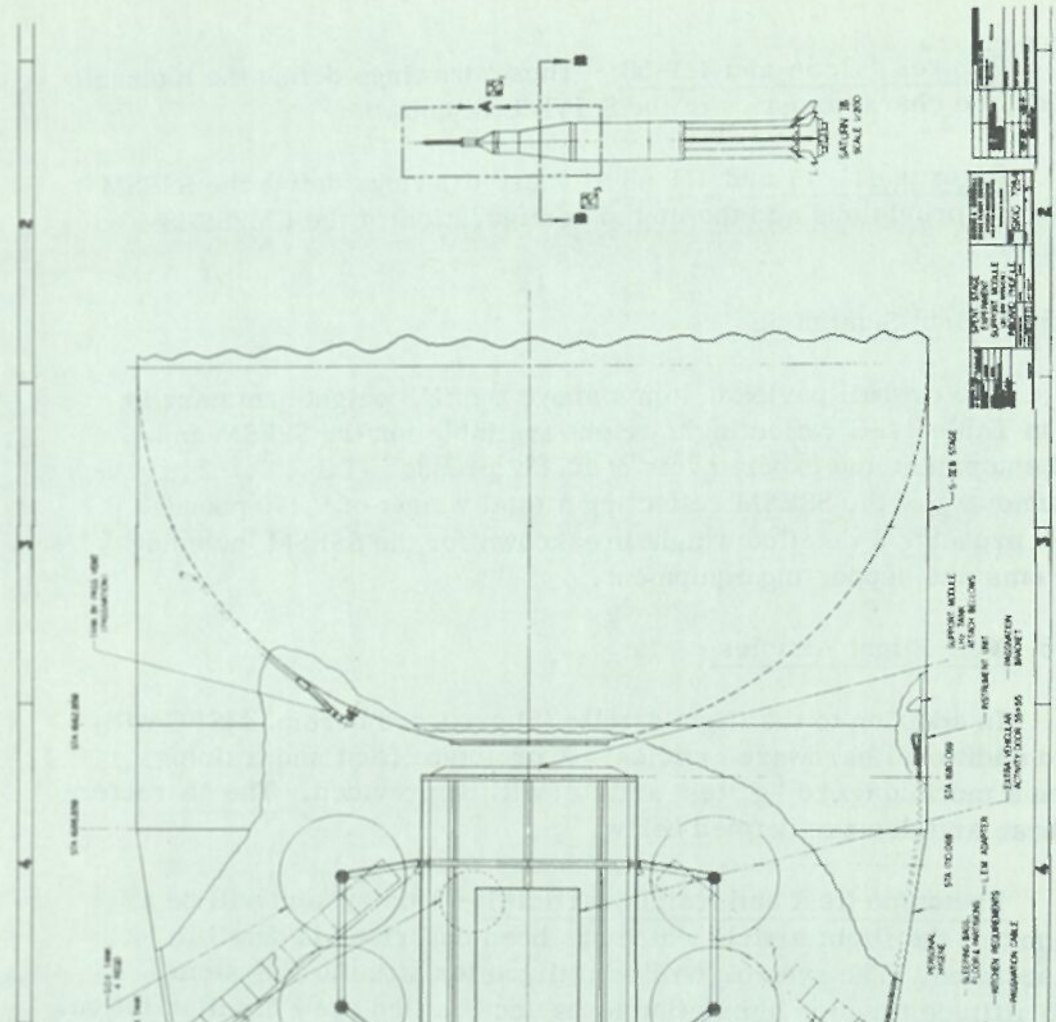


FIGURE 4.1-2. SSESM (20-DAY MISSION) IN BOARD PROFILE (SH. 1)

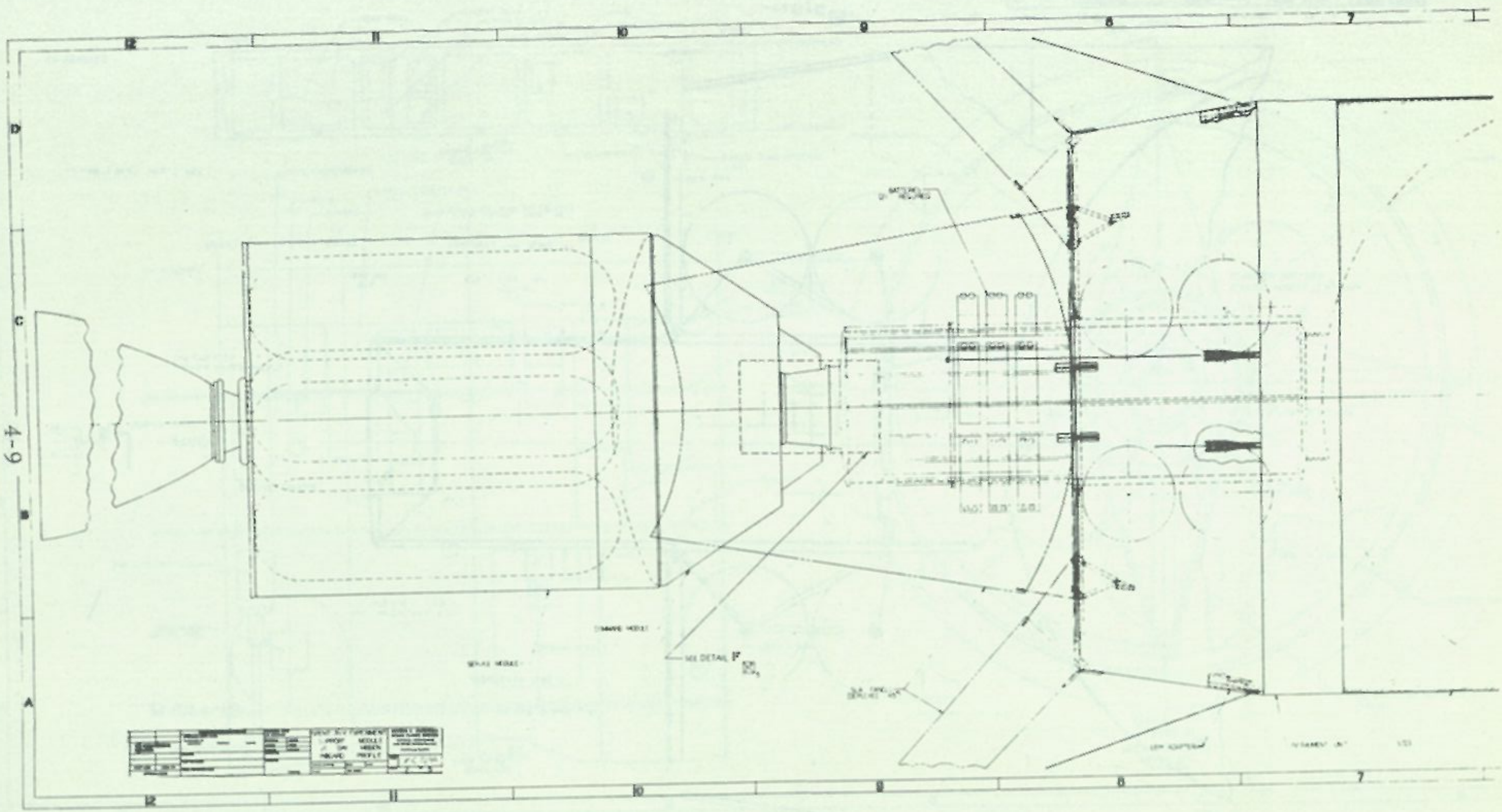


FIGURE 4.1-3. SSESM (20-DAY MISSION) IN BOARD PROFILE (SH. 2)

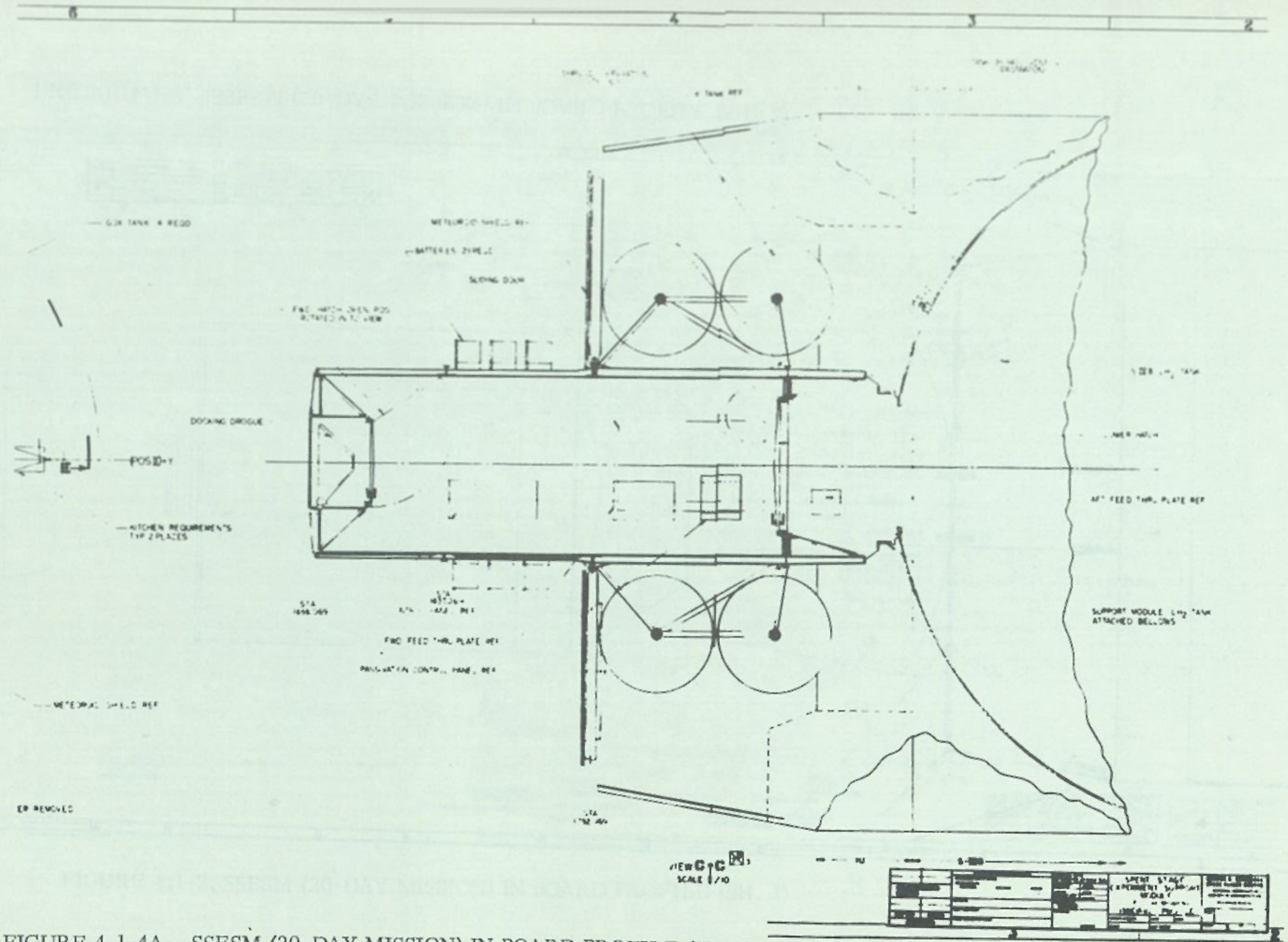


FIGURE 4.1-4A. SSESM (20-DAY MISSION) IN BOARD PROFILE (SH. 3)

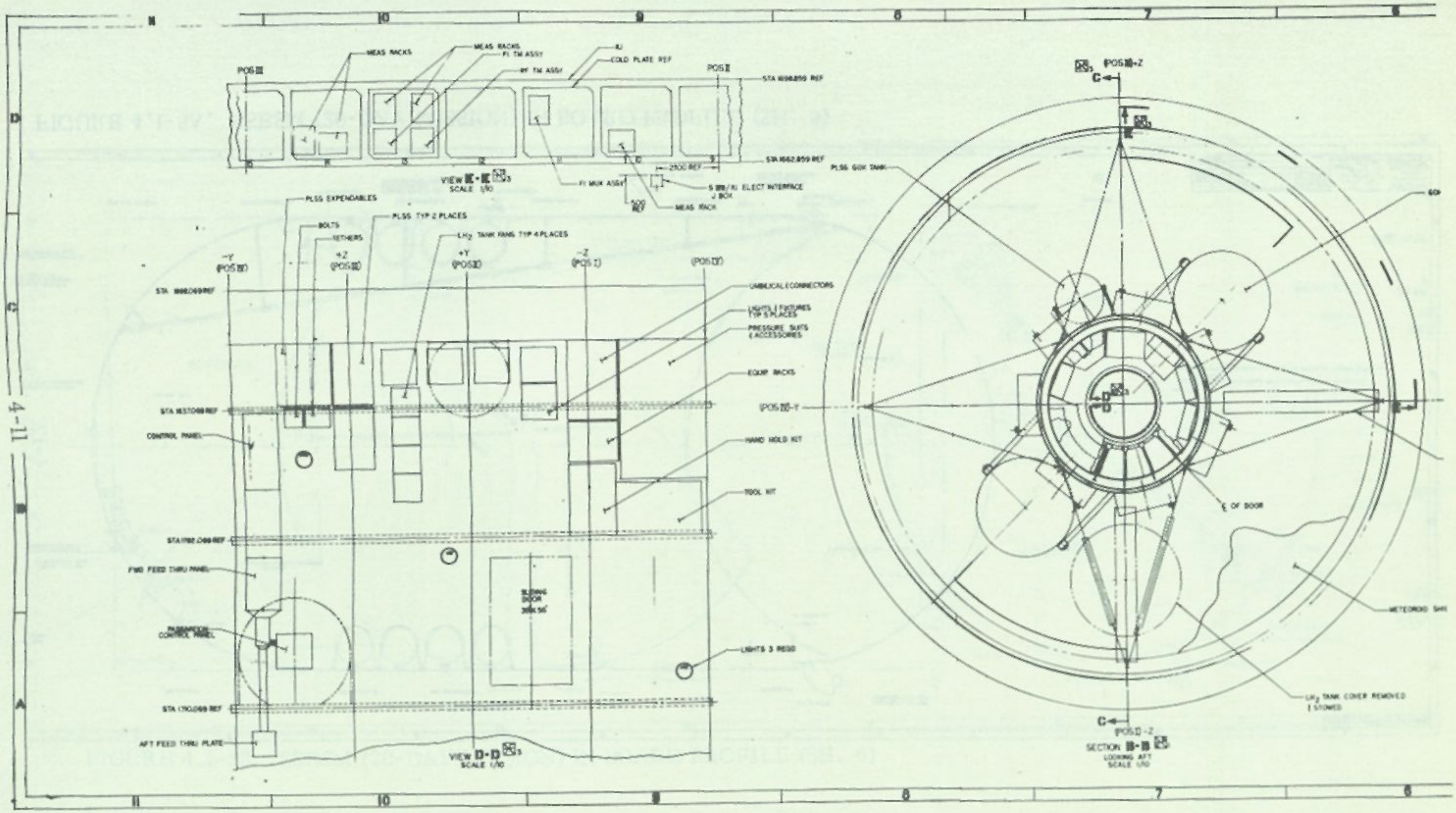


FIGURE 4.1-4B. SSESM (20-DAY MISSION) IN BOARD PROFILE (SH. 4)

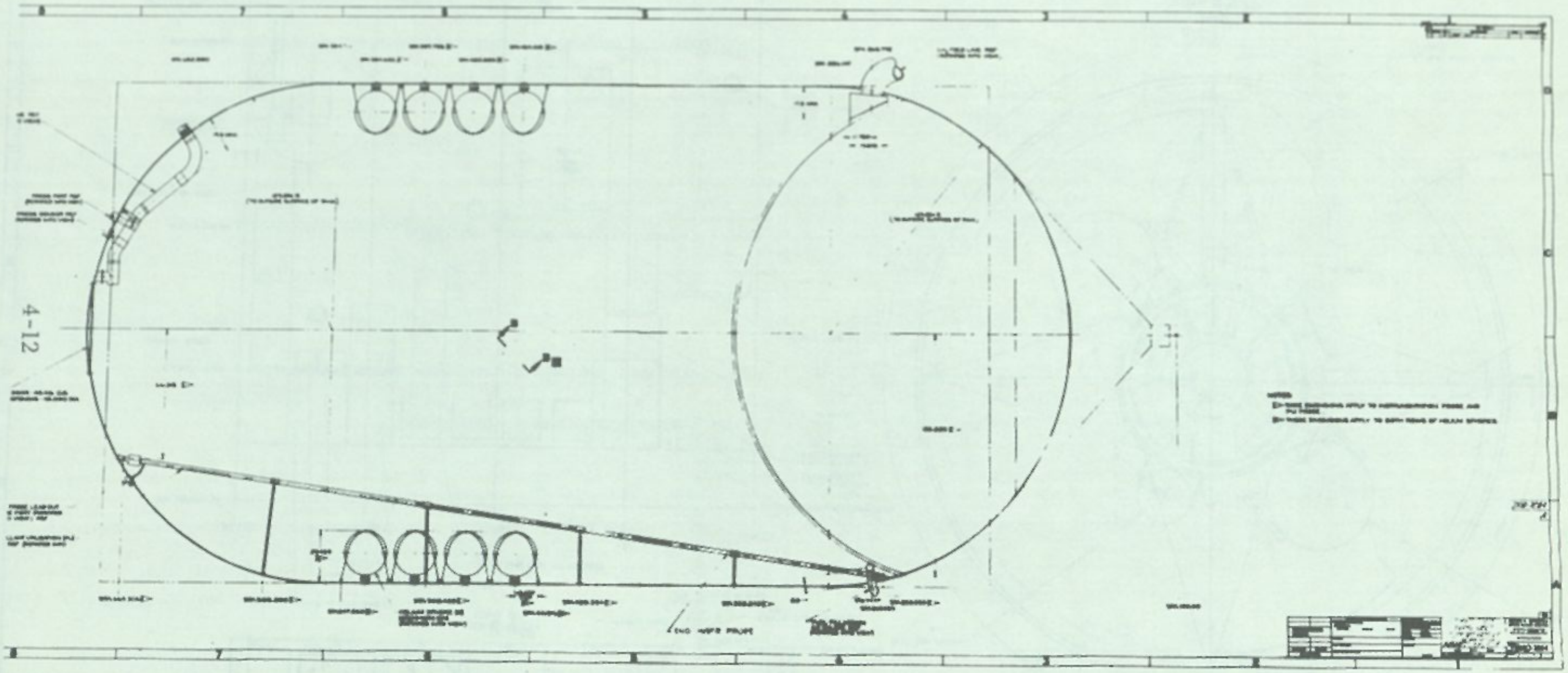


FIGURE 4.1-5A. SSESM (20-DAY MISSION) IN BOARD PROFILE (SH. 5)

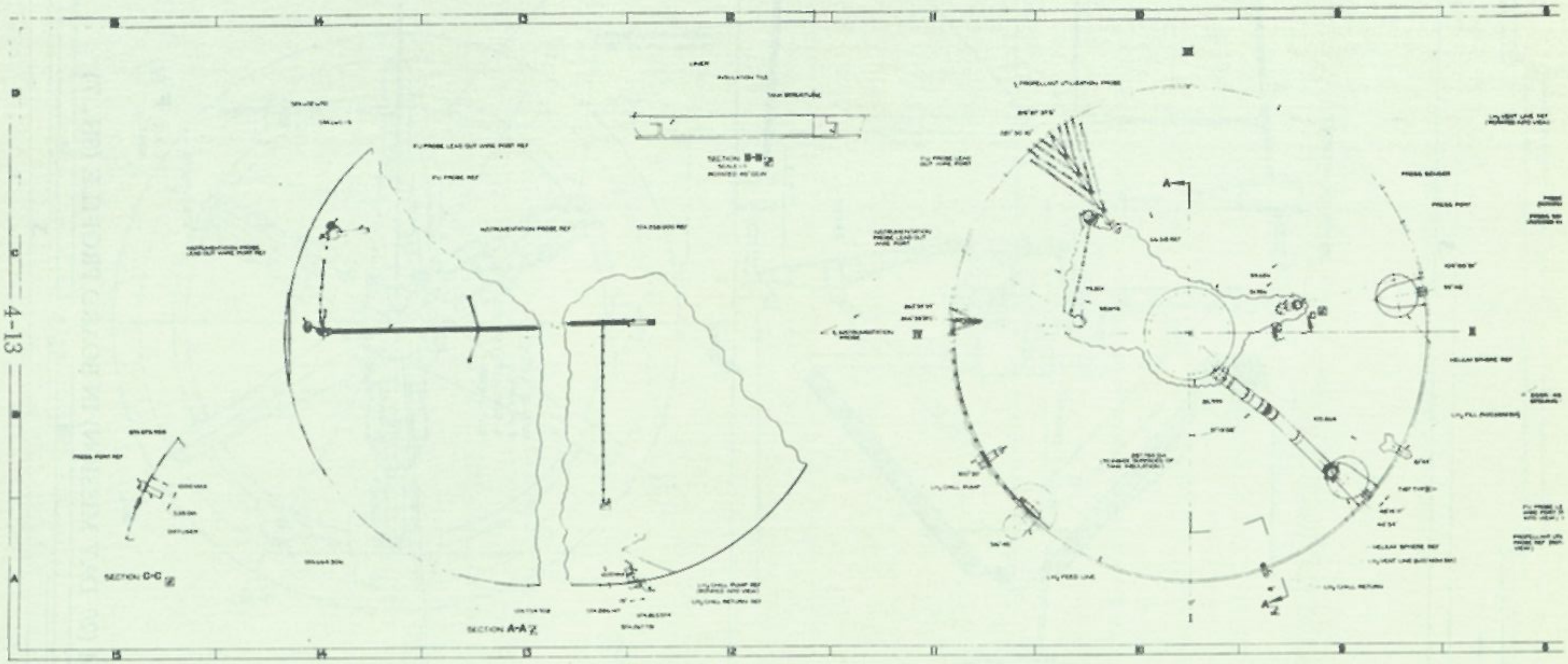


FIGURE 4.1-5B. SSESM (20-DAY MISSION) IN BOARD PROFILE (SH. 6)

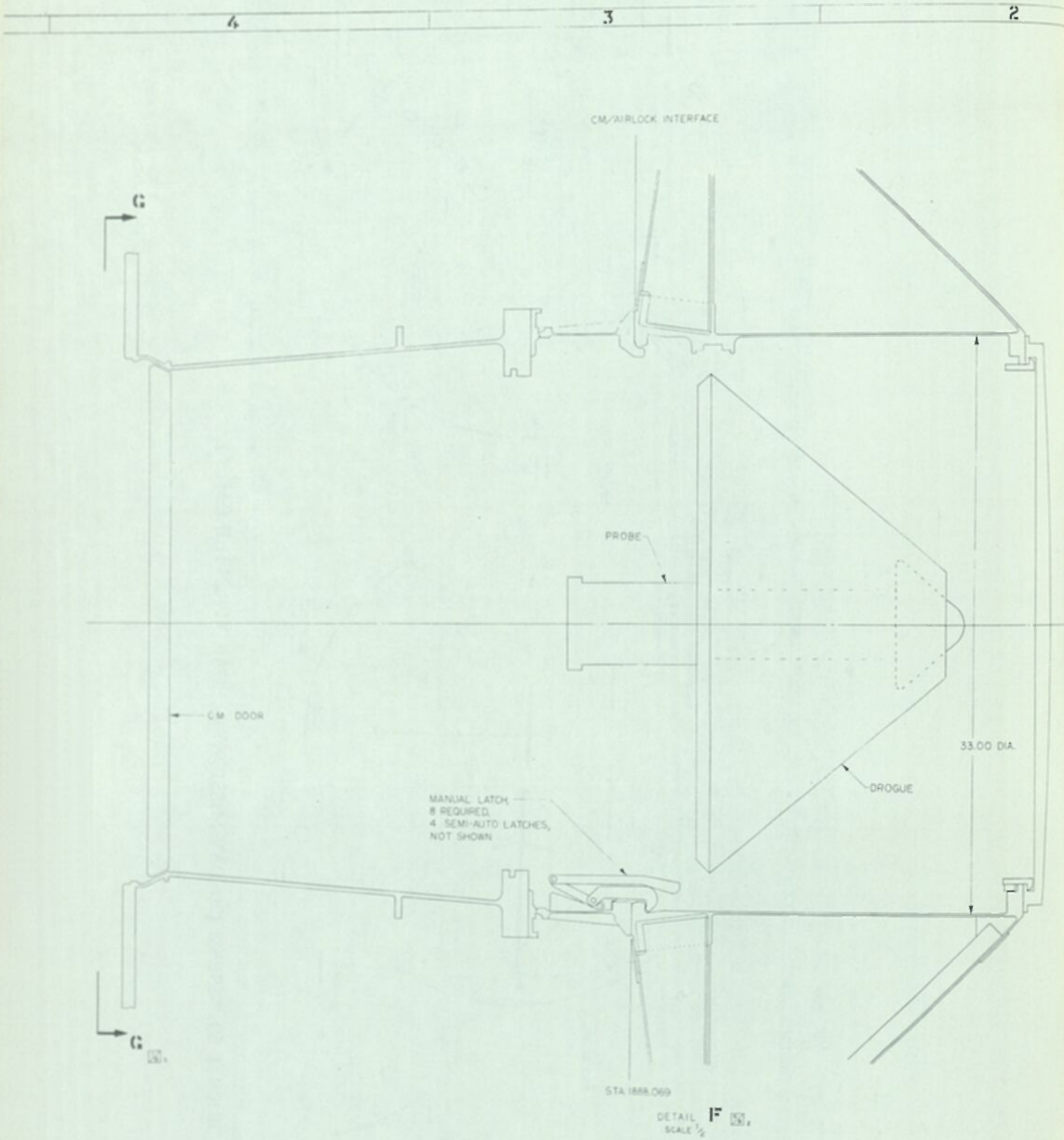


FIGURE 4.1-6A. SSESM (20-DAY MISSION) IN BOARD PROFILE (SH. 7)

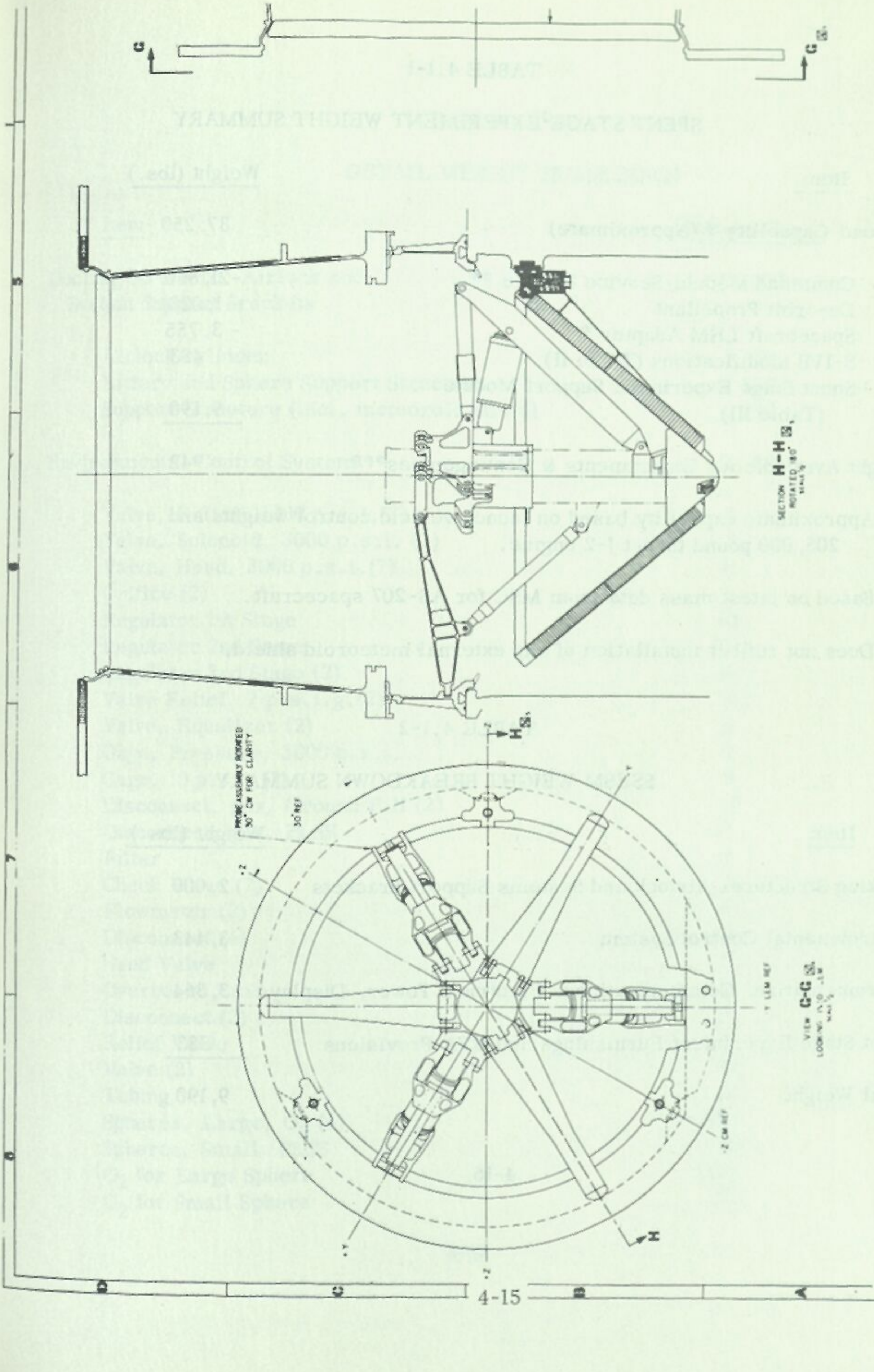


FIGURE 4.1-6B. SSESM (20-DAY MISSION) IN BOARD PROFILE (SH. 8)

TABLE 4.1-1

SPENT STAGE EXPERIMENT WEIGHT SUMMARY

Item	Weight (lbs.)
Payload Capability * (Approximate)	37,250
Command Module/Service Module **	-21,860
De-orbit Propellant	- 1,020
Spacecraft LEM Adapter **	- 3,755
S-IVB Modifications (Table II)	- 483
Spent Stage Experiment Support Module (Table III)	- 9,190
Weight Available for Experiments & Contingencies***	942

* Approximate capability based on launch vehicle control weights and 205,000 pound thrust J-2 engine.

** Based on latest mass data from MSC for AS-207 spacecraft.

*** Does not reflect installation of any external meteoroid shield.

TABLE 4.1-2

SSESM WEIGHT BREAKDOWN SUMMARY

Item	Weight (lbs.)
Docking Structure, Airlock and Systems Support Brackets	2,000
Environmental Control System	3,443
Instrumentation, Communications, Electrical Power, Displays	3,364
Spent Stage Experiment Furnishings and EVA Provisions	383
Total Weight	9,190

TABLE 4.1-3

DETAIL WEIGHT BREAKDOWN

Item	Weight (lbs.)
Docking Structure, Airlock and System Support Brackets	(2,000)
Airlock cylinder	1,150
Battery and Sphere Support Structure	360
Support Structure (incl. meteoroid shield)	490
Environmental Control System	(3,443)
Valve, Relief, 3000 p.s.i.(3)	9
Valve, Solenoid, 3000 p.s.i. (2)	3
Valve, Hand, 3000 p.s.i.(7)	11
Orifice (2)	1
Regulator 1st Stage	10
Regulator 2nd Stage	10
Regulator 3rd Stage (2)	16
Valve Relief, 7 p.s.i.g.(2)	4
Valve, Equalizer (2)	4
Gage, Pressure, 3000 p.s.i.	1
Gage, 10 p.s.i.(3)	3
Disconnect, gox, Ground Fill (2)	2
Disconnect, gox, PLSS	1
Filter	1
Check Valve (7)	3
Flowmeter (2)	1
Disconnect (4)	2
Hand Valve	2
Overboard Line	2
Disconnect (2)	1
Relief Valve	3
Valve (2)	5
Tubing	56
Spheres, Large, O_2 (4)	1960
Spheres, Small, PLSS	50
O_2 for Large Sphere	1222
O_2 for Small Sphere	60

TABLE 4.1-3 (Cont'd.)

Item	Weight (lbs.)
Instrumentation, Communications, Electrical Power, Displays	(3,364)
Measuring Racks, ECS and Housekeeping (2)	42
Multiplexer, 270 Mux,	21
FM/FM Transmitter	14
RF Transmitter	13
Measuring Racks, Experiment (3)	63
Transducers	50
Batteries	2,940
Control Panel	50
Voltage Sensors (2)	1
Display Panel	15
Power Supply	5
Wiring, Plug, etc. (includes Battery Cabling)	150
Spent Stage Experiment Furnishings and EVA Provisions	(383)
LH_2 Tank Fans (6)	30
Portable Task Lamps (3)	2
Airlock Lamps (3)	2
Lights and Fixtures (S-IVB)	10
Tether Kit	8
Reactionless Tool Kit	9
Astronaut O_2 and Life Support Pkg. (3 lines)	20
Portable Life Support System (2)	128
PLSS Expendable (3)	8
Pressure Suit (2)	64
Thermal-Meteoroid Garment (2)	46
Constant Wear Garment (4)	12
Suit Umbilical Connect	3
Equipment Racks (6)	20
Bolt Storage	1
Tool Kit	20
Total	9,190

4.1.7 Interface Requirements

Interface Areas - Four major areas of interfacing are required: (1) Spacecraft to SSESM; (2) SSESM to Instrument Unit; (3) SSESM to S-IVB Stage; and (4) SSESM to KSC Facilities. Areas 1, 2, and 3 are shown on Figure 4.1-7 titled Saturn IB SSE AS-209 Interface Requirements (orbital phase). Area 4 is shown on Figure 4.1-8 titled Saturn IB SSE AS-209 Interface Requirements (KSC Phase).

Extended documentation will be based on these interface areas. An outline of these interface area contingencies are:

1. Spacecraft* to SSESM

a. Command Module docking ring to docking tunnel mounted as integral component of SSESM.

b. SSESM to SLA (LEM attach points).

c. SSESM originated electrical cables to SLA (bonded cable support).

* Consists of CSM and spacecraft LEM adapter (SLA).

2. SSESM to Instrument Unit

a. SSESM associated electrical equipment to IU coldplates.

b. SSESM originated electrical cabling to IU coldplates.

c. Space Envelope Requirements.

3. SSESM to S-IVB Stage

a. SSESM bellows to LH_2 tank.

b. SSESM passivation cable to S-IVB forward skirt interconnect bracket.

c. SSESM originated cabling, fluid lines and equipment to LH_2 tank.

4. SSESM to KSC Facilities

a. SSESM to handling equipment.

b. SSESM MSFC furnished checkout and control equipment to KSC electrical and pneumatic sources.

Interface Tooling - MSC and/or the spacecraft contractor will provide the airlock/spacecraft interface tooling. The S-IVB stage contractor will provide the airlock/S-IVB adapter interface tooling.

Field Splice Connecting Hardware - MSFC will supply the connecting hardware for all airlock unit field splices. The interface hardware will be specified and documented on vehicle assembly documentation by the S-IB stage contractor. The hardware will be delivered to Cape Kennedy in compliance with the vehicle assembly schedule for AS-209.

Interface Control Procedures - Interface control procedures will be under the cognizance of MSFC. All interfaces will be controlled in accordance with Interface Control Documents (ICD's) in the Apollo Intercenter Interface Control Document Log (IA01) and the Saturn Interface Control Document Log (IS01). Supplementary documentation will be prepared as required.

4.1.8 Composite Documentation Plan

All documentation for the experiment shall be prepared in accordance with existing MSFC procedures and shall be released through normal channels as defined in MSFC Drafting Manual. There shall be a minimum of additional documentation developed. All existing S-IVB stage and Instrument Unit drawings will be modified to reflect incorporation of the experiment. The experiment shall have a system specification and subordinate specifications as required to fully reflect configuration and meet the minimum requirements of NPC-500-1. Test plans and procedures will be prepared in accordance with the requirements of NPC-500-10.

A documentation tree shall be prepared to reflect all specifications, test plans, technical documentation, and manufacturing procedures with procedures for preparation, schedules, and responsibilities.

