NASA TECHNICAL MEMORANDUM

October 1974

NASA TM X-64824

(NASA-TM-X-64824) MSFC SKYLAB STRUCTURES AND MECHANICAL SYSTEMS MISSION EVALUATION (NASA) 348 p HC $9.50 CSCL 22C

N75-10968

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G3/12 53892

MSFC SKYLAB STRUCTURES AND MECHANICAL SYSTEMS MISSION EVALUATION

Skylab Program Office

NASA

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

TECHNICAL REPORT STANDARD TITLE PAGE

1. REPORT NO.	2. GOVERNMENT ACCESSION NO.	3. RECIPIENT'S CATALOG NO.
NASA TMX-64824		

4. TITLE AND SUBTITLE	5. REPORT DATE
MSFC Skylab Structures and Mechanical Systems Mission Evaluation	October 1974
	6. PERFORMING ORGANIZATION CODE
7. AUTHOR(S)	8. PERFORMING ORGANIZATION REPORT #
9. PERFORMING ORGANIZATION NAME AND ADDRESS	10. WORK UNIT NO.
George C. Marshall Space Flight Center Marshall Space Flight Center, Alabama 35812	11. CONTRACT OR GRANT NO.
	13. TYPE OF REPORT & PERIOD COVERED
12. SPONSORING AGENCY NAME AND ADDRESS	Technical Memorandum
National Aeronautics and Space Administration Washington, D. C., 20546	14. SPONSORING AGENCY CODE

15. SUPPLEMENTARY NOTES

Prepared by Structures and Propulsion Laboratory, Structures Division

16. ABSTRACT This report presents a performance analysis for structural and mechanical major hardware systems and components. Development background testing, modifications, and requirement adjustments are presented where considered pertinent.

Functional narratives are provided for comparison purposes as are predicted design performance criterion. Each item is evaluated on an individual basis: that is, (1) history (requirements, design, manufacture, and test); (2) in-orbit performance (description and analysis); and (3) conclusions and recommendations regarding future space hardware application.

Overall, the structural and mechanical performance of the Skylab hardware was outstanding. There were anomalies, some expected, some unexpected, but none of such consequence that on-orbit workarounds and/or new hardware development did not solve the problems.

EDITOR'S NOTE

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17. KEY WORDS	18. DISTRIBUTION STATEMENT
	UNCLASSIFIED - UNLIMITED R. G Eudy R. G Eudy

19. SECURITY CLASSIF. (of this report)	20. SECURITY CLASSIF. (of this page)	21. NO. OF PAGES	22. PRICE
UNCLASSIFIED	UNCLASSIFIED	348	NTIS

MSFC - Form 3292 (Rev December 1972)

For sale by National Technical Information Service, Springfield, Virginia 22151

ACKNOWLEDGEMENTS

The data used to compile this report were furnished by the George C. Marshall Space Flight Center Astronautics Laboratory, the McDonnell Douglas Astronautics Company (Eastern and Western Divisions), the Martin Marietta Corporation, and the Teledyne Brown Engineering, Aerospace Support Division.

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SECTION I. SUMMARY

A. Skylab Requirements and Configuration

Skylab structural and mechanical design considered operational requirements including the natural and imposed environmental conditions during transportation, handling, prelaunch, launch, ascent, and manned and unmanned space flight phases of the Skylab mission. Shirt sleeve and hard-suited operations internal to Skylab and hard-suited operations external to it were planned.

Control of internal noise levels, internal and external contamination, radiation transmission through the structure, and venting and dumping operations on-orbit was required. Design criteria to minimize the probability of meteoroid penetration were utilized. Windows were required for flight crew external viewing, experiment operations, and photography. Two scientific airlocks to support experiment operations were required, one with a solar orientation and one with an anti-solar orientation. A trash airlock that would allow waste disposal without compromising pressure vessel integrity was also required. Crew mobility and stability aids, including work station platforms, were required both internally and externally.

The as-flown Skylab (dry workshop) evolved from the wet workshop configuration initially planned: the docking adapter, the telescope mount, and the airlock. In order for the wet workshop (Figure I-1) to achieve the orbital altitude and inclination required, it was necessary to load the S-IVB stage with fuel and oxidizer; hence, the wet workshop designation. A flow-through gridwork floor, gridwork partitions fold-down brackets and beta cloth liners were developed.

With the availability of a Saturn V vehicle, a payload increase was made possible and the wet workshop evolved into the as-flown Skylab (Figure I-2). Structural and mechanical modifications were required for the workshop, docking adapter, and airlock. Resupply was not required. Food, water, clothing, and other crew expendables were stowed in the workshop. The fixed airlock shroud, in addition to providing load bearing structure, was designed with pressure vessels to contain atmospheric gases.

Docking requirements were reduced and the docking adapter evolved into a two-port configuration. It also became a control center for the telescope mount. In addition, earth observation experiments, control panels, and tool and miscellaneous spares containers were installed.