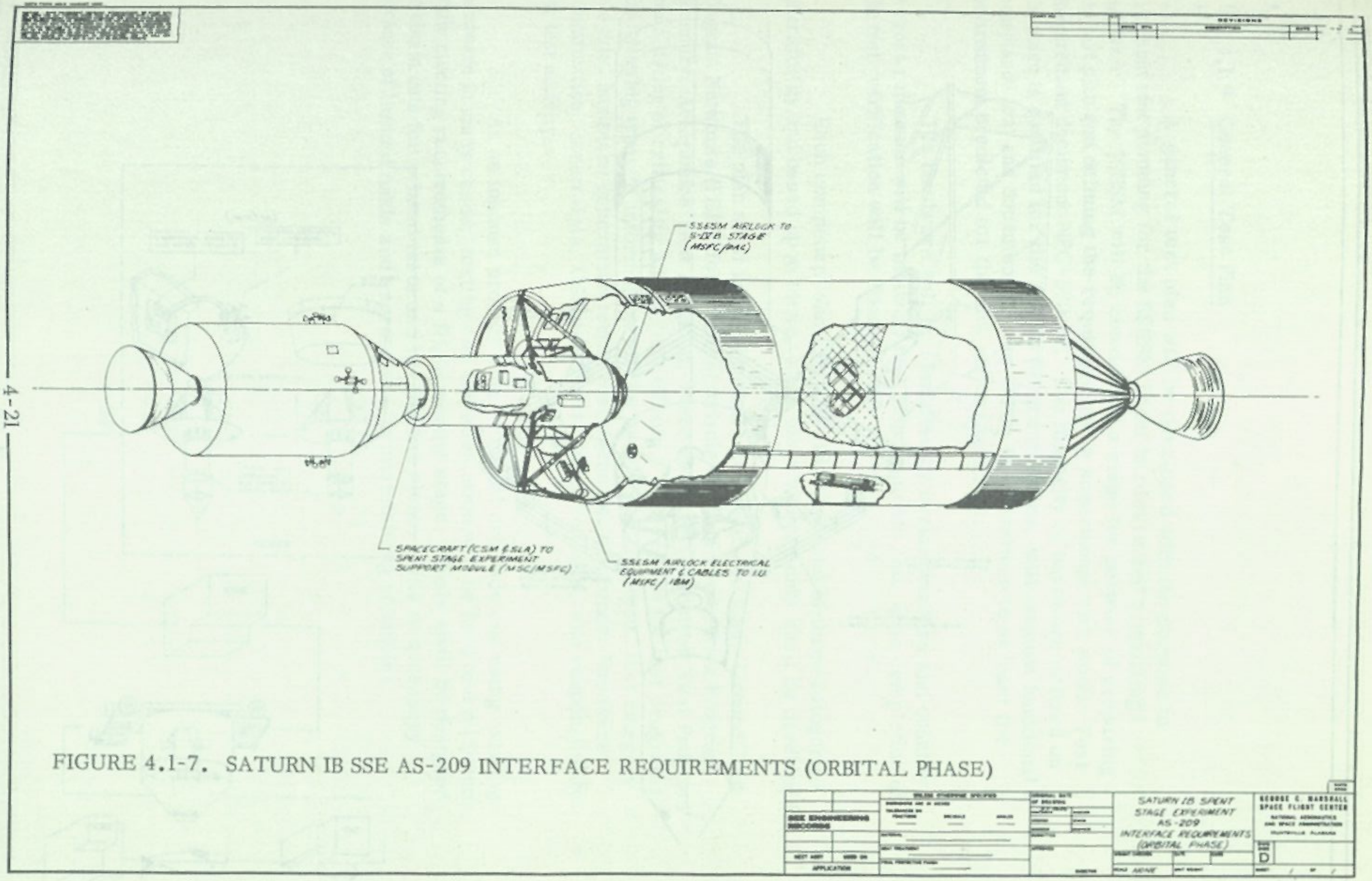


FIGURE 4.1-7. SATURN IB SSE AS-209 INTERFACE REQUIREMENTS (ORBITAL PHASE)

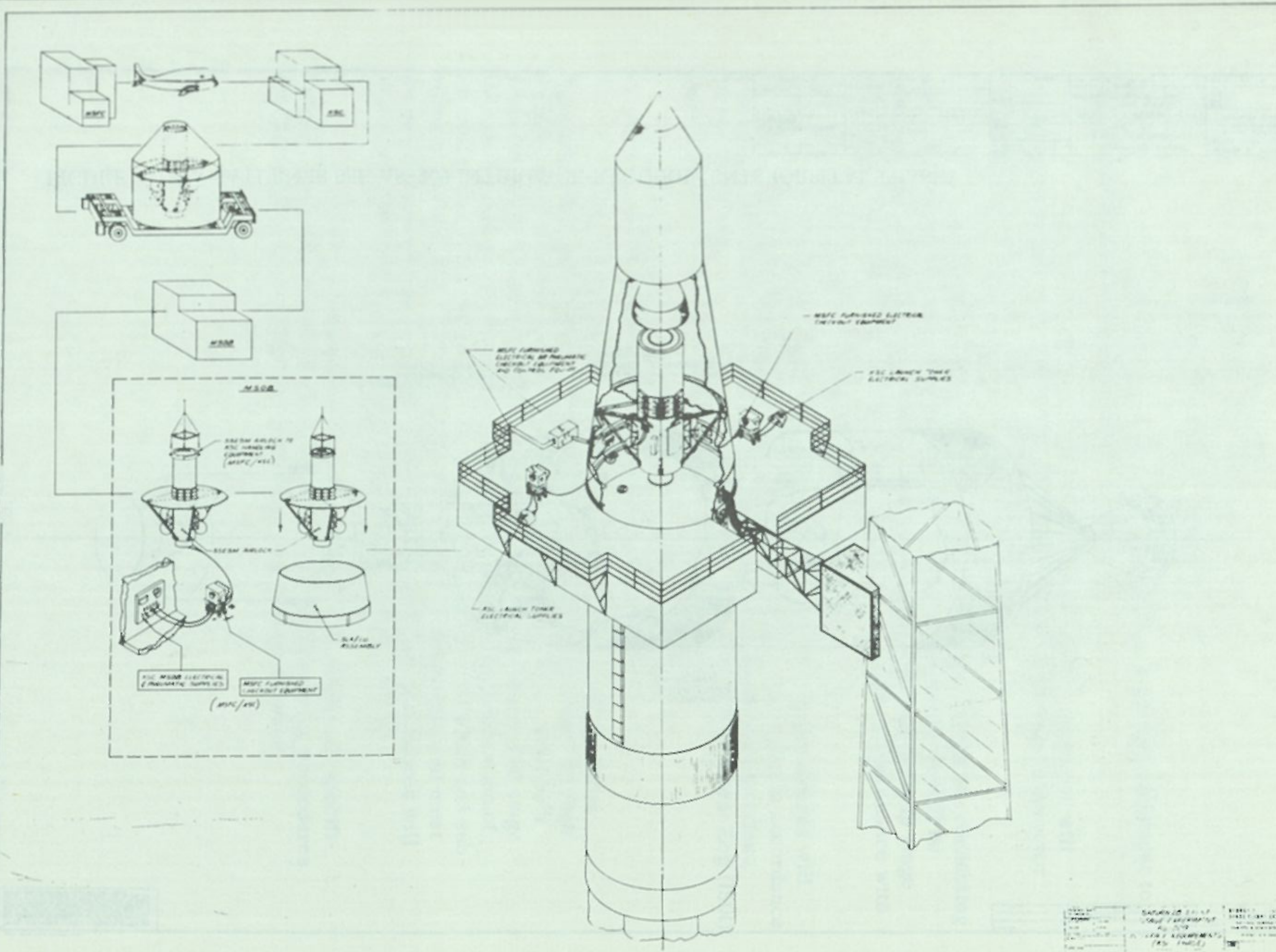


FIGURE 4.1-8. SATURN IB SSE AS-209 INTERFACE REQUIREMENTS (KSC PHASE)

4.1.9 General Test Plan

A general test plan will be developed with the purpose to document test planning for the SSESM flight hardware and supporting hardware. The SSESM will be considered a stage for purpose of preparing the test plan and defining the requirements in accordance with Apollo Test Requirement document NPC-500-10. The majority of hardware utilized on the stage is qualified for the Apollo program; hence, will require functional acceptance test and documentation to verify qualifications to at least the environment predicted for the new zones (location).

The hardware will be identified by serial numbers and quantity. A cross reference will be made to test requirements and, upon completion of the test, verification will be documented.

Each component shall be classified into one of three categories of criticality and based upon these, rigid test requirements shall be developed.

The plan shall be established into major sections: Ground Test Program Networks (PERT form of test article flow), Qualification Program Summary, Acceptance Test Program, Piece Part and Component Test Program and a listing of criticality and failure effects. For each of the test programs the following types of information will be provided: Test type, test category and title, hardware generation level, test document reference, hardware identification, constraints, GSE and facility requirements, and responsibility for test activity.

As an inhouse project with a critical schedule and using existing hardware in many cases, testing and documentation will be held to a minimum while meeting requirements of a flight worthy stage. Tests shall be designed to obtain data for related tests and not for one alone. This is necessary because of limited funds and urgent need for maintaining schedules.

4.2 STRUCTURE

4.2.1 Description

The SSESM, as a structure, consists of a number of basic subsystems as listed below and as shown in Figure 4.2-1:

1. Airlock cylinder
2. Flexible boot
3. Support structure
4. Forward meteoroid protection
5. Pressure container support structure
6. Electrical batteries support structure

Airlock Cylinder - The airlock cylinder is the pressurized portion of the module, providing the connection of the CSM to the S-IVB hydrogen container. The structure is approximately 210 inches long, 65 inches in diameter and has three hatches and doors respectively; the forward hatch is 32 inches in diameter and it is assumed, that the basic LEM hatch can be used without modification. The side door has a clear opening of 55 inches by 36 inches and opens to the inside of the cylinder. The aft hatch has an opening of 48 inches in diameter, and also opens to the inside of the cylinder. The cylinder is of welded construction, using Al 2219-T87. The forward bulkhead incorporates the docking adapter, which is an integral part of this bulkhead. The adapter is 32 inches in diameter, approximately 20 inches long and has all necessary features for incorporating the CSM drogue and latching mechanism. The docking adapter loads (pressure and docking) are distributed to the outside cylinder by integral stiffeners and struts from the lower end of the docking adapter. The forward portion of the airlock cylinder is stiffened by 16 longerons running from the forward bulkhead to the center ring at which the horizontal airlock support struts are attached. The skin of the cylinder is milled from approximately 1 inch plate stock, the longerons are attached to milled ribs with mechanical fasteners. Lugs for tapped holes (from the inside) are provided to give maximum variability for internal attachments of components.

A smaller ring on the forward portion of the cylinder provides attachment for the battery assembly support structure. The larger

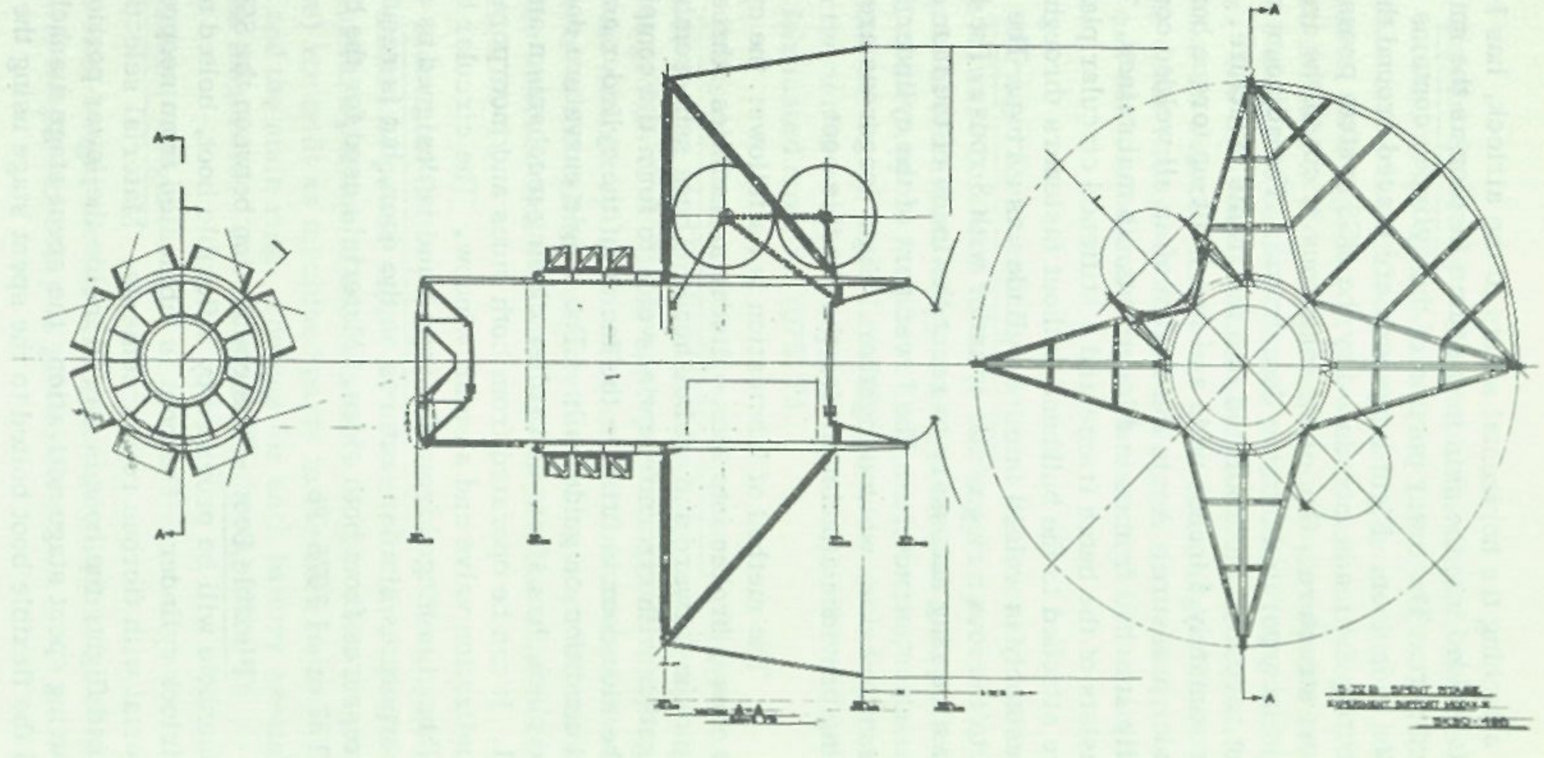


FIGURE 4.2-1. S-IVB SPENT STAGE EXPERIMENT SUPPORT MODULE (SK 30-498)

center ring, attaching the horizontal struts to the airlock, has I-cross section and is welded into the skin in the same fashion as the smaller ring at the forward part. The lower portion of the cylinder contains the large opening for the side door. Eight longerons are placed around the circumference, distributing the loads introduced by the ECS system pressure containers and the support structure. One of the longerons is cut in the area of the door opening and is incorporated into the door frame. The adjacent two longerons, though not cut, are also incorporated into the frame structure. The frame itself is approximately 8 inches wide and is a built-up torque box to limit the deflection under pressure. Again the design of an all welded construction, with externally attached frames and longerons, is maintained. The lower bulkhead consists of the hatch frame and a stiffened circular plate. The stiffeners are attached to the bulkhead without fasteners through the pressure skin. The assembly is welded into the cylinder as a ring. The hatch frame is supported to the lower ring of the cylinder with 8 rods. The bulkhead serves also as the ring necessary to react the loads of the four diagonal struts of the support structure. The lower part of the cylinder is again formed from milled skin, welded together. Eight longerons are provided and an end ring provides attachment for the flexible boot.

The method of fabrication is as follows: the cylinder is, as discussed above, broken into four cylinder assemblies, three ring assemblies and the forward and aft bulkhead. These subassemblies will be welded together with circumferential welds to form the complete airlock cylinder. The side door is fitted to the inside of the cylinder and is supported in the opened condition on guide rails. The single curvature door is designed as a stiffened plate, has a cam and roller closing mechanism and a non-metallic seal. It can be operated from both sides and incorporates a pressure equalization valve and a small window. The circular hatch in the lower bulkhead is hinged from the frame and is designed as a stiffened plate. It incorporates similar features as the door, but is required to seal under pressures from both sides. Materials used for the basic doors are Al 2219-T87 or Al 7075-T6.

<u>Flexible Boot</u> - The connection between the SSESM and the spent stage manhole will be provided by a flexible boot, bolted to the lower ring of the airlock cylinder. The boot is fabricated from neoprene or other suitable material with fibrous reinforcements. Material selection is pending. During powered flight, the boot is stored inside the lower portion of the cylinder. During spent stage activation, the spent stage manhole cover is removed and the flexible boot bolted to the spent stage using the manhole bolt holes.

Support Structure - To attach the support module to the four hard points, located in the SLA (Spacecraft-LEM Adapter), four outriggers are connected to the airlock cylinder. The horizontal structure consists of 8 I-beams, held together at the outside points by four fittings. These fittings also connect to the counterparts in the shroud and are machined parts. The other ends of the beams connect to the center ring in four places with pinned joints. The horizontal beams also provide attachment for the meteoroid shield and its support structure. The four diagonal struts, extending from the outer fittings to the lower ring on the cylinder, are made from tubes and associated end fittings to the ring. The material used is 7075-T6.

Forward Meteoroid Protection - Additional meteoroid protection for the forward bulkhead of the S-IVB stage is provided by a bumper sheet, made of aluminum honeycomb 1-inch thick, supported by channels attached to the support module. This shield forms a closed compartment around the lower half of the SSESM and a door to permit extravehicular activities is provided adjacent to the cylinder side door. The structure is fabricated from Al 7075-T6.

Pressure Container Support Structure - The necessary pressure containers for the slug pressurization of the module and spent stage are stored in groups of two in an appropriate support, mounted to the lower part of the airlock cylinder. The support structure is basically a truss and connects to the cylinder at only four places, at the center and lower rings and the intersection with the longerons. The trusses are fabricated from Al 7075-T6.

Electrical Battery Support Structure - The necessary batteries for the electrical power supply are mounted with a special structure to the forward portion of the cylinder. Eight of the longerons, (every second one) are built as machined parts in the area between the smaller forward and the main ring. Shelves for each battery assembly are mounted to the longerons. The material used is Al 7075-T6.

4.2.2 Structural Weights

The total weight for the structure and all parts as described is 2,000 pounds. A weight summary is given below:

1.	Airlock cylinder (Including flexible boot)	1,150 lbs.
2.	Battery and pressure container support structure	360 lbs.
3.	Support structure (Including meteoroid shield)	490 lbs.
	Total	2,000 lbs.

4.2.3 Structural Test Requirements

The structural test requirements are to structurally qualify the SSESM prior to launch by verifying the integrity of the structure under simulated flight load environments and to substantiate the basic design assumptions and methods of analysis. These tests will be performed on the test and training article.

Pressure Tests - on the airlock cylinder will require three separate pressure cycles in order to subject the door and hatches to the required magnitude and direction of pressure, and to qualify the total airlock cylinder.

Static Load Tests - on the entire configuration, applying both the static load and corresponding dynamic (equivalent static) load will simulate the most critical flight condition.

Docking Load Tests - will qualify the docking structure to the given docking load requirements.

Component Tests - will verify locking mechanism, leak rate and deflection on the door and hatch.

Vibration and Acoustic Tests - The assembly vibration tests will employ sinusoidal sweep and random dwell tests in three axes on an assembled prototype airlock structure with all associated equipment and hardware installed. Those items of equipment required to function during launch and boost should be functionally checked during this test. Items which cannot be made available for this test should be simulated as closely as possible with dummy items.

The assembly acoustic tests will check the airlock assembly described above in a reverberant acoustic field to specified procedures. Functional tests should be made on appropriate equipment.

The meteoroid shield acoustic test will subject a representative section of the meteoroid shield to the acoustic environments to assure reliability. Plane wave and/or reverberant field acoustic tests will be specified.

Detail requirements for vibration and acoustic testing are given in the Appendix.

4.3 ENVIRONMENTAL CONTROL SYSTEM (ECS)

4.3.1 System Description

1. Concept

The spent stage environmental control system (ECS) is designed to provide airlock/lab atmosphere contamination and thermal control via a combination of active and passive techniques; thereby utilizing, to the maximum extent possible, off the shelf flight qualified hardware. These techniques afford a minimum approach to satisfy shirt sleeve environment maintenance with the intent to be compatible with launch schedules at minimum costs.

The large volume of the S-IVB LH_2 tank is unique to 2-3 man spacecraft and permits control of metabolic CO_2 generation within allowable pressure limits by scheduling leakage compatible with occupancy schedules. The H_2O vapor generation can be kept within limits by a combination of atmosphere leakage/O_2 make-up and adsorber utilization. Thermal control of the atmosphere can be kept within limits via judicious establishment of surface coatings and vehicle orientation. The maintenance of electrical equipment temperatures can also be realized with use of passive techniques rather than utilizing active coolant loops.

The ECS is designed to the following parameters:

1. Atmosphere Environment Parameters

a. Temperature	65±25°F
b. Pressure	3.5 to 5.5 p.s.i.a.
c. Composition	Oxygen (one gas)
d. Relative Humidity	30 to 70%
e. CO_2 Limit	.147 p.s.i.a.
f. Temperature Chg/RPL Gas	70±5° F
g. Temperature Suit Loop Gas	50±5°F

2. Metabolic

a. CO_2 Production Rates	2.4 lb/man-day
b. H_2O Production Rates	3.2 lb/man-day
c. O_2 Consumption	2 lb/man-day
d. Heat Production	425 BTU/man-hour

2. Slug Pressurization Mechanics

Continuous Leakage Concept - The atmospheric conditioning system must perform the function of removing carbon dioxide and water vapor such that the percentage of each is within metabolic limitations. This function can be accomplished by venting the atmosphere (O_2 + CO_2 + H_2O) over board and replinishing with pure O_2 (see Figure 4.3-1) at a rate dependent upon the amount of time for astronaut occupancy and time desired prior to exceeding allowable limits. The volume of the occupied space becomes significantly important when establishing time to available contaiminent limits; hence the large volume of the S-IVB LH_2 tank and controlled level of astronaut occupancy afford the so called "slug" concept consideration.

Studies show that the water vapor partial pressure approaches the maximum metabolic limits at a rate exceeding the carbon dioxide concentration. (see Figure 4.3-2 and 4.3-3). As subsequently discussed an adsorber can be easily used to circumvent exceeding the maximum water vapor limit, therefore, the CO_2 limit was utilized to establish mission duration and leakage criteria. Also it should be noted from Figures 4.3-2 and 4.3-3 that the unoccupied period allows a time for the contamination level to reduce.

The mission duration dependency upon occupancy cycle and leakage rate is depicted by the data of Figure 4.3-4. As can be observed from the figure a leakage rate of 30 lbs/day will allow a occupancy cycle (2 astronauts) of 4 hours in the Lab followed by 9 unoccupied hours for a total of 18 days. As a note, 28 mission days can be realized by reducing the occupancy cycle to 4/11 hours.

Blow Down Concept - The continuous leakage concept employs venting oxygen over board at varying (increasing) percentages of carbon dioxide. By allowing the CO_2 to reach the maximum or near maximum limit, then venting the Lab atmosphere down to the minimum total pressure limit followed by re-establishing total pressure (see Figure 4.3-5) with make-up oxygen, a total weight savings could be realized. A weight saving of 100 pounds of O_2 could be realized by this technique (see Figures 4.3-6 and 4.3-7); however, this is not currently planned until astronaut time lines and minimum stage leakage values are established.

Moisture Removal - Moisture removal in current spacecraft systems is accomplished via condensing heat exchangers which utilize coolant loops for a heat sink. In the absence of coolant loops and for desired simplification, moisture removal to accommodate the 30 to 70% relative humidity comfort zone is to be obtained by the use of an adsorbent.

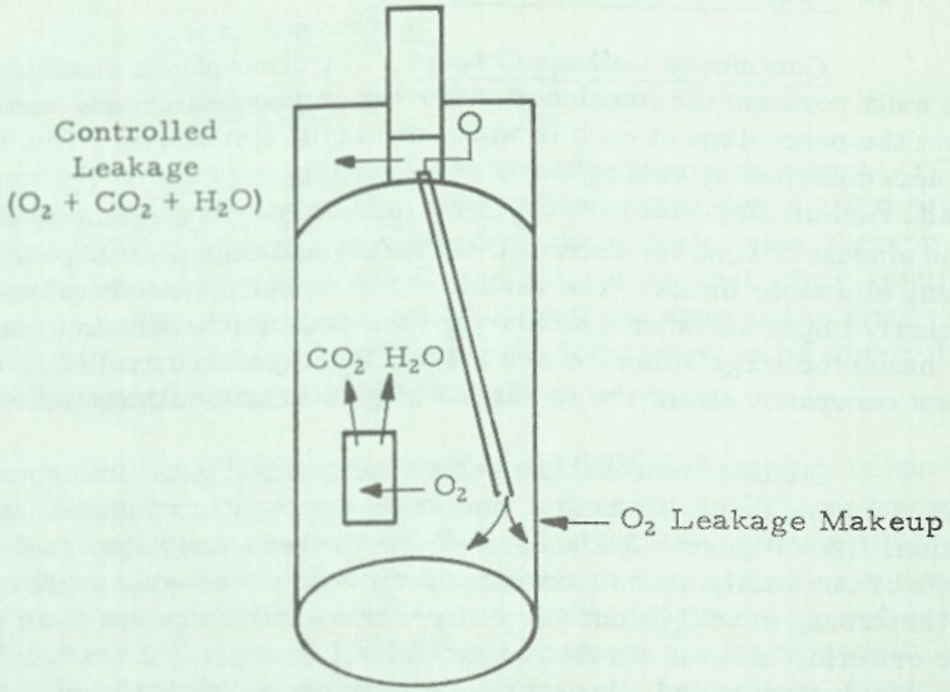


FIG. 4.3-1 CONTROLLED LEAKAGE CONCEPT

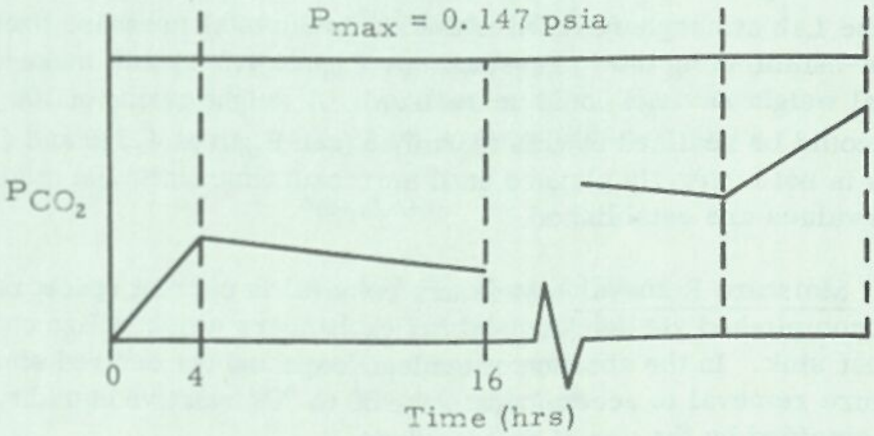


FIG. 4.3-2 CO_2 CONCENTRATION

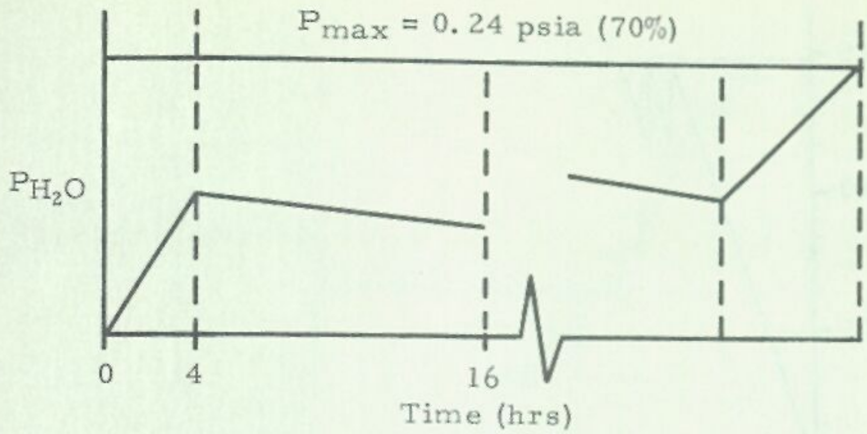


FIG. 4.3-3 H_2O CONCENTRATION

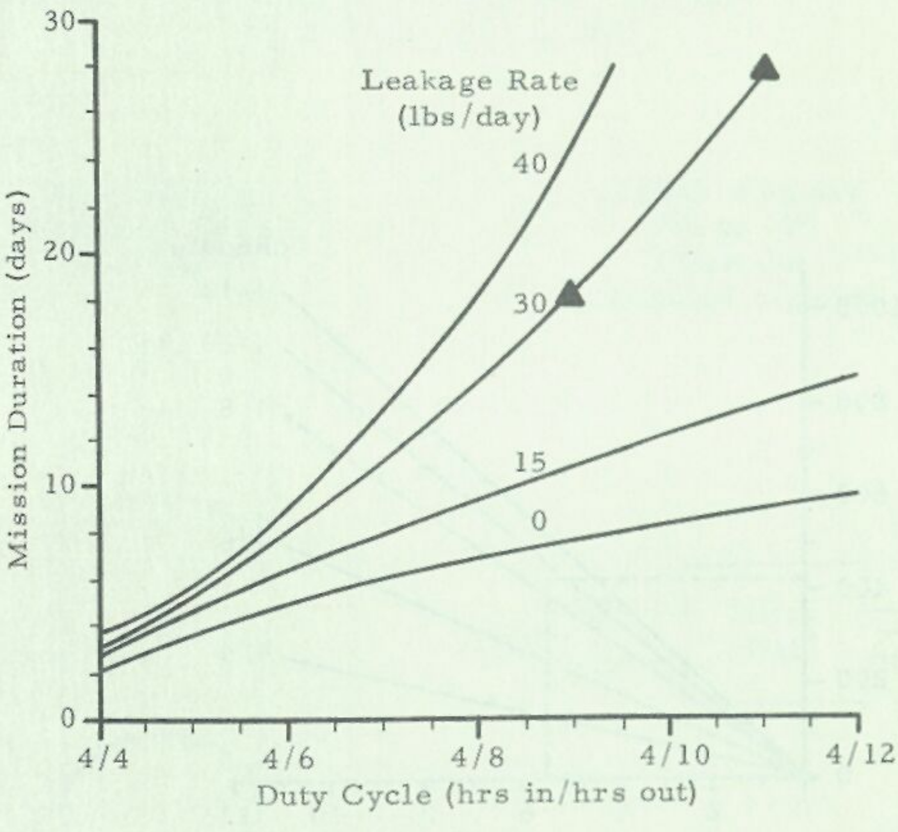


FIG. 4.3-4 MISSION DURATION DEPENDENTS

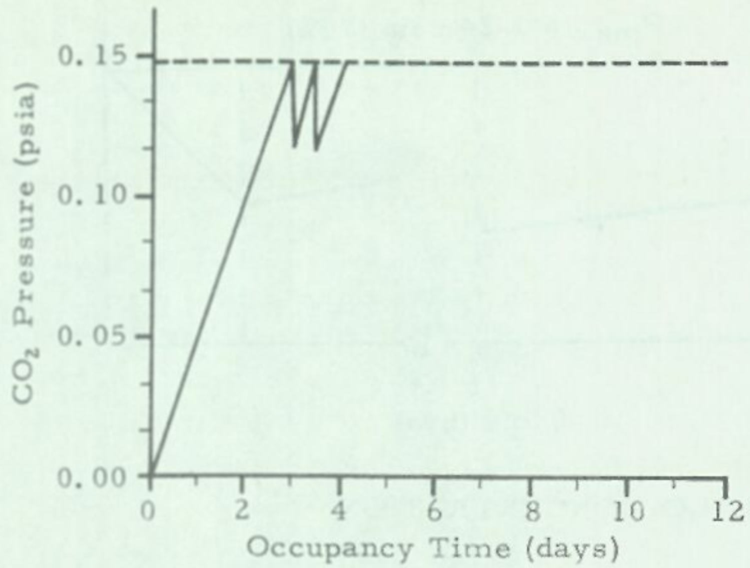


FIG. 4.3-5 BLOWDOWN CO_2 CYCLE

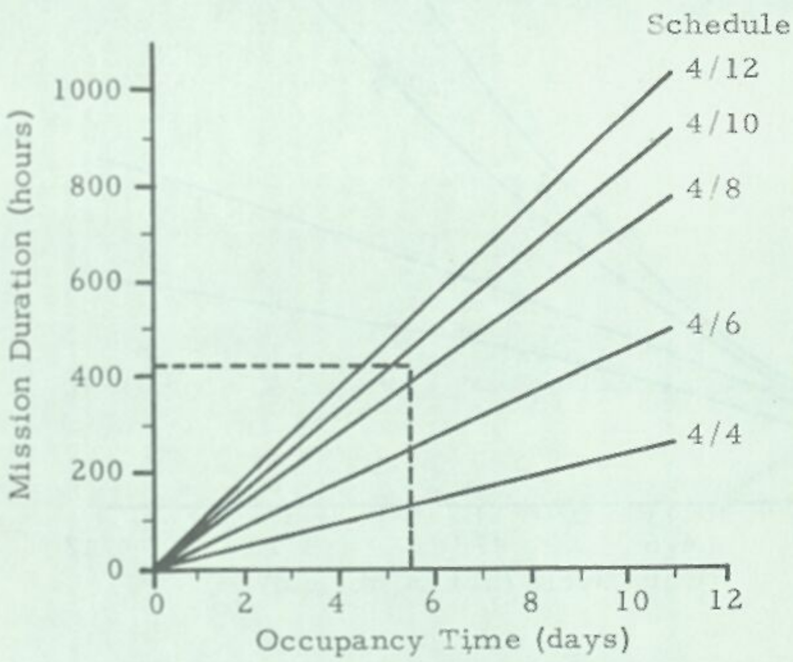


FIG. 4.3-6 OCCUPANCY SCHEDULE AND TIME RELATIONSHIP

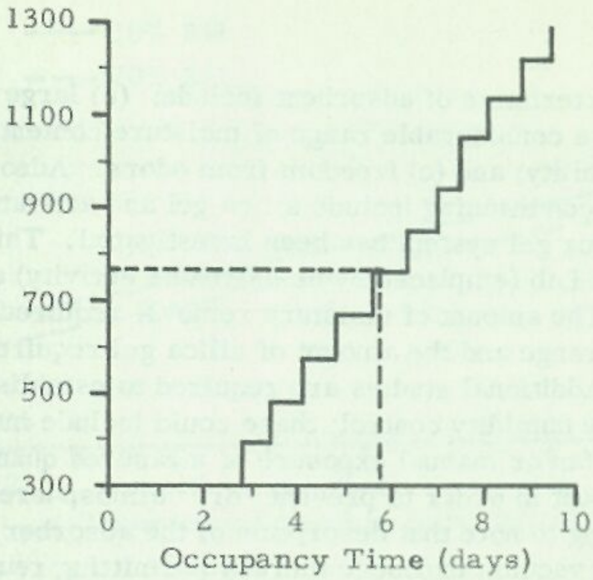


FIG. 4.3-7 OCCUPANCY TIME VS. TOTAL O_2 WEIGHT REQUIRED

Metabolic H_2O	3.2 lbs/man day
Relative Humidity	30% to 70%
Adsorber	Silica Gel Activated Alumina

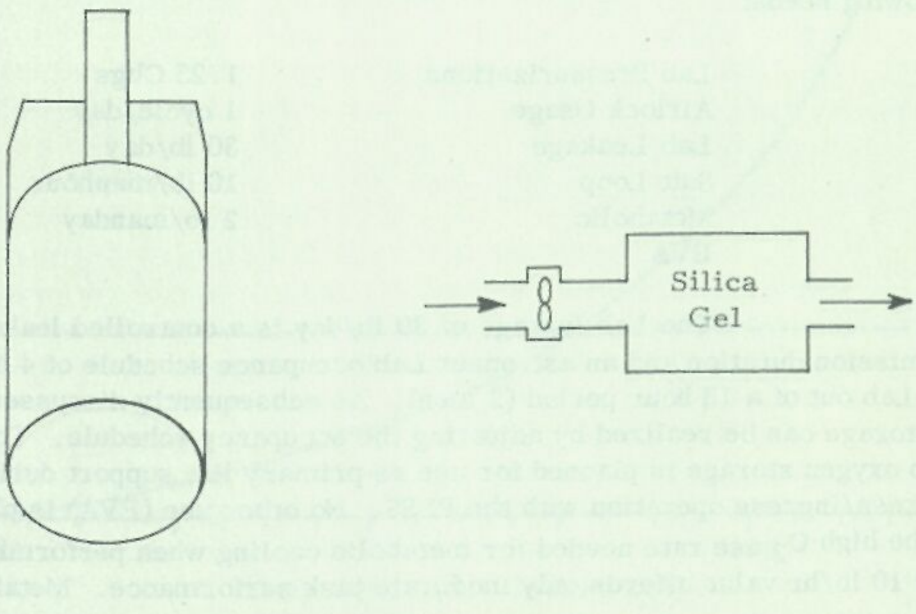


FIG. 4.3-8 SCHEMATIC OF MOISTURE REMOVAL

Desirable characteristics of adsorbent include: (a) large hydroscopic depression over a considerable range of moisture content; (b) chemical and physical stability; and (c) freedom from odors. Adsorbents used in commercial air conditioning include silica gel and activated alumina. The use of a silica gel system has been investigated. This system could be located in the Lab (emplaced by an astronaut activity) as shown in Figure 4.3-8. The amount of moisture removal required to achieve the 30 to 70% R.H. range and the amount of silica gel required is shown in Figure 4.3-9. Additional studies are required to establish operation concept to insure humidity control; these could include intermittent manual operation of the fan or manual exposure of measured quantities of pre-packaged adsorbent in order to prevent "dry" atmosphere conditions. It is also interesting to note that desorption of the adsorber could be accomplished by vacuum exposure thereby permitting reuse, however, a significant weight savings is not involved.

3. Oxygen Storage

The state (cryogenic/gaseous) of storage for the atmospheric oxygen storage is influenced by many parameters and/or engineering trade-offs. The MSFC 20-day concept employs gaseous storage for the reasons of simplicity, low cost and reliability. As an example, GSE cryogenic servicing capability, a key consideration, is not needed; the bottles can be charged a considerable time prior to launch and manually disconnected. The oxygen storage requirements are dictated by the following needs:

Lab Pressurizations	1.25 Chgs
Airlock Usage	1 cycle/day
Lab Leakage	30 lb/day
Suit Loop	10 lb/manhour
Metabolic	2 lb/manday
EVA	

The Lab leakage of 30 lb/day is a controlled leakage dictated by mission duration and an astronaut Lab occupance schedule of 4 hours in the Lab out of a 13 hour period (2 men). As subsequently discussed, variations in storage can be realized by adjusting the occupancy schedule. The suit loop oxygen storage is planned for use as primary life support during airlock engress/ingress operation with the PLSS. No other use (EVA) is planned due to the high O_2 use rate needed for metabolic cooling when performing tasks. The 10 lb/hr value affords only moderate task performance. Metabolic oxygen

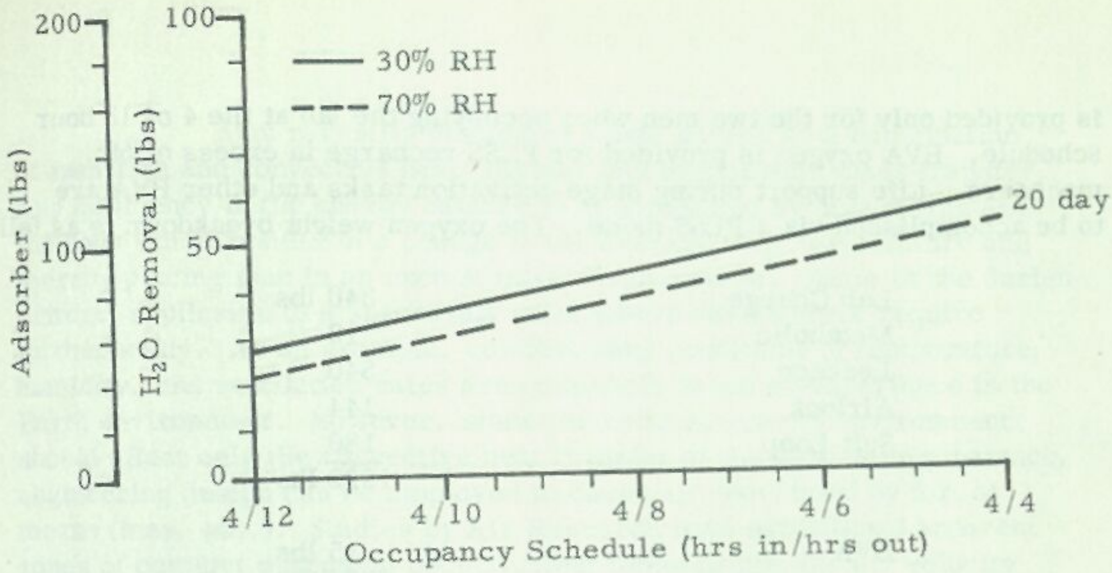


FIG. 4.3-9 WATER VAPOR REMOVAL AND ABSORBENT REQUIREMENTS

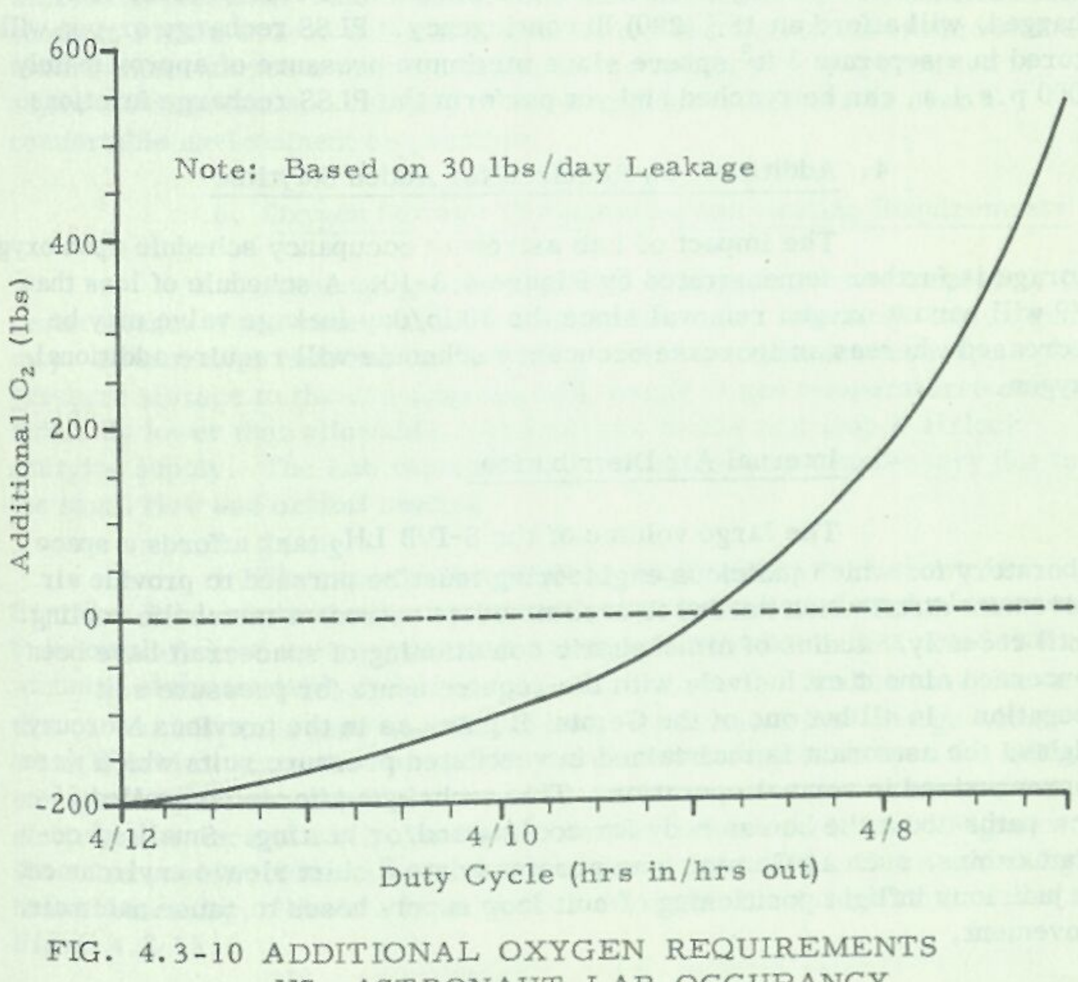


FIG. 4.3-10 ADDITIONAL OXYGEN REQUIREMENTS VS. ASTRONAUT LAB OCCUPANCY

is provided only for the two men when occupying the lab at the 4 of 13 hour schedule. EVA oxygen is provided for PLSS recharge in excess of 60 manhours. Life support during stage activation tasks and other EVA are to be accomplished via a PLSS mode. The oxygen weight breakdown is as follows:

Lab Charge		340 lbs
Metabolic		18
Leakage		540
Airlock		144
Suit Loop		180
		1222 lbs
EVA		15 lbs
	Total	1237 lbs

The O_2 storage is to be accomplished via 19.5 ft^3 containers which with initial and rest pressure of 3000 p.s.i.a. and 50 p.s.i.a. respectively will afford a useable weight of 360 lbs per sphere . Four spheres, fully charged, will afford an 18% (220) lb contingency. PLSS recharge oxygen will be stored in a separate 3 ft^3 sphere since minimum pressure of approximately 1000 p.s.i.a. can be reached and yet perform the PLSS recharge functions.