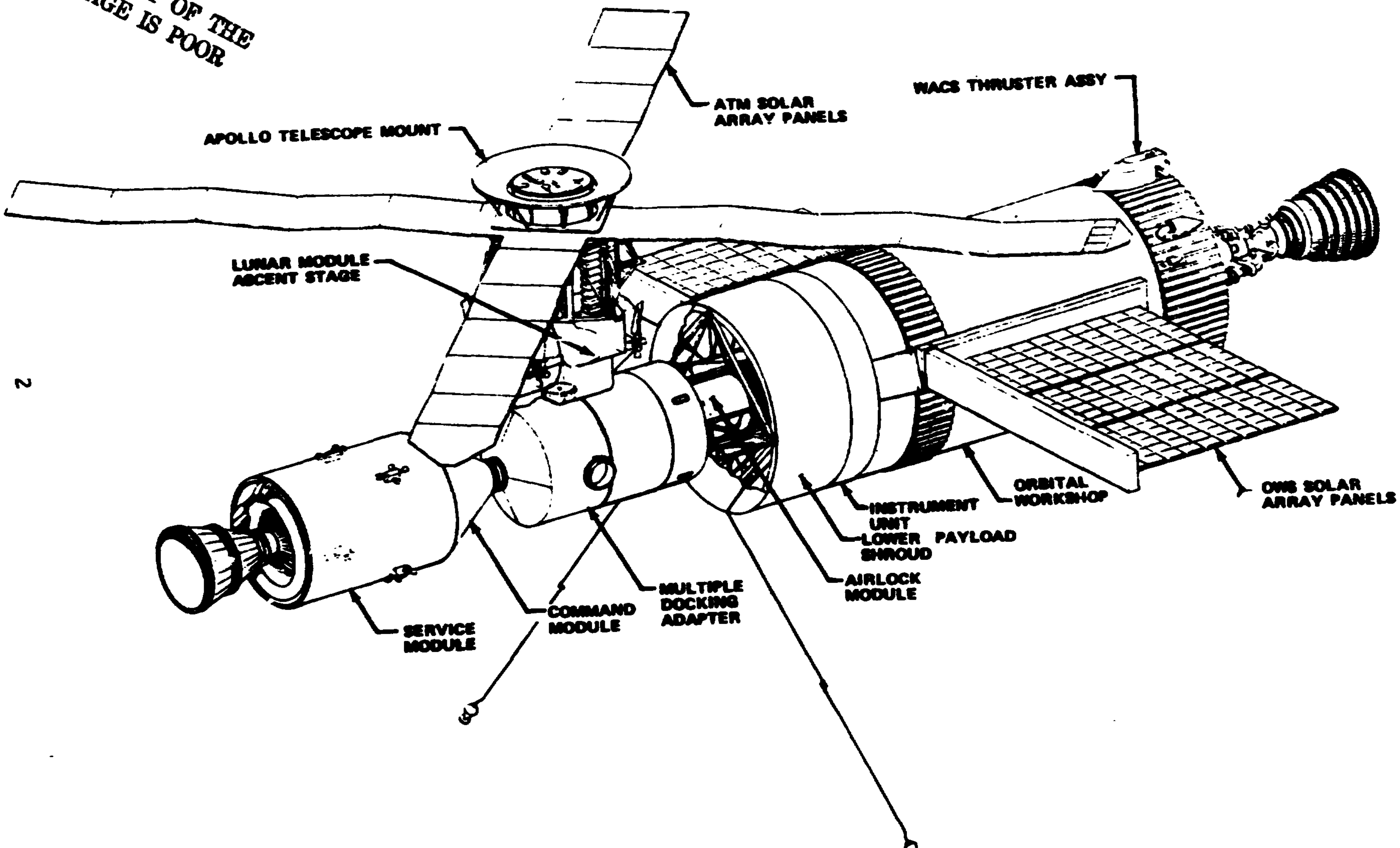


Figure I-1. Wet Workshop

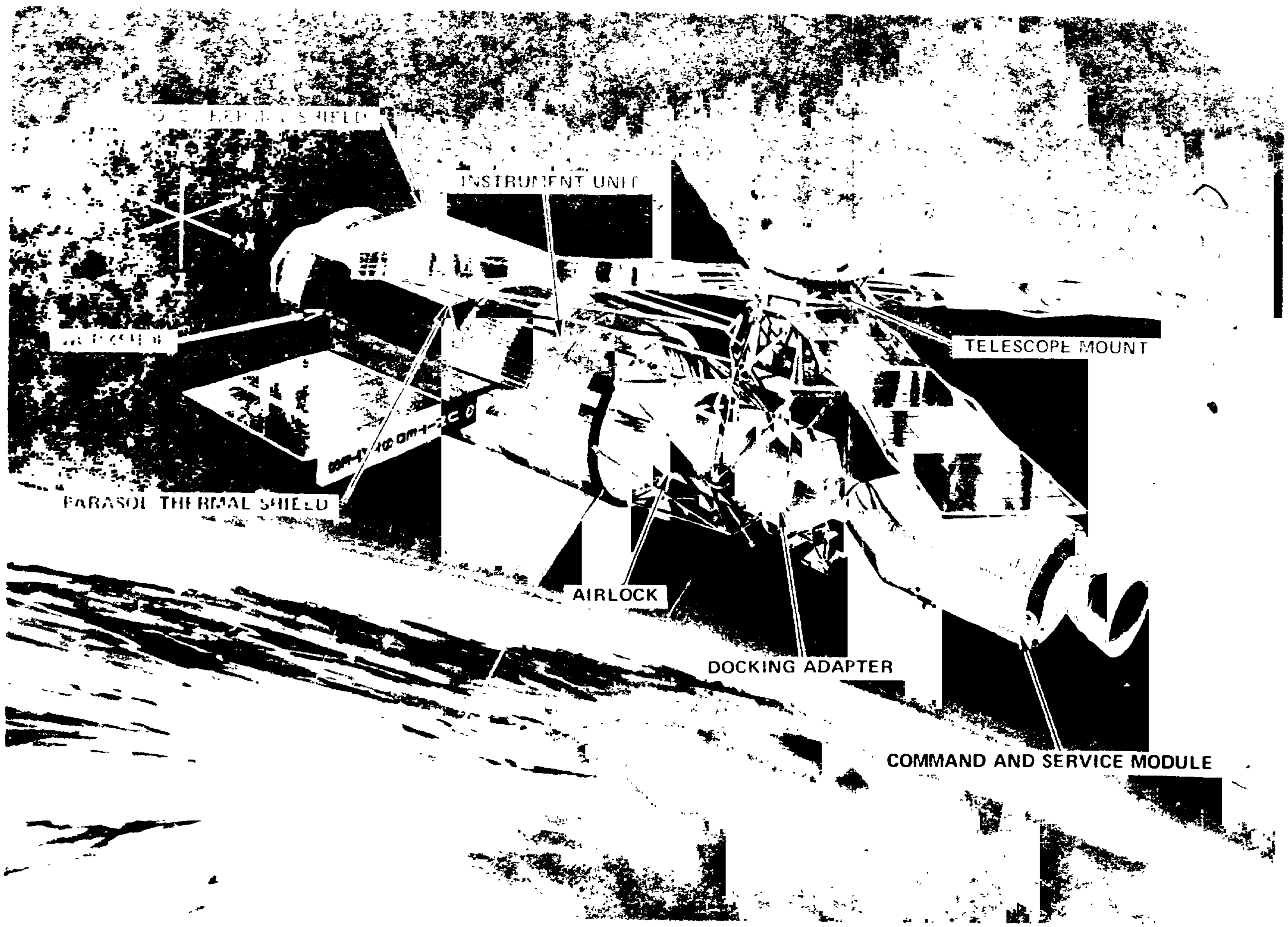


Figure I-2 As-Flown Skylab

The payload shroud was redesigned longer, with increased structural strength and improved capability, to attenuate sound pressure for protection of the enclosed modules. The combined fixed airlock shroud and payload shroud enclosed all Skylab hardware except the instrumentation unit and workshop from the time of stacking until the payload shroud was jettisoned on orbit.

B. Structures and Mechanical Hardware Synopsis

The Skylab mission consisted of four launches: SL-1, the unmanned Saturn Workshop launched by a Saturn V vehicle; SL-2; SL-3; and SL-4, three crews of three men launched by Saturn IB vehicles. The first manned phase was delayed because of loss of the workshop meteoroid shield, loss of workshop solar array wing 2, and failure of workshop solar array wing 1 to deploy. During the mission, other anomalies had smaller, but significant impact relative to ground support by the structures/mechanical group as well as other support groups, including such items as malfunctioning rate gyros which required development of a stable platform for rate gyro six-pack installation; coolanol leakage which required development of coolant reservicing equipment; and sticking thermal control valves which required development of a heating kit for use in the suit cooling water loop.

Workshop meteoroid shield design was inadequate. Lack of atmospheric pressure relief between the meteoriod shield and the workshop skin forced the meteoroid shield out of the boundary layer and exposed its leading edges to the slipstream. At approximately 63 sec after lift-off, Mach 1, and 8,700 m altitude, atmospheric drag tore the meteoroid shield from its mountings. While tearing free, the meteoroid shield disturbed the mountings of solar array wing 2 and wrapped a strap of debris over solar array wing 1 which later prevented its deployment. When the Saturn S-II stage retrorockets were fired, wing 2 was literally "blown" into space. The S-II retro-rocket firing caused an immediate rise in solar array system wing 2 temperature measurements because of impingement, followed by vehicle body rate changes, a momentum change when wing 2 hit the 90° stops, and a final momentum kick when wing 2 tore free at its hinge link.

The adequacy of Skylab pressure vessel design was verified throughout the Skylab mission, and especially during the period in which the meteoroid shield was lost. At that time, telemetry data of dynamic measurements such as vibration, acceleration, attitude error, and acoustics showed strong distrubances. During the same period, a clockwise rotation of 3.0 deg/sec peak amplitude was sensed by the roll rate gyro and a sensor in the instrument unit showed a maximum peak-to-peak shock of 17.2 g's, values above the criteria used in Skylab design. The fact that Skylab survived these stress conditions and the thermal stress that occurred on orbit are both tributes to its structural integrity.

Various types of deployment mechanisms were used in Skylab design. Much of the hardware deployed or ejected had a greater mass than similar hardware used in other space programs and required designs with greater structural strength and mechanical force to ensure deployment. No anomalies were attributed to the deployment mechanisms. Although the adequacy of mechanisms used for workshop meteoroid shield and solar array deployment cannot be totally evaluated, there were no indications that these designs would not have performed as required had the meteoroid shield anomaly not occurred.

Some of the doors exposed to the space environment exhibited problems during the Skylab mission. The S190 window cover was operated through 100 trouble-free cycles and is recommended for future applications.

Because of design redundancy and the capability of crewmen to perform troubleshooting and in-flight maintenance, mechanical hardware operation was adequate. Numerous fluid systems problems occurred, but these were surmounted either by using redundant systems, modifying operating procedures, or by onboard maintenance. Leakage of fluids and probable fluid system contamination caused a majority of mechanical hardware problems.

The structures/mechanical area provided mission support at the Huntsville Operations Support Center throughout the Skylab mission for action requests and mission action requests. A number of workarounds were devised to support day-to-day activities aboard Skylab. Troubleshooting procedures were developed for use on orbit, and tests using backup and other hardware were performed, both at MSFC and at contactors' facilities.

The initial thermal shield installation on-orbit was a JSC-developed parasol-type device deployed through the workshop solar scientific airlock. During extravehicular activities, the second Skylab crew deployed the MSFC developed twin-pole thermal shield over the parasol. The initial device was effective in reducing Skylab temperatures and the second achieved, and allowed maintenance of, internal temperatures at a nominal 72 °F except for the high beta angle operations during the third manned phase.

Analyses of the initial Skylab problems, the conceiving, designing and materials selection, and the all-out hardware development, qualification, packaging, development of deployment techniques and procedures (in the MSFC neutral buoyancy simulator), and delivery efforts for the thermal shields required only 10 days to complete. Evaluation of these efforts and their products can be measured in terms of mission success, for without the thermal shields, the workshop would not have been habitable.

Simultaneously with thermal shield efforts, analyses, design, selection, development, qualification, packaging, and delivery of solar array release tools were accomplished. Techniques and procedures were developed and verified in the neutral buoyancy simulator at MSFC. Effectiveness of this ground support activity was verified when the first flight crew successfully released workshop solar array wing 1. The tools provided and the extravehicular activities accomplished by the crew resulted in an increase of electrical power by about 50 percent.