4. Additional O_2 Required for Added Staytime

The impact of Lab astronaut occupancy schedule upon oxygen storage is further demonstrated by Figure 4.3-10. A schedule of less than 4/9 will permit oxygen removal since the 30 lb/day leakage valve may be decreased whereas an increase occupancy schedule will require additional oxygen.

5. Internal Air Distribution

The large volume of the S-IVB LH_2 tank affords a space laboratory for which judicious engineering must be pursued to provide air movement distribution needed to accommodate astronaut metabolic cooling. Until recently, studies of atmospheric conditioning of spacecraft have been concerned almost exclusively with the requirements for pressure suit operation. In all but one of the Gemini flights, as in the previous Mercury flights, the astronaut is maintained in ventilated pressure suits which were unpressurized in normal operation. This technique affords controlled flow paths about the human body for cooling and/or heating. Small spacecraft cabins, such as Gemini, can accommodate a shirt sleeve environment by judicious inflight positioning of suit loop supply hoses to cause cabin air movement.

Man normally dissipates waste energy by a combination of radiation and convective heat transfer and evaporation mass transfer. Any deficiency in the energy balance is accompanied by heat storage in the body which results in a change in the average body temperature and thereby placing man in an uncomfortable atmosphere. Some of the design criteria applicable to a spacecraft shirt sleeve environment require further study. As an example, comfort zone conditions of temperature, humidity, and ventilation rates are commonly based on experience in the Earth environment. However, since the reduced gravity environment should affect only the convective heat transfer of the total energy balance, engineering design can be employed to cause air movement by forced means (fnas, etc.). Studies by Air Research have established apparent zones of comfort depending upon ambient temperature and air velocity at given conditions of inside cabin wall temperature and atmosphere temperature (see Figure 4.3-11). Fans such as the Gemini cabin fan/ heat exchanger assembly (stripped of the heat exchanger) would afford centerline velocity as shown in Figure 4.3-12. This velocity also varies with radial position. Placement of four fans inside the Lab as schematically shown in Figure 4.3-13 will afford near complete circulation of the volume. By making these fan positions adjustable, as astronaut may be permitted to adjust the direction and velocity at work stations to achieve as near a comfortable environment as possible.

6. Oxygen Storage Temperature and Heating Requirements

Although gaseous storage of oxygen is to be employed, the requirements of use necessitate warming of the gas. Althrough the oxygen may be stored at near the desired temperature, expansion from the high pressure storage to the use-pressue will result in gas temperatures considerably lower than allowable, particularly for the suit loop & airlock charging supply. The Lab replenish O_2 heating may be unnecessary due to the small flow and orbital heating.

The use and temperature requirements are shown in Figure 4.3-14. Individually thermostatically controlled electrical heaters are to be employed for use oxygen temperature maintenance. It is noted that warming of the initial Lab O_2 charge is not proposed. Studies have shown that orbital LH_2 tank side wall heating will warm the initial charge within a single orbit. Since the use temperature is also dependent upon the initial storage temperature, studies have been performed to assess the need for electrical heaters in the O_2 bottles. Limited results depicted by Figure 4.3-15 show that passive thermal control techniques are suitable, thereby voiding heater necessity. These sphere studies employed the assumptions shown in Figure 4.3-15.

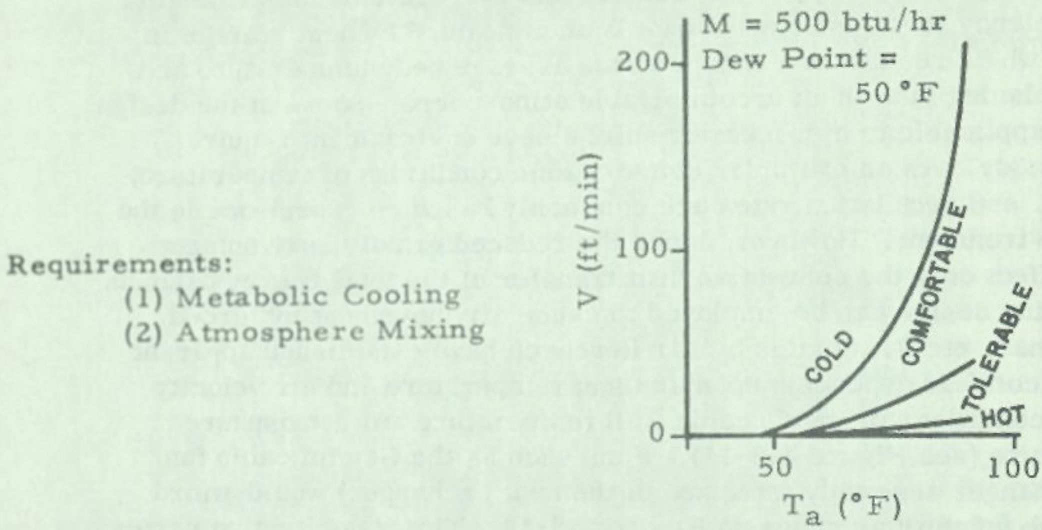


FIG. 4.3-11 ASTRONAUT COMFORT ZONE FOR FORCED CONVECTION

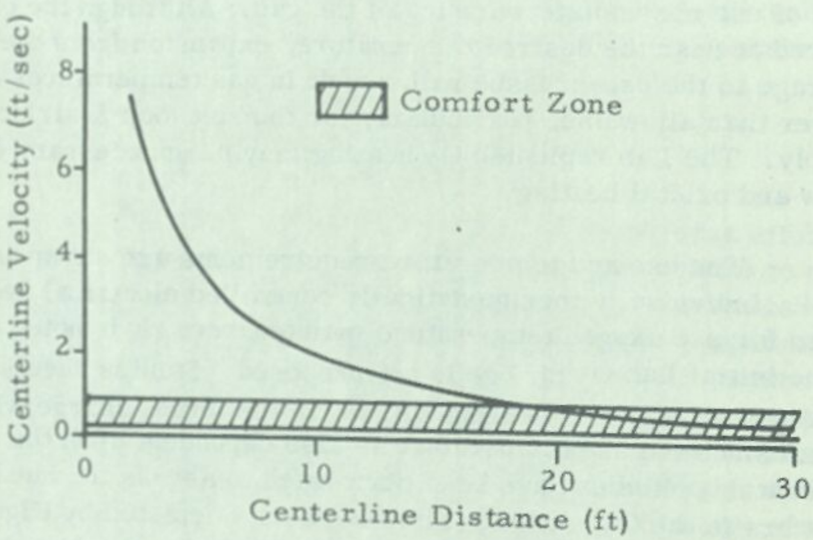


FIG. 4.3-12 GEMINI CABIN FAN VELOCITY PROFILE

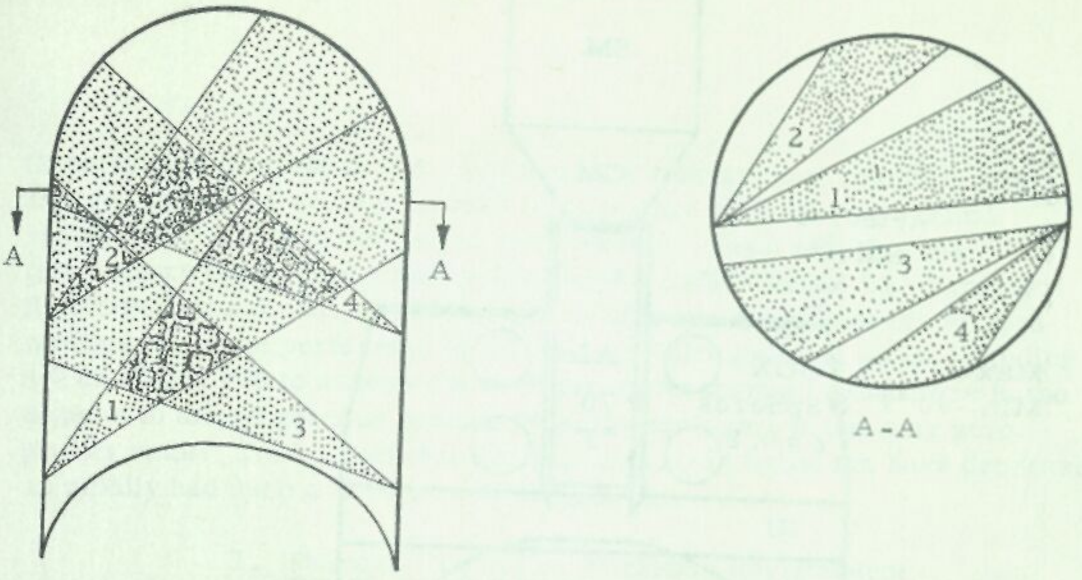


FIG. 4.3-13 SCHEMATIC OF FAN COVERAGE

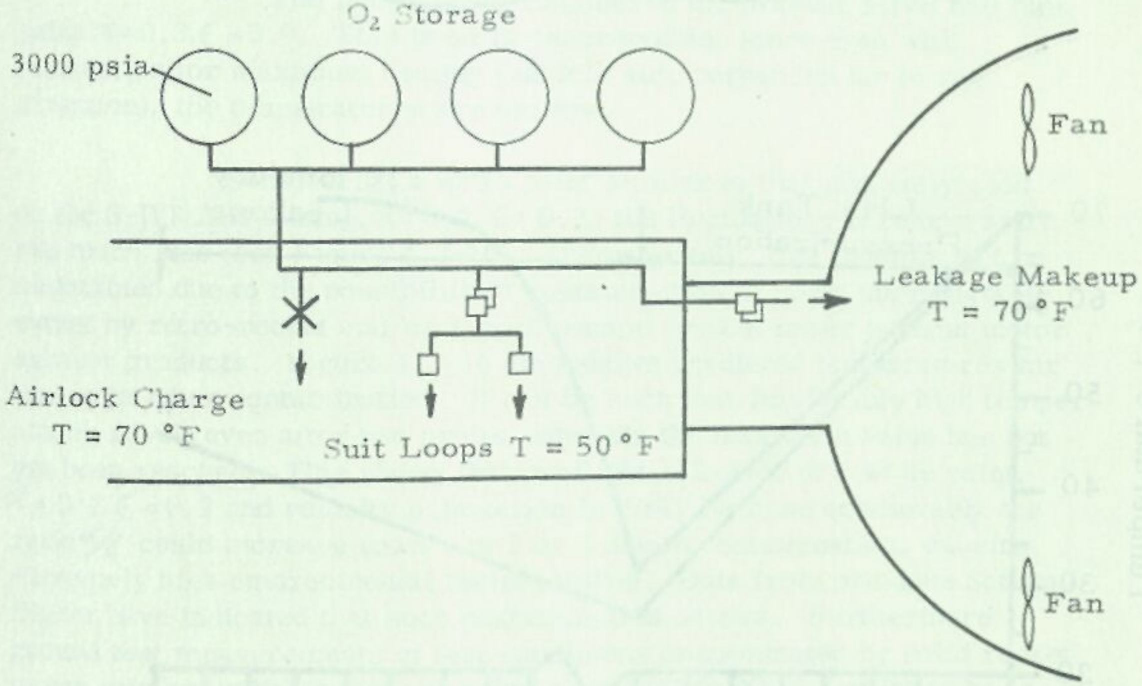
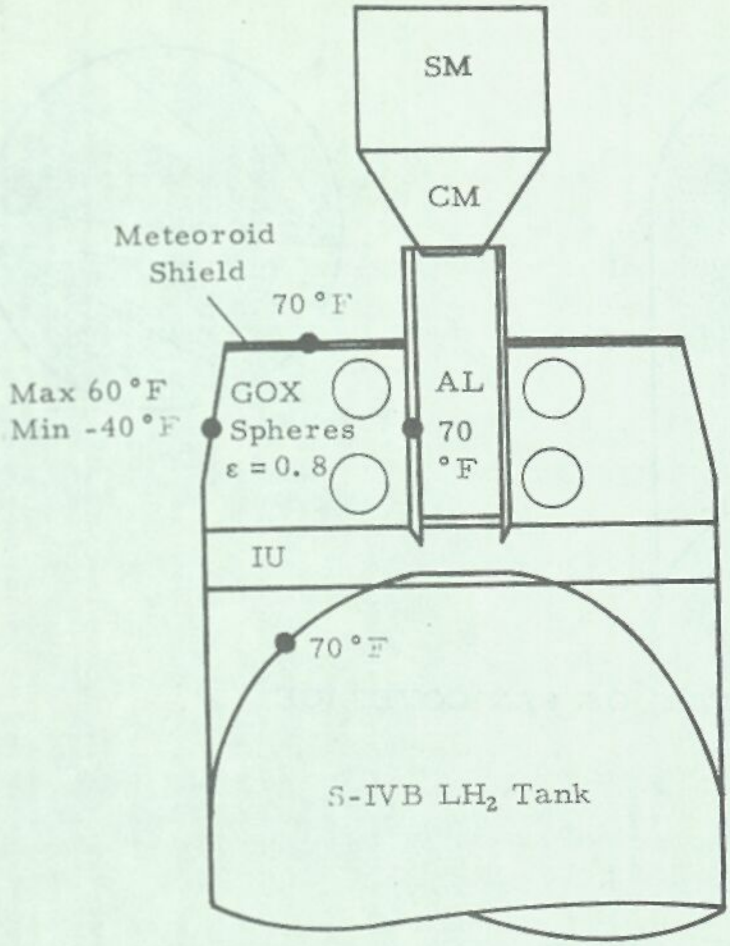


FIG. 4.3-14 OXYGEN USE AND TEMPERATURE REQUIREMENTS



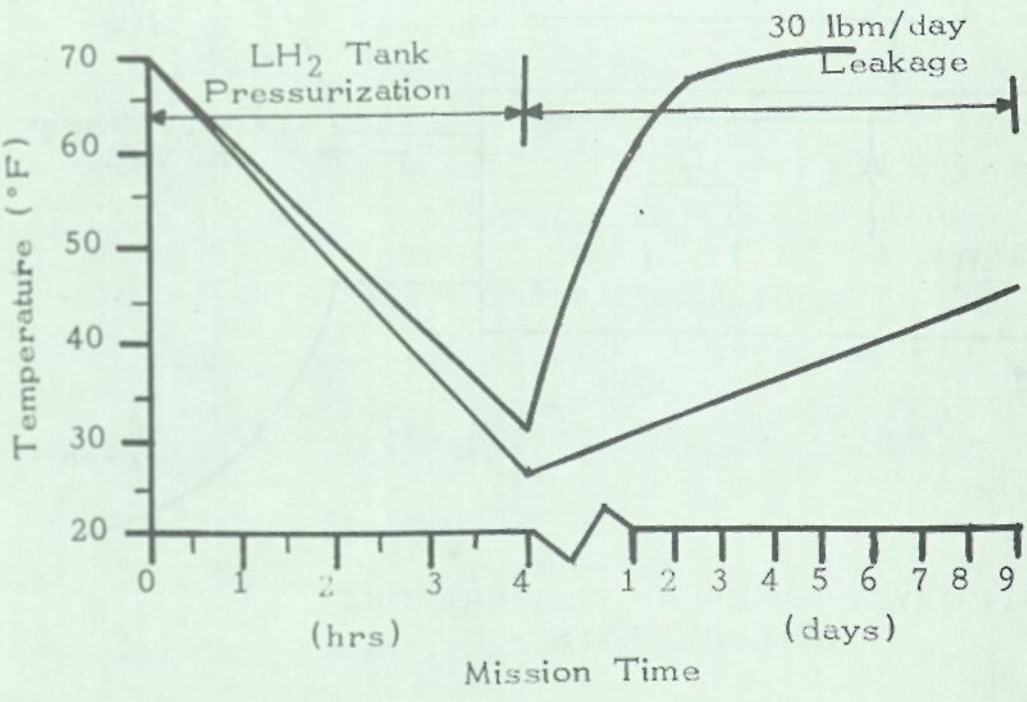


FIG. 4.3-15 GASEOUS BOTTLE TEMPERATURE RESPONSE

Of particular interest is the minimum SLA temperature condition which corresponds to an average orbital temperature with the longitudinal vehicle axis held paralled to the solar vector. Even with this low temperature the oxygen rises in temperature after initial lab charging. Rigorous thermal analysis of the entire SLA area cognizant of use rate profiles are to be performed to further establish painting needs. Studies are also required to assess the need for static thermal conductors in the spheres to afford gaseous heating from the tank wall in the near zero gravity state. The temperature of Figure 4.3-15 would not have decreased as rapidly had such a device been considered.

7. Thermal Control of Workshop Environment

Use of active cooling and/or heating of the internal gas atmosphere is not compatible with either schedule or available funding. Therefore, studies have concentrated on passive thermal control methods. That is, maintaining gas temperatures within acceptable limits (specified as 65±25° F by MSC) through use of thermal control coatings on the S-IVB fuel tank, and through vehicle orientation relative to the Earth and Sun.

The initial study considered the present S-IVB fuel tank paint, $\alpha = 0.3, \epsilon = 0.9$. This paint is unacceptable, since even with orientation for maximum heating (vehicle side perpendicular to sun direction), the temperatures are too low.

Considering a white paint similar to that presently used on the S-IVB APS fairing, $\alpha = 0.2, \epsilon = 0.2$, the fluctuations in temperature are much less (see Figure 4.3-16). Unfortunately, this paint cannot be maintained due to the possibility of contamination to some unpredictable extent by retro-rocket and/or launch escape system tower jettison motor exhaust products. Figure 4.3-16 also shows predicted temperatures for this paint after contamination. It can be seen that intolerably high temperatures result even after two orbits, wherein the maximum value has not yet been reached. This shows that preflight selection of a white paint, $\alpha = 0.2, \epsilon = 0.2$ and velocity orientation is risky because concievably the ratio α/ϵ could increase from 1 to 2 or 3 due to contamination, causing extremely high environmental temperatures. Data from previous Saturn I flights have indicated that such contamination occurs. Furthermore, ground test measurements of test specimens contaminated by solid rocket motor exhaust indicate that the solar absorptivity (α) of a white paint is increased to a greater extent that the infrared emissivity (ϵ).

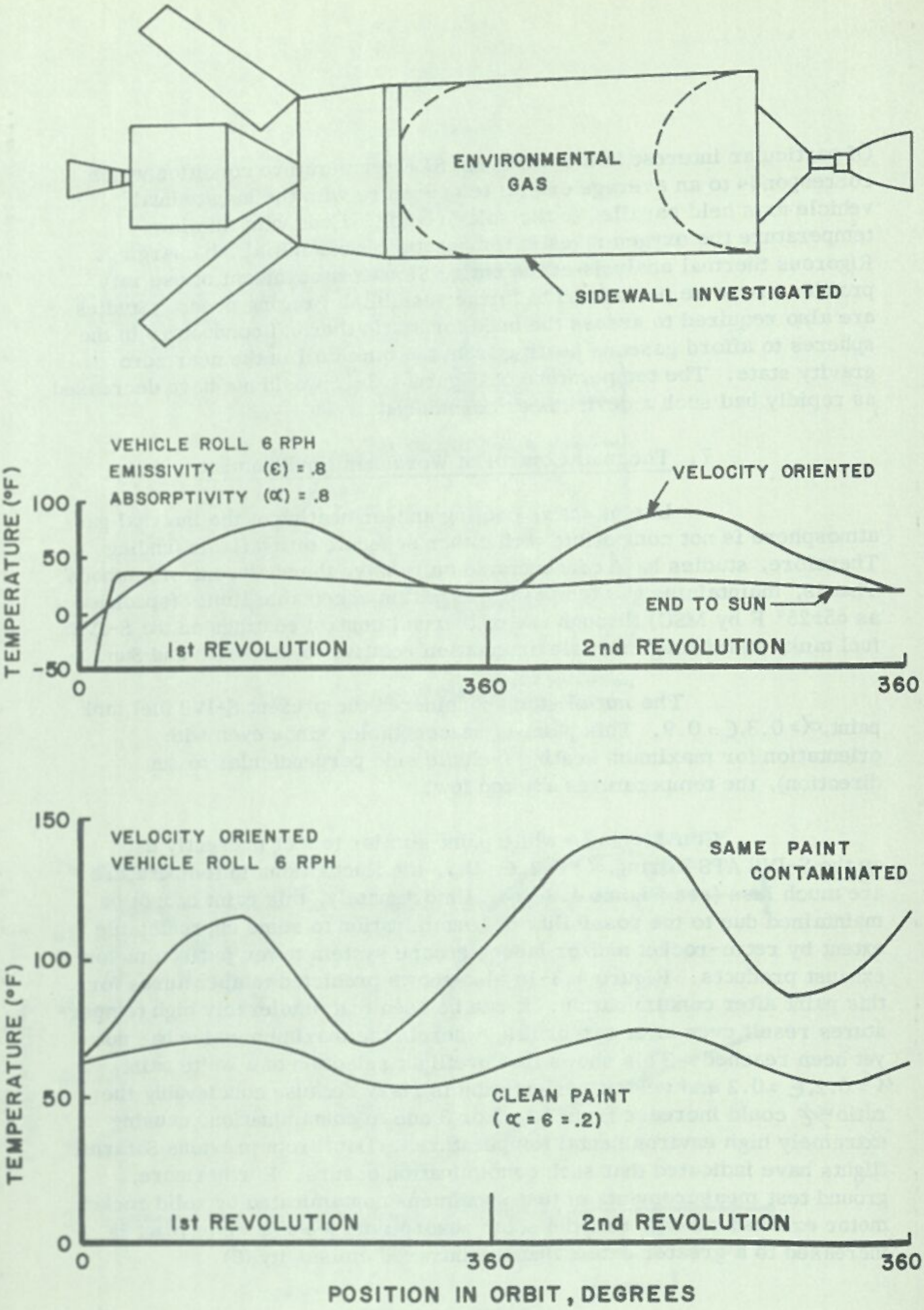


FIG. 4.3-16 ENVIRONMENTAL GAS TEMPERATURES

Figure 4.3-16 shows predicted temperatures for $\alpha = 0.8$, $\epsilon = 0.8$ for orientation of the vehicle longitudinal axis parallel to the sun direction (end toward sun) and also velocity oriented. From this figure, it can be seen that temperatures can be reduced substantially as orientation is changed from velocity to end toward sun. The present thinking is that a coating with $\alpha = 0.8$, $\epsilon = 0.8$ should be used in order to minimize changes in surface properties due to contamination, and that the vehicle should be oriented relative to the sun such as to maintain acceptable temperatures. Presently, studies are being made to verify the feasibility of this approach.

A comfortable environment requires that the inner tank wall temperature, as well as the gas, be maintained at near 70°F. A study showed that even with the internal gas temperature maintained at 70°F, the wall temperature varied from 25 to 130°F. Vehicle rotation of 6 RPH about the longitudinal axis (roll) reduced the maximum temperature from 130 to 85° F and increased the minimum from 25 to 40°F. Therefore, it is recommended that the vehicle be rotated (roll) at 6 RPH.

8. Electrical Equipment Temperature Control

To further establish the feasibility of eliminating the active coolant loops the operating temperature of the electrical equipment must be established and compared with allowable "skin" temperature limits. The planned electrical equipment and their schematic locations are shown on Figure 4.3-17. Analysis of the batteries have been pursued to assess passive thermal control feasibility and is shown in Figure 4.3-18. The three noted cases of Figure 4.3-18 basically show that control can be achieved passively. Of particular interest is the minimum heating condition (Case 3) where the batteries are off. This data shows the temperature response to be sluggish and thereby not requiring heating on the dark side of the orbit. However, covering the batteries with a material with low infrared emissivity may be needed during the stage activation period and for the inactive batteries. Also of significance is the analysis showing that the batteries can be kept sufficiently cool when operating and the vehicle is oriented for maximum solar irradiation (Case 2).

The components mounted on the I.U. have not been studied; however, it is known that the current I.U. thermal conditioning systems release approximately 30% of the total component heat load by radiant means. With considerably reduced heat load, passive thermal control is expected to be adequate. Additional in-depth studies are planned.

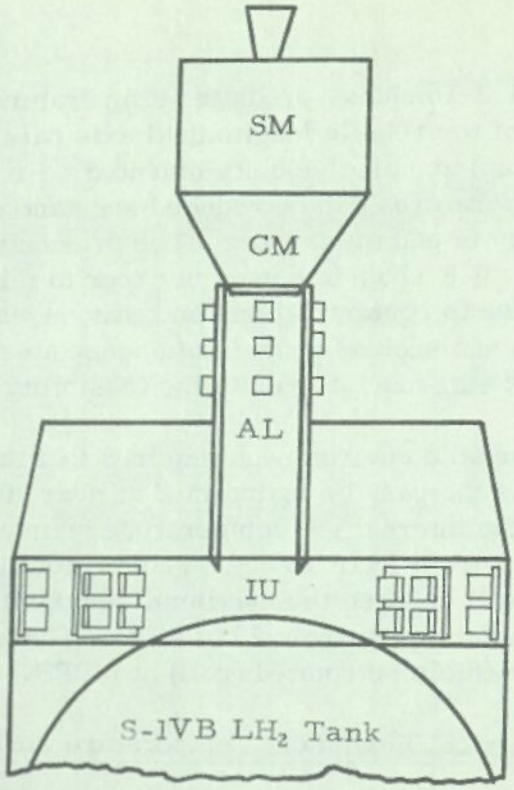


FIG. 4.3-17 ELECTRICAL EQUIPMENT TEMPERATURES

Case 1 Side to Sun Spinning, $\varepsilon = 0.5$, $\alpha = 0.5$, $q = 100$ watts
2 Side to Sun Spinning, $\varepsilon = 0.8$, $\alpha = 0.2$, $q = 100$
3 End to Sun Spinning, $\varepsilon = 0.8$, $\alpha = 0.2$, $q = 0$

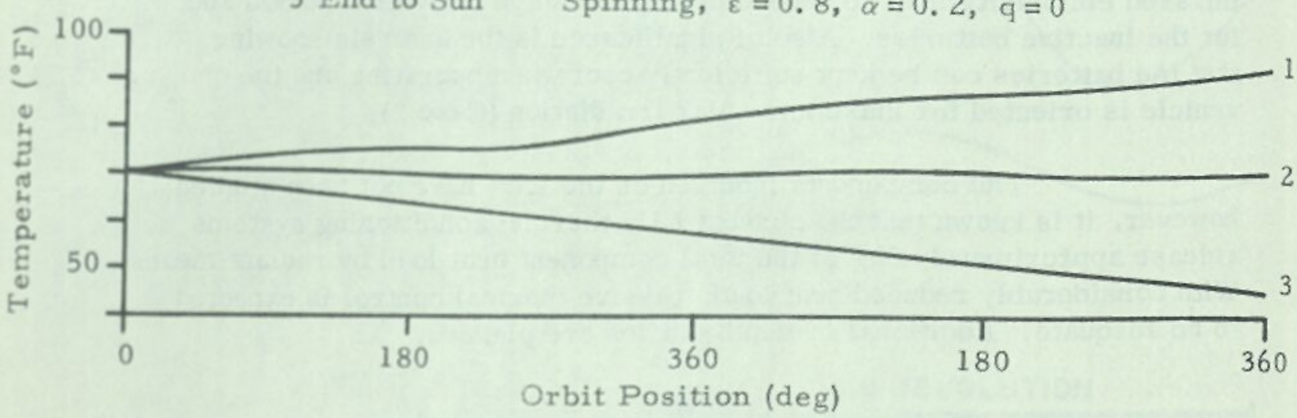


FIG. 4.3-18 BATTERY TEMPERATURE DURING ORBIT

9. Electrical Energy Requirements

As noted in previous sections, electrical energy is needed to provide heat for warming the life support oxygen. In addition, power is needed to power the tank fans for air mixing and astronaut comfort conditioning in addition to control and display power. The power requirements are estimated to be as follows:

	LOAD	DUTY	AVG	KWHR
Tank fans	220 watts	38%	84 watts	36.40
Suit loop O_2	400	5	20	8.64
Lab O_2	20	100	20	8.64
Airlock O_2	450	2	9	3.88
Control & Display	50	38	20	8.64
Total	1140		153	66.20

The duty cycle was established by considering the estimated use period needs subject to additional study.

10. Prelaunch Purging/Thermal Conditioning

The compartment formed by the SLA, I.U., and S-IVB stage forward skirt will require purging to reduce explosive hazard potentials. The injection of GN_2 gas in sufficient quantities will afford O_2 concentrations less than 4% with consideration of air infiltrations. By placing covered openings in the mete oroid shield, the current Apollo spacecraft and I.U. purges, shown in Figure 4.3-19, will afford the required environment inclusive of temperature control.

4.3.2 ECS System Functional Description

1. Function

The two major subsystems comprising the ECS are the stage and airlock atmosphere system (SAAS) and the PLSS Recharge System. The functions of the SAAS are to be provided pressurizing gox for maintaining the LH_2 tank atmosphere, to supply gox to the astronauts' space suits through umbilical lines, and to provide gox for recharging the airlock when used for extravehicular activity (EVA). The function of the PLSS recharge system is to provide gox for recharging the astronauts' portable life support system backpacks for use in EVA. Figure 4.3-20 presents a functional schematic diagram of the systems

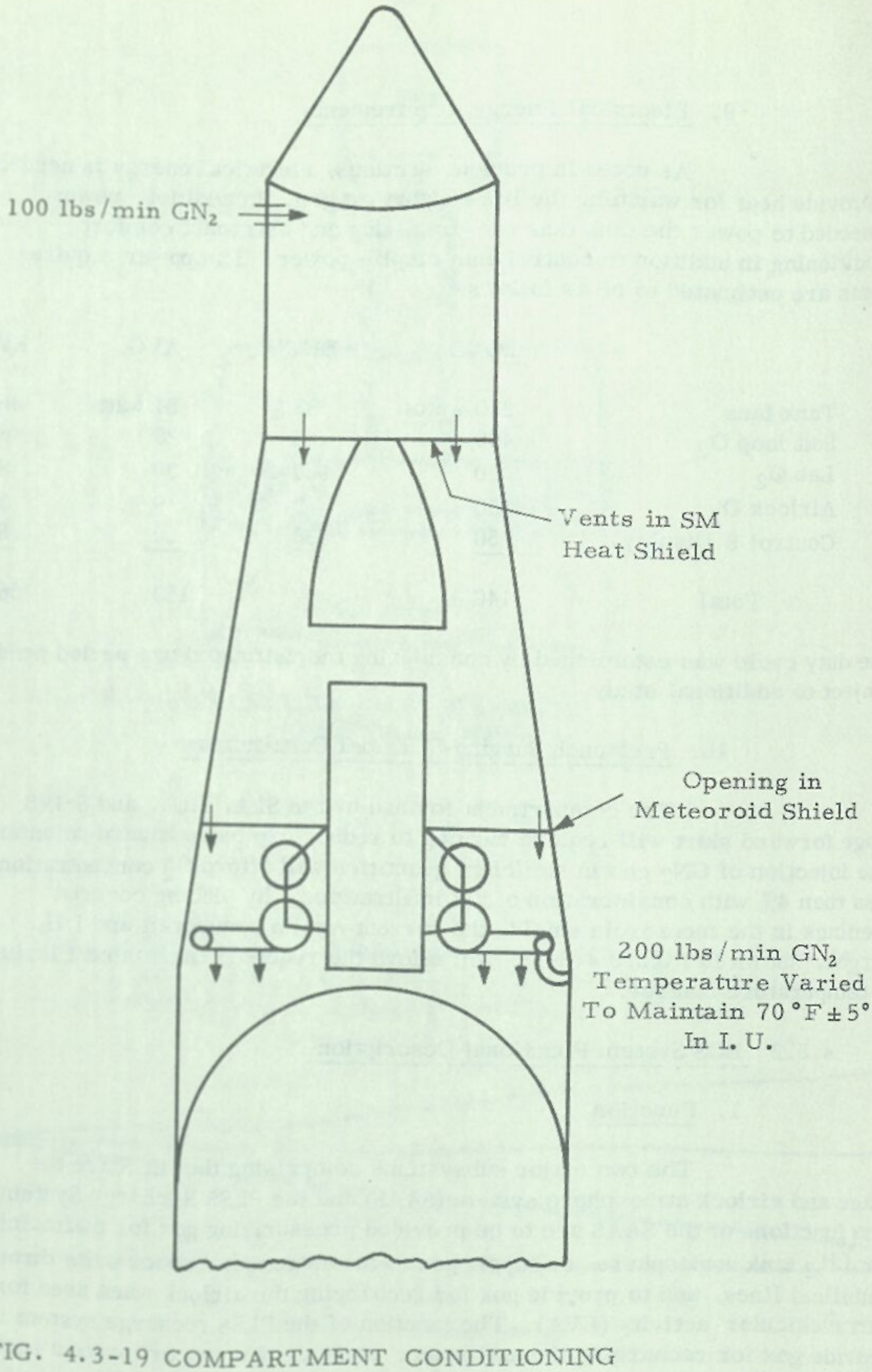


FIG. 4.3-19 COMPARTMENT CONDITIONING

2. Stage and Airlock Atmosphere System

Fill and Dump - Gox for the SAAS is stored in four 19.5 cubic foot, 3000 p.s.i.a., Inconel 718, high pressure oxygen storage spheres (1A), (1B), (1C) and (1D), Figure 4.3-20, which are located outside the airlock on the supporting structure. Fill is accomplished through the fill self-sealing quick disconnect coupling (15A), filter (17), and sphere fill check valves (20B) and (20C). Each relief valve (3A) and (3B) prevents the overpressurization of two spheres. Gox also flows through isolation check valves (20A) and (20D) to the isolation hand valve (5A) which is closed until after CSM turn around and docking. The isolation check valves prevent a loss of the entire SAAS gox supply in the event of a leakage in one of the spheres. Sphere dump valve (4A) is provided for the dumping of gox in the event of a launch abort. Since there is no requirement for system dump during boost or in orbit, the Sphere Dump Valve is manifolded back to the fill line, where the self-sealing quick disconnect coupling seals the line upon umbilical disconnect.

Pressure Test - The airlock is pressurized with gox on the ground through a self-sealing quick disconnect coupling and a pressure leak check is performed utilizing a ground test airlock relief valve which is coupled to a quick disconnect coupling attached to the outlet of the airlock relief valve (11A). After the pressure test has been performed, the ground test airlock relief valve will be removed and the airlock relief valve will maintain the airlock at 5.0 p.s.i.g. during the remainder of the mission.

Operation - The SSESM is pressurized initially prior to launch and, after docking has been completed, airlock operations are initiated by opening the CSM hatches and equalizing pressure across the airlock upper hatch. This is accomplished by utilizing the upper hatch equalization hand valve (12A) and upper hatch airlock pressure gage (P-4).

Airlock Depressurization - Once the airlock has been entered, it will be necessary to perform EVA before proceeding with the mission . To perform EVA, the airlock must be depressurized. After closing the upper hatch and donning and testing their space suits, the astronauts will depressurize the airlock by opening the airlock bleed hand valve (10). When the airlock internal pressure is sufficiently low, the side hatch may be opened and EVA performed.

Airlock and Stage Pressurization - After the airlock has been mated to the LH_2 tank forward bulkhead and the astronauts have reentered and sealed the airlock, pressurization of the assembly may begin. Pressurization is accomplished by opening the isolation hand valve (5A) which allows 3000 p.s.i.a. oxygen to pass through the tank flow control quick charge orifice (6B) to the tank quick charge hand valve (21E). High pressure oxygen also flows through the first stage regulator (8) where the pressure is reduced from 3000 p.s.i.a. to 100 p.s.i.a. The flow then continues through the airlock

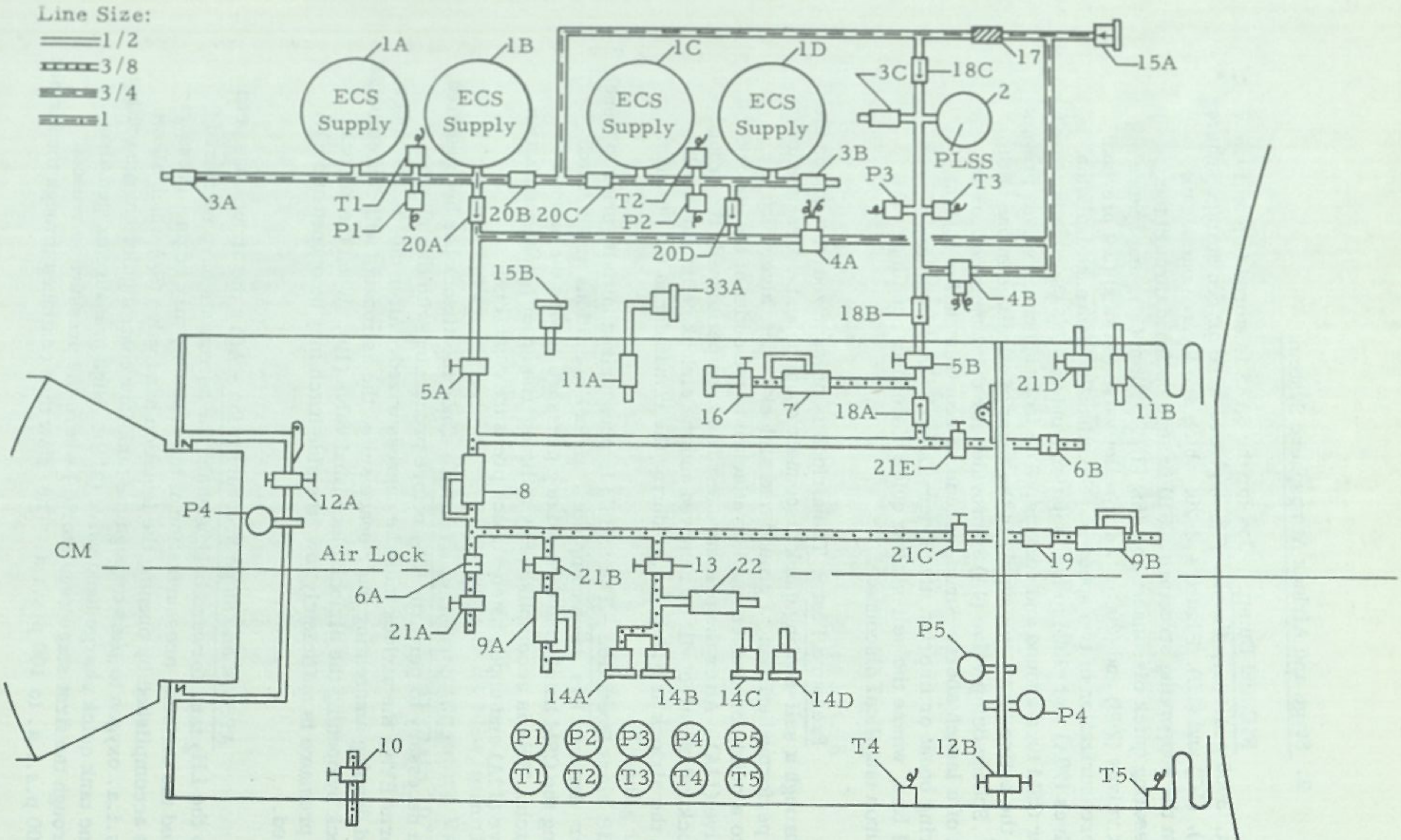


FIGURE 4.3-20. OXYGEN SUPPLY SYSTEM

flow control quick charge orifice (6A) to the airlock quick charge hand valve (21A), and also to the airlock second stage regulator hand shutoff valve (21B) and the tank second stage regulator hand shutoff valve (21C). Once the isolation hand valve is opened, pressurization may be accomplished by opening the tank quick charge hand valve and the airlock quick charge hand valve.