Malfunctioning rate gyros in the telescope mount required that a mounting location for installation of additional rate gyros be selected and that a stable platform for mounting them be developed, qualified, and launched with the second Skylab crew. The selected installation location was in the docking adapter between longerons 4 and 5, immediately aft of cabin fan 2. The second Skylab crew mounted the platform and gyros to brackets that had previously supported a storage container. Mounting misalignments were well within tolerance, providing an evaluation of ground support effectiveness, accuracy of drawings, and stable platform development and test.

The six operating gyros increased the noise level at their location beyond specified acoustic criteria in some frequency ranges, with the greatest deviation being 7 dB at 500 Hz. Overall sound pressure level was required to be not greater than 72.5 dB, and this criterion was not exceeded. The noise level caused no crew discomfort, but the atmosphere circulation velocity required for rate gyro cooling caused minor annoyance while crewmen worked at the telescope mount control and display panel.

Ground tests showed the desirability of two-fan gyro cooling during the decreased pressure conditions when Skylab was unmanned, and between the second and third manned mission phases. With the aid of a universal camera mount and equipment straps, a portable fan from the workshop was attached to the telescope mount control and display foot restraint and secured to preclude damage from docking loads.

Due to leakage the primary coolant loop had to be shut down during the second manned phase. The leakage problem required design, development and qualification of a coolanol reservicing kit that was launched with and used by the third Skylab crew. Penetration into the coolanol system was made with a saddle valve developed by NASA. After the crew reserviced the primary loop with 7.7 lb of coolanol, full capability of the system was restored. Secondary coolant loop reservicing was not required.

The airlock 47 °F thermal control valves "B" in each coolant loop stuck during the first Skylab extravehicular activity. A heater and associated equipment were designed and flown up with the second Skylab crew to add heat to the system in 250 W increments, up to 1,000 W, should the thermal control valves again move to and remain in a colder than desirable flow position. Although the heater was launched with the second Skylab crew, it was never used since the system performed adequately through the remainder of the Skylab mission.

SECTION II. INTRODUCTION

A. Purpose

The purpose of this report is to provide a structural integrity and mechanicl systems and component performance evaluation of the Saturn Workshop hardware for the Skylab mission, including prelaunch. The evaluation provides discipline oriented insight into normal hardware performance as well as anomalous performance. Evaluations are based on flight data, test data, crew reports, and photographs. It is intended that the report will become a permanent record of hardware performance and base from which future spacecraft structures and mechanical systems will be developed.

B. Scope

Evaluation is made for each Saturn Workshop module and the pertinent equipment and systems therein. Anomalies are discussed within the appropriate evaluation section.

The following objectives were primary in the preparation of this performance evaluation report:

1. Identification of the requirements and configurations applicable to the structural and mechanical items evaluation of the Saturn Workshop modules.

2. Recording of the criteria and/or parameters which accurately and completely evaluate/demonstrate that the mechanical and structural capabilities of the specific items under the cognizance of the Structures and Mechanical Mission Support Group were, or were not, within the requirements specified.

3. Providing constructuve recommendations relative to hardware application for future projects based on assessment of the performance of the Saturn Workshop modules.

SECTION III. APPLICABLE DOCUMENTS*

Document Identification	Issue/Rev Date	Document Title
CP003M00023	March 1969	Part I End Item Detail Specification (Prime Equipment) Performance and Design Requirements, E1003023, Saturn Payload Shroud for the Saturn I Workshop and Apollo Telescope Mount Missions
CP114A1000026	N/A	MDA Contract End Item CEI Specification
CP2080J1C	November 1969	Contract End Item Detail Specification (Prime Equipment), Performance/Design Requirements
DAC-56618A	September 1969	Quality Program Plan
DAC-56620C	May 1971	Acoustic Shock and Vibration Test Criteria
DAC-56689A	January 1970	Configuration Management Plan
DAC-56697A	September 1969	Test Plan
DAC-56601A	September 1969	Reliability Program Plan
DAC-5672A	September 1969	Government Furnished Property Requirements
ED-2002-1209-9	August 1973	Skylab Interior Acoustic Environment Roport
E451-5102	March 1974	AM Coolant System Package

*Many of these publications are revised periodically. The latest edition should be consulted.

Document Identification	Issue/Rev Date	Document Title
E453-8	April 1973	Airlock Environmental Control System Component Data
E935	July 1971	Airlock Qualification Status Report
E946	November 1971	Airlock Performance/ Configuration Specification (CEI Specification) Flight Article No. 1 and No. 2.
G499	N/A	Airlock Equipment Acceptability Review
IN-ASTN-AD-70-1	N/A	Preliminary Vibration Acoustic and Shock Specification for Components on Saturn V Workshop
IN-ASTN-AD-70-2	N/A	Preliminary Loads Analyses for the Saturn V Dry Workshop
MDC-E0047	December 1973	Contract End Item Detail Specification Performance, Design and Test Specification for the AAP Saturn Workshop Payload Shroud (Part I and II)
MDC G0017	September 1969	Dynamic Test Article Program Requirements
MDC G0174	February 1970	Engineering Mockup-One-G Trainer Program Requirements

Document Identification	Issue/Rev Date	Document Title
MDC G0837	September 1971	Operational Nomenclature
MDC G0945	June 1971	Critical Components List
MDC G5170	May 1974	Skylab-Orbital Workshop Final Technical Report - 3 Vol.
N/A	June 1973	Airlock Design Data Book
RS003M00003	June 1972	Cluster Requirements Specification

SECTION IV. SKYLAB VENTILATION SYSTEM HARDWARE

The Skylab ventilation system is unique in that it is common to and spans all habitable modules, the only structures/mechanical system to do so. The ventilation system circulates conditioned and revitalized atmosphere at required velocities throughout the Skylab pressurized volume to maintain a habitable environment. The ventilation schematic is shown in Figure IV-1. The cabin and portable fans provide high velocity movement of atmosphere. Twenty-seven fans, 3 portable and 24 installed in ducting, are required.

A. General Requirements

Noise levels are required to be less than 55 dB. Flow rate and pressure requirements for each fan at 5 psia are:

LOW SPEED:	105 $\pm$15 cfm at 0.04 in. H_2O pressure
HIGH SPEED:	180 $\pm$18 cfm at 0.08 in. H_2O pressure
	150 $\pm$18 cfm at 0.15 in. H_2O pressure
	100 $\pm$18 cfm at 0.26 in. H_2O pressure
	75 $\pm$18 cfm at 0.28 in. H_2O pressure
	50 $\pm$18 cfm at 0.30 in. H_2O pressure
	40 $\pm$18 cfm at 0.31 in. H_2O pressure

An odor removal canister, in combination with a fan, is provided in the workshop waste management compartment for odor control. Other workshop fans include three clusters of four fans each in workshop ducts and three portable fans. Seven fans are installed in combination with the seven cabin heat exchangers in the airlock, and one fan is in the airlock interchange duct. Two cabin fans for circulating docking adapter atmosphere and one for circulating command module atmosphere are used.

B. Development and Testing

The fans were developed for the Apollo program and adapted for Skylab usage. Qualification included vibration to 106.5 g rms, shock to 1500 g, altitude to 1.93 x 10^{-8} psia, humidity to 95 percent with temperature cycling, temperature extremes to 165 and -65 °F, 100 percent oxygen atmosphere at 5.5 psia, 5 percent salt fog at 95 °F for 48 hr, 28 hr of exposure to sand and dust, and pressure and flow rate tests. Demonstrated life included 3,360 hr of operation and 500 on/off cycles.

C. Mission Performance

Fan performance criteria were satisfied by on-orbit operations. Measured noise levels in Skylab were 55 to 60 dB showing adequate noise suppression during fan operation. The waste management compartment fan and canister assembly effectively controlled odors. The longer third manned phase and fan operation in the docking adapter, to aid gyro six-pack cooling during the unmanned period between the second and third manned phases, resulted in more run time than initially planned. Since it was possible that the docking adapter fan could have operated under pressure loss conditions, a test was conducted at MSFC during the Skylab mission to determine fan operating capability in a vacuum. Two fans were operated in a chamber at approximately five microns pressure for 384 and 1,164 hr, respectively. Ambient temperature was maintained at 75 °F and maximum case temperature was 89 °F. Both fans operated with no apparent degradation to operating characteristics. The fan that operated 384 hr was disassembled and bearing examination showed no degradation.

Cooling bay fans had the highest anticipated on/off cycle rate, but actual cycles were very low because of the meteoroid shield thermal problem and lowered thermostat settings which caused nearly continuous running.

D. Anomalies

Throughout Skylab no fan life problems occurred, but flowmeter readings were of doubtful accuracy. For example, the airlock interchange duct fan was replaced on mission day 44 of the second manned phase in an attempt to increase indicated flow, but when it failed to do so it was concluded that the original fan was satisfactory and that the flowmeter was in error. Also, the four fans in the airlock used for workshop cooling were replaced on mission day 63 of the third manned phase, again with failure to increase indicated flow. This led to the same conclusions; the fans were adequate but flowmeter readings were in error.

E. Recommendations

Small items of debris resulting from crewmen living and working in space occasionally drifted into the Skylab atmosphere. Such items were moved by the circulating atmosphere to the screens/filters of the circulation system and were held, as shown in Figure IV-2, until removal by the crew. This feature helped to maintain Skylab orbital cleanliness and should be considered in the design of future, larger volume spacecraft.

It is further recommended that for future applications, similar to these described in this section, fan on/off ground command capability should be considered since control during contingencies may be required. Also, filters for heat exchanger mounted fans should be sized to preclude contamination buildup and blockage of air flow through the heat exchangers.

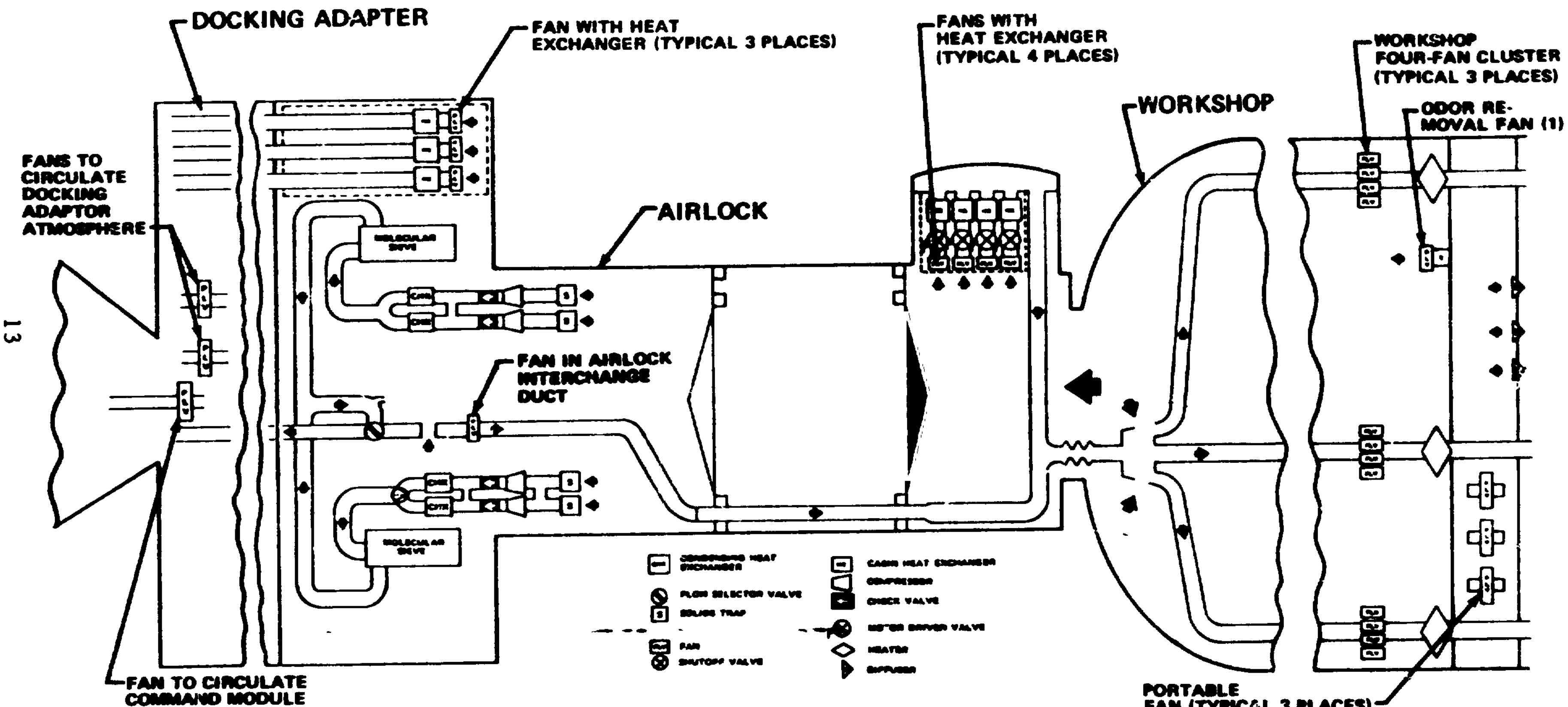


Figure IV-1. Ventilation System Schematic

Figure IV 2. Circulation System Screen/Filter

SECTION V. SKYLAB MODULES AND HARDWARE

All the major Skylab modules and hardware items related specifically to module structural/mechanical performance are evaluated in this section.

A. Airlock

1. Basic Requirements and Configuration Evolution. The airlock provides: a habitable, interconnecting pressure vessel between the docking adapter and the workshop; the atmospheric nitrogen supply; support for intervehicular activities; an airlock to support extravehicular activities. The as-flown airlock (Figure VA-1) is an item that was carried over from the wet workshop to the dry workshop. At the time of carry over, one of the four airlock trusses had a removable link; two others were modified to this configuration to allow mounting of six nitrogen bottles on the three removable link trusses. The docking adapter interface ring was strengthened and gussets were added to the structural transition section stringers. Other modifications such as penetrations, welds, and revised rivet patterns were also accomplished.

2. Structures. Commensurate with design requirements and criteria, the airlock pressure vessel consists of a structural transition section, a tunnel section, or lock compartment, and a flexible tunnel extension or bellows. It structurally supports the docking adapter, accepting its loads at the interface ring and transmitting them and its own to the fixed airlock shroud by way of four fusion-welded, aluminum tubing truss assemblies. Sealant was applied to the interface rings to maintain internal pressure. The bellows provide a flexible pressure vessel interface to the workshop.

The structural transition section volume is 288 ft^3 contained in a welded aluminum, stressed-skin, semimonocoque cylinder 47-in. long and 120 in. in diameter. It reduces to 65 in. at the tunnel section. The stingers and longerons resistance welded externally to the skin, carry overall axial loads and body bending loads with intermediate internal rings added for support. Loads are transferred to truss fittings by way of eight intercostals.

The tunnel section is a semimonocoque, aluminum cylinder 65 in. in diameter, divided by two internal bulkheads with mating pressure hatches. The 31-in. long forward compartment interfaces with the structural transition section, the 80-in. long center tunnel (lock) compartment interfaces with the bellows and an octagonal airlock ring. Seven external shear webs and the octagonal ring provide attachment and shear continuity between the tunnel section and the trusses.

Accommodation of deflection with minimum load transfer between the airlock and workshop is provided by the 42.5-in. diam, 13-in.