Airlock and Stage Pressure Control - When visual observation of the airlock and tank pressure gages confirms that the pressure is up to 5.0±0.2 p.s.i.a., the quick charge valves are closed and the tank and airlock second stage regulator hand shutoff valves are allowing 100 p.s.i.a. gox to flow into tank second stage regulator (9B) and airlock second stage regulator (9A), respectively. In these regulators, gox pressure is reduced from 100 p.s.i.a. to 5.0±0.2 p.s.i.a. Pressure relief capability is provided in both the airlock and tank by airlock relief valve (11A) and lower trunk relief hand valve (11B), respectively. A lower trunk bleed hand valve (21D) is provided as a backup to the lower trunk relief hand valve. The lower bulkhead tank pressure gage (P-5), lower bulkhead airlock pressure gage (P-4), and lower hatch equalization hand valve (12B) permits equalization of pressure across, and use of, the lower hatch.

Umbilical Extra Vehicular Activity - To provide a backup capability for EVA using the SAAS gox system, umbilical attach points have been provided. To utilize this capability, the isolation hand valve (5A) must be open to allow 3000 p.s.i.a. gox to flow through the first stage regulator (8), where the pressure is reduced to 100 p.s.i.a., to the manual O_2 metering valve (13). When umbilical EVA is required, the suit umbilical will be plugged into either suit umbilical self-sealing quick disconnect coupling (14A) or (14B), and suit exhaust self-sealing quick disconnect coupling (14C) or (14D) will be attached to the outlet port of the suit. The manual O_2 metering valve will be opened and adjusted to provide 3.5 p.s.i.g. in the suit. The low pressure suit umbilical relief valve (22) will maintain the umbilical line pressure at 3.5 p.s.i.g. Umbilical capability is designed to be used in checkout operations and as a backup to the PLSS.

3. Portable Life Support System Recharge System

Fill and Dump - Gox for the PLSS recharge system is stored in one 3-cubic foot, 3000 p.s.i.a., high pressure PLSS oxygen storage sphere (2) which is located outside the airlock on the supporting structure. Fill is accomplished through fill self-sealing quick disconnect coupling (15A), filter (17), and PLSS sphere fill check valve (18A). Relief valve (3C) prevents over-pressurization of the sphere or system. Gox also flows through isolation check valve (18B) to isolation hand valve (5B) which remains closed until after CSM turnaround and docking. The isolation check valve prevents loss of SAAS gox in the event of a PLSS vent valve malfunction or a system leak. PLSS sphere dump valve (4B) is provided for the dumping of gox in the event of a launch abort. Since there is no requirement for system dump during boost or in orbit, the sphere dump valve is manifolded back to the fill line where the self-sealing quick disconnect coupling seals the line upon umbilical disconnect.

Operation - When an astronaut's PLSS requires recharging, the cap will be removed from the PLSS recharge valve and coupling assembly (16) and the PLSS will be coupled to the outlet. To utilize the recharge system, the isolation hand valve must be opened. This permits 3000 p.s.i.a. gox to flow to the regulator (7) where the pressure is reduced to 900 p.s.i.a. The valve on the PLSS recharge valve and coupling assembly will be opened and 900 p.s.i.a. gox will flow into the PLSS. Upon completion of a PLSS recharge cycle, both the isolation hand valve and the valve on the PLSS recharge valve and coupling assembly must be closed. An intersystem check valve (18C) in the interconnecting line allows gox flow from the SAAS system to charge the astronaut's PLSS, if necessary; but prevents flow out of the PLSS recharge system when gox in the SAAS spheres is used.

4. Pressure Display

A pressure display panel containing pressure and temperature gages is provided in the airlock. The gages are as follows:

No.	Direct Reading	Remote Reading
P1		From transducer on spheres 1A & 1B
P2		From transducer on spheres 1C & 1D
P3		From transducer on PLSS sphere
P4	Integral in airlock	
P5		From tank
T1		From transducer on spheres 1A & 1B
T2		From transducer on spheres 1C & 1D
T3		From transducer on PLSS sphere
T4	Integral in airlock	
T5		From tank

Gages P1, P2, T1, and T2 are used to compute total amount of SAAS gox aboard. Gages P3 and T3 are used to compute total amount of PLSS gox aboard. Gages P4, P5, T4, and T5 are used for monitoring environmental conditions.

5. CO_2 Control

Several methods are available to monitor leakage or CO_2 levels. Flowmeters (19) in the gox supply lines would give an indication leakage. Calculating the amount of gox onboard from the available pressure and temperature displays and recording the amounts will give an indication of the gox loss. CO_2 partial pressure sensors will also give an indication of the CO_2 concentration. S-IVB LH_2 tank leakage will be minimized by plugging the tank outlets. The leakage will then be established by adjusting handvalve (210).

TABLE 4.3-1

ECS SYSTEM COMPONENT REQUIREMENT

ITEM NO.	NOMENCLATURE	SIZE
1A	Bottle (ECS Supply)	19.5 Ft^3
1B	Bottle (ECS Supply)	19.5 Ft^3
1C	Bottle (ECS Supply)	19.5 Ft^3
1D	Bottle (ECS Supply)	19.5 Ft^3
2	Bottle, gox, PLSS	3.0 Ft^3
3A	Valve, Vent	0.5" Tube Size
3B	Valve, Vent	0.5" Tube Size
3C	Valve, Vent	0.5" Tube Size
4A	Valve, Solenoid	0.75" Tube Size
4B	Valve, Solenoid	0.75" Tube Size
5A	Valve, Hand Op., ECS Shutoff	0.5" Tube Size
5B	Valve, Hand Op., PLSS Shutoff	0.5" Tube Size
6A	Orifice, Airlock Quick Fill	0.38 Tube Size
6B	Orifice, Workshop Quick Fill	0.38 Tube Size
7	Regulator, PLSS Supply	0.25 Tube Size
8	Regulator, 1st Stage ECS Supply	0.25 Tube Size
9A	Regulator, Airlock Pressure	
9B	Regulator, Workshop Pressure	
10	Valve, Airlock Vent	1.0"
11A	Valve, Relief	
11B	Valve, Relief	
12A	Valve, Equalization, Airlock Fwd	
12B	Valve, Equalization, Airlock Aft	
13	Valve, Metering, Manual O_2	
14A, B, C, D,	Disconnect Couplings	
15AB	Disconnect Couplings	
16	Valve, Backpack Shutoff	
17	Filter, Supply Fill	0.75" Tube Size
18A	Valve, Check	0.50" Tube Size
18B	Valve, Check	0.50" Tube Size
18C	Valve, Check	0.50" Tube Size
19	Flowmeter, Workshop Supply	
20A	Valve, Check	0.75" Tube Size
20B	Valve, Check	0.75" Tube Size
20C	Valve, Check	0.75" Tube Size
20D	Valve, Check	0.75" Tube Size
21A	Valve, Hand Op., Airlock Quick Fill	0.38" Tube Size

TABLE 4.3-1 (Cont'd)

ITEM NO.	NOMENCLATURE	SIZE
21B	Valve, Hand Op., Airlock Regulator S.O.	0.38" Tube Size
21C	Valve, Hand Op., Airlock Workshop S.O.	0.38" Tube Size
21D	Valve, Hand Op., Workshop Vent	0.38" Tube Size
21E	Valve, Hand Op., Workshop Quick Fill	0.38" Tube Size
22	Valve, Suit Relief	0.38" Tube Size
P-1, 2, 3	Transducer, Pressure	—— ——
P-4, 5, 6	Gage, Pressure	—— ——
T-1, 2, 3, 4, 5	Pickup, Temperature	—— ——

4.4 ELECTRICAL SYSTEM

The electrical system is composed of power sources, a lighting system, control panel, portable display panel, and cabling (Figures 4.4-1A and 4.4-1B). These elements are described in the following paragraphs along with the electrical portion of the S-IVB stage passivation subsystem, measurement and instrumentation subsystem, and electrical interfaces. The electrical support equipment is also covered in the subsequent paragraphs.

4.4.1 Power Sources

Power sources consist of twenty-one 28-volt batteries, of silver-zinc oxide type, and rated at 500 ampere hours each. Battery dimensions are approximately 8.5 by 10.5 by 23 inches, and weigh approximately 140 pounds each. Twenty of the batteries are arranged in banks of ten each, Figure 4.4-2, which will provide 5,000 ampere hours of power per bank. The two banks will be used on a predetermined sequence established to utilize the available power most efficiently and reliably. Present estimated load on the system is approximately 7,000 ampere hours, exclusive of experiment power, as shown in the load profile of Figure 4.4-3. The twenty-first battery is provided as a source of emergency power and as a reference voltage source. Two low voltage detectors, one on each battery bank bus, are used to detect low voltage of these banks. Emergency power will automatically be applied to the emergency bus when low voltage is detected on either battery bank bus. Capability of manually switching to emergency power is also provided.

4.4.2 Lighting System

A lighting system is required to provide lighting associated with the SSESM. Expected lighting requirements consist of acquisition lights (40 watts), airlock lights (40 watts), and LH_2 tank lights (400 watts) which will be divided into two loads of approximately 200 watts each.

The tank and airlock lights will consist of 28 Vdc fluorescent fixtures with an overall efficiency, including power converters, of approximately 40 lumens per watt. Each fixture will weigh approximately 2 pounds, have a volume of approximately 50 cubic inches, and be sized between 10 to 40 watts, depending on the final lighting patterns required.

All lights are connected to the power buses through circuit breakers. These circuit breakers can be used to turn the lights on and off. Sensing of low voltage by a voltage detector will cause automatic transfer of one airlock light and one LH_2 tank light to the emergency bus. This provides the astronauts with emergency lighting for return to airlock in case of a power failure.

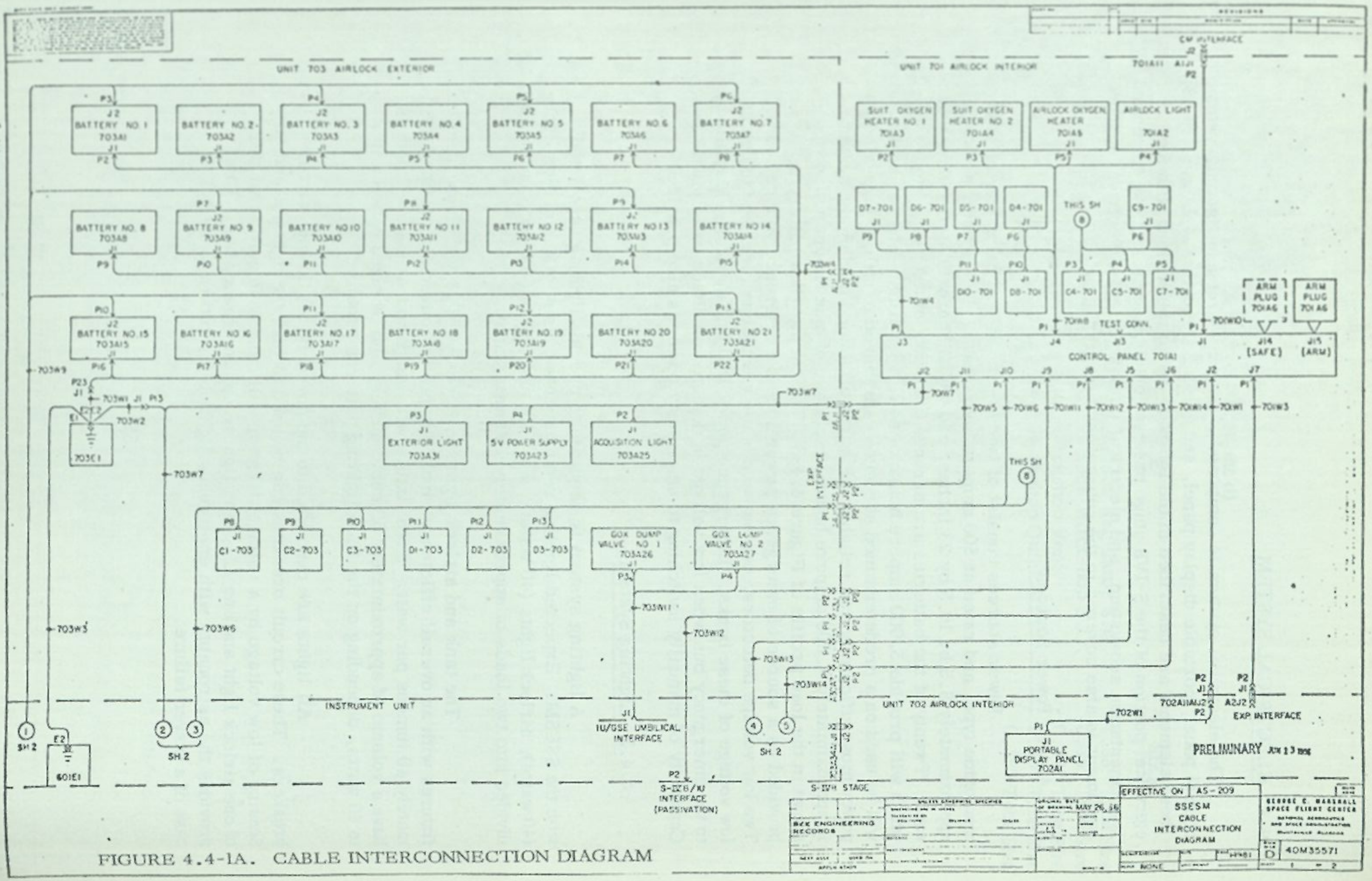


FIGURE 4.4-1A. CABLE INTERCONNECTION DIAGRAM

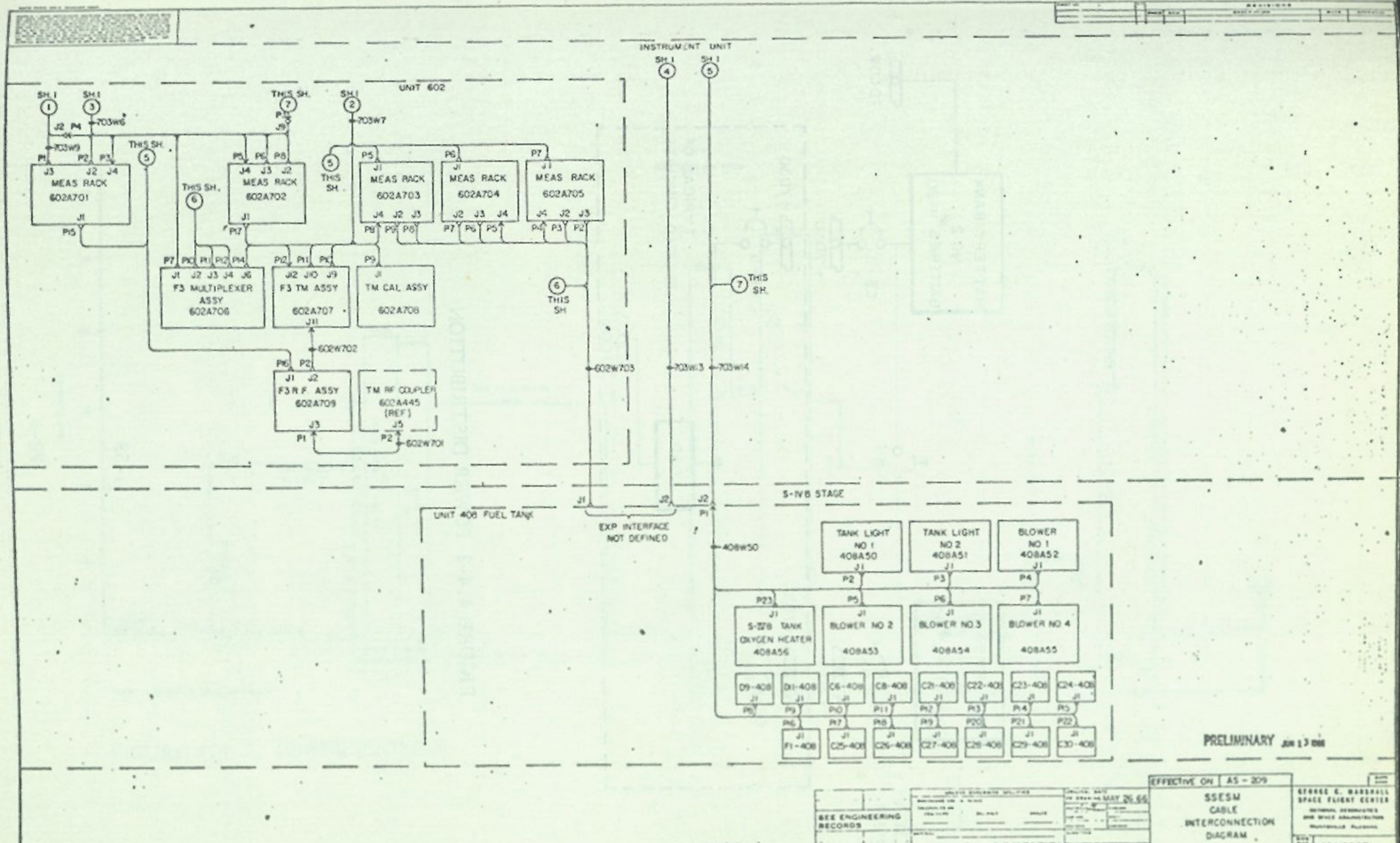


FIGURE 4.4-1B. CABLE INTERCONNECTION DIAGRAM (CONT'D.)

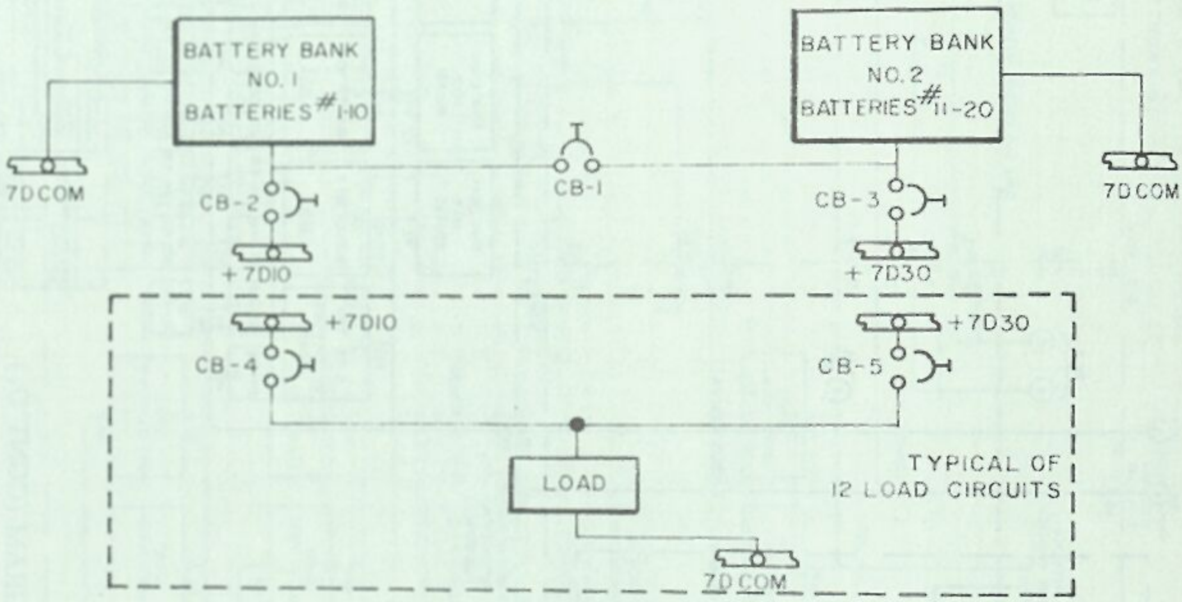


FIGURE 4.4-2 POWER DISTRIBUTION

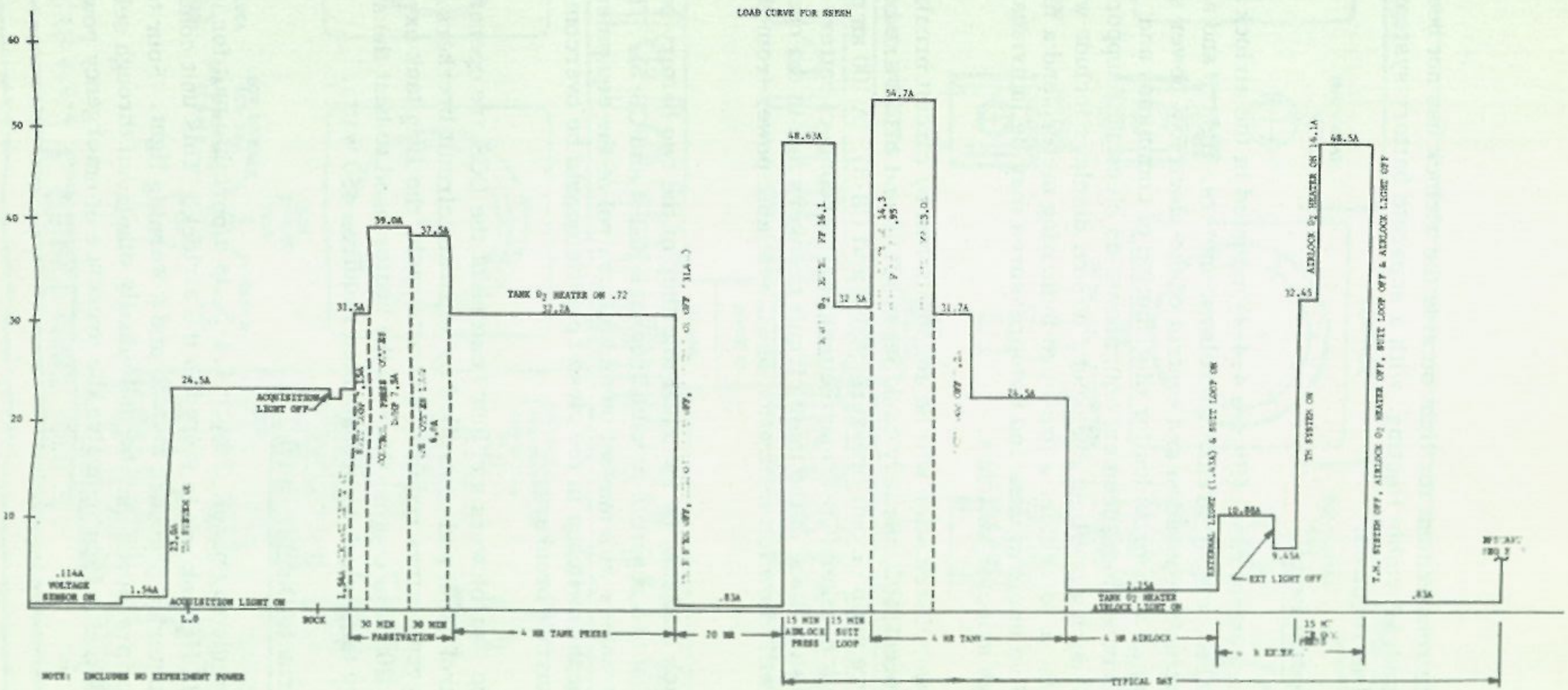


FIGURE 4.4-3. POWER PROFILE

The requirement for light outside the airlock has not been completely defined. Portable lighting, with a separate battery system, will be included as required.

4.4.3 Control Panel

The control panel (Figure 4.4-4) mounted in the airlock unit, consists of switches, circuit breakers, relays, meters, lights, and a distribution system for operation and control of the electrical power subsystem. This panel also provides for distribution of commands and measurements between measurement equipment and electrical support equipment (ESE) during preflight checkout. Visual displays include warning lights, ammeters, a voltmeter, a pressure indicating meter, and a temperature meter. Five temperatures and five pressures may be individually displayed through selector switches.

Power distribution will be accomplished by circuit breakers as shown in Figure 4.4-2. Battery bank buses +7D10 and +7D30 are controlled by two 100 ampere main circuit breakers (CB-2 and CB-3). A 100 ampere battery tie circuit breaker (CB-1) may be used to interconnect battery banks No. 1 and No. 2 should one of the main circuit breakers fail in the open position. This will allow full utilization of all available power from the batteries.

Each load may be switched to either of the two battery bank buses by means of two separate circuit breakers (CB-4 and CB-5). This provides the astronauts with manual on and off control of the equipment. Failure of one of the switches in the closed position could be overcome by using the main circuit breakers.

The four blowers and four heaters of the ECS are operated through the control panel and controlled by separate circuit breakers. The two suit oxygen heaters require 225 watts each, the LH_2 tank oxygen heater requires 20 watts continuous, and the heater used to heat the airlock oxygen each time the airlock is pressurized requires 450 watts.

4.4.4 Portable Display Panel

The display panel, Figure 4.4-5, is a portable unit for use inside the S-IVB LH_2 tank and is stored in the airlock. This unit contains a temperature meter, a pressure meter, and a warning light. Four temperatures and four pressures may be individually displayed through selector switches. The warning light indicates the presence of emergency power.

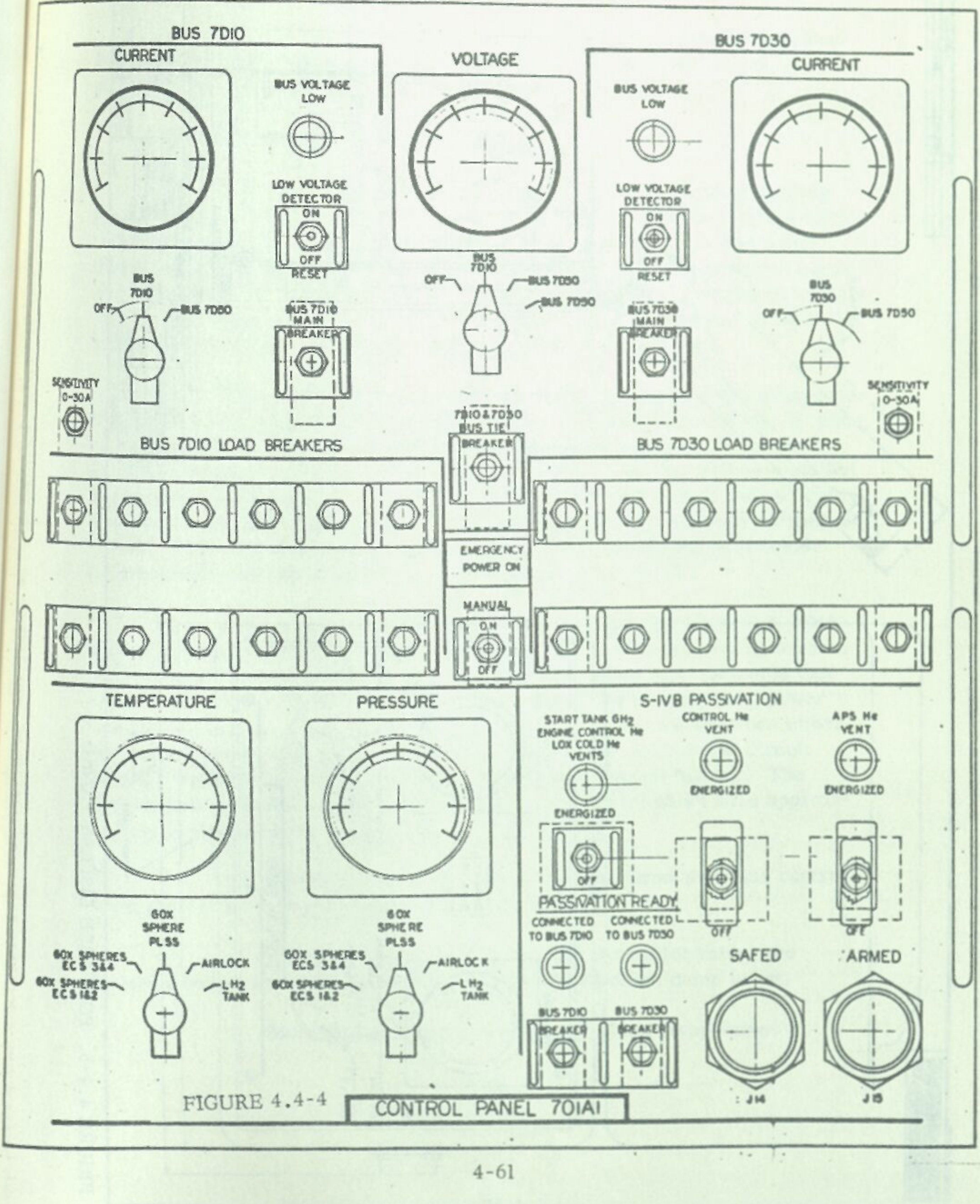


FIGURE 4.4-4 CONTROL PANEL 701A1

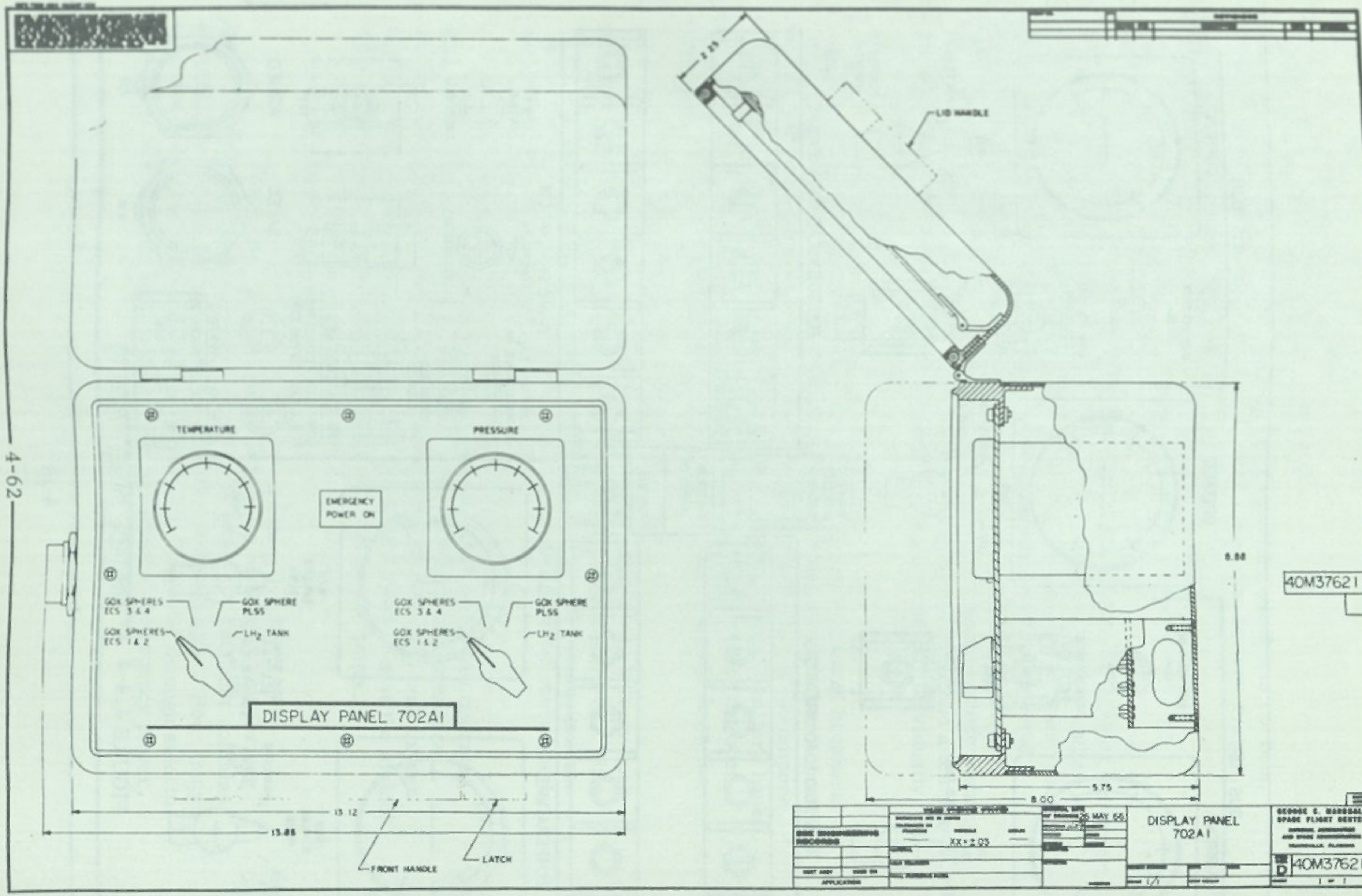


FIGURE 4.4-5. PORTABLE DISPLAY PANEL

4.4.5 Cabling

Cabling consists of that necessary for interconnecting power sources, the control panel, and the display panel with lighting, blowers, measuring racks, sensors, and telemetry.

4.4.6 S-IVB Stage Passivation Subsystem

The passivation electrical subsystem consists of an arming plug, manual control switches, and indicator lights which are located on the airlock control panel. In addition, the system consists of a cable which interfaces with various valves on the S-IVB stage. Existing stage circuitry and sequencing will be used to vent residual lox and LH_2 from the propellant tanks. The stage destruct system will be passivated by turning the destruct system power off through an RF command from range control.

Premature activation of the S-IVB stage passivation subsystem is prevented by removing an arming plug from the arm position which opens circuits to all the valves. The arming plug will be installed in the arm position during prelaunch checkout as required. The plug will be installed in the safe position prior to liftoff. The astronaut will install the arming plug in the arm position after S-IVB burn when the stage is ready for passivation. The receptacle that arms the passivation subsystem will double as prelaunch test points when the arming plug is removed.

Two indicator lights are provided on the diaplay panel. One light indicates that the arming plug is installed and that the circuit breaker supplying power from bus +7D10 is closed. The other light indicates that the arming plug is installed and that the circuit breaker supplying power from bus +7D30 is closed. The stage is ready for passivation when either indicator light is illuminated. The redundant power source and circuit breaker will be used only in the event of primary circuit failure. The passivation system controlled from the airlock control panel uses approximately 14 amperes when all valves are energized.

The following three four-pole, double-throw switches control components in the stage passivation subsystem:

1. Switch S1 controls the start tank vent pilot valve, the helium emergency control solenoid, and the cold helium dump valve.

2. Switch S2 controls the ambient helium dump valve.

3. Switch S3 controls both APS No. 1 and APS No. 2 fuel tank and oxidizer tank helium vent valves.

The operational procedure requires that valves controlled by switch S1 be activated before the remaining valves. To ensure adherence to the operational procedure, switch covers must be removed from switches S2 and S3 before they are actuated.

During checkout, each bottle to be dumped by the passivation circuitry will be pressurized with pneumatics. The bottle dump valves will then be energized from the airlock control panel. Observation of the porting of gases from the bottle vents, and telemetry monitoring of the bottle pressure decay, will be used to verify integrity of the passivation circuitry. The S-IVB/ESE circuits in parallel with the passivation circuits are to be isolated during functional testing of the passivation system.

4.4.7 Measurement and Instrumentation Subsystem

The F3 RF assembly is supplied 28 Vdc power by redundant load buses through two circuit breakers. The circuit breakers will be used to alternate load buses furnishing power to the RF assembly. The circuit breaker supplying power from the +7D10 battery bank bus is closed on the ground prior to liftoff. During checkout, power to the RF assembly is inhibited by ESE control during periods of RF silence.

The F3 TM assembly, the 5 Vdc power supply, and two measuring racks are supplied power through another set of circuit breakers and through an inhibit relay. Power is supplied to these components in the same manner as that supplied to the F3 RF assembly.

The remaining three measuring racks are supplied power through another set of circuit breakers. Power is supplied to these racks by two circuit breakers which alternate the battery bank buses that supply power. There are no inhibit relays in the power supply circuit to the three measuring racks. The three measuring racks will be turned on as required for checkout and after the system is in orbit.

The 5 Vdc power supply furnishes 5 Vdc to the F3 TM assembly, all measuring racks, and 11 pressure measurements. Nine measurements, specified in the instrumentation program and components list, are capable of being switched by ESE controlled relays from telemetry to ESE via hardwire for checkout.

Certain critical measurements are paralleled to telemetry and to the control panel. Measurements routed to the control panel are monitored visually.

4.4.8 Interface Requirements

Marshall Space Flight Center will control the following electrical interfaces in addition to the Saturn IB launch vehicle interfaces:

1. Airlock to spacecraft.

2. Airlock to S-IVB stage (passivation).

3. Airlock to IU (RF output to IU antennas).

4. Airlock to ESE (utilize connection No. 14 of IU/ESE interface control document).

4.4.9 Electrical Support Equipment

The Electrical Support Equipment (ESE) will provide means of controlling and monitoring the on-board equipment as required during the checkout and launch up to the time of umbilical separation. Insofar as possible the SSESM ESE will be separate from the existing ESE and facilities, to minimize interfacing problems and the resulting impact on the existing contractor supplied ESE. System documentation will be an inhouse effort.

The following is a list of the required new hardware and the source of procurement:

1. SSESM Control and Monitor Panel - This panel will contain the necessary lights, meters, switches, etc. as required. This panel will be designed, documented, and procured as an inhouse effort.

2. SSESM Distributor - This distributor will contain the relay logic and patching necessary for the system. The basic hardware already exists as surplus from other programs and will only require that the necessary documentation and patching be accomplished inhouse.

3. 15 Amp Power Supply in AGCS - At the present time there is an existing spare module available in the Saturn IB ESE at KSC. It is proposed to utilize this module. This will require an interface change with the resulting change in the documentation supplied by G.E..

4. DEE-6 Recording - The existing Saturn IB system has the capacity to accept the additional requirements. This will require an interface change with the resulting change in the documentation supplied by G.E..