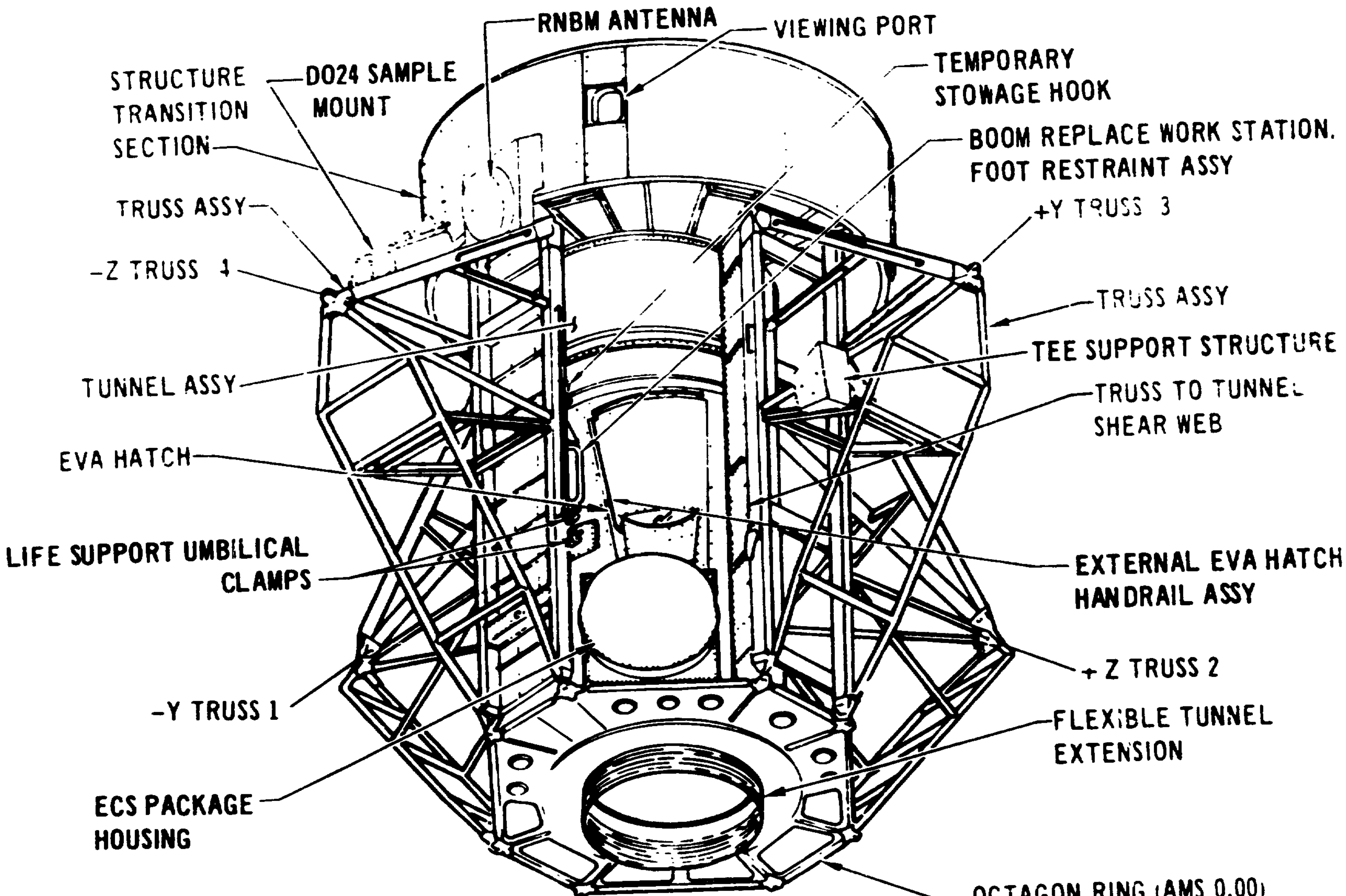


Figure VA-1. Airlock Module Structure

long convolute, flexible bellows. Internal fluorocarbon coating provides a redundant bellows seal and a fiber glass laminate cylinder inside the bellows protects it from damage.

a. Pressure vessel. The required airlock relief valve upper limit pressure is 6.2 psia. While under static loads, the flight-type wet workshop configuration static test article was proof pressure tested to 1.4 times the limit pressure. Burst pressure tests were also conducted on this article to 2.0 times limit pressure with no external loads applied. A proof pressure test to 1.4 times the limit load was conducted on the flight article with no external loads applied.

b. Bending moment and axial loads. The airlock specification weight was 15,416 lb and design requirements stipulated that it be able to support a docking adapter weight of 14,000 lb. Lift-off weights for the two modules were 15,166 and 13,645 lb, respectively. The structure was designed for exposure to specified thermal, acceleration, shock, random vibration, pressure, and acoustic environments. The minimum required factor of safety on limit loads combined with limit operating pressure for the airlock structure is 1.25 for unmanned conditions and 1.36 for manned conditions, commensurate with criteria imposed for tested structure. Nitrogen bottle support structures, excluding trusses, are designed to a factor of safety of 2.0 (unmanned condition) to preclude the need for static test.

The airlock static test article, a production type structure, was tested at the MSFC static test facility.

3. Natural Environments Design. Airlock requirements for maintaining habitable volume in the space environment, including radiation protection, are provided by structural design. Appropriate protection against particulate matter, excessive humidity, rain, ground winds and flight winds are primarily a function of the payload shroud, the fixed airlock shroud, and the KSC facility, including facility gasses. The meteoroid protection requirement for the airlock provided for the structural transition section by the radiator and for the remainder of the airlock, and the instrumentation unit and workshop dome, by the fixed airlock shroud and meteoroid curtains. These flexible curtains, made primarily of Viton rubber impregnated fiber glass, have gold-coated interiors and off-white fiber glass exteriors. They are stretched conically between the structural transition section and the fixed airlock shroud except for the extravehicular activity bay quadrant. Here, the tunnel section is protected by two curtains stretched from shear webs to exterior truss members, and the workshop dome is protected by a curtain stretched between the airlock octagonal bulkhead and the fixed airlock shroud lower intermediate ring. Curtains are also stretched fore and aft of the airlock hatch, fastening to the structural transition section and the octagonal bulkhead. The airlock hatch is protected by a rigid fiber glass installation. No occurrence of meteoriod penetration was detected during the Skylab mission.

Requirements to distribute breathable atmosphere level during operations at KSC and to permit purging of the airlock and docking adapter during prelaunch are provided by the aft compartment purge fitting. Venting at KSC during ascent and on-orbit was a function of the docking adapter. Prelaunch atmosphere and purge flow was successfully provided by a "drag-on" hose at KSC.

Structural design limits internal radiation to 0.6 rad/day. On-orbit radiation measurements taken in the airlock show the average internal radiation to be approximately 0.10 rad/day, which is well within the design requirements.

4. Mechanical Components.

a. Hatches. An extravehicular activity hatch and two internal hatches were required for the airlock tunnel section. Specifications state that all three hatches have a window and that they be capable of being restrained in the open position.

(1) Extravehicular Activity Hatch. This pressure hatch was required to provide astronaut access to the exterior of the vehicle. Operation of the opening mechanism by a pressure suited crewman from the inside by applying a maximum force of 45 lb was a design requirement. The single stroke hatch handle is equipped with a positive lock for holding it in the closed position. Following closeout at the pad, exit was through the extravehicular hatch. The handle lock could not then be engaged; the first crew engaged the lock during activation. Thereafter, each usage required lock release prior to moving the handle through a 153° arc to actuate the 12 latch assemblies for opening or closing. After each closing the handle lock is reengaged. A catch latch, incorporated to restrain the hatch slightly open, is provided to prevent complete opening until pressure equalization.

The extravehicular activity hatch was partially qualified by similarity to the Gemini hatch. Additional Skylab test requirements consisted of meteoroid impact simulation, handle force tests, and retest of the hatch seal. Also, the static test article hatch was installed during testing at MSFC when the airlock was pressurized to 12.4 psig, and during the vibroacoustic test at JSC. During these tests no problems were attributed to the hatch.

Planned hatch operations during the Skylab program were 50 cycles on the ground and 6 cycles on orbit. The hatch was actually used on-orbit eight times. During the first crew debriefing it was stated that forces and hatch operation were essentially the same as on the trainers and that hatch size was adequate for all equipment and personnel transfers.

The catch latch did not operate as designed on opening, but adequately held the hatch in the slightly open position prior to closing. Apparently, initial rapid motion caused the hatch to move past the catch prior to capture. However, during the slower closing operation, engagement was normal. Inability of the catch to engage during opening presented no operational problems according to the crewmen. The retainer rod was adequate in performing its function of holding the hatch fully open.

During testing to Skylab mission requirements, hatch handle forces were found to be unacceptable. Further evaluation of the problem showed the seal to have a low and inconsistant state of cure resulting in excessive outgassing, inconsistant hardness, undesirable surface adhesion, poor bond integrity, and unacceptable permanent set. Seal procurement specifications were changed to correct these problems and to provide better control of seal thickness dimensions. The test fixture seal indentor was rubbed with teflon and a release agent (CAMIE NO. 2000) was applied to the seal segment surface. This configuration was subjected to a development test at 170 °F at 10^{-6} torr for a period of 30 days and another seal segment was qualified at 10^{-6} torr with temperatures at 120 °F for 30 days and at 75 °F for 66 days. Neither seal segment showed appreciable deterioration, and no evidence of adhesion appeared.

The planned hatch seal replacement was accomplished just prior to Vertical Assembly Building (VAB) closeout at KSC. On orbit visual inspections by Skylab crewmen while the hatch was still open following extravehicular activities, showed no evidence of adhesion or appreciable set or wear. Prevention of leakage on-orbit was adequate based on the small, overall cluster leakage rate.

(2) Internal hatches. The two internal airlock hatches were required to provide a primary pressure seal at the two lock compartment intermediate bulkheads. However, an operational change was made to utilize the workshop dome hatch as a pressure seal in lieu of the aft hatch which resulted in more volume in the lock compartment to accommodate film and container transfer between the lock and the telescope mount. Extension of the lock compartment to include the flexible tunnel extension provided the increased volume. Thus, the aft airlock hatch was used only as a backup or secondary pressure seal during Skylab orbital operations.

The two hatches were required to seal bulkhead openings of 4.73 in. in diameter. Their opening mechanisms were required to be operable from either side by a pressure suited crewman applying a maximum force of 35 lb. Leakage could not exceed 825 scc/m at 6.2 psig gaseous nitrogen.