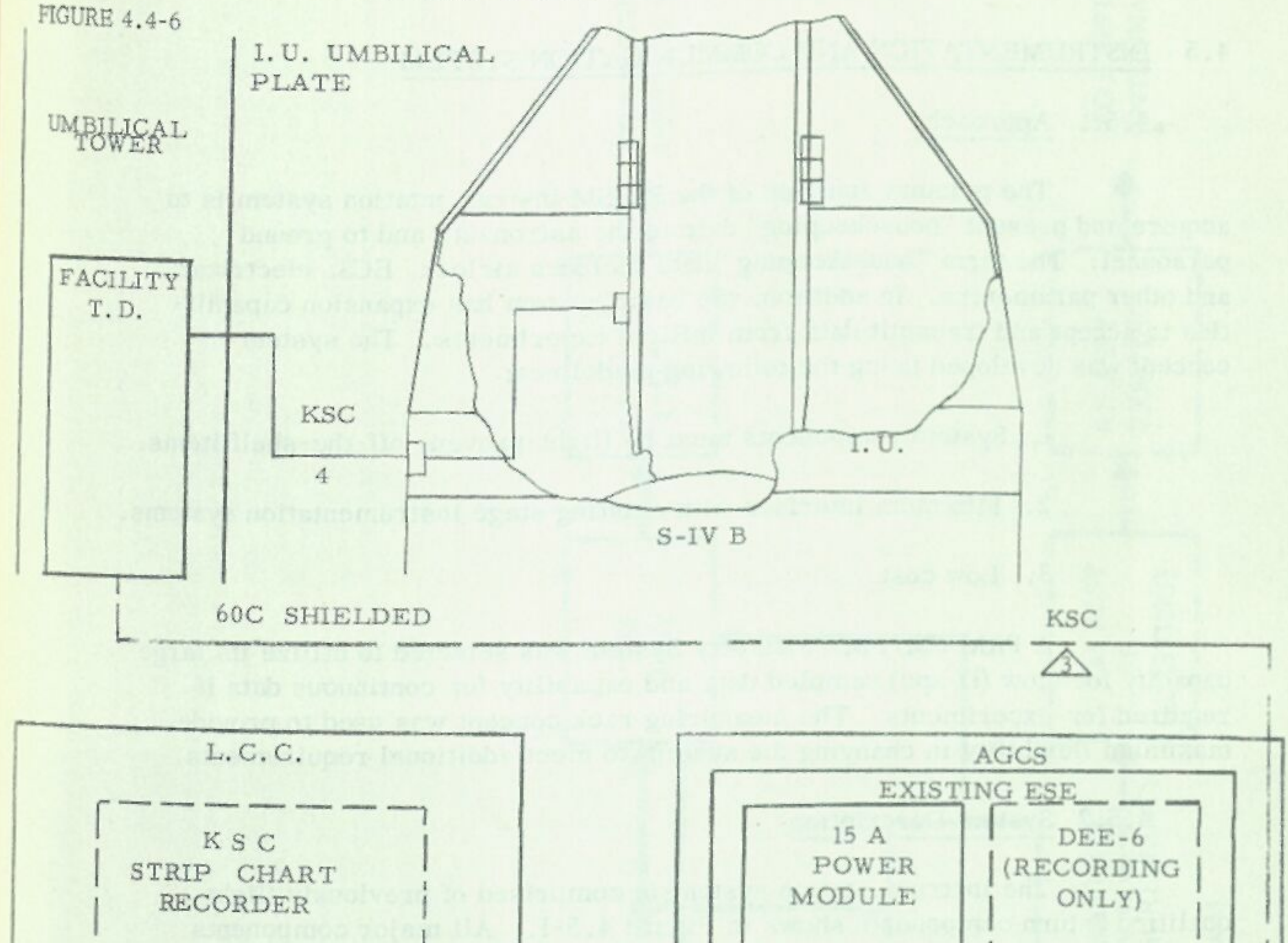
5. KSC Strip Chart Recording - KSC must make available the necessary recording facilities. Assuming that the KSC facility has the small number of spares that will be required then this will only mean changing the interface control document (Figure 4.4-6).

6. Interior Cables - A total of 6 interior type cables will be required and these will be supplied inhouse. It is planned that cables of the correct type and length can be found as surplus for a portion of these cables.

7. Facility Cables - One 60c cable from the LCC to the AGCS must be made available by KSC. One 60c cable from the AGCS terminal distributor up the tower to the I.U. umbilical level must be made available by KSC. One 60c cable across the swing arm to the I.U. umbilical must be built. These cables have been supplied by KSC in the past and it is anticipated that KSC would supply this.

It is planned that only one set of the above equipment will be built. This set will be supplied in time for checkout of the SSESM and that the ESE will then be shipped to KSC for the checkout or launch. The necessary facility power, cables, and recording equipment will be supplied inhouse.

FIGURE 4.4-6



1 Control panel and distr. will be an MSFC in-house design and documentation effort

2 These cables will be supplied by MSFC

3 These cables will be made available by KSC

4 These cables must be built by KSC

4.5 INSTRUMENTATION AND COMMUNICATION SYSTEM

4.5.1 Approach

The primary function of the SSESM instrumentation system is to acquire and present "housekeeping" date to the astronauts and to ground personnel. The term "housekeeping" data includes airlock, ECS, electrical, and other parameters. In addition, the basic system has expansion capabilities to accept and transmit data from inflight experiments. The system concept was developed using the following guidelines:

1. System components must be flight-proven, off-the-shelf items.

2. Minimum interface with existing stage instrumentation systems.

3. Low cost.

A PAM FM/FM Telemetry System was selected to utilize its large capacity for slow (12 sps) sampled data and capability for continuous data if required for experiments. The measuring rack concept was used to provide maximum flexibility in changing the system to meet additional requirements.

4.5.2 System Description

The instrumentation system is comprised of previously flight qualified Saturn components shown in Figure 4.5-1. All major components of the instrumentation system are located on available IU mountings. The IU antennas will be utilized if the SLA panels are folded back so that they do not interfere with RF transmission; otherwise, separate SSESM antennas will have to be mounted on the forward end of the airlock.

The purposes of the on-board instrumentation are to provide:

1. On-board displays to the astronaut to determine SSESM condition.

2. Data regarding the operation of on-board systems and experiments for real-time and postflight analysis.

3. Data prior to launch which is used for checkout of the SSESM systems.

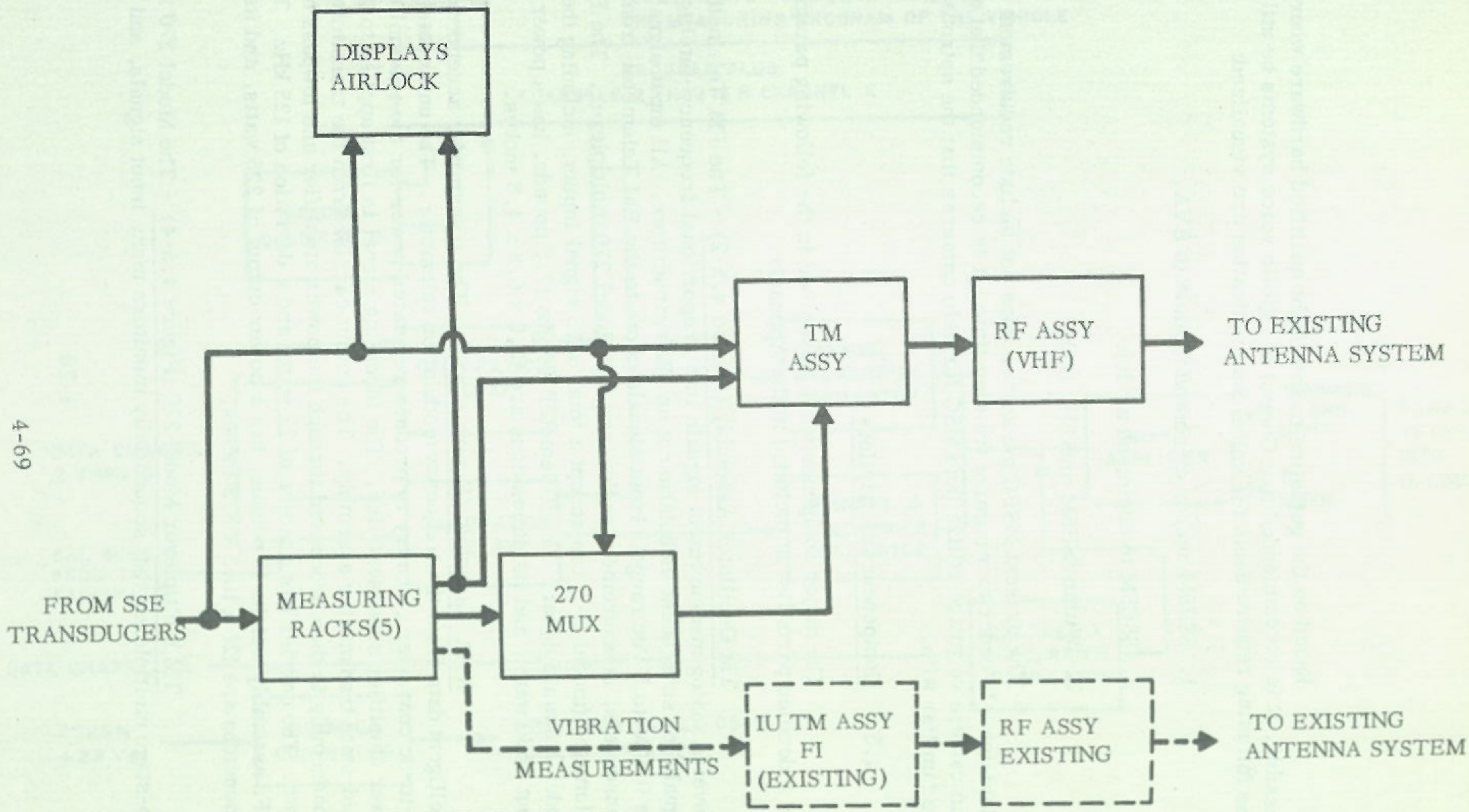


FIGURE 4.5-1. INSTRUMENTATION SYSTEM

Based on the guideline to use flight qualified hardware wherever possible, it is recommended that Gemini or Apollo voice systems be utilized. The following requirements for voice communication are visualized:

1. SSESM and/or command module to EVA.

2. SSESM to command module.

3. Within SSESM and LH_2 tank.

The present MSFC guidelines does not include requirement for television. In order to minimize training time, it is recommended that any film camera coverage utilize (and MSC furnish) cameras that the astronauts are familiar with.

4.5.3 Component Descriptions

The major components are described in the following paragraphs and a detailed parts list is included in the Appendix.

TM Oscillator Assembly (Figure 4.5-2) - The FM Telemeter converts analog measurement signals into proportional frequency-intelligent signals for subsequent modulation of an FM transmitter. All input signals are 0 Vdc to 5 Vdc range. Input signals come to the FM Telemeter from transducers, measurement racks, and the Model 270 multiplexer. The FM Telemeter (model C-1) can accept a total of 15 signal inputs, including the Model 270 multiplexer. The Telemeter weighs 17.5 pounds, has a power input of 30 watts, and its dimensions are 12.0 x 6.8 x 4.8 inches.

R.F. Assembly (Figure 4.5-3) - The transmitter accepts frequency-intelligent data from the frequency modulation telemeter. The unit contains dc-to-dc converter circuitry to produce regulated power for the transmitter and power amplifier subassemblies. The incoming signal is first applied to the solid-state transmitter assembly. The output carrier from the transmitter subassembly is then processed through the power amplifier and lowpass output filter. The carrier has a power of 22 watts and a deviation of 125 kHz. The R.F. assembly weighs 15 pounds, has a power output of 225 watts, and its dimensions are 128 x 10.5 x 3.5 inches.

TM Multiplexer Model 270 (Figure 4.5-4) - The Model 270 is a two-stage multiplexer that sequentially monitors many input signals, and

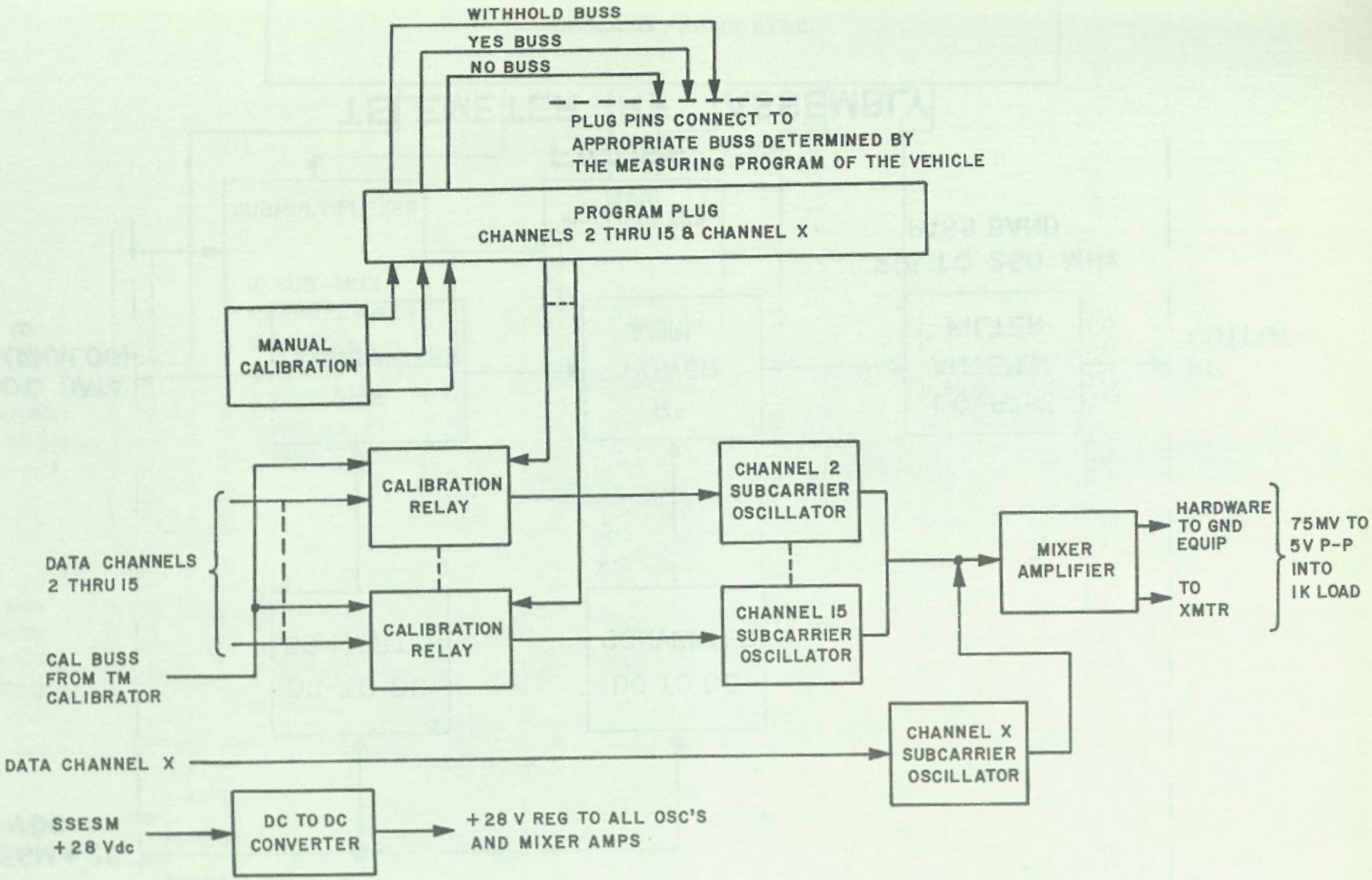


FIG. 4.5-2 TM OSCILLATOR ASSEMBLY C-1

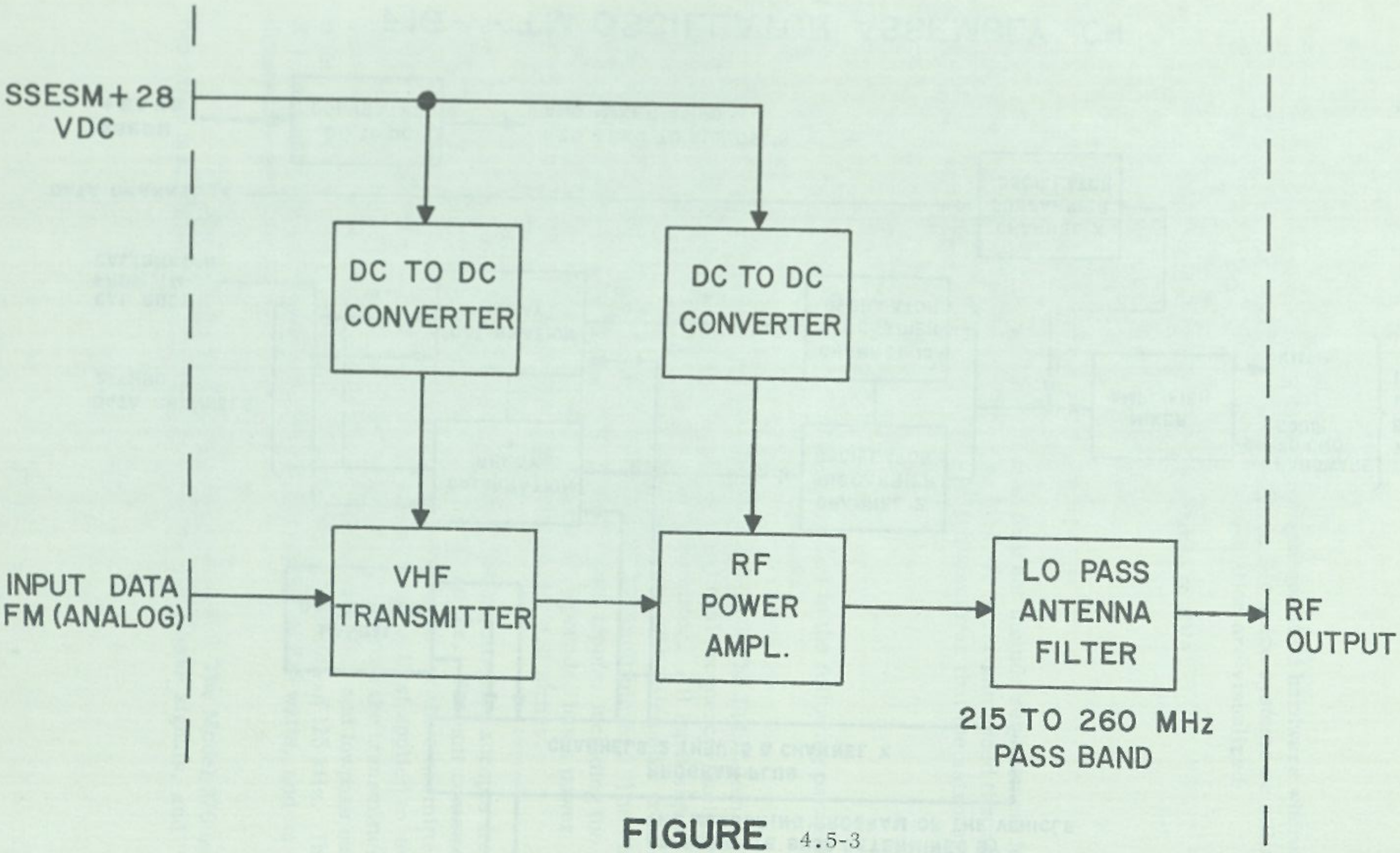


FIGURE 4.5-3
TELEMETER RF ASSEMBLY

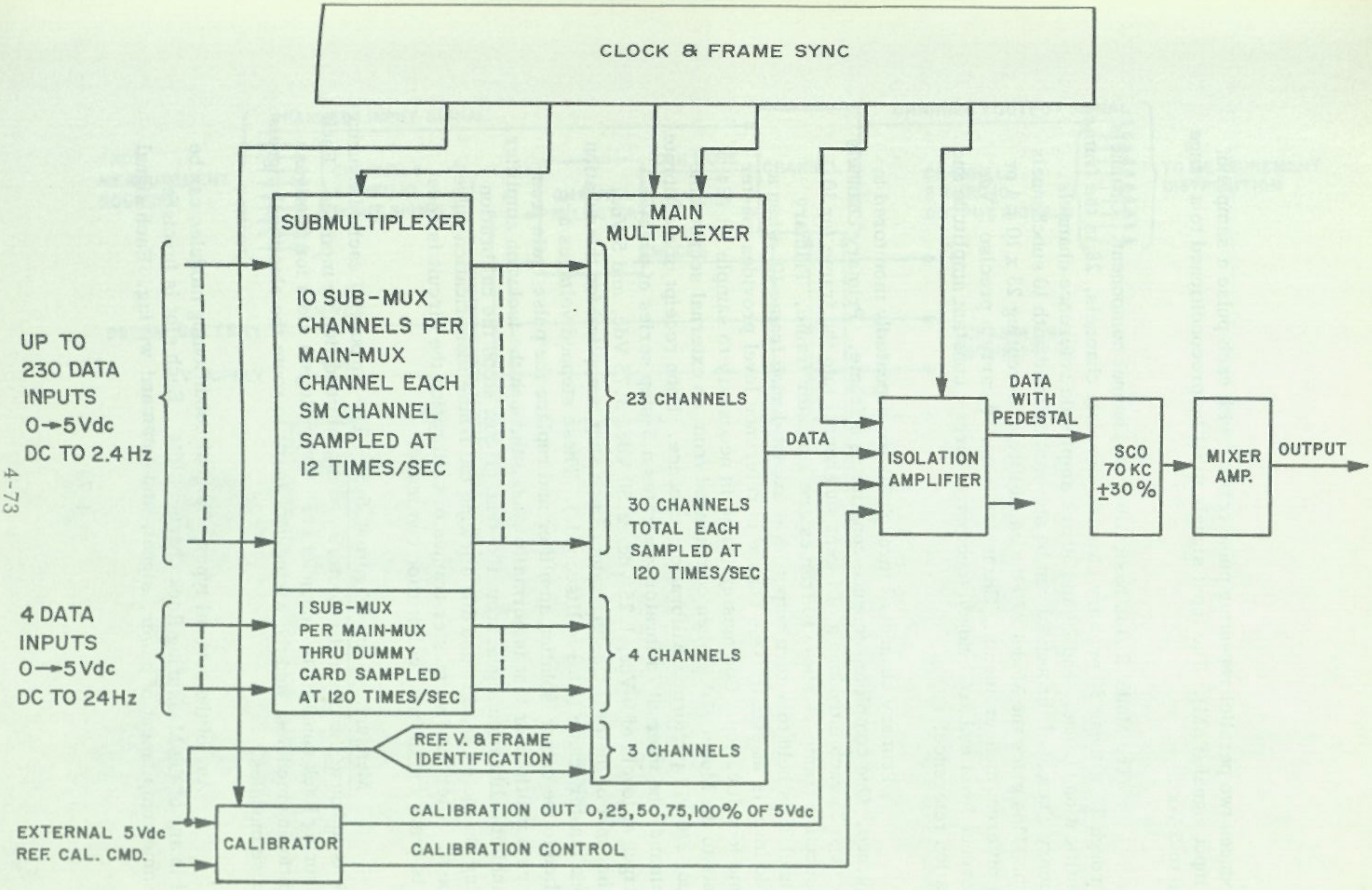


FIG. 4.5-4 TM MULTIPLEXER MOD. 270

produces two parallel repeating pulse trains, with each pulse a sample of an input signal (PAM). The input signals must be preconditioned to a range of 0 to 5 Vdc.

The Model 270 is basically a 30-channel component. Channels 1 through 27 of these 30 primary channels are data channels, 28 is the frame identification channel, and 29 and 30 are amplitude reference channels. Primary channels 1 through 23 can be sub-multiplexed with 10 subchannels each. This gives the Model 270 the capability of accepting 23 x 10 + 4 or 234 different measurements. Channels 29 and 30 carry a precise 5 Vdc reference level and are bridged together to form a constant amplitude and location reference.

Primary channels 1 through 30 are repeatedly monitored in sequence. One complete sequence constitutes a frame. Primary channels 1 through 23 each introduce a different subchannel into the frame for 10 consecutive frames. These 10 frames are a master frame. Primary channel 28 is held to a zero output level except during frame 10, when a 5 Vdc level is inserted. This change in reference level provides master frame identification. One master frame is necessary to sample all 234 inputs to the Model 270. When commanded from an external source, the Model 270 will perform a calibration sequence. Upon receipt of a calibrate command, the internal calibrator generates a 5-step series of precise voltages at levels of 0 Vdc, 1.25 Vdc, 2.50 Vdc, 3.75 Vdc, and 5 Vdc. At the start of the next master frame, each step is applied for the duration of one master frame (83.3 milliseconds). These stepped voltages are applied to the output isolation amplifier and replace the pulse train from the main multiplexer that is normally routed through the isolation amplifier. An inhibit calibration signal from the sync circuit stops the calibration during channels 28, 29, and 30, allowing the frame identification pulses to pass. After the five master frames of calibration the circuit is reset and is ready for the next calibration command.

Measuring Rack (Figure 4.5-5) - The purpose of each Measuring Rack is to house the channel selectors and signal conditioning modules. Each Measuring Rack contains two channel selectors, plug-in slots for 20 signal conditioning modules, and the wiring necessary to route the electrical signals to these modules.

Any of the several types of signal conditioning modules can be used in any of the Measuring Rack channel slots. Each slot is furnished the same complement of power, signal, and command wiring. Each signal

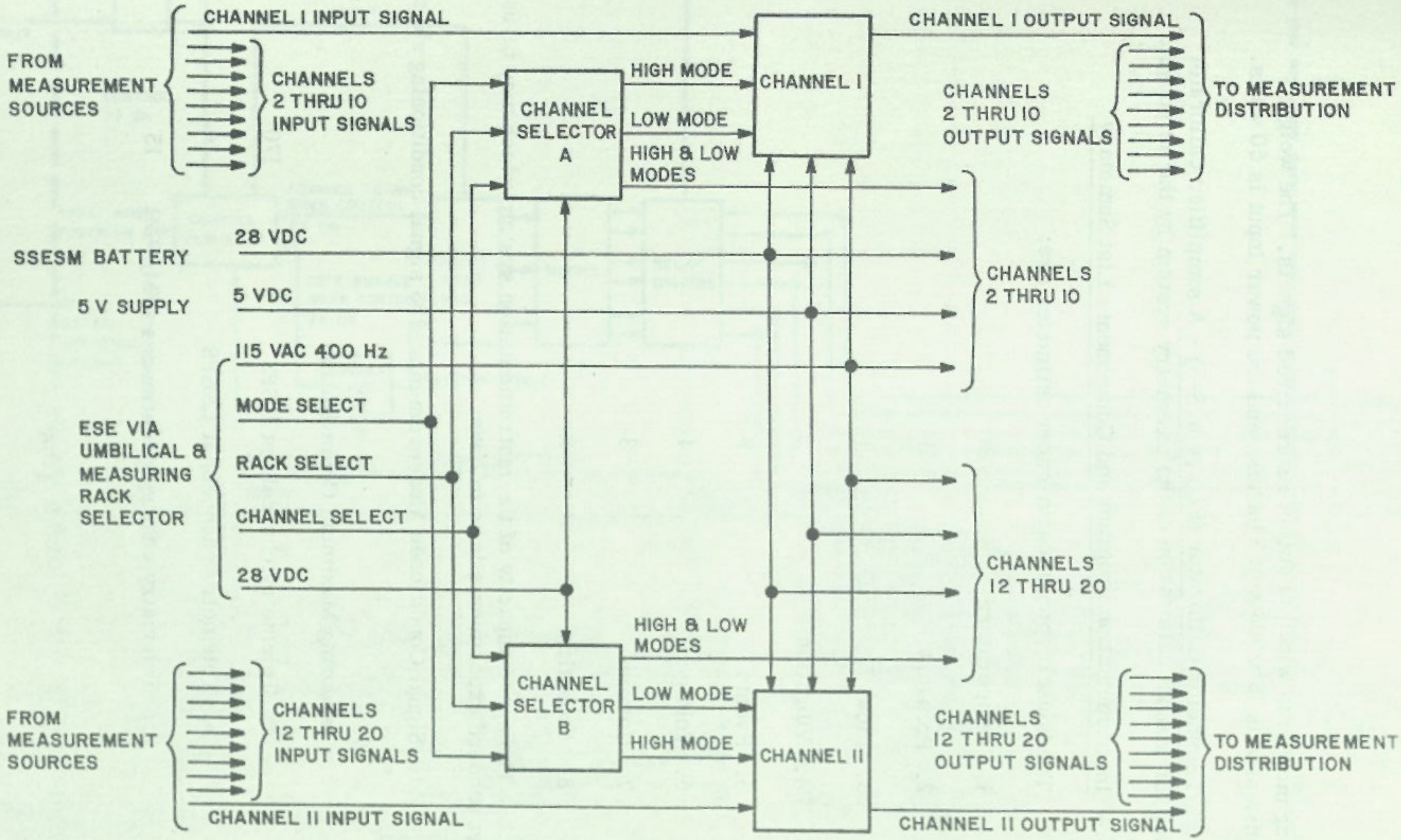


FIGURE 4.5-5 MEASURING RACK FUNCTIONAL DIAGRAM

conditioning module is wired to accept its required signals. The weight is 21 pounds, size is 13 x 9.8 x 6.5 inches, and the power input is 60 watts.

Telemeter Calibrator (Figure 4.5-6) - A simplified calibrator will be used for manual calibration of the telemetry system by the astronaut.

4.5.4 Instrumentation Program and Components List Summary

The housekeeping measurement estimates are:

1.	Temperature	30
2.	Pressure	11
3.	Flow	1
4.	Vibration	5
5.	Events	9
6.	Voltage	4
7.	Current	3
8.	Acoustics	2

The spare capacity of the instrumentation system which may be used to support onboard experiments is as follows:

1. Signal Conditioned Measurements - 55 signal conditioning slots available.

2. Assuming Maximum Commutation

a.	Telemetry channels at 12SPS	170
b.	Telemetry channels at 120SPS	4
c.	Telemetry channels continuous (FM/FM)	15

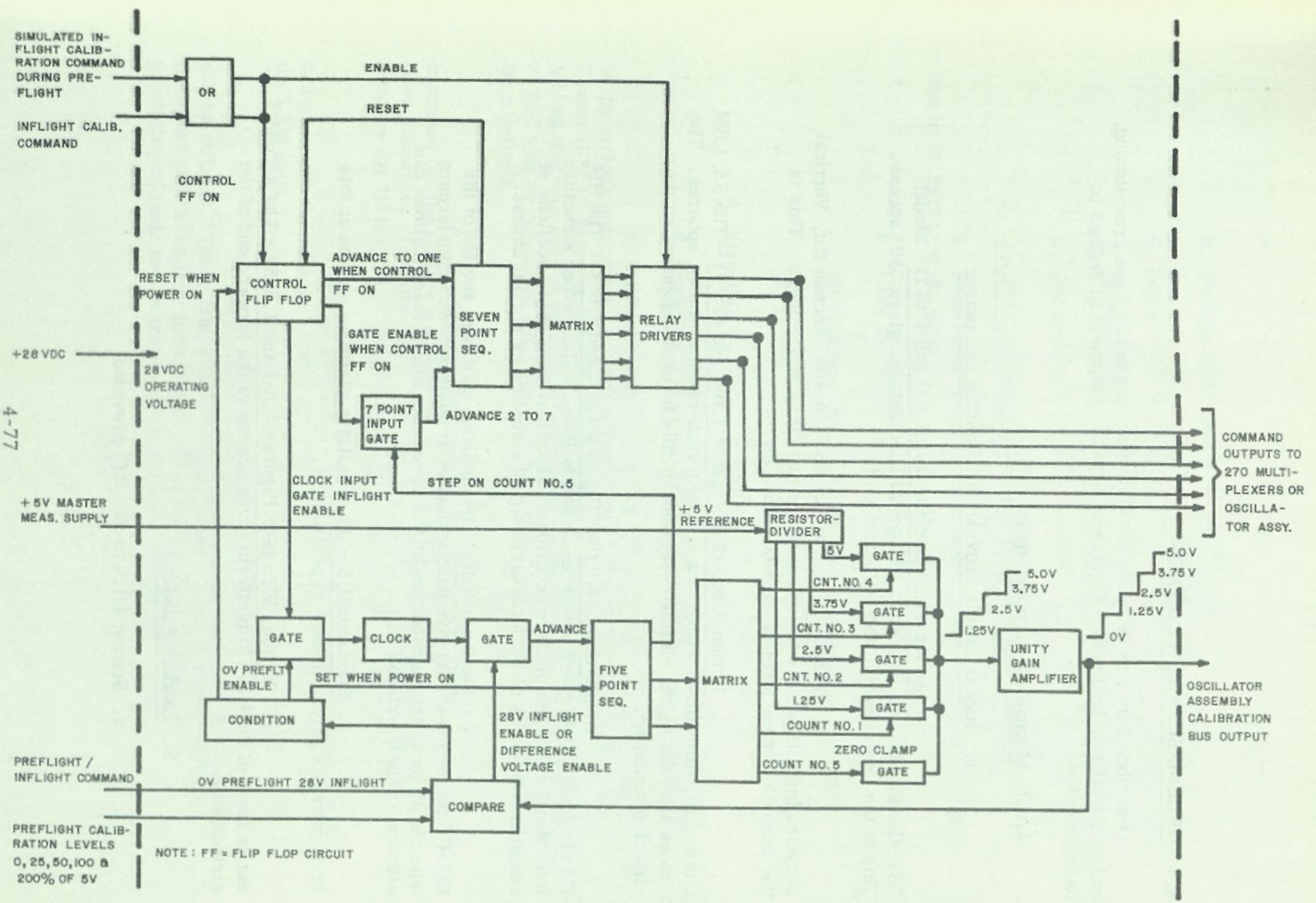


FIG. 4.5-6 TM CALIBRATOR ASSEMBLY

4.6 GROUND SUPPORT EQUIPMENT

Described here are the mechanical support equipment, gox provisioning, and umbilical requirements. Electrical support equipment is defined in paragraph 4.4.

4.6.1 Mechanical Support Equipment

1. Access Levels Provided by Existing Equipment

a. Station 441.0 (See Figures 4.6-1 and 4.6-2) - Model DSV-4B-402, Access Kit, Vertical Forward Interstage, gives 360° access. This is the lower level of the 402 access kit.

b. Station 477.0 - Model DSV-4B-402, Access Kit, Vertical Forward Interstage, has the capability of providing 360° access. This is the second or upper level of the 402 access kit.

c. Station 525.0 (See Figures 4.6-1 and 4.6-3) - The SA/LEM internal platform (lower) provides access for checkout of the O_2 spheres. The access provided by the platform sections at this level is dependent on the airlock orientation.

d. Station 603.0 (See Figures 4.6-1 and 4.6-4) - The H14-176 platform (NAA) gives 360° access at this level. The airlock structure interferes with three platform sections. Modification of these sections or complete removal will be necessary to give complete or partial access.

e. Station 639.0 - The auxiliary platform section of the H14-176 platforms (NAA) give partial access at position + Y. Interference with the airlock structure occurs at this level, but minor modifications can eliminate this interference.

f. Station 660.5 - The auxiliary platform section at this level provides partial access at position -Y.

g. Station 697.5 (See Figures 4.6-1 and 4.6-5) - The second major level of the H14-176 platform gives access to the forward end of the airlock unit.

2. Ladders Provided

a. Station 441.0 to 525.0 (3 provided)

b. Station 525.0 to 603.0 (1 provided)

c. Station 603.0 to 639.0 (1 provided)

d. Station 603.0 to 660.5 (1 provided)

e. Station 638.5 to 697.0 (1 provided)

3. Access Door Locations (center line and size of doors)

a. Instrument unit (IU) area, station 485.5. The access door size is 32.89W X 32.5L (REF).

b. LEM Area (2 provided):

(1) Position + Z, station 634.88, 34.0W X 34.0L (REF)

(2) Position - Z, station 634.88, 28.0W X 34.0L (REF)

4. Handling Equipment Required

a. Handling Sequence - is described in the separate Appendix.

b. Airlock Handling Equipment (See Figure 4.6-6) - The assembly of the airlock unit within the LEM Adapter will be performed at Kennedy Space Center (KSC). Investigation into the possibility of using handling equipment and tooling that was used in manufacturing for the assembly of the unit and handling at KSC will be conducted after manufacturing procedures and requirements have been defined.

c. Component Handling Equipment (See Figure 4.6-7) - The batteries for the airlock unit will be installed during the last day of the count-down sequence. Transportation from the storage to the vehicle, handling fixtures for lifting, and transportation inside the vehicle must be provided.

Two access doors are possible means of transferring the batteries into the vehicle: One in the instrument unit area and the other in the LEM area.

Existing dollies for battery transportation to the vehicle will be utilized. These transporters are provided for S-IVB and IU battery transport and the battery installation sequences would determine the possibility of their use in transporting the airlock batteries.

By entering through the access door in the IU area the component hoist can be used and the hoist at the 603.0 level can be utilized to transfer the batteries to the level of installation. This procedure can be used only if a trap door is provided in the shield platform. By entering through the access door in the LEM area they will be at the level of installation.

A mechanical hoisting unit must be provided for battery installation. The points of installation, weight of the batteries, and the criteria of the bulkhead protection study (D5-12802) required a mechanical means of handling these components.

Fixtures must be provided for battery handling that are compatible with the hoisting unit. Modifications to existing designs for the S-IVB or IU battery fixtures may be possible.

4.6.2 Gox Provisioning

A gox supply at ambient (80°F) temperature is required for: (1) fill of the four 3,000 p.s.i. storage bottles; (2) 20 p.s.i.g. leak check; and (3) vent capability to 5 p.s.i.g. for flight.

4.6.3 Umbilical Requirements

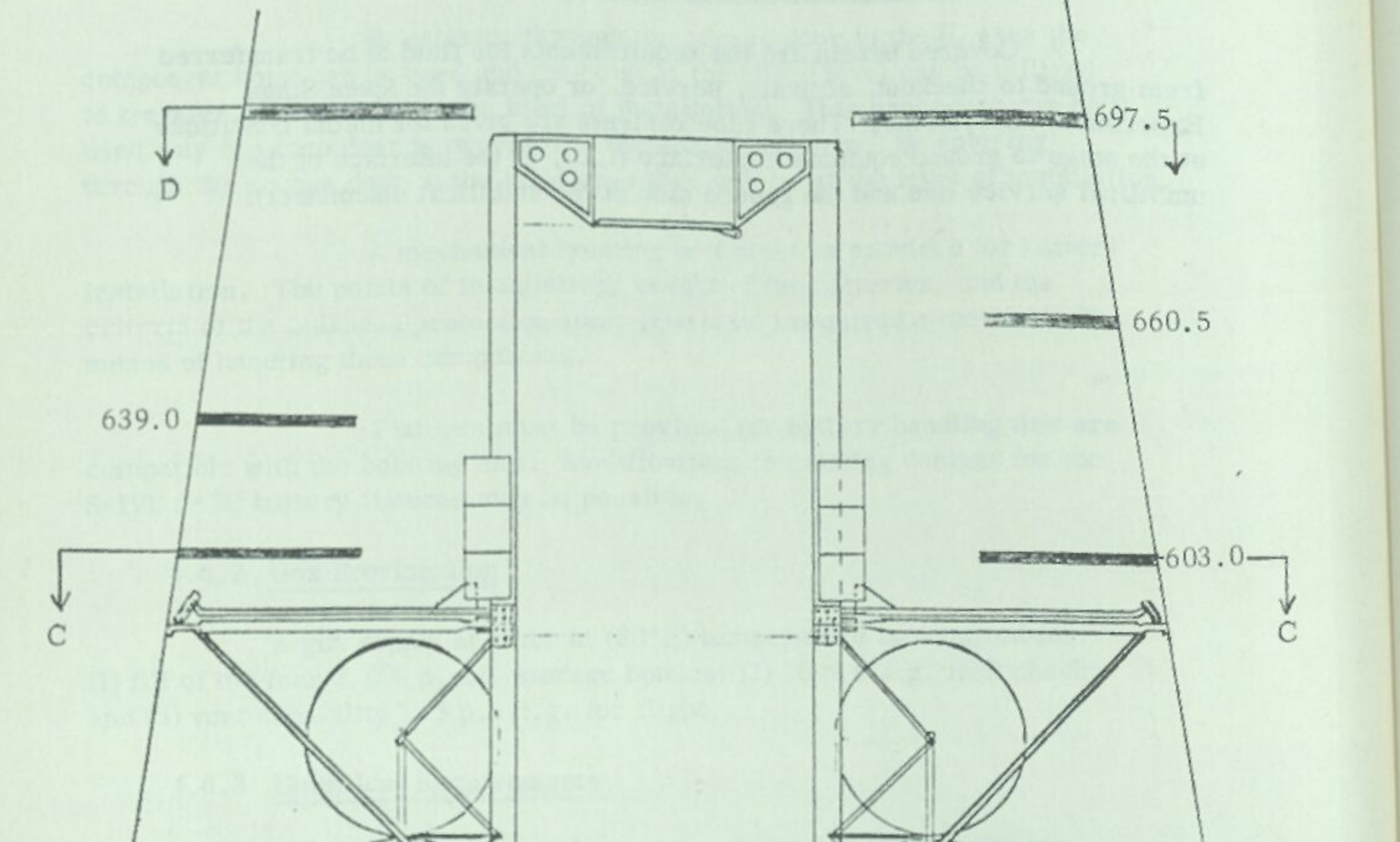
Quick disconnect couplings are required to service the gox spheres and the airlock. Present requirements indicate a 3/4-inch coupling will be required to purge and pressurize the gox spheres and two 1/2-inch coupling for the airlock purge, pressurization, and vent system. It is assumed that couplings designed and qualified on the Saturn V umbilical program will be adequate for these requirements.