Ground testing included temperatures at 20, 120, and 160 °F; proof pressure at 8.7 psig; altitude at 9.3×10^{-7} psia; leakage at the rate of 190 scc/m; and 507 hatch and mechanism cycles, including high, low and ambient temperature at ambient pressure, and 24 cycles at 9.3×10^{-7} psia. Hatch seals were qualified concurrently with the extravehicular activity hatch seal since they were basically alike. Anticipated on-orbit usage was nine open/close cycles for the forward hatch and three cycles for the aft hatch. The forward hatch was opened nine times and closed eight. The aft hatch, once opened, was restrained in the open position throughout the remainder of the Skylab mission. Internal hatch operations presented no problems. Crewmen reported that the hatches were adequate in size and easier to manage in zero g than they had been in the trainers.

b. Pressure equalization valves. Three identical valves are used in the airlock: one in each internal hatch plus the lock compartment vent valve, mounted on the tunnel wall adjacent to the extravehicular activity hatch. With air at 70 $\pm$20°F and inlet pressure of 5.0 psia, flow was required to be 10.0 lb/min with valve outlet pressure of 2.6 psia maximum. Maximum torque for valve handle operation was required to be 40 in.-lb with a pressure of 6.2 psid.

A disc in each butterfly-type valve can be actuated from either side by a handle and shaft arrangement that is coupled to the valve shaft by bevel gears. The disc is offset from the valve shaft around which it rotates for opening and closing. When open, effective flow area is 1.44 in^2. The valve is held either fully open or fully closed by ball type detents that lock the handle shaft. Detents are unlocked by pressing a button on either handle. If the valve is left in a partially open position, it is spring-loaded to close which allows engagement of the detent. Valve length overall is 9.63 in. and the mounting flange is approximately 63 in. in diameter.

Internal hatch pressure equalization valves were required to be open at launch to allow tunnel venting through the closed hatches during ascent. During extravehicular activities preparation, the workshop dome hatch and the forward airlock internal hatch and valve were closed before manually operating the lock compartment vent valve to depressurize the lock. Complete venting of the lock compartment from the nominal 5.0 psia to external ambient was precluded by gas exhausting from crew suits such that the compartment retained pressure of about 0.15 psia until the extravehicular activity hatch was opened. Ground testing included proof pressure to 12.4 psig, burst pressure to 24.8 psig, twenty-eight 24-hr temperature/humidity cycles, pressure/temperature tests, leak tests not to exceed 0.19 scc/m gaseous nitrogen at 6.2 psig, 500 operating cycles at 5.0 psia outlet pressures, natural environments, strength and shock tests, cold soak, and random vibration to 10 g rms overall. The valves were also tested with the airlock structure during static testing at MSFC and vibroacoustic tests at JSC.

It was estimated that 25 total valve cycles would be required on-orbit. The forward hatch pressure equalization valve and the airlock vent valve were each cycled eight times. Neither valve malfunctioned, but the airlock vent valve accumulated ice on the inlet screen and thus restricted gas flow during depressurization. A 0.025-in. diameter, 304 stainless wire, #8 mesh screen was assembled to a valve cap and flown on-orbit by the second manned crew for use on the airlock vent valve during subsequent depressurization activities (Figure VA-2). This screen reduced the gas flow restriction; ice crystals formed in the central screen area, only, leaving the outer portion free of ice to allow atmospheric flow. When pressure decreased to about 1 psi, the screened cap was removed to achieve an increased atmospheric flow rate through the existing #6 mesh screen in the vent valves.

c. Windows and Covers. Seven windows were required for the airlock. In addition to the windows in the three hatches, four 8 by 12-in. oval windows were required to be spaced at 90° intervals, 37°52' off the vehicle axes, around the aft portion of the structural transition section. Provisions to cover and uncover these windows with a thermal barrier were required. Optical qualities were required to be compatible for use by the crew during visual observations. All seven windows were required to be double pane and the five windows designed for external viewing were required to have valves for venting the space between windows panes. These valves were opened at vehicle final closeout at KSC to allow venting of the dry argon gas between the panes during vehicle ascent, and were closed by the first manned crew during activation to preclude "breathing" caused by internal vehicle pressure fluctuations.

The structural transition section windows, shown in Figure VA-3, are protected externally by sliding, fiber glass-reinforced, plastic-laminate covers operated by an internal crank assembly which is locked (open or closed) by a quick-release pin. Each exterior window pane is 0.42-in. thick vycor glass and its internal surface has a thin gold coating to reflect infrared light to reduce heating of the inner pane, thereby avoiding a touch temperature problem. Each interior pane is 0.24-in. thick tempered glass with its external surface having an ultraviolet reflective coating to avoid ultraviolet triggering of the caution and warning fire alarm. The nominal space between panes is 0.25 in. The panes are individually sealed to preclude atmospheric leakage. Limit pressure is ± 6.2 psid in the compartment and 14.7 psid trapped between panes.

A fracture mechanics analysis was conducted on the windows. A sympathetic shatter test demonstrated that the outer pane remained intact and maintained pressure up to 25 psid even though the inner pane was purposely broken by impact. Burst pressure tests were conducted in fixture duplicating actual installation. The inner pane space was pressurized to rupture. All burst tests indicate failure at approximately 65 psid. No fogging of the windows occurred during ambient or low

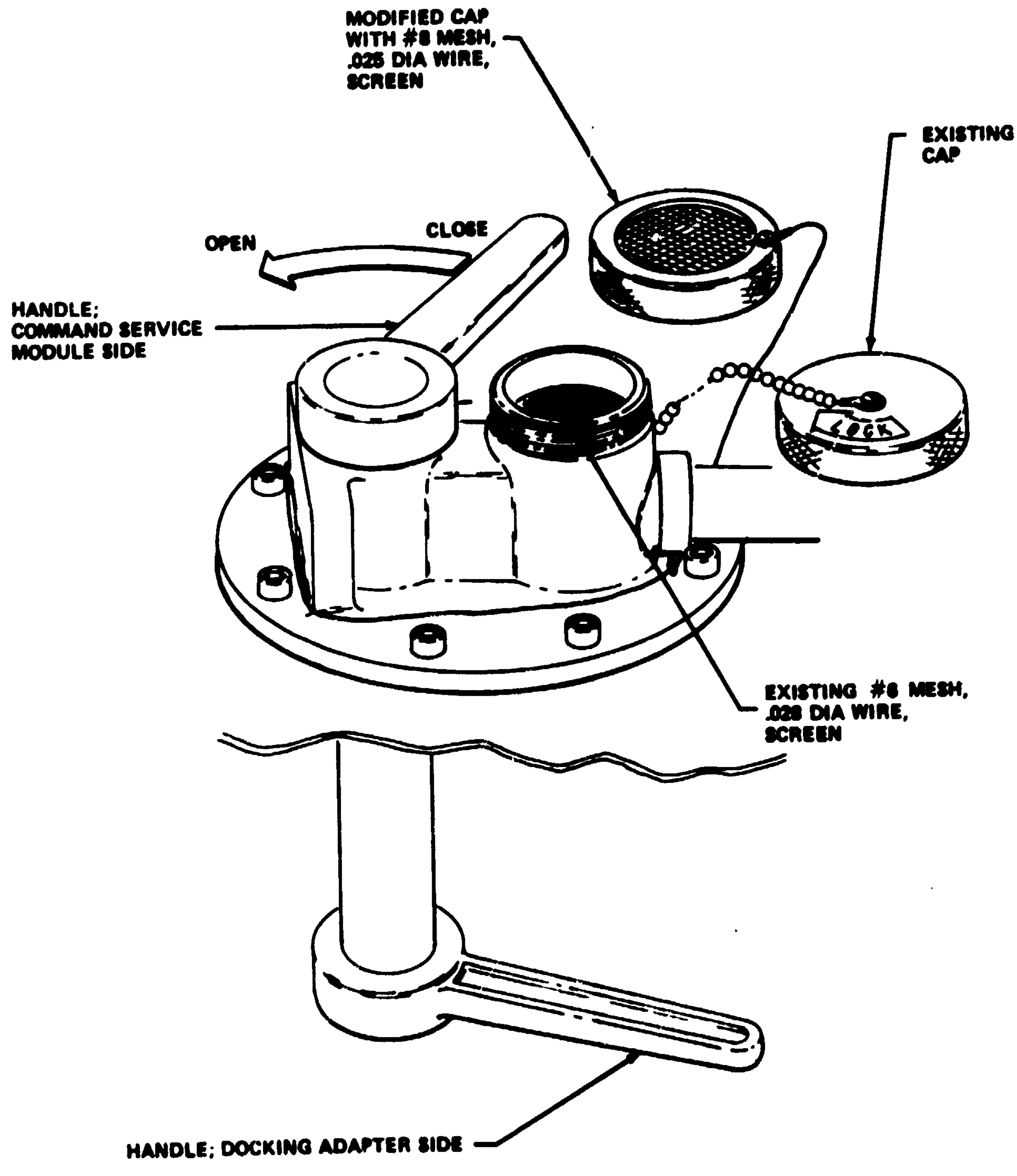


Figure VA-2. Butterfly Valve Assembly (with Modified Cap)