Drag on lines with manual connections will be utilized to service the experiment with disconnect accomplished manually sometime prior to launch. Since there will not be an umbilical plate to interface with the Kennedy Space Center facility lines, a coordination effort will be required to assure the proper lines are available to the quick disconnect couplings to provide for manual connection and drag on.

The requirement exists for only one electrical umbilical connector and this is presently planned to be met by utilizing the spare connector in the instrument unit umbilical plate. A coordination effort will be required to assure this requirement is met and the IU umbilical modified to include this connector. Specific electrical support equipment requirements are discussed in paragraph 4.4.

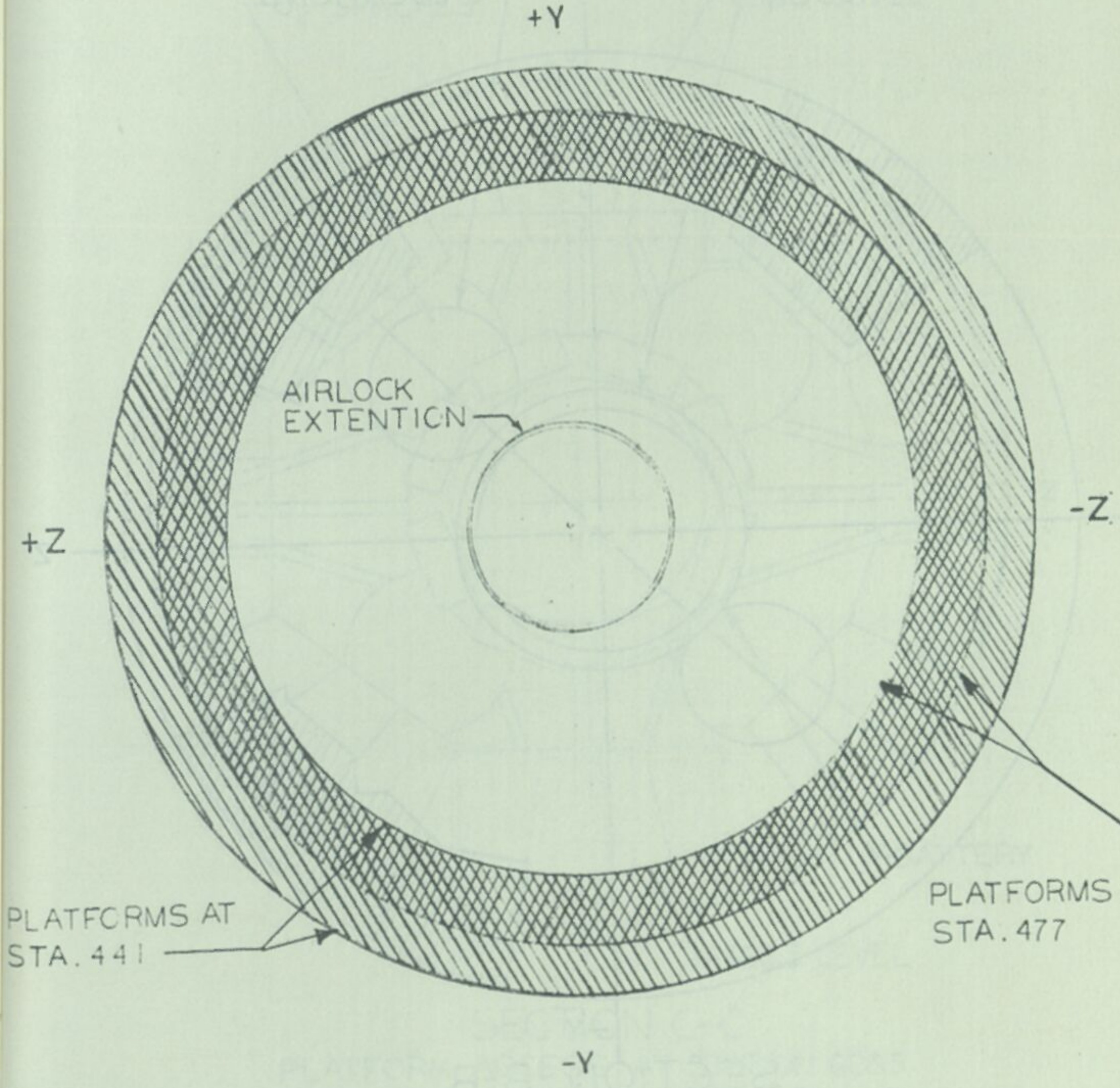
4.6.4 Fluid Requirements (Figure 4.6-8)

Covered herein are the requirements for fluid to be transferred from ground to checkout, activate, service, or operate the Spent Stage Experiment subsystems. These requirements are given for media conditions at the stage to ground equipment interface (i.e., at the interface of the umbilical service line and the ground side of the umbilical disconnect).



AIRLOCK UNIT ROTATED 45° FOR CLARITY

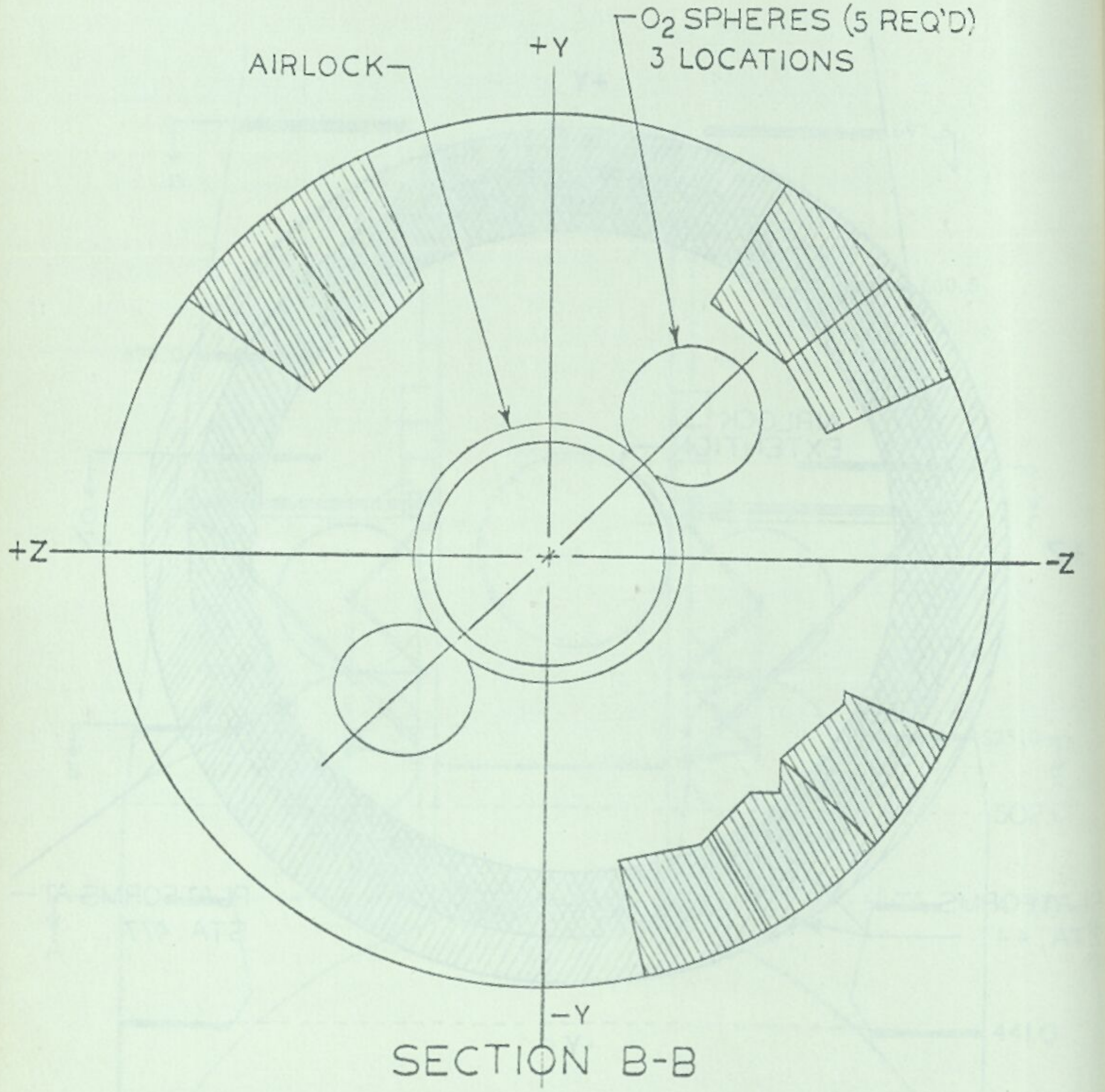
FIGURE 4.6-1



SECTION A-A

360° PLATFORM ACCESS AT
STATIONS 477 AND 441

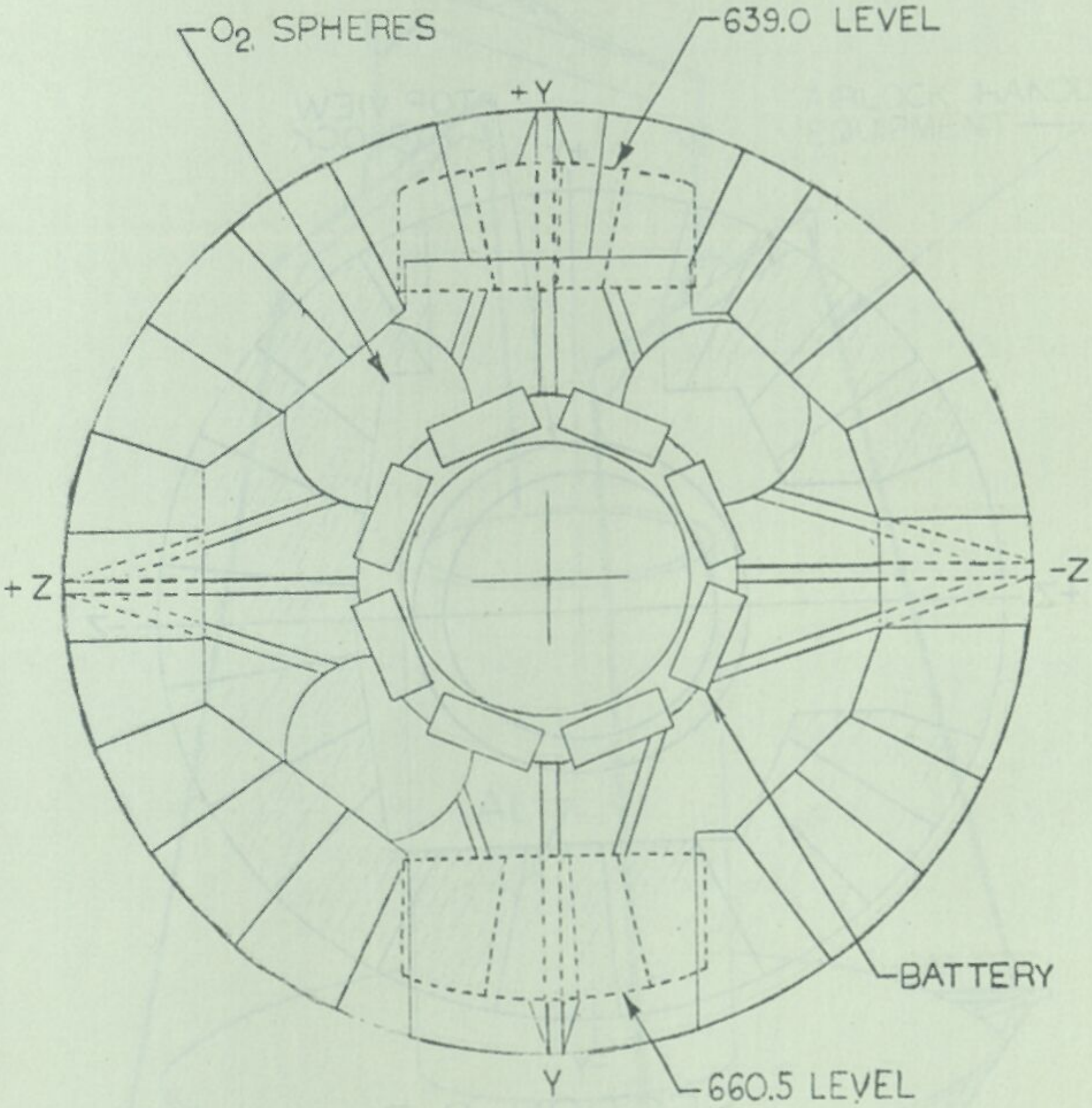
FIGURE 4.6-2



SECTION B-B

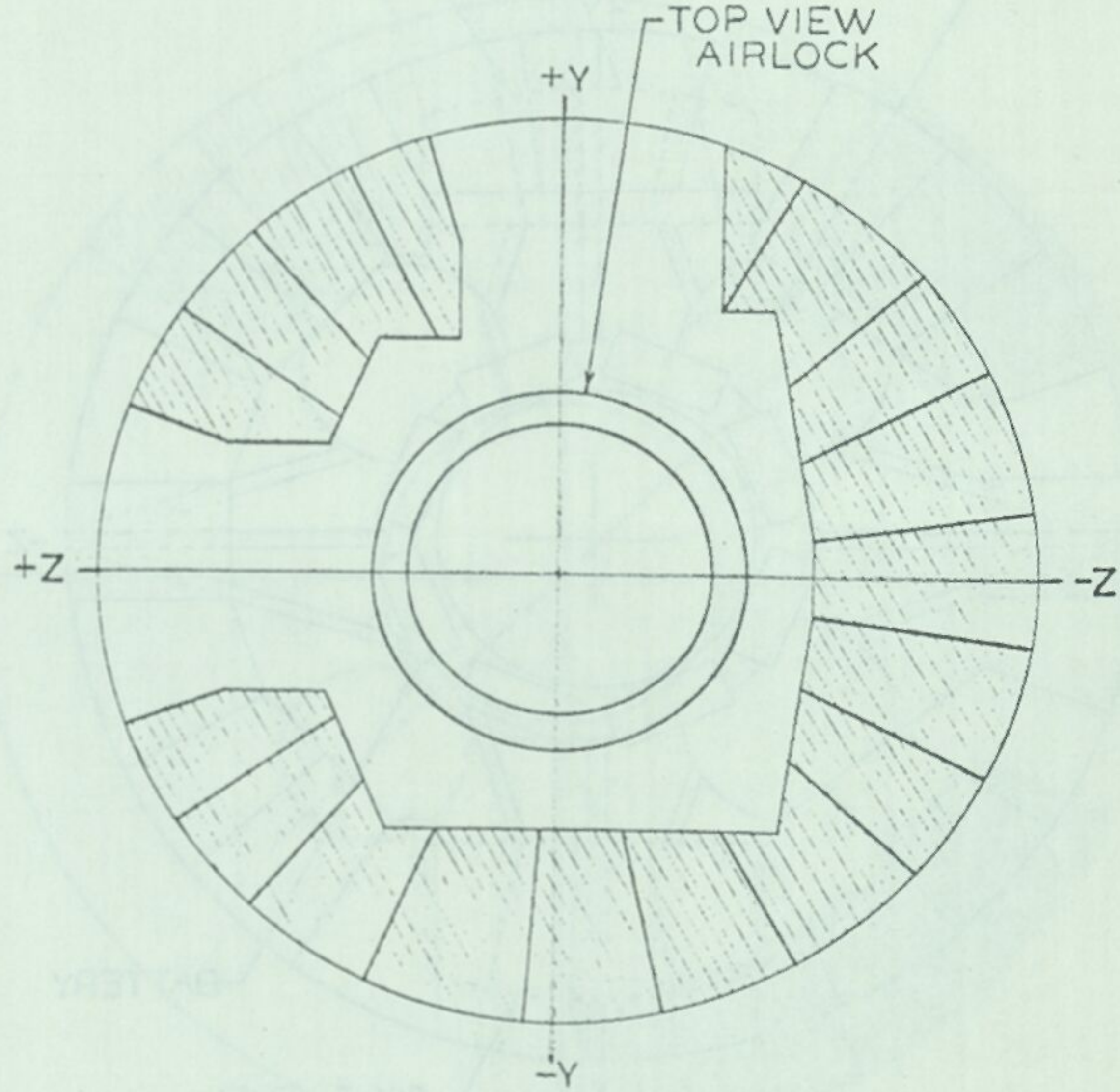
PLATFORM ACCESS AT STA 525.0

FIGURE 4.6-3



SECTION C-C
PLATFORM ACCESS AT STATION 603.5
WITH LEVELS 639.0 AND 660.5 SHOWN
BY DASHED LINES

FIGURE 4.6-4



SECTION D-D

PLATFORM ACCESS AT STA. 697.5

FIGURE 4.6-5

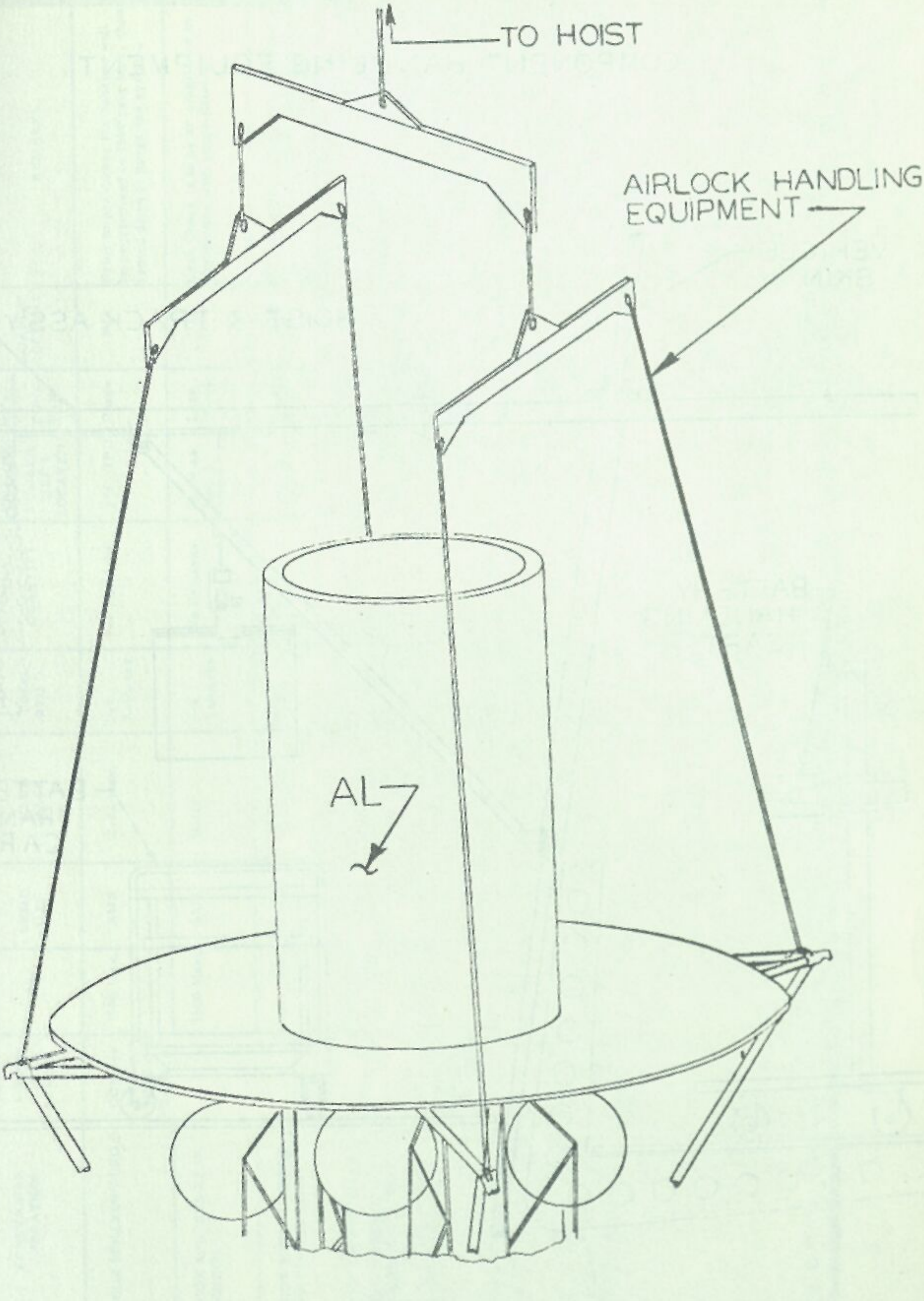


FIGURE 4.6-6 AIRLOCK HANDLING EQUIPMENT

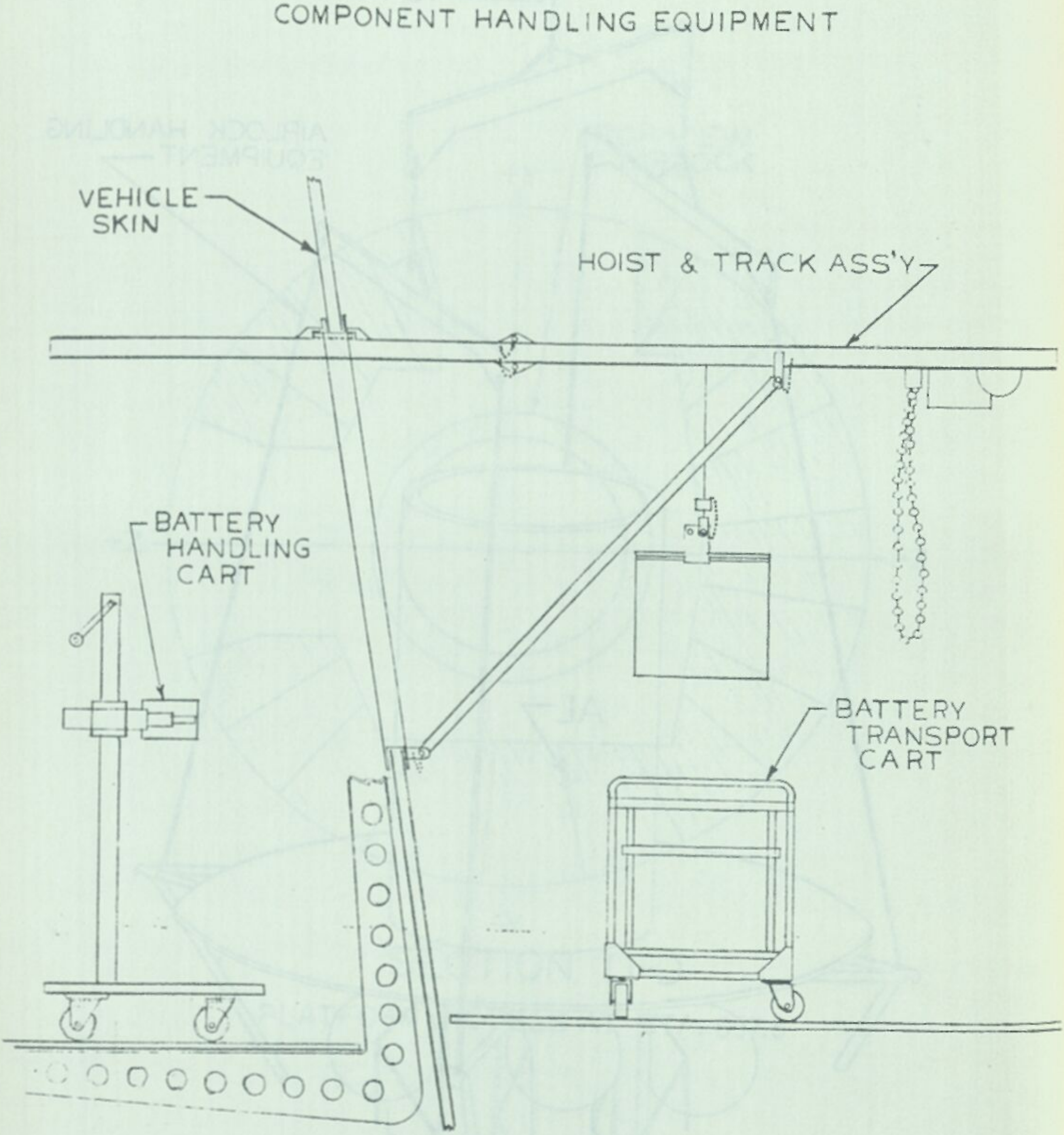


FIGURE 4.6-7 COMPONENT HANDLING EQUIPMENT

STAGE	ITEM	FUNCTIONAL OPERATION	FUNCTIONAL REQUIREMENTS									
			MEDIA & SPECIF	PRESS	TEMP. (°F)	FLOW RATE	TIME REQ'D	QUANTITY	UMB. REQ'MTS NUMBER SIZE & LOCATION	UMB. REQ'MTS FUNCTION	LOADING ACCURACY	REMARKS
Spent Stage Experiment	1	GOX SPHERES PURGE	GOX per MSFC-SPEC-399 Grade B	1500 Max.	AMB	12.22 Max.	See Remarks	See Remarks	3/4" TBD	Purge	N/A	Prior to pre-press the spheres will be pressurized to 1500 psig Max. then vented down to purge the system.
	2.	GOX SPHERES LEAK CHECK		1500 Max.	AMB	Static	See Remarks	See Remarks	Same as Item 1	Leak Check	N/A	Leak Check is to be accomplished in conjunction with purge (Item 1)
	3	GOX SPHERES PRE-PRESSURIZATION		1500 Max.	70 ± 20	12.22 Max.	TBD	TBD	Same as Item 1	Press. Flow	N/A	Prior to propellant loading the sphere will be pressurized to 1500 psig max. Pre-press. capability should be maintained until pressurization
	4	GOX SPHERES PRESSURIZATION		3100 Max.	70 ± 20	12.22 Max.	120 Max.	1280	Same as Item 1	Press. Flow	N/A	Pressurization required until mass is loaded.
	5	AIRLOCK PURGE	GHe per MSFC-SPEC-364	20 Max.	AMB	TBD.	TBD.	TBD	TBD	Purge	N/A	The airlock will be pressurized to 20 psig max. then vented down to purge the system.
	6	AIRLOCK LEAKCHECK		20 Max.	AMB	Static	See Remarks	See Remarks	Same as Item 5	Leak Check	N/A	Leak Check is to be accomplished in conjunction with purge (Item 5)
	7	AIRLOCK PRESSURIZATION	GOX per MSFC-SPEC-399 Grade B	5 Max.	70 ± 20	TBD	TBD	TBD	Same as Item 5	Press. Flow	N/A	Airlock will be pressurized to 5 psig max. prior to cryogenic propellant loading.

Issue Date:	Prepared by: M-P&VE-EF	Approved:

FIGURE 4.6-8. FUNCTIONAL REQUIREMENTS - FLUID

SECTION V. MANUFACTURING AND QUALITY & RELIABILITY ASSURANCE PLAN

5.1 MANUFACTURING PLAN

This manufacturing plan describes the proposed manufacturing sequence of the Spent Stage Experiment Support Module (SSESM). The basic discussions on the following pages include: (a) description of the general configuration of the SSESM; (b) description of the basic fabrication and assembly procedures for the canister portion of the module; (c) description of the fabrication and assembly procedures for the module support structure; (d) description of the assembly procedures for the canister; and (e) outline of the proposed assembly and procedure required to assemble the SSESM.

5.1.1 General

Canister. (See Figures 5.1-1 and 5.1-4.) The canister is 65 inches in diameter and approximately 204 inches long with vertical T-section stringers riveted to milled lands on the outside of the skin panels for stiffness. Four skin sections are required for the canister. Each skin section is comprised of four skin segments. The two forward skin sections (No. 1 and No. 2) are milled to provide 16 vertical stringers on the outside surface. The number one skin section is approximately 51 inches wide and 45 inches long. The number two skin section is approximately 51 inches wide and 50 inches long. The number three skin section is composed of four skin segments approximately 51 inches wide and 68 inches long. Two of the skin segments will be mechanically milled with vertical stringers and the remaining two skin segments will be milled with vertical weld pads. The number four skin section will be composed of four skin segments approximately 51 inches wide and 28 inches long and will be of the same configuration as the skin segments for the number one and number two skin sections. All skin segments will be contour formed and age hardened in a simultaneous operation. Three I-beam shaped rings will be spaced between the four skin sections and welded in place. The forward end of the canister will be equipped to receive the CSM and will have a sealed hatch for access to the CM. The aft end will have a bellows assembly for attachment to the S-IVB workshop and will also have a sealed hatch for access to the S-IVB spent stage. Also in the aft portion of the canister will be an escape hatch for access to the LEM adapter area.

Module Support Structure. (See Figures 5.1-1 and 5.1-2.) The module support structure consists of four tripod-style strut assemblies. Each strut assembly consists of a vertical strut and two horizontal struts with machined attach fittings. The vertical strut is manufactured from aluminum alloy tubing which is approximately three inches in outside diameter with a 1/4-inch thick wall. The two horizontal struts are manufactured from aluminum alloy I-beams which have three-inch wide flanges with a thickness of 1/4-inch. Each of the four strut assemblies were assembled in an assembly fixture.

5.1.2 Typical Skin Section Fabrication and Assembly

Skin Segment Fabrication - The proposed procedure for the task is as follows:

1. Locate plate material on skin mill; pull vacuum on plate material using the skin mill vacuum chucks, locking the plate material in place for milling.

2. Mill plate material to the required configuration.

3. Repeat operations for the three remaining skin segments.

4. Locate the milled skin segment on the age form fixture; clamp the skin to the age form fixture, forming the skin to contour.

5. Place the skin and age form fixture into the autoclave; age form the skin to the required condition.

6. Repeat operations for the three remaining skin segments.

Skin Section Assembly - The proposed procedure for this task is as follows:

1. Adjust the dimensions on all skin segments, allowing for weld shrinkage to determine the amount of trim at the edge of each skin segment and to enable the completed skin section to be welded to the I-beam and end rings.

2. Position a 90-degree skin segment on air bearing blocks on skin section assembly fixture.

3. Align the skin segment, pull vacuum, and lock in place.

4. Mount the router head on weld manipulator and make a vertical cut on the right edge of the skin segment to the predetermined dimension.

5. Release and rotate the skin segment 90-degrees clockwise.

FIGURE 5.1-1.

SPENT STAGE EXPERIMENT SUPPORT MODULE (SSESM)

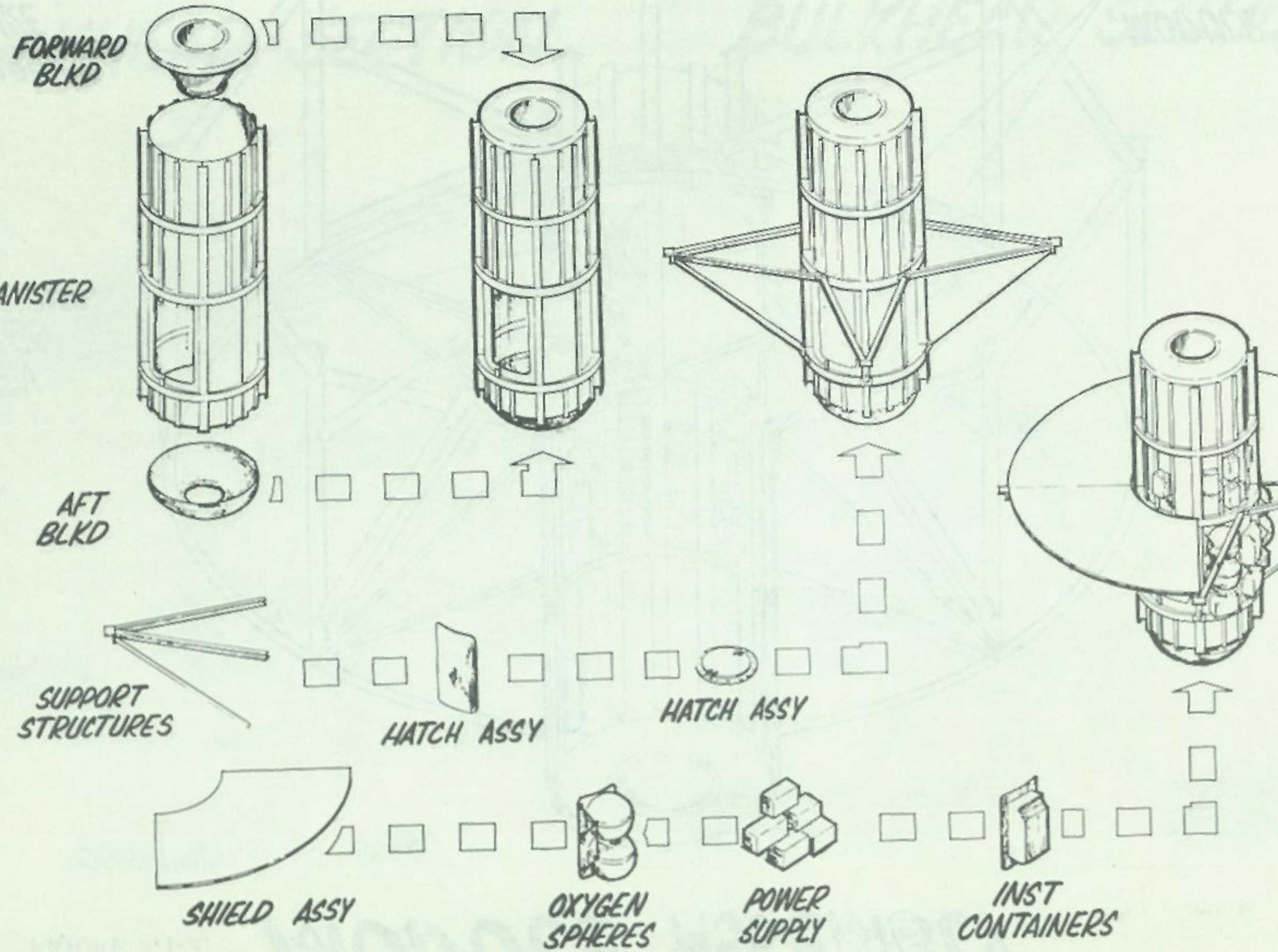


FIGURE 5.1-2. MODULE ASSEMBLY

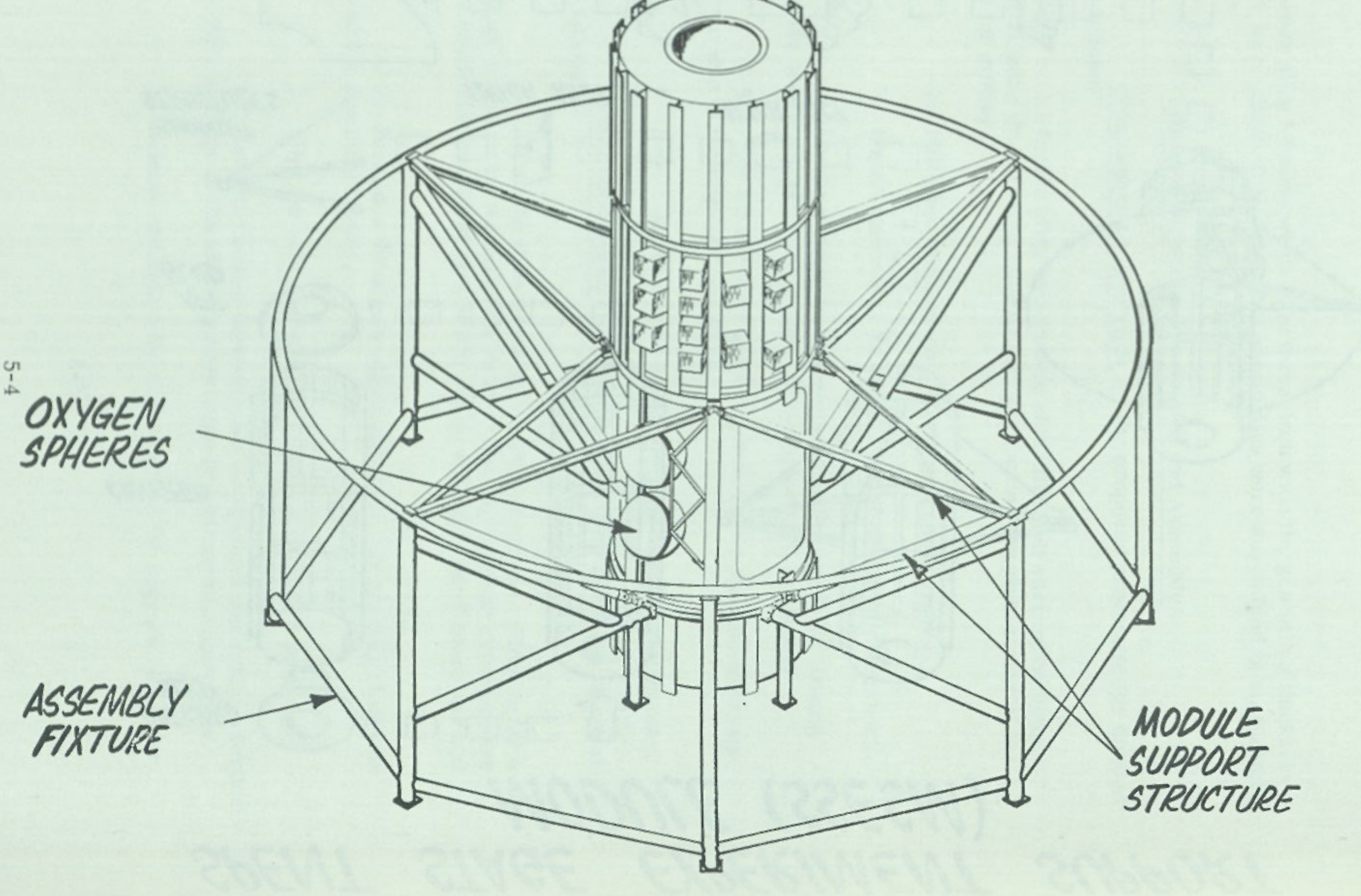


FIGURE 5.1-3.

MODULE AFT BULKHEAD SECTION

MODULE FORWARD BULKHEAD SECTION

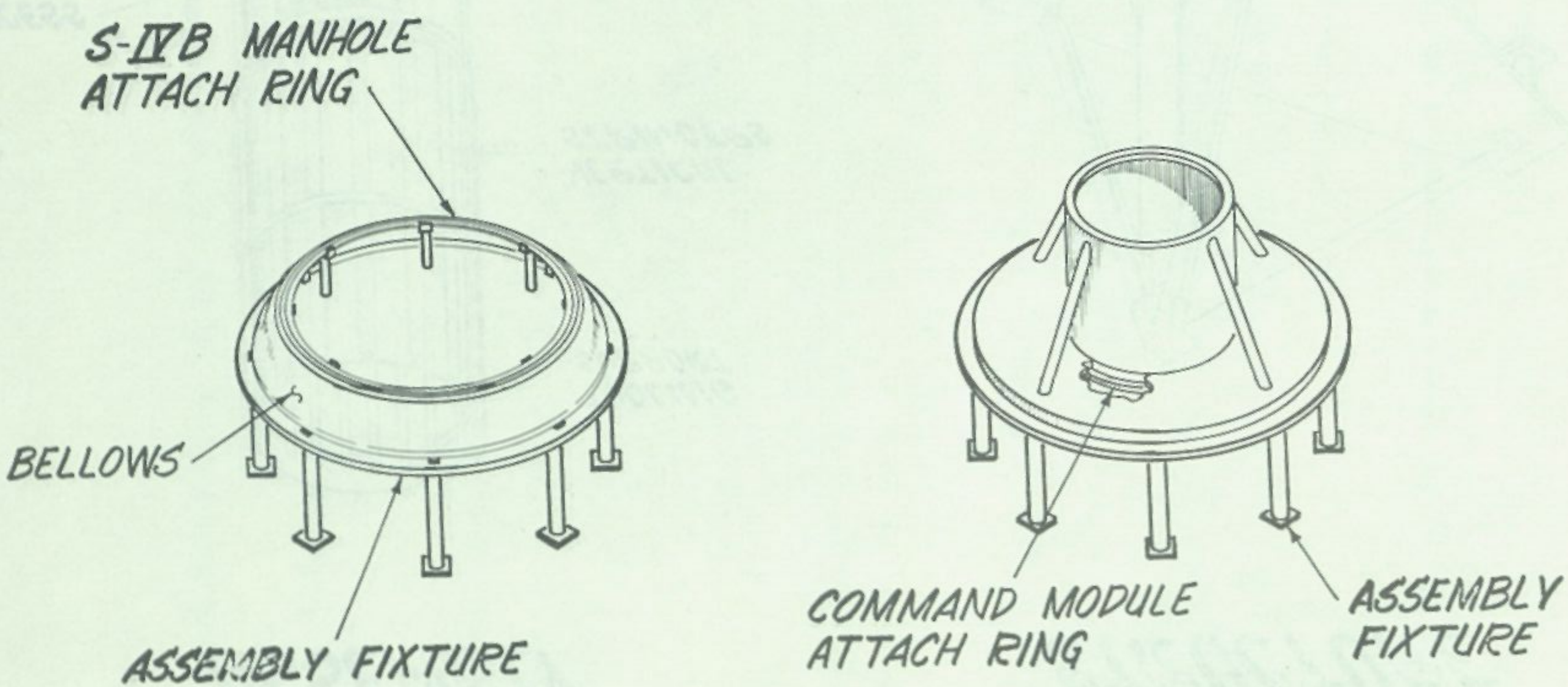
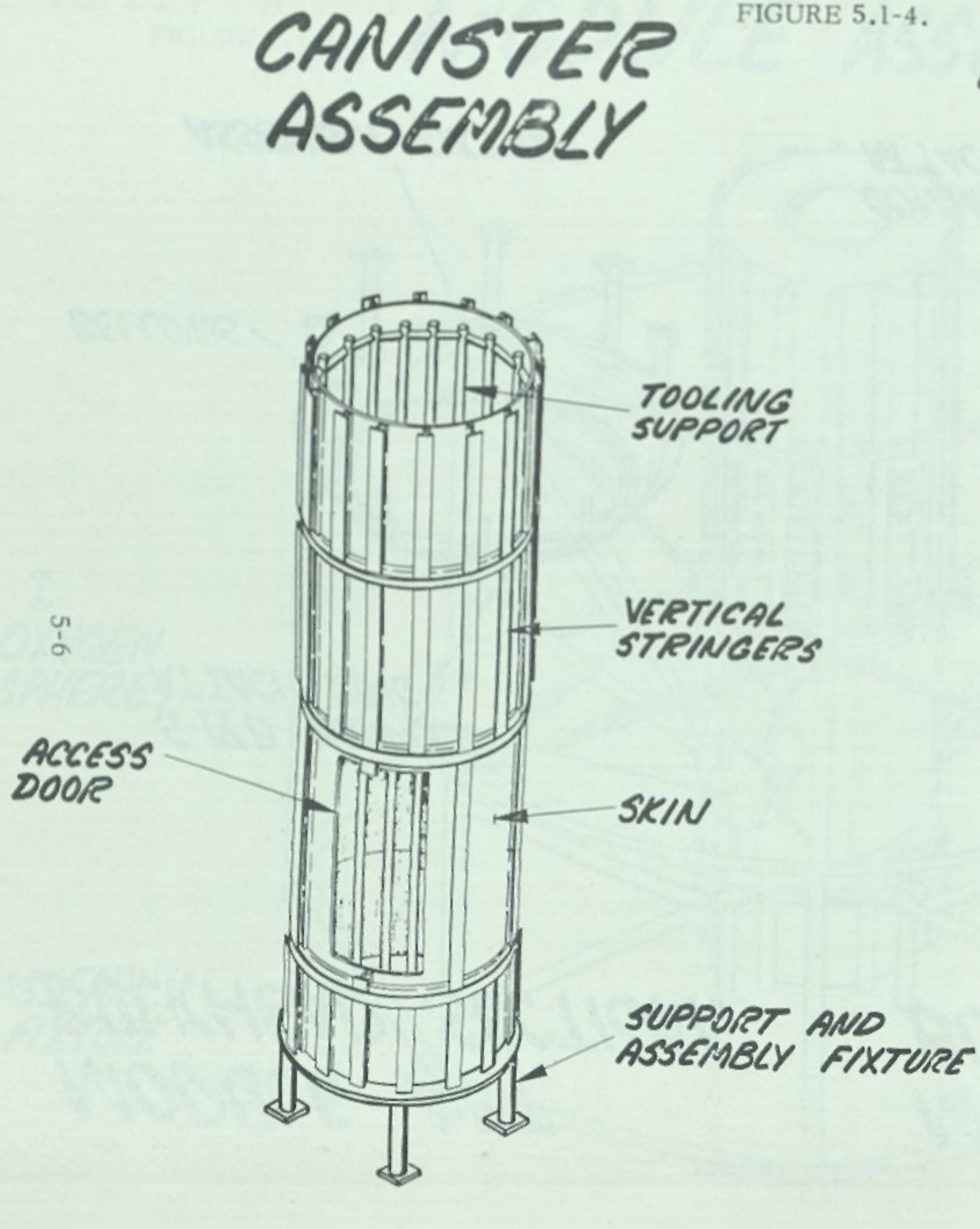
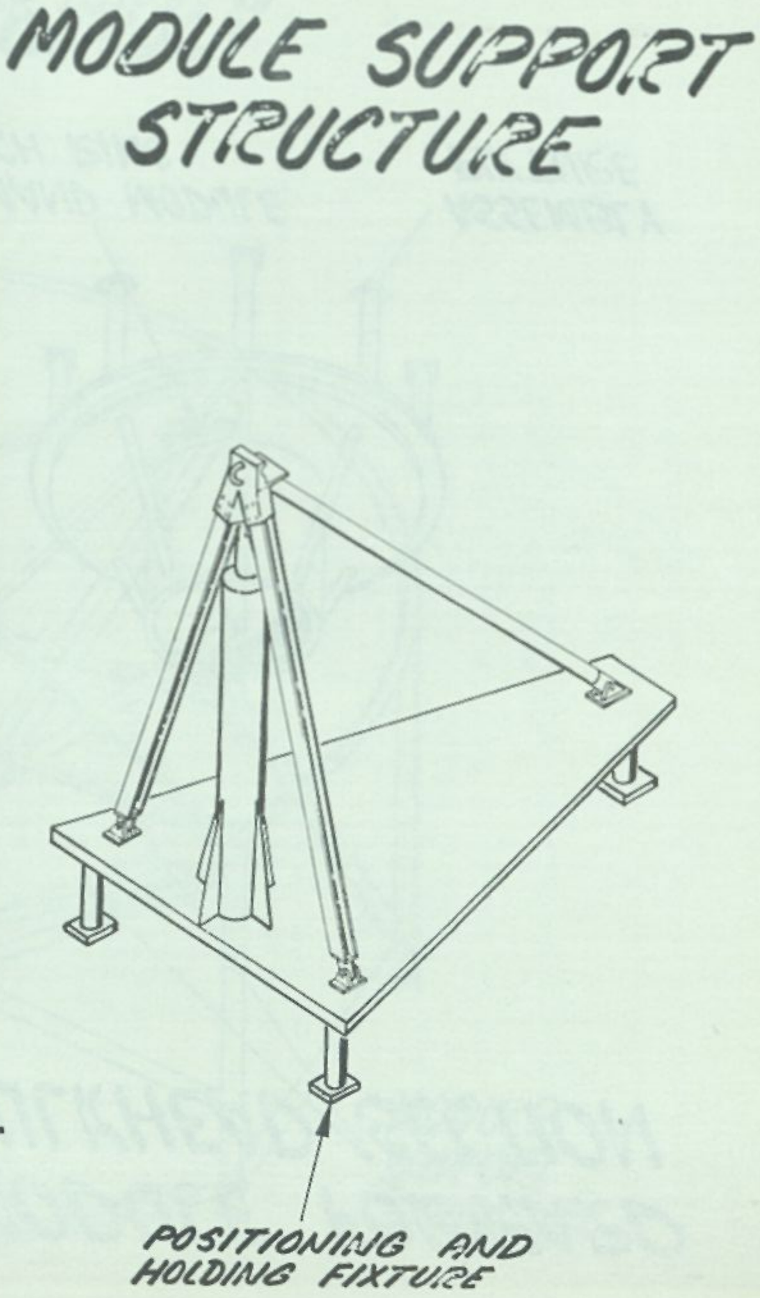


FIGURE 5.1-4.


CANISTER
ASSEMBLY
TOOLING
SUPPORT
VERTICAL
STRINGERS
ACCESS
DOOR
SKIN
SUPPORT AND
ASSEMBLY FIXTURE


MODULE SUPPORT
STRUCTURE
POSITIONING AND
HOLDING FIXTURE

6. Repeat similiar procedure to cut left edge of section.

7. Hoist the next 90-degree skin segment in place locating the right edge of the skin segment against the skin segment already installed on the weld fixture.

8. Pull vacuum and lock segment in place.

9. Release and rotate trimmed segment to facilitate cleaning.

10. Make a vertical cut on the right edge of the untrimmed skin section.

11. Release and rotate segment to facilitate cleaning.

12. Clean the weld edges of the two skin segments by removing the conversion coating, inside and outside.

13. Make three test welds on samples. Inspect and make proper settings.

14. Locate the two skin segments for welding. Pull vacuum and lock the skin segment in place.

15. Install the finger clamping device, tack weld the two skin segments together, and remove the finger clamping device.

16. Tack weld tabs on top and bottom of the weld joint.

17. Automatically weld the two skin segments together.

18. Shave the weld bead to 0.015-inch high using the microshaver.

19. Release and rotate the segments 90 degrees clockwise and remove weld tabs.

20. Repeat proceeding operations outlined above for each of the two remaining 90-degree skin segments.

21. Repeat previous procedures in a similiar manner for the two welded skin segments.

22. Release and remove the skin section from the weld fixture.

23. Machine the top and bottom edges of the skin section parallel and to the required length using router fixtures.

5.1.3 I-Beam Rings Fabrication

The three I-beam rings are utilized in the assembly of the canister and are completely machined parts. The aft I-beam ring is machined to include the aft bulkhead escape hatch fitting.

5.1.4 Forward Bulkhead Section Fabrication and Assembly (See Figure 5.1-3)

Command Module Attach Ring Fabrication - The attach ring will be machined complete from raw material that is 44 inches outside diameter by seven inches thick by seven inches wide.

Bulkhead Center Fitting Ring Fabrication - The center fitting ring will be machined complete from raw material that is 44 inches outside diameter by seven inches thick by seven inches wide.

Bulkhead Outer Fitting Ring Fabrication - The outer fitting ring will be machined complete from raw material that is 72 inches outside diameter by seven inches thick by seven inches wide.

Bulkhead Skin Section Fabrication - The skin section will be fabricated from plate material and formed to the required dimensions. After forming, the skin material will be routed to the final dimensions.

Bulkhead Center Tunnel Fabrication - The center tunnel will be fabricated from two pieces of plate material which will be formed to 30 inches inside diameter. The two 180-degree cylinder halves will be vertically welded together to form the 360-degree 30-inch inside diameter center tunnel. The tunnel will then be trimmed to the required length.

Bulkhead Reinforcement Struts Fabrication - The reinforcement struts attach between the center fitting ring and the outer fitting ring. The raw material for the reinforcement struts will be sawed to the required length.

Forward Bulkhead Section Assembly - The proposed procedure for this task is as follows:

1. Position the outer fitting ring in place on the forward bulkhead section assembly fixture and clamp in place.

2. Position the command module attach ring in place on the assembly fixture and clamp to the center post of the assembly fixture.

3. Locate the skin section in place on the outer fitting ring and the command module attach ring and clamp in place.

4. Weld the skin section to the outer fitting ring and to the command module attach ring.

5. Locate the center tunnel in place on the command module attach ring and clamp in place.

6. Weld the center tunnel to the command module attach ring.

7. Locate the center fitting ring in place on the aft end of the center tunnel; clamp in place.

8. Weld the center fitting ring to the aft end of the center tunnel.

9. Locate the reinforcement struts in place between the center fitting ring and the outer fitting ring; clamp in place.

10. Weld the reinforcement struts to the center fitting ring and to the outer fitting ring.

NOTE: The forward escape hatch will be attached during final assembly of the module.

11. Remove the forward bulkhead section from the assembly fixture.

5.1.5 Aft Bulkhead Section Fabrication and Assembly. (See Figure 5.1-3)

Bellows Fabrication - The bellows will be a complete fabricated procured item.

S-IVB Manhole Attach Ring Fabrication - The attach ring will be machined to the required configuration from raw material.

Aft Bulkhead Section Assembly - Following is the proposed procedure for this task:

1. Locate bellows in place on the assembly fixture and clamp in place.

2. Locate the attach ring in place on the aft end of the bellows and clamp in place.

3. Drill required holes through the attach ring and bellows.

4. Remove the attach ring from the bellows.

5. Apply adhesive to the attach ring and bellows mating surfaces.

6. Relocate the attach ring in place on the bellows; attach with the required attaching hardware.

7. Remove aft bulkhead section from the assembly fixture.

5.1.6 Module Support Structure Fabrication and Assembly (See Figures 5.1-2 and 5.1-4)

Vertical Strut Fabrication - The vertical struts are fabricated from aluminum alloy tubing which is approximately three inches in outside diameter with a 1/4-inch thick wall. The struts will be sawed to the required length and the attach ends are sawed to the required configuration for attachment to the support structure fittings.

Horizontal Strut Fabrication - The horizontal struts are fabricated from aluminum alloy I-beams which have three-inch wide flanges with a thickness of 1/4-inch. The struts will be sawed to the required length and the attach ends are sawed to the required configuration for attachment to the support structure fittings.

Support Structure Attach Fitting Fabrication - The attach fittings for the support structure will be machined complete from aluminum alloy forging.

Assembly of Strut Assemblies - The proposed procedure for this task is as follows:

1. Locate four support structure attach fittings in place on the positioning and holding fixture; clamp in place.

2. Locate the two horizontal struts and the vertical strut in place between the attach fittings on the positioning and holding fixtures.

3. Secure the horizontal and vertical struts to the attach fittings with the required hardware.

4. Remove the strut assembly from the positioning and holding fixture.

5. Repeat operations previously outlined for three remaining strut assemblies.

Micro-Meteoroid Shield Fabrication - Four 90-degree sections of honeycomb material will be utilized for the micro-meteoroid shield. The lower skin, which will be 0.010-inch thick, will be placed in a bond form fixture; the honeycomb material will then be placed onto the lower skin; the adhesive and the upper skin will then be located in place. The entire assembly will then be placed in the autoclave for bond forming. After bond forming, the honeycomb panel (90-degree section) will be sawed to the required dimensions.

5.1.7 Assembly of the Canister (See Figure 5.1-4)

The proposed assembly procedure is as follows:

1. Locate and secure the number two skin section in place on the tooling ring of the turntable, with the aft end down.

2. Mechanically clean the top edge of the number two skin section and the aft edge of the forward I-beam ring for welding.

3. Install a roundout ring-backup bar in the forward end of the number two skin section.

4. Locate the forward I-beam ring in place on the number two skin section and expand the roundout ring-backup bar against the I-beam ring and number two skin section.

5. Prepare test weld samples and verify the weld settings before each weld; analyze the results before proceeding.

6. Weld the I-beam ring and the number two skin section together.

7. Repeat the preceeding operations outlined until the number one, number three, and number four skin sections and the remaining two I-beam rings are welded together and inspected.

NOTE: The vertical T-section stringers will be welded and riveted to the lands on the skin sections during assembly of the module.

5.1.8 Module Assembly (See Figures 5.1-1 and 5.1-2)

The proposed assembly procedure is as follows:

1. Mechanically clean the forward end of the canister and the aft end of the forward bulkhead outer fitting ring for welding.

2. Locate and clamp the forward bulkhead in place.

3. Weld the forward bulkhead to the forward end of the canister.

4. Locate the aft bulkhead in place. Apply adhesive and secure the aft bulkhead (bellows and S-IVB manhole attach ring) to the aft end of the canister.

5. Locate and clamp the four I-beam vertical struts in place on the number three skin section of the canister.

6. Weld the I-beam vertical struts to the vertical weld lands of the number three skin section.

7. Locate and clamp the vertical T-section stiffeners in place on the milled lands of the canister.

8. Rivet the vertical T-section stiffeners to the milled lands of the canister rivets.

9. Locate and clamp the four support structure strut assemblies in place.

10. Weld the support structure fittings to the I-beam rings on each end of the number three skin section.

11. Locate and clamp the four 90-degree honeycomb panels of the micro-meteoroid shield in place on top of the horizontal struts of the module support structure.

12. Drill and ream the required holes through the honeycomb panels and the horizontal struts and install the required hardware.

13. Locate the escape hatch (to the LEM adapter area) in place on the number three skin section of the canister, lay out the opening, and remove the escape hatch.

14. Lay out and saw the required opening in the number three skin section for the escape hatch.

15. Install the sliding track for the escape hatch on the inside of the canister.

16. Install the forward escape hatch in place on the forward bulkhead section.

17. Install the aft escape hatch in place on the aft bulkhead section.

18. Install the escape hatch in the opening in the number three skin section of the canister.

19. Locate and attach the 21 batteries around the outside of the canister between the first and second I-beam rings on the number two skin section.

20. Locate and attach four 42-inch diameter oxygen bottles (two each 180 degrees apart) on the outside of the number three skin section of the canister.

21. Apply Alodine coating and paint the module in accordance with the spacecraft color code.

5.2 QUALITY AND RELIABILITY ASSURANCE PLAN

5.2.1 General

This plan describes those inspections, analyses, and tests planned to provide maximum assurance of the acceptability of the SSESM.

5.2.2 Analyses and Inspection Operations

Source Control - Source control is required to assure the most efficient interfacing of quality assurance testing operations performed by vendors and MSFC. Contractual documents, work statements, etc., will be reviewed for adequate quality requirements and source controls to be imposed on the selected supplier. Generally, Government concern for control of procurement sources will be described in one of the following documents: NPC 200-2; NPC 200-3; NPC 250-1; others as called for by contract.

Receiving Inspection - Receiving analysis activities and requirements are to plan and perform effective receiving inspection, analysis, and testing operations which will assure the degree of quality for all procured items satisfactory for the purpose intended, and that only those materials and items that meet the required standards and specifications are procured and stocked for the SSESM. In order to assure the receipt of acceptable raw material and hardware at MSFC, it is required that all such items intended for the SSESM be routed through Receiving Inspection.

Articles shall not be accepted unless they are qualified or designated for qualification. Hardware for the SSESM program will be qualified in accordance with test requirements and schedules established by the contract. Facilities and trained personnel for inspection and analysis are available and will be committed to this program.

Specifically, the criteria to be applied to the various materials include:

1. Structural shapes: dimensions, physical tests of samples, composition, heat treatment, identification, and certification.

2. Plate and sheet metal: dimensions, physical test of samples, composition, heat treatment, documentation submittal flatness, surface scratches, protective coatings, identification.

3. Pressure tubing: dimensions, roundness, concentricity, surface (outer and inner) composition, physical properties, identification, and certification.

4. Bellows: visual, cure date, dimensions of attaching surfaces, identification, and certification.

5. Fasteners: visual, physical tests of samples, thread forms and size, identification, and certification.

6. Machined and formed parts: dimensions, surface finish, cracks, visual, identification and certification.

7. Pressure vessels: dimensions, certification, and other documentation, identification, cleanliness, proof pressure, surface finish, protective finish, location and dimensions of bosses and parts, and thread form and size.

Component and Subassembly Analysis - As previously noted in planning material, the key to successful SSESM will be the individual reliability of the functional components comprising the module's systems. It is essential, therefore, that a most discriminating and demanding component functional test be performed on all hardware prior to installation on the module. Components shall be functionally tested in as near test mission environment as practical. The Test and Training module and flight SA-209 module canister assemblies will be leak tested following assembly. The Test and Training module canister assembly will also be inspected and tested following structural testing.

Mechanical Components Tests - The purpose of these tests is to establish confidence in the ability of each component to perform satisfactorily when incorporated into the module. It is therefore necessary to functionally test these components in as near a mission mode as possible. The basic type of tests shall include:

1. A visual inspection;

2. Cleanliness of components;

3. Components conformance to documentation;

4. Mark assembly data and cure date of oldest seal on component;

5. Check safety wiring, lubrication, and electrical connector pins;

6. Check in instrumented electrical and mechanical test setup;

7. Operating pressure tests;

8. External leakage;

9. Internal leakage;

10. Valve operation.

Electrical/Electronic Component and Subsystem Functional Testing - These tests and the selection, evaluation, maintenance and control of the associated test equipment are discussed below:

1. Functional testing is necessary to insure that adequate manufacturing procedures were utilized to produce an acceptable component or package. Performing functional vertification testing under simulated operating conditions assures its ability to satisfy mission requirements.

2. Electrical functional tests shall be performed on all electrical/electronic components and subsystem such as telemetry packages; measuring devices consisting of AC and DC amplifiers; pressure and temperature transducers; communication systems components; power supplies; heaters; lights; blowers; and all other instrumentation components or subsystems that comprise the life support system, airlock system, docking structure and complete experiment packages or systems as required.

3. The selection, evaluation, approval, maintenance and control of the test equipment used for functional testing will be in accordance with the requirements of NASA Quality Publication NPC-200-2, Section 9. Test equipment will be an order of magnitude more accurate than the specified tolerance on parameters it is measuring or providing.

Failure Analysis - The objectives of a failure investigation program are to determine the specific cause and origin of the various failures that occur to components and to eliminate further failures of the same or related natures. A failure analysis will be conducted on all components that fail after assembly to the module. Failure of components during receiving inspection and subsequent bench functional tests will also require failure analysis when the failure indicates a design or quality problem, the failure occurs to a critical component or involves long lead time components, or failure of components has been experienced on previous lots of a particular component. Components will be submitted for failure investigation for any of the following conditions: (1) actual observance of component failure; (2) suspicion of component failure with reasonable basis.

A program is presently established to feed back information and take corrective action on troubles, malfunctions, deficiencies, and failures discovered during inspection and test at the plant, field site, etc.

Fabrication Analysis - These operations provide the mechanical and electrical inspections and analysis to be performed during the various fabrication and assembly operations on the SSESM. End items or intermediate operations shall be subjected to the tests and inspections which are appropriate to determine acceptability. In-process inspection shall only be used where the quality of the part or operation cannot be verified by an end item inspection.

During fabrication all drawings, specifications, processes, procedures, and integration analysis planning shall be reviewed continuously in order to eliminate errors and omissions and improve efficiency without compromising quality.

Sheet Metal or Machined Parts - A complete inspection of each part such as the canister skin shall be performed. The item shall be inspected for conformance to applicable drawings and specifications and shall include: dimensional analyses, hardness tests, dye penetrant of formed areas, surface finish, cleanliness, X-ray, and other non-destructive testing.

Structural Fabrication - Major subassemblies including the forward and aft bulkheads and canister assembly shall be inspected to include dimensional analysis, rivets and fasteners for proper installation, interference fit holes for proper diameter prior to insertion, physical appearance and surface finish, alignment, and status compliance to applicable drawings and specifications.

Welding - Preparation for the welding on the canister assembly and other welded structures will be in accordance with standard procedures and must be inspected prior to commencing the welding operation. Test specimens of the type of joint to be welded will be made prior to the welding operation. Strength tests and bead analyses will be performed, and failure to meet the minimum requirements of any specimen will be cause for recommending production welding not to take place. Upon successful completion of the specified tests an inspection tests report containing the essential information will be completed and signed by the quality control representative.

All welds will be radiographically inspected. Examination of welds will demonstrate that the inspection technique positively establishes the defects. Where radiographic inspection is determined to be inconclusive, ultrasonics, etch, and dye-penetrant shall be used. Visual inspection of all welds will be made.

Cabling and Trunking - All cabling and trunking shall be inspected for compliance to applicable documentation and proper functioning. These inspections shall include visual inspection of the completed assembly for proper length, lacing and ties, connector condition, hy-ring installation, shield breakout, cable size, ground wire installation and potting or molding. Functional tests shall be made of each cable, trunk, J-box to include continuity, leakage between pins, and insulation resistance.

Electrical Connectors - All connectors attached by methods such as potting and molding or crimping shall be inspected for proper assembly. A visual inspection shall be made of each crimped connector for damage of conductor or terminal, proper gap, deformation, tarnish, proper pin taper, broken strands and functional insertion capability. A visual inspection of potted connectors shall be performed for appearance, surface condition, bond integrity, and alignment of contracts.

Cleaning - Rigid tubing, flexible hose assemblies, containers, components, and SSESM pressure bottles shall be inspected to MSFC specifications for cleanliness by monitoring the operation, recording and analyzing operational data, and by laboratory analyses of samples pulled from the operation. A visual inspection shall be performed on completed items.

Assembled SSESM Analysis - The assembled module analysis shall be accomplished immediately following completion of assembly. This analysis consists of a series of nonfunctional analytical operations that are performed to assure the delivery of an end item conforming to design requirements. Additionally, these operations establish the base line status of the module and are prerequisite to the module functional test.

All Spent Stage Experiment Support Modules will be subjected to the assembled module analysis. The test and training article will be examined closely for damage resulting from structural and vibration testing. The Zero "g" mockup will only undergo an examination of sufficient depth to assure completeness and astronaut safety.

Electrical Installation Analysis - This operation will: (1) Ascertain that all cables have been installed, routed, tied, and laced in accordance with applicable installation drawings, and applicable requirements; (2) verify that all cable and connector reference designation markers are legible and correspond with reference designation markers on mating component connectors; (3) verify that all electrical components have been installed as specified; (4) verify that grounding methods used are in accordance with the installation drawings and applicable provisions.

Module Electrical Systems Continuity/Compatibility Tests - Continuity/compatibility testing of electrical/electronic systems utilized in the airlock, environmental control, experimental and life support systems shall be performed to meet test parameters specified by design and/or quality assurance requirements.

Torque Verification - The torque values of all pressure system connections utilizing gaskets will be checked. Pressure system connections utilizing a metal-to-metal contact will be checked following any major structural movement resulting from transportation, attitude change, or structural test. All types of bolted connections having metal-to-metal contact will be checked at least once after the initial installation. Connections utilizing gaskets must be checked at scheduled intervals.

Component Identification - Age control of individual elastic type parts will be maintained by verifying that the cure data limits are not exceeded.

Pressure System Continuity - All tubing (including pneumatic, hydraulic, propellant systems, etc.) will be traced from end to end to assure that the system is properly installed and to determine if tubing and/or components have been damaged during or subsequent to installation.

Weighing - The complete assembly will be weighed by a single suspension load cell at the time of removal from the assembly fixture and after experiments have been installed. Electronic load cells will be of tension type and all operations will meet the basic requirements presented in procedure 6-OH-MA-5A. A weight log will be initiated following completion of the weighing operation. Weighing will require a maximum of one day.

Preparation of Shipment - Upon successful completion of checkout and tests at MSFC, the SSESM shall be prepared for shipment. The preparation for shipment shall be monitored and a final visual inspection performed to verify that the SSESM has been properly prepared for shipment to the test site.

Receiving Inspection at Vacuum Facility and KSC - Personnel shall be sent to the vacuum facility and KSC to perform receiving inspection and damage assessment of the SSESM and to witness unloading and preparation for the transfer of the SSESM to the vacuum chamber and launch site. Status information and test results gathered during fabrication, assembly, and checkout at MSFC shall be consolidated and transmitted to the vacuum facility and KSC test personnel prior to receipt of the SSESM itself.

Pressure Functional Analysis - The objective of performing the following pressure and functional tests on the mechanical systems of the support module is to assure the integrity and functional capability of the mechanical systems. The checkout shall be an operational test performed in a pressure cell.

1. Test to verify pressure switch operation, leak test at system pressure, check for internal and external leakage, verify actuation and deactuating pressure settings, and make a break repeatability of the switches.

2. Test to check minimum pressure required for component operation and relief settings of high pressure regulator and system relief valves. In addition, check response timing and repeatability of component operation; test system and components at normal system operating pressure, and repeatability of control component.

3. A leak check of all lines, fittings, and connections shall be conducted under a suitable test pressure using lead detector solution or tracer gas as applicable.

4. A support module functional analysis will be performed whereby pressure chambers are attached to each end of the module and side hatch, and the pressure differentials established sequentially in each module chamber to simulate operation in the space environment.

5. Heaters - apply power and cycle each unit three times measuring voltage, amperage, actuations-deactuations.

Environmental Checkout and Analysis in a Vacuum Chamber - Following the ambient analyses described above at MSFC, the SSESM will be shipped to a suitable man-rated vacuum chamber. A man shall enter the chamber and perform a complete functional and operational check of all mechanical and electromechanical systems including access hatches.

5.2.3 Systems Electrical Checkout

A comprehensive checkout program will be performed to verify satisfactory operation of the individual systems or subsystems, and to ensure that no problems result from interaction of the systems. Test operations will incorporate fail-safe provisions which assure return to a safe condition in the event of power failure or other emergency. The tests are summarized below.

Power Distribution Tests - These tests will verify the proper control and distribution of electrical power in the SSESM. Bus resistance measurements will be made prior to application of power, and the independence of the busses will be checked. Redundant circuitry will be verified, where possible.

Component Functional Tests - These tests will verify the operation of all controlled components from the instrument panels. In addition, the various inteface functions - S-IVB, CSM, and GSE, will be checked. Among the components tested are the gox Dump Valves, the acquisition light, the airlock lights, the oxygen heaters, the tank lights, the blowers, and the display panel.

Measuring System Tests - These tests are performed to verify the calibration of all transducers and signal conditioners on the SSESM and to assure conformance to proper channel assignments. In all practicable cases, the systems will be operated or stimulated for every flight measurement.

Telemetry System Tests - These tests will determine that the telemetry system operates in compliance with applicable specifications while installed in the SSESM and controlled by its electrical networks. Calibration of sub-carrier oscillators will be checked and adjusted as necessary.

Experiment Tests - These tests will be conducted primarily to verify the interface between the experiments and the SSESM systems. These tests may include power distribution, operation of the experiment or portions thereof, and retrieval of experiment data through the SSESM telemetry systems.

Electromagnetic Compatibility (EMC) Tests - These tests will essentially parallel other testing operations and will determine if all electrical, electronic, and electromechanical systems and subsystems will operate, both individually and simultaneously, without degraded performance due to EMC.

5.2.4 Facilities

Facilities and inspection stations presently exist that would provide adequate space for performing all tests and inspections except as required for vacuum chamber testing.

5.2.5 Schedule

A schedule is presented in Figure 5.2-1 summarizing the operations associated with the Quality and Reliability Assurance Plan.

Q AND RAL OPERATIONS FOR SPENT STAGE EXPERIMENT SUPPORT MODULE

FIGURE 5.2-1

SECTION VI. RESOURCES AND SCHEDULES

6.1 RESOURCE REQUIREMENTS

This section defines the resources required by MSFC to design, fabricate and test the SSESM design described in this proposal. All MSFC manpower required for the SSESM will be available within the current manpower allotments of the organizations involved, with assignments being made within each area of responsibility as required to support the proposed effort. As the project status advances to include systems and operational support, it is planned that personnel participation will increase and the additional manpower required will be phased in from existing personnel.

The resources requirements peak during the following phases: Tool fabrication, parts fabrication, and structural assembly, which all occur during the 2nd and 3rd quarters of FY-67; manpower requirements will extend into the 3rd and 4th quarters of FY-68 for the purpose of reduction of flight data and mission reporting.

The total material cost of the proposed SSESM is $3, 275, 000. This cost includes:

1. Mechanical GSE.
2. Installation Hardware and Testing.
3. Structural Components for Testing.
4. Vibration and Acoustical Test.
5. Sphere Development.
6. LSS Components Development.
7. ECS for Test and Flight Article.
8. Experiment Installation.
9. Electrical Power System.
10. Control Panels.

11. Integration of Experiments.

12. Modification of Existing Consoles, Distributors.

13. Electrical Simulators for CSM and S-IVB.

14. Miscellaneous Cables and Break-In Boxes.

15. Weight and Alignment Tooling.

16. Handling and Pressure Test Fixtures.

17. Hardware for Mock-up, Test Article, and Flight Article.

The hardware cost of the zero-g mock-up is $87.5K, the test article is $138K, and the flight article is $138K. The test article is identical to the flight article except for some qualified hardware.

The material cost for an additional flight article would be $743,000. This cost includes:

1. Hardware for Flight Article.

2. Environmental Control System.

3. Electrical Power System.

4. Life Support Equipment.

5. Integration of Experiments.

These costs do not include experiment procurement.