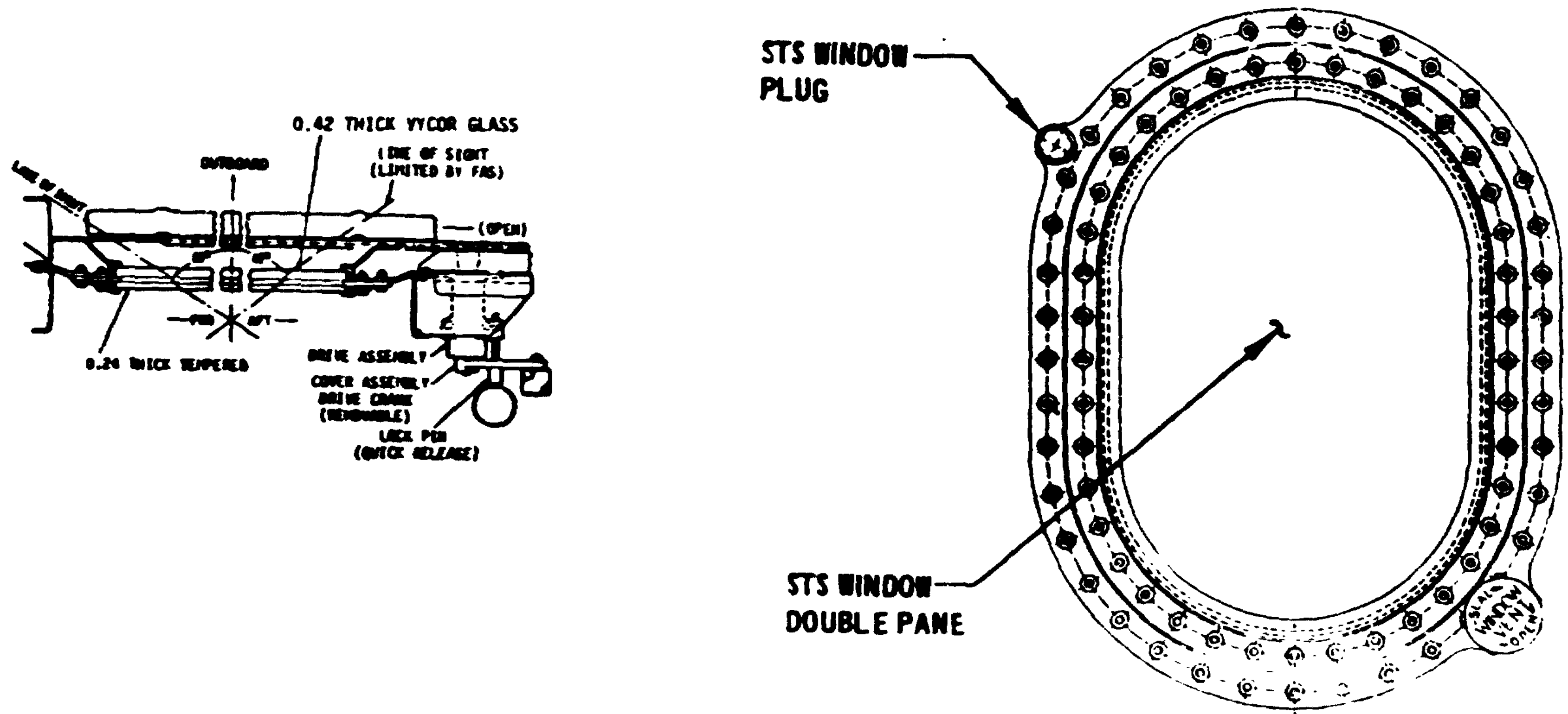


Figure VA-3. Structural Transition Section Window Assembly temperature fogging tests. Maps of each window were prepared showing the location and size of all microscratches and leaks. Protective covers were then installed over the inner panes and the outside protective covers closed. These covers were removed and the mapping rechecked at KSC just prior to final closeout. Only one new small scratch was discovered and none of the others had enlarged, so the windows were considered satisfactory for launch.

Additional window assembly qualification tests included launch vibration with leakage measured before and after vibration, leakage at -30 °F., proof pressure at 8.4 psig, and internal cabin burst pressure to 12.4 psig. Cover assembly qualification included random vibration at 6.8 g, leak test of cover drive assembly, and 300 operating cycles at 9.49×10^{-4} torr. Each flight window pane was proof-loaded at acceptance to 30 psig, and all window assemblies were proof tested to 8.4 psig installed in the module.

Window size and shape are adequate for on-orbit use according to crew comments during debriefing, although interference from exterior structure precluded some photographic usage. Window fogging on the earth side of the vehicle occurred after 3 to 4 hr exposure with covers open. No fogging was observed in window assemblies on the sun side of the vehicle. When earth side covers were closed, windows would free themselves of fog in 2 to 3 hr.

Window covers were closed during sleep periods to decrease the light level in the workshop sleep compartment. Window cover mechanisms became increasingly harder to operate as the mission progressed with the #3 cover mechanism being the most difficult to operate.

The extravehicular activity hatch window is identical to the hatch windows used in the Gemini program except for the ultraviolet infrared coatings and the addition of a trapped volume vent valve. The outer pane is 0.380-in. Vycor and the inner pane is 0.220-in. tempered glass with a 0.250 spacing between the two. The ground verification of the hatch window paralleled that for the structural transition section windows.

The extravehicular activity hatch window was not used. A stowed cover was installed over the window by the first Skylab crew and was not removed throughout the three manned mission phases.

The 8.5-in.-diam interior hatch windows are covered by a protective mesh on either side. The window on the forward hatch was used by the third crewman for viewing the two extravehicular crewmen in the lock compartment of the structural transition section during extravehicular activities. Size was considered adequate and installation necessary to support extravehicular operations. These windows had no cavity bleed valve and showed no signs of fogging.

The most likely cause of condensation that formed on the interior surface of earth-side structural transition section exterior window panes is the breathing of humidity laden atmosphere through the cavity vent valves caused by minor vehicle pressure fluctuations and thermal cycling of the window structure. Initial design of the windows provided lockup of 14.7 psia dry argon in the cavity. A later modification provided pressure bleeding. Had initial design been followed, the probability of condensation would have been decreased. Future design should consider alternatives such as: dry gas lockup; partial cavity bleeding such that lockup pressure would be about 6.0 psid at lowest internal pressure (0.5 psia) and about 1.5 psid at nominal operating range (5.0 psia); window heaters; or periodic dry gas purging.

d. Discone Antennas. Two discone antennas are mounted on the fixed airlock shroud for telemetry transmission and command reception (Figure VA-4). Since the deployed telescope mount and solar arrays have a degrading effect on radiation patterns, the antennas are mounted on folding booms to deploy the discones away from the cluster. The booms fold in two segments for stowage inside the payload shroud for launch and ascent, and are released on ground command through the airlock digital command system after the shroud is jettisoned. The primary release logic is powered from airlock power bus #1 to release both antennas. A redundant system was powered from bus #2 and triggered by a backup command. An alternate method of deploying the antennas was available to the crew onboard. The deployment mechanism has two spring powered rotary joints which supply energy for boom extension, control the rate of deployment, and lock the boom in the extended position. After full deployment, a locking pin triped a microswitch which transmited a signal to verify full boom extension. One rotary joint connects the two boom segments together and rotated through approximately 180° while the other mounts the boom assembly to the upper ring of the fixed airlock shroud and rotated through approximately 90°.

Ground verification tests were conducted on the rotary joints, release actuator, release module assembly, and the deployment assembly. Test environments for these components were vibration, life, humidity, temperature, acceleration, and functional. After the humidity qualification test, the rotary joints failed to operate because of corrosion. This condition was caused by a stainless steel roll pin pressed through an aluminum bearing bushing and left unprotected from the exposed environment. Corrosion was also evident on the internal shaft where the nickel plating separated from the aluminum shaft because of poor adhesion on the sharp corners. The solution was to have the rotary joint sharp edges rounded and replated. The music wire springs were replaced with stainless steel springs. The stainless steel roll pin was pressed half-way into the bearing bushing and sealed on the back side. After rework was completed, the unit was successfully retested. System performance was verified when a complete deployment assembly was functionally tested to measure time to deploy and rates, using air bearings to support the weight of discone and booms.