Refer to the following tables and figures for a further breakdown of the resource requirements:

1. Table 6.1-1 - Resource Requirements by Laboratory

2. Table 6.1-2 - Resource Requirements by Article and Function

3. Figure 6.1-1 - Manpower Requirements

4. Figure 6.1-2 - Material Requirements

TABLE 6.1-1

RESOURCES REQUIREMENTS BY LABORATORY

	FY - 1966			FY - 1967			FY - 1968			T O T A L		
LAB	C.S. (MY)	S.C. (MY)	MAT. ($x10^3)	C.S. (MY)	S.C. (MY)	MAT. ($x10^3)	C.S. (MY)	S.C. (MY)	MAT. ($x10^3)	C.S. (MY)	S.C. (MY)	MAT. ($x10^3)
R-ME	1.8	1		105	7.5	649	14.3	1		121.1	9.5	649
R-TEST	.3	.8		15.8	15.8		2	2	50	18.1	18.6	50
R-P&VE	7.8	8.5	150	26.5	49.8	1,646	12.3	20.3		46.6	78.6	1,796
R-AERO	1.5	1.3	--	17.3	11.3	--	6	4		24.8	16.6	---
R-QUAL	1.5	--	--	20.5	--	132	8.8	--	--	30.8	--	132
R-ASTR	2.5	1.2	155	7.8	3.7	470	1.6	1.2	23	11.9	6.1	648
T O T A L	15.4	12.8	305	192.9	88.1	2,897	45	28.5	73	253.3	129.4	3,275

C.S. ----Civil Service
S.C. ----Support Contractor
MAT. ----Materials
MY ----Man-Years

Additional Flight Article would cost $743,000

TABLE 6.1-2

RESOURCES REQUIREMENTS BY ARTICLE AND FUNCTION

I T E M	T O T A L S		
	CIVIL SERVICE MAN YEARS	SUPPORT CONTRACTOR MAN YEARS	MATERIALS $ THOUSANDS
RDT&E	84	84.4	1,316
MOCK-UP	16.2	1.7	123
TEST & TRNG ARTICLE	55	9.9	795
FLIGHT ARTICLE	57	9.1	919
INTEGRATION OF EXPERIMENTS	12.1	7.7	122
MISSION PLANNING	29	16.6	--
T O T A L S	253.3	129.4	3,275

Additional Flight Article would cost
$743,000

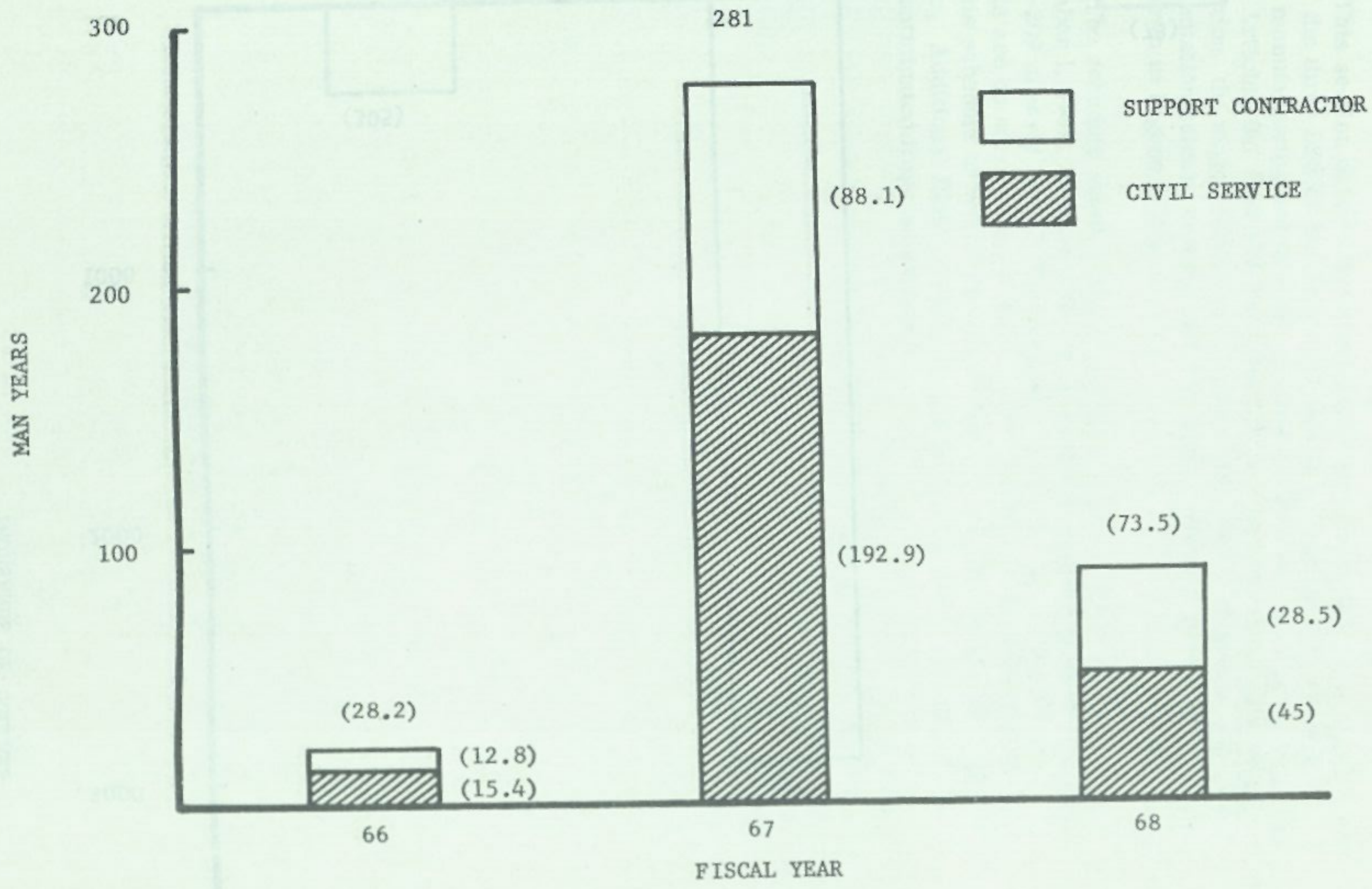


FIGURE 6.1-1. MANPOWER REQUIREMENTS

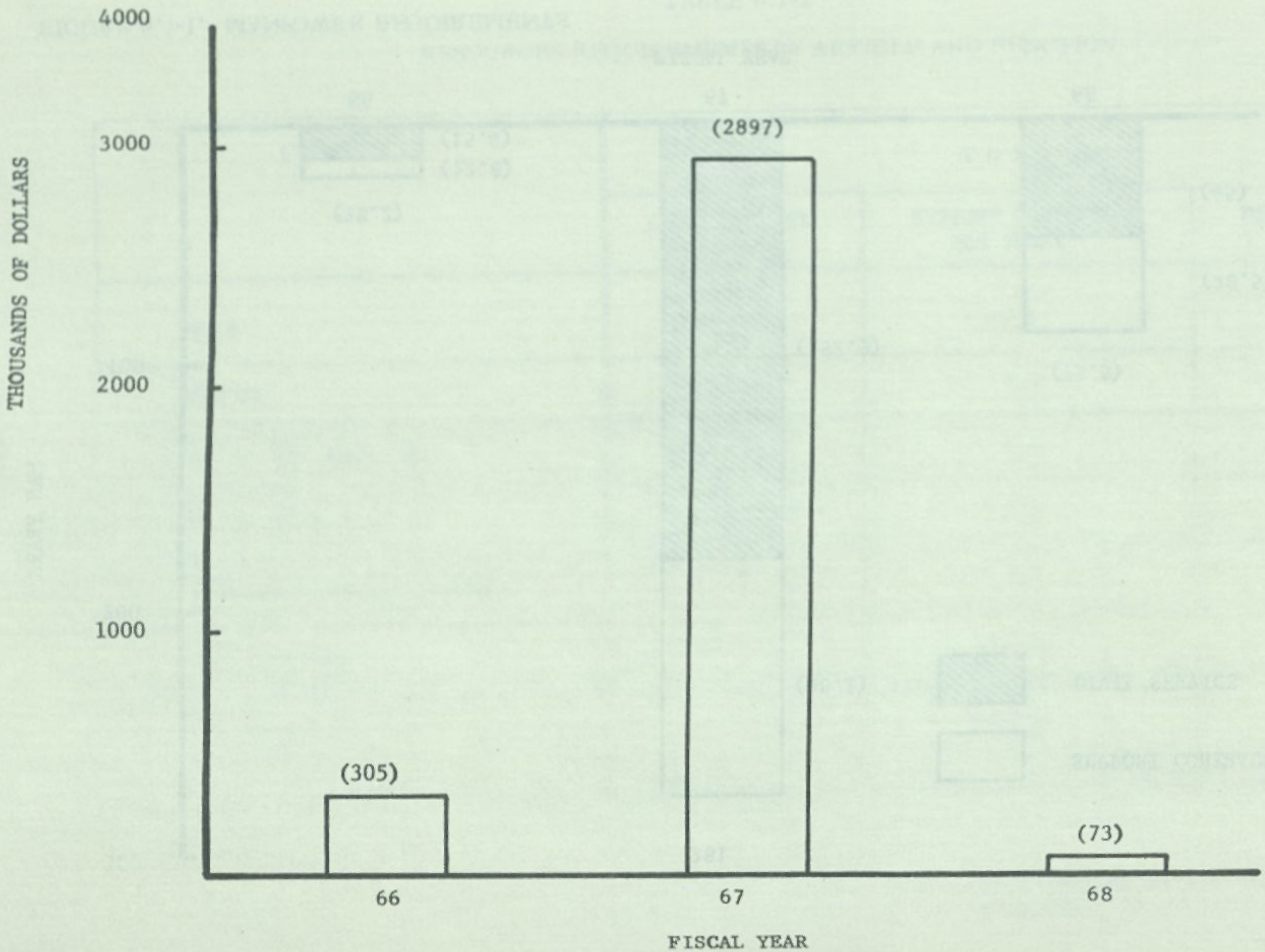


FIGURE 6.1-2. MATERIAL REQUIREMENTS

6.2 SCHEDULE

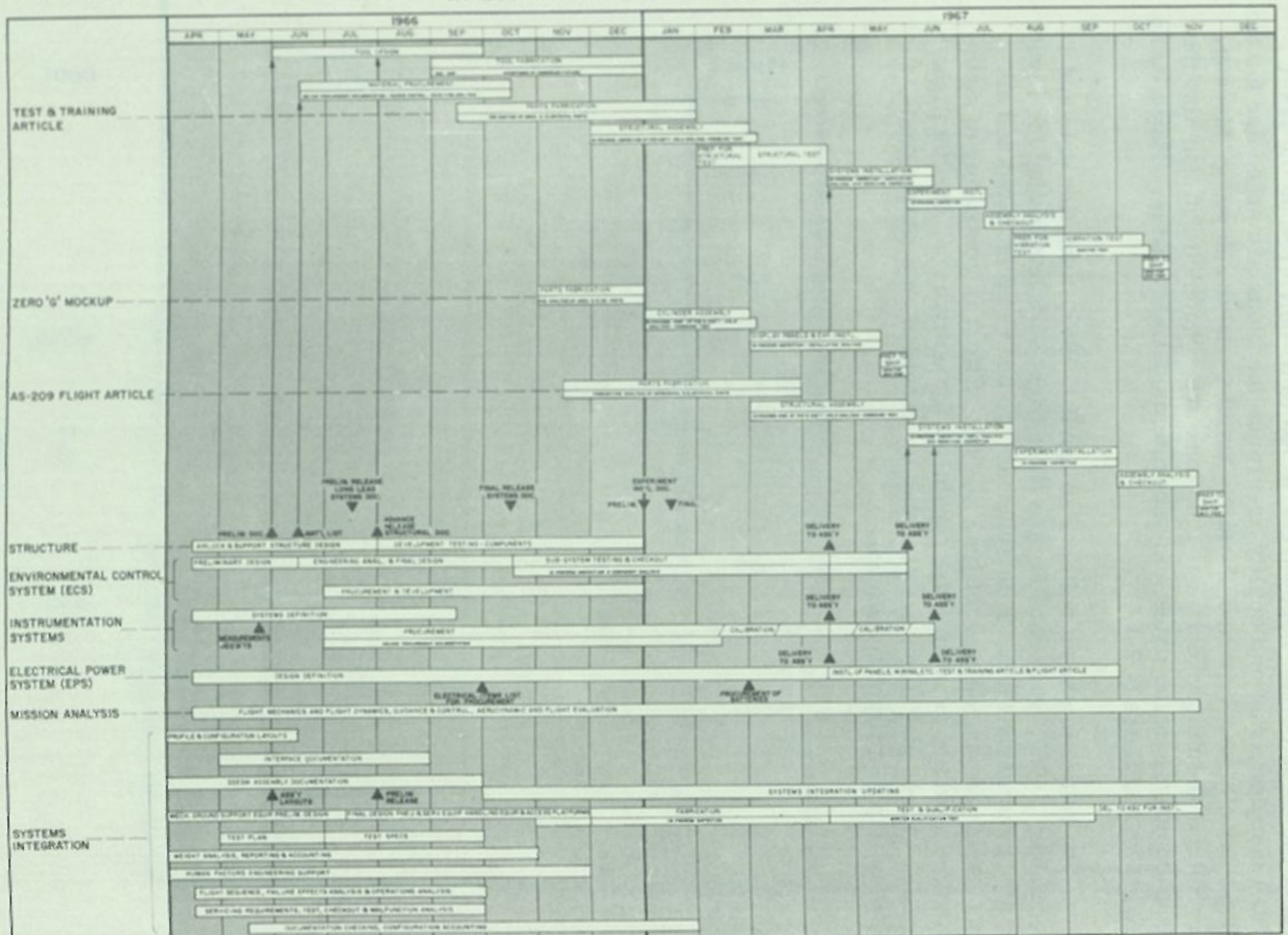
This section defines the overall schedule for design, fabrication, and test of the three SSESM hardware articles to be delivered by the MSFC. Detail manufacturing and quality assurance functions are scheduled for the Flight Article, the Test and Training Article, and the Zero "g" Mockup. In addition, the major technical functions are scheduled defining also the documentation release dates and subsystems delivery dates. These items are shown in Figure 6.2-1.

The schedule shown reflects an April 1, 1966, initiation date and a December 1, 1967, ship date for the first flight article allowing for flight on AS-209 in early 1968. This schedule has been implemented and all aspects are on schedule as of June 1, 1966. Effort is continuing to maintain this schedule to assure the capability for delivery as AS-209 flight article. Additional flight articles can be provided on a timely basis to meet anticipated flight schedules.

FIGURE 6.2-1

SCHEDULE

AS-209 SSESM IN-HOUSE MSFC CONCEPT



SECTION VII. MANAGEMENT PLAN

7.1 SCOPE

This section briefly outlines the major management functions and organizational interfaces involved with the development of the SSESM. These interfaces are shown in the function chart, Figure 7.1-1. It shows how MSFC will maintain control of the various SSESM activities.

7.2 MANAGEMENT RESPONSIBILITIES

MSFC will maintain overall responsibility for the design, fabrication, and assembly, and test of the SSESM. These activities would be in addition to the MSFC responsibility for the overall planning, systems design, and integration for the S-IVB Spent Stage Experiment. Overall MSFC responsibility for the SSESM development will be vested in the Industrial Operations (IO) Saturn/Apollo Applications Office (S/AA). However, the actual development will be accomplished by the various Research & Development Operations (R&DO) Laboratories through a technical control element responsible to the Director, R&DO.

The S/AA Office will be responsible for overall funding and scheduling for the S-IVB Spent Stage Experiment. The office will also be responsible for all inter-Center and NASA Headquarters management interfaces. The S/AA Office will relay funding and overall direction applicable to the development of the SSESM to the designated technical control element representing R&DO. This technical control element will exercise technical and resources management for the SSESM development among the various R&DO Laboratories. This element will also establish and maintain the funding distribution, schedules, and technical interfaces.

Normal technical interfaces among the three involved Centers will be handled through the existing Apollo Interface Panels. Specific MSFC technical data requirements on the procurement of Gemini or Apollo components will be submitted to the applicable MSC program office through the MSFC S/AA Office.

7.3 FUNCTIONAL RESPONSIBILITIES

Design, development, and manufacturing of the SSESM will be accomplished within the MSFC R&DO Laboratories. Identification of the overall

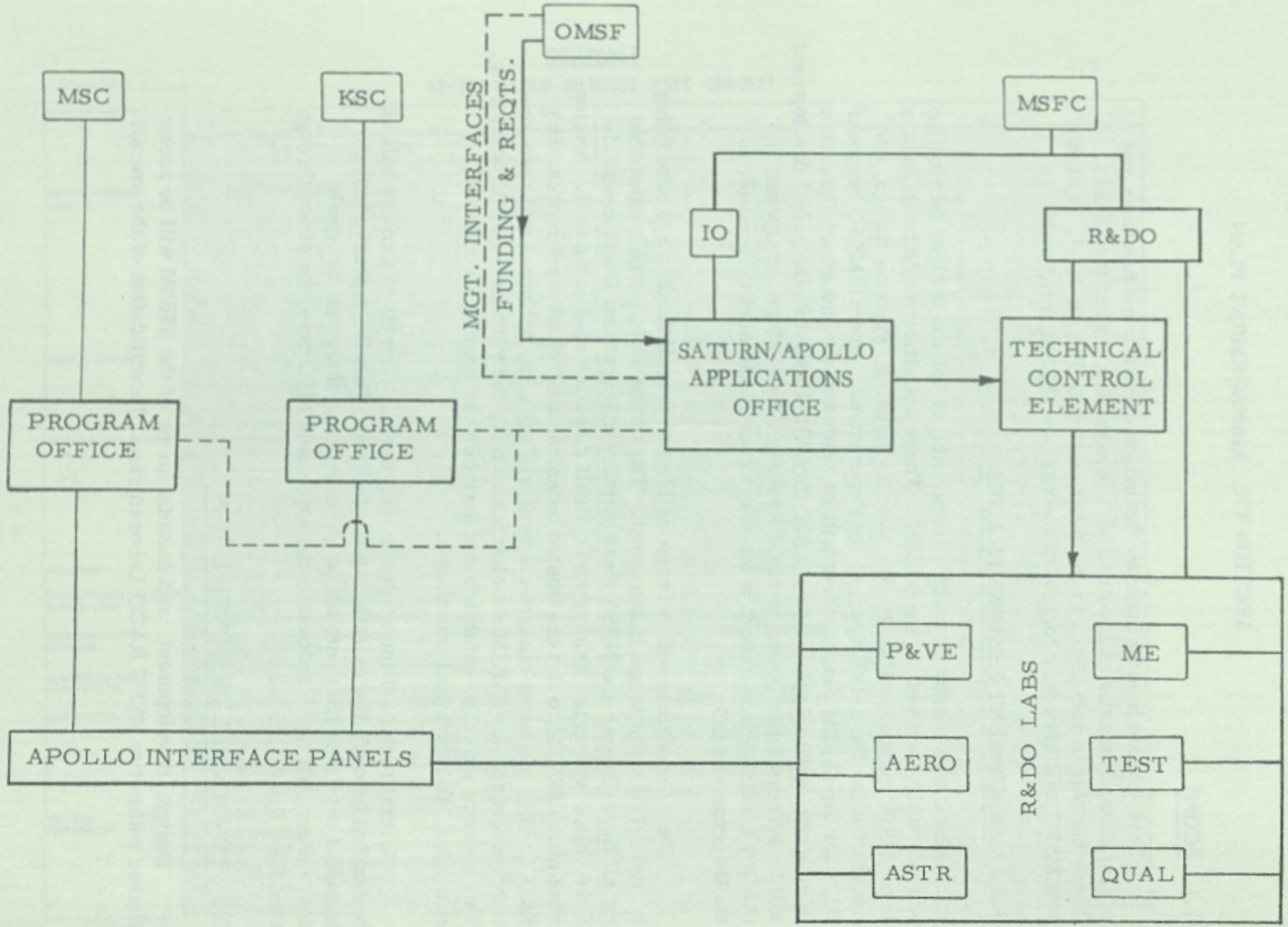


FIGURE 7.1-1. SESSM ORGANIZATIONAL INTERFACES

functional responsibilities among the various laboratories is shown on Table 7.2-1. Each area has been further subdivided into the various sub-element tasks. These tasks are then assigned through the normal organizational arrangement: Division, Branch, and Section. Experienced technical and management personnel are available at all these levels to accomplish these tasks, and additional technical Laboratories and Divisions are available to support this program in specific technical areas as requirements develop.

TABLE 7.2-1

MSFC LABORATORY FUNCTIONAL RESPONSIBILITIES

PROPULSION & VEHICLE ENGINEERING LABORATORY

1. Vehicle Systems Division: Systems engineering

2. Structures Division: Structural design

3. Propulsion Division: Environmental control subsystem

4. Materials Division: Adaptation and selection of materials

AERO-ASTRODYNAMICS LABORATORY

1. Dynamic & Flight Mechanics Division: Flight mechanics and dynamics analyses

2. Flight Test Analysis Division: Flight evaluation

ASTRIONICS LABORATORY

1. Instrumentation & Communications Division: TM instrumentation and communication; voice communication

2. Electrical Systems Integration Division: Electrical systems (power, networks, control panels); electrical support equipment; lighting system design

MANUFACTURING ENGINEERING LABORATORY

1. Planning and Tool Division: Tool planning, design and processing; planning and processing of integration effort.

2. Manufacturing Development Division: Module and tooling manufacture; experiment integration.

3. Industrial Support Branch: Component procurement.

TABLE 7.2-1 (Cont'd.)

TEST LABORATORY

1. Components and Subsystems Division: Operation of altitude chamber.

2. Control and Instrumentation Division: Special instrumentation development.

QUALITY & RELIABILITY LABORATORY

1. Source control and receiving and inspection.
2. Fabrication analysis.
3. In process inspection.
4. Assembled SSESM checkout.
5. End item assembly analysis.
6. Component and subassembly checkout.
7. Failure effects analysis.

SECTION VIII. ALTERNATE DESIGNS AND SYSTEM FLEXIBILITY

8.1 GENERAL

The basic SSESM design proposed has, as previously described, substantial capabilities for supporting crew and experiments for up to a 20-day mission. To maintain simplicity and low program cost this approach requires no major interfaces with the CSM and consequently makes no provisions for extending the CSM life time beyond its inherent capability. The basic design does lend itself to alternate approaches, flexibility for growth, sophistication and flexibility for mission extension. These adaptations can be accomplished by several methods and in all instances utilize the nucleus of the basic design. The alternate designs, with predominate emphasis upon ECS and fuel cell reactant storage alternates, are discussed in the following paragraphs.

8.2 30-DAY SYSTEM CONCEPT

8.2.1 Approach

This design approach, proposed by MSC, constitutes supply fuel cell reactants and life support oxygen to the CSM by a fluid umbilical attached to the CSM service umbilical, and primary electrical energy furnished to the SSESM power distribution system from the CSM. Elimination of primary power sources on the SSESM permits the installation of cryogenic containers. The required umbilical attachments would be accomplished by EVA.

8.2.2 ECS

As previously noted and implied the "slug" ECS concept is proposed on the 20-day concept for reasons which include the following:

1. 209 schedule adherence;
2. Low costs;
3. Simplicity;
4. Reliability;
5. Minimum hardware with maximum utility of available components;
6. Weight capability permits non-optimized approach.

However, extended missions deem the slug leakage contaminate control concept to be excessively weight penalizing for desirable astronaut lab occupancy schedules. A comparison of an atmospheric revitalization system with the slug approach, shown in Figure 8.2-1 determines a respective weight requirement of 575 and 1950 pounds for an astronaut occupancy cycle of 4/11 hours. A similar comparison for the 20-day system shown in Figure 8.2-2 shows the slug to be acceptably competitive for the shorter mission. The 1375 pound weight penalty for 30 days is considered excessive thereby establishing the need for atmospheric revitalization. Existing flight equipment such as the Gemini ECS suit loop module would be employed. This module would be stripped of excess equipment and would serve only to remove CO_2 from the Lab/airlock volume. The heat exchanger normally used for water vapor removal would be inactive due to the absence of SSESM active coolant loops. The silica gel concept of H_2O removal used on the 20-day concept would remain.

8.2.3 Electrical Power System

Changes to the basic 20-day SSESM power system design will involve the removal of most batteries which now supply power to the SSESM. The remaining batteries will be used to satisfy telemetry requirements from liftoff to docking and emergency power source requirements during orbit. The retained batteries will provide power for operation of the motor-driven switch and the SM umbilical separation circuits without sending control power through the CM/SSESM interface. The SM umbilical separation circuits may be operated directly through a modified SSESM control panel utilizing SSESM battery power (see Figure 8.2-3).

8.2.4 Cryogenic Storage

For the extended mission the life capability of the cryogenic vessel becomes a formidable concern. Analyses were performed to determine the amount of cryogen remaining in the 14-day Gemini RSS vessels at various mission periods. These vessels were initially considered primarily due to potential spare/test equipment availability. As can be observed in Figure 8.2-4, relief venting causes the stored mass to decrease quite rapidly although mission duration was initiated with a one atmosphere ullage pressure. The hydrogen vessel shows a maximum mission capability of 20 days obviously establishing the need for potential solution investigation. These potential solutions include:

1. Flow management between the CSM and SSESM;
2. Radiation shields;
3. Boil-off shields;
4. Slush;
5. Improved cryogenic insulation for storage vessel.

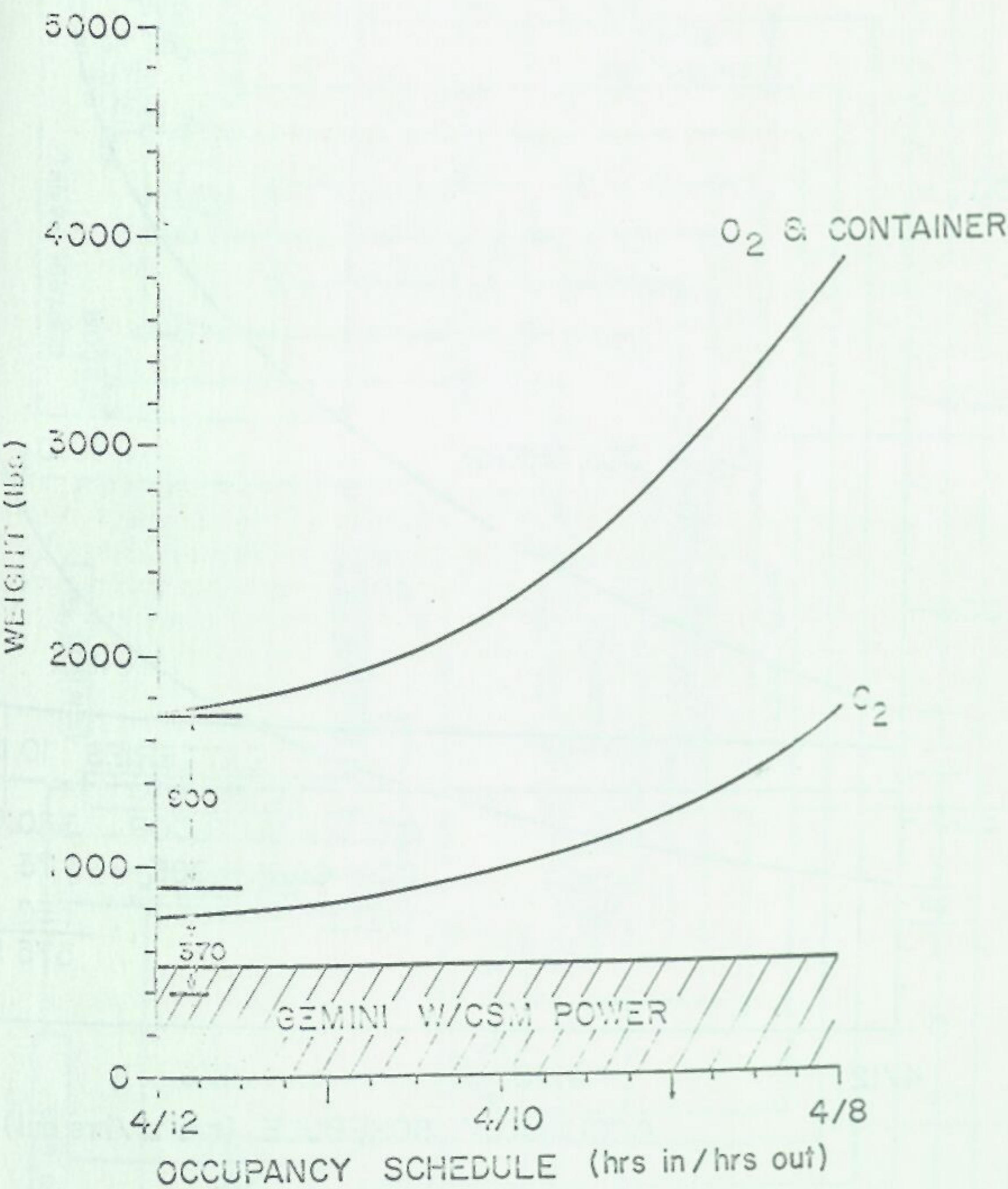


FIGURE 8.2-1 ATMOSPHERIC REVITALIZATION VS. SLUG CONCEPT- 30 DAY MISSION

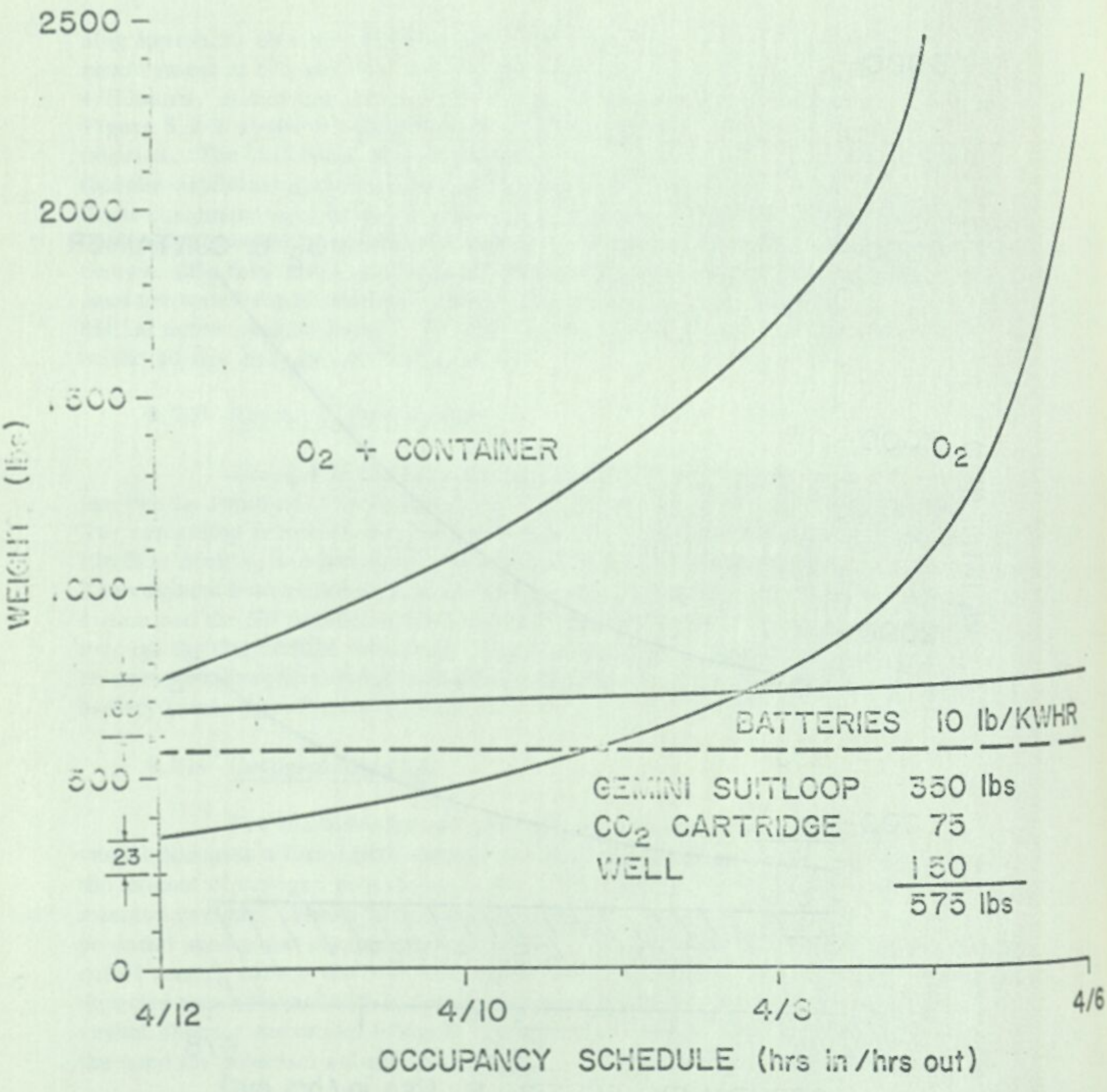
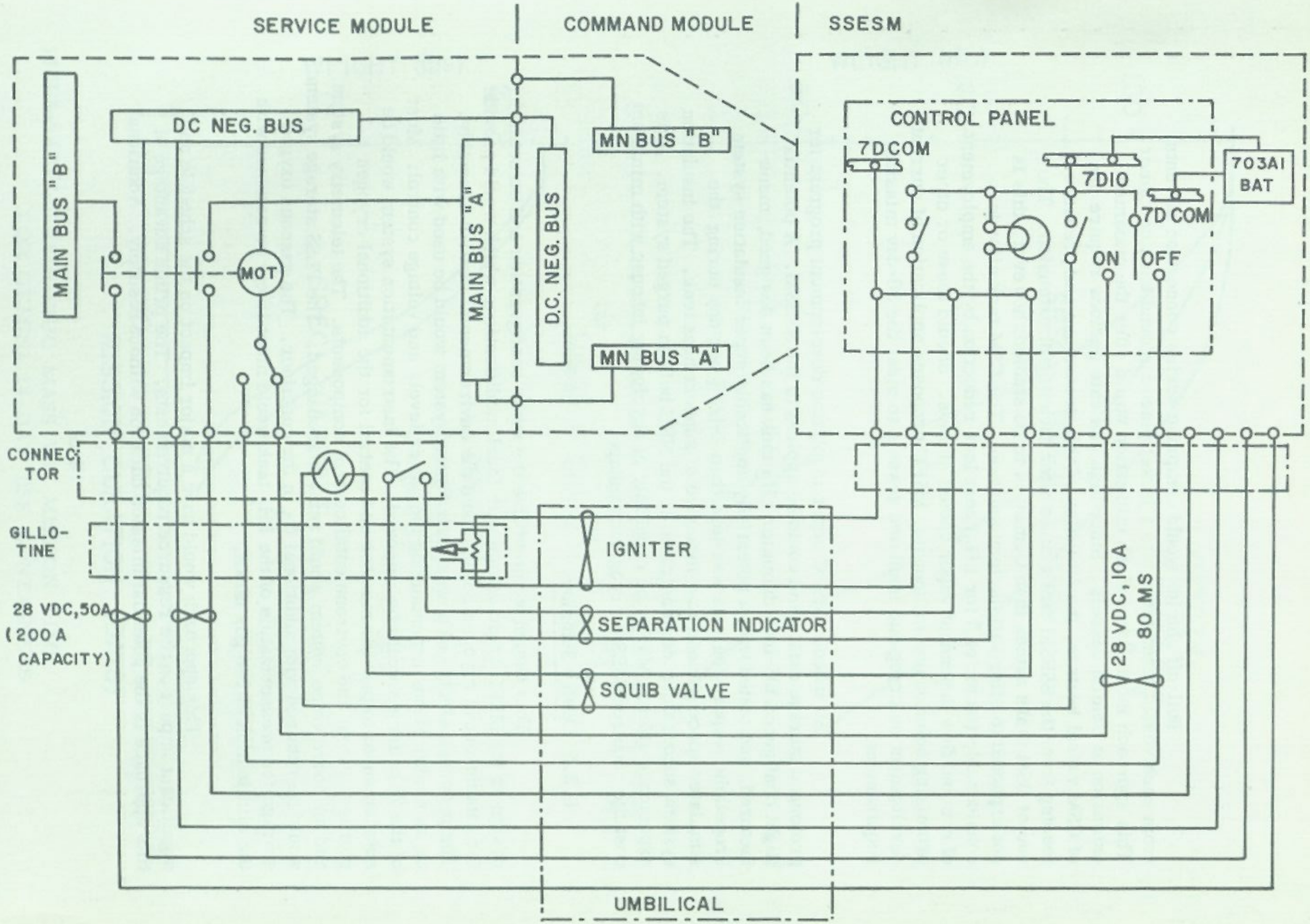


FIGURE 8.2-2 ATMOSPHERIC REVITALIZATION VS. SLUG CONCEPT - 20 DAY MISSION

FIGURE 8.2-3 **ELECTRICAL SCHEMATIC - CSM/SSESM UMBILICAL**



Boil-off shields would comprise series connection of vents from each vessel terminated into a jacket placed around each container. This approach inclusive of slush utilization would offer the maximum extension of Gemini vessels. Since high use rate periods require use of CSM vessel heaters, the omission of this heater use and pressure feeding from the SSESM tanks could offer meaningful extension. The use of CSM tanks rather than Gemini is to be studied; however, this is not expected to offer satisfactory solution. The CSM tank affords considerable (factor of 3 for LH_2) heat leak reduction by the employment of demand flow dependent vapor cooled shields. Should these or other promising solutions be inadequate, MSFC proposes application of current development on cryogenic insulated vessels to meet the 30-day mission requirement.

An inhouse MSFC super insolation development program for prolonged storage exists and could be applied to the SSESM. A potential flight configured 105-inch diameter LH_2 tank has been designed, manufactured, and tested with a potentially applicable super insulation system. Propellant evaporation rate was less than 2-1/2% per day during the simulated space vacuum test inclusive of support heat leak. The insulation system successfully demonstrated is the NRC helium purged system. This experience affords MSFC the capability of developing inhouse with minimum cost the required SSESM cryogenic tankage.

8.2.5 Design Summary

This design would utilize the same configuration and structural design as the MSFC basic design with local modifications which would provide for attachment of cryogen bottles, and the environmental control assembly. The proposed electrical power distribution system would be used with little or no modifications depending on the power levels and voltage control. Most of the 21 batteries would be removed. The instrumentation system would be supplemented to provide monitor and control for the additional cryogen storage system and environmental control components. The telemetry system and on-board voice system would remain unchanged. The PLSS storage system would be retained with additional O_2 in the container. The gaseous oxygen storage for pressurization of the LH_2 tank would most likely be retained with the elimination of two gox tanks.

This approach would have a major impact on the schedule and significant impact on the resource requirements. The prime advantage of this approach is the potential mission duration without resupply. Additional

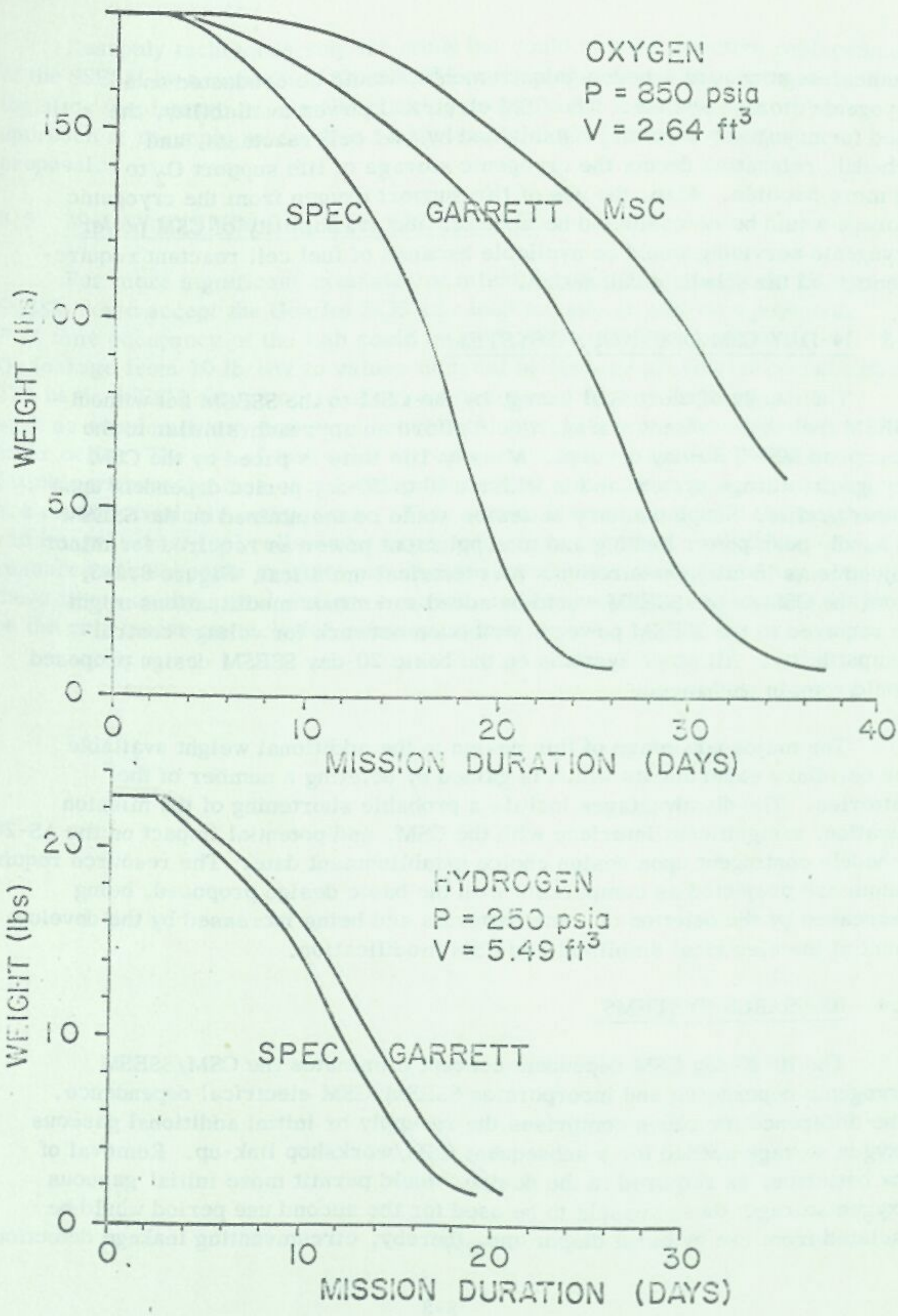


FIGURE 8.2-4 CRYOGENIC MASS VS. MISSION TIME FOR GEMINI 14-DAY RSS CONTAINERS

studies, cognizant of schedule requirements, would be conducted on a cryogenic storage system. The CSM electrical power availability, the need for cryogenic servicing established by fuel cell reactants, and schedule relaxation deems the cryogenic storage of life support O_2 to be more feasible. Also, the use of life support oxygen from the cryogenic storage would be re-evaluated because of: the availablility of CSM power; cryogenic servicing would be available because of fuel cell reactant requirements; and the schedule relaxation.

8.3 14-DAY CSM DEPENDENT SYSTEM

The supply of electrical energy by the CSM to the SSESM but without SSESM fuel cell reactant storage would afford an approach similar to the described MSFC 20-day concept. Mission life time is paced by the CSM cryogenic storage system and is within a 10 to 20-day period dependent upon power profile. Supplementary batteries would be maintained on the SSESM to handle peak power loading and to supplement power as required for minor adjustments in mission duration. An electrical umbilical, Figure 8.2-3, from the CSM to the SSESM would be added and minor modifications might be required to the SSESM power distribution network for voltage control compatibility. All other systems on the basic 20-day SSESM design proposed would remain unchanged.

The major advantage of this design is the additional weight available for corollary experiments which is gained by deleting a number of the batteries. The disadvantages include a probable shortening of the mission duration, a significant interface with the CSM, and potential impact on the AS-209 schedule contingent upon design choice establishment date. The resource requirements are projected as comparable with the basic design proposed, being decreased by the deletion of some batteries and being increased by the development of the electrical umbilical and CSM modification.

8.4 REUSABLE SYSTEMS

The 10-20 day CSM dependent concept eliminates the CSM/SSESM cryogenic dependence and incorporates SSESM/CSM electrical dependence. The difference for reuse comprises the resupply or initial additional gaseous oxygen storage needed for a subsequent CSM/workshop link-up. Removal of the batteries, as required in the design, would permit more initial gaseous oxygen storage; these vessels to be used for the second use period would be isolated from use by burst diaphrams, thereby, circumventing leakage detection.

Resupply techniques require study but could include effective replacement of the SSESM by a version of the SSESM containing gaseous oxygen vessels, logistics furnishings, and a CM docking probe on the lower section. This approach at resupply would also be applicable to the basic 20-day design proposal.

8.5 20-DAY SYSTEM GROWTH FLEXIBILITY

For more significant extension of mission capability, the basic 20-day SSESM could accept the Gemini ECS suit loop module as add-on equipment. Full time occupancy of the Lab could be realized in addition to reduction of O_2 leakage from 30 lb/day to values dictated by leakage prevention techniques. The basic SSESM design could also accept a power system sophistication such as supplementary power and recharge capability from the addition of solar cells. The use of fuel cells to replace the battery system would be feasible without the loss of a development effort since the proposed batteries are readily available components. This would, however, require the purchase and integration of fuel cells and a cryogenic storage system. Significant resource and schedule impacts are anticipated with the inclusion of any of these items on the initial design; however, it is expected that resources would be the only major obstacle for subsequent missions.