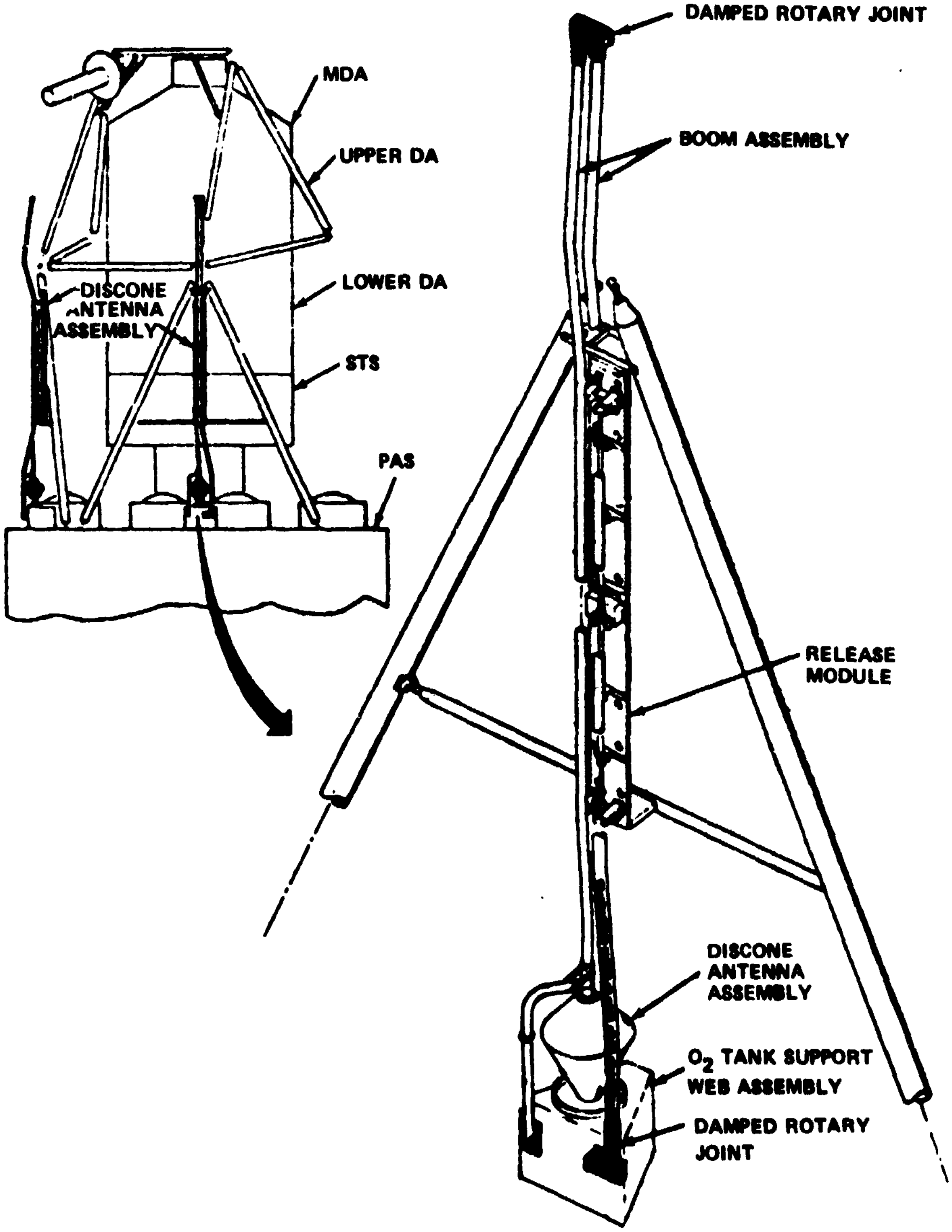


Figure VA-4. Discone Antenna System

The JSC vibroacoustic test article incorporated a production release module, production booms with dummy rotary joints, and a mass simulation of the antenna. No problems or anomalies were encountered. The flight booms were subjected to a full deployment functional test at the manufacturer's facility using the air bearing test setup at final acceptance. A trial release of the final installation was performed at KSC to assure free movement and availability of deployment torque, but full deployment was not feasible in that configuration. A problem developed at KSC when it was discovered that the nylon webbing, which restrains the booms in the stowed position, stretched under the rigging tension. This stretching could allow the booms to move under launch vehicle dynamics with possible structural damage. The nylon webbing was replaced with stranded steel cable, the system rerigged, and the release mechanism rechecked with no further anomalies.

The discone antenna deployment sequence was initiated after payload shroud jettison and the arming of the deploy buses. Telemetry indicated actuation of the deploy circuits at 16 min 10.1 sec and at 16 min 34.1 sec. The first actuation released both antennas. Discone antennas #1 and #2 (Figure VA-5) were fully deployed at 16 min 54.2 sec and 16 min 52.5 sec, respectively. Ground deployment times were slightly longer probably caused by friction under 1 g envirnoment.

5. Environmental Control System. Some of the environmental control system components evaluated in this report are adopted from previous space programs and some were developed specifically for their intended Skylab use.

a. Oxygen/nitrogen system. The system provided nitrogen and oxygen gas supplies; limited the airlock, docking adapter, and workshop atmospheric pressure to a maximum of 6 psig during orbit; and provided a means of transferring atmospheric gas from the airlock to the workshop pressure of 5.0 ±0.2 psia during normal operation with an oxygen partial pressure of 3.6 ±0.3 psia and the difference made up of nitrogen. The system also supported intra and extravehicular activities, pressurization for experiments, and remote pressurization of the airlock, docking adapter, and workshop.

The oxygen/nitrogen system schematic is shown in Figure VA-6. Six cylindrical oxygen tanks are mounted to the fixed airlock shroud and six spherical nitrogen tanks are mounted to airlock trusses. Gases are routed into the habitable volume through penetrations in the airlock structure.

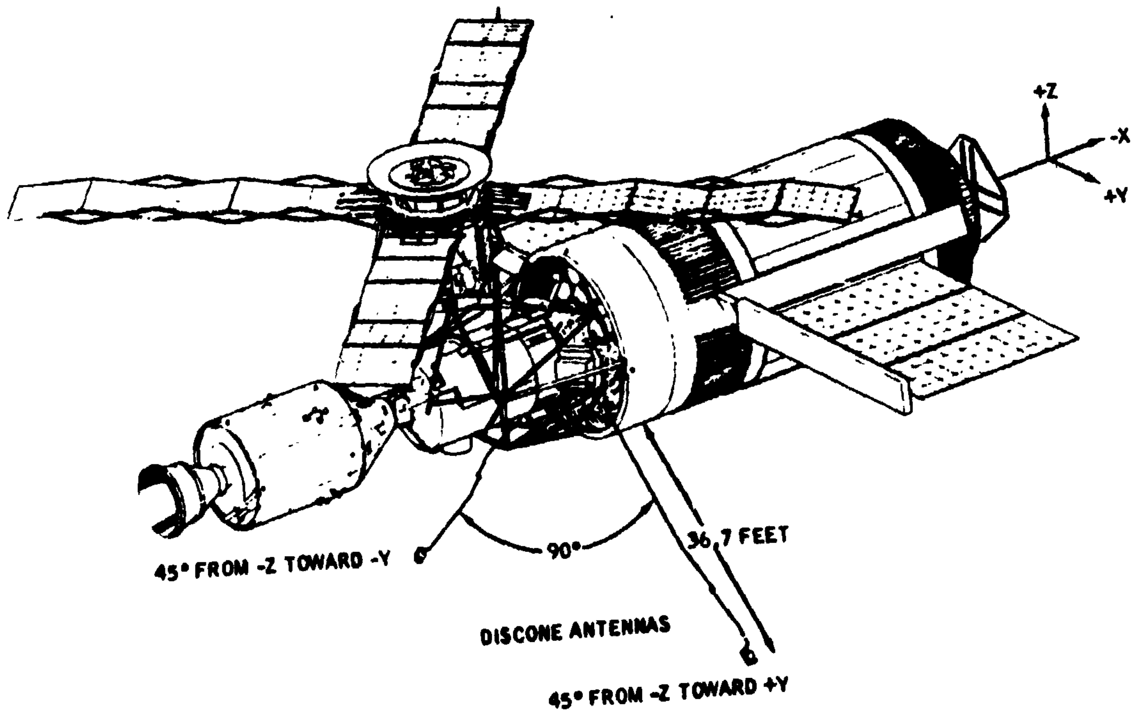


Figure VA-5. Discone Antennas Fully Deployed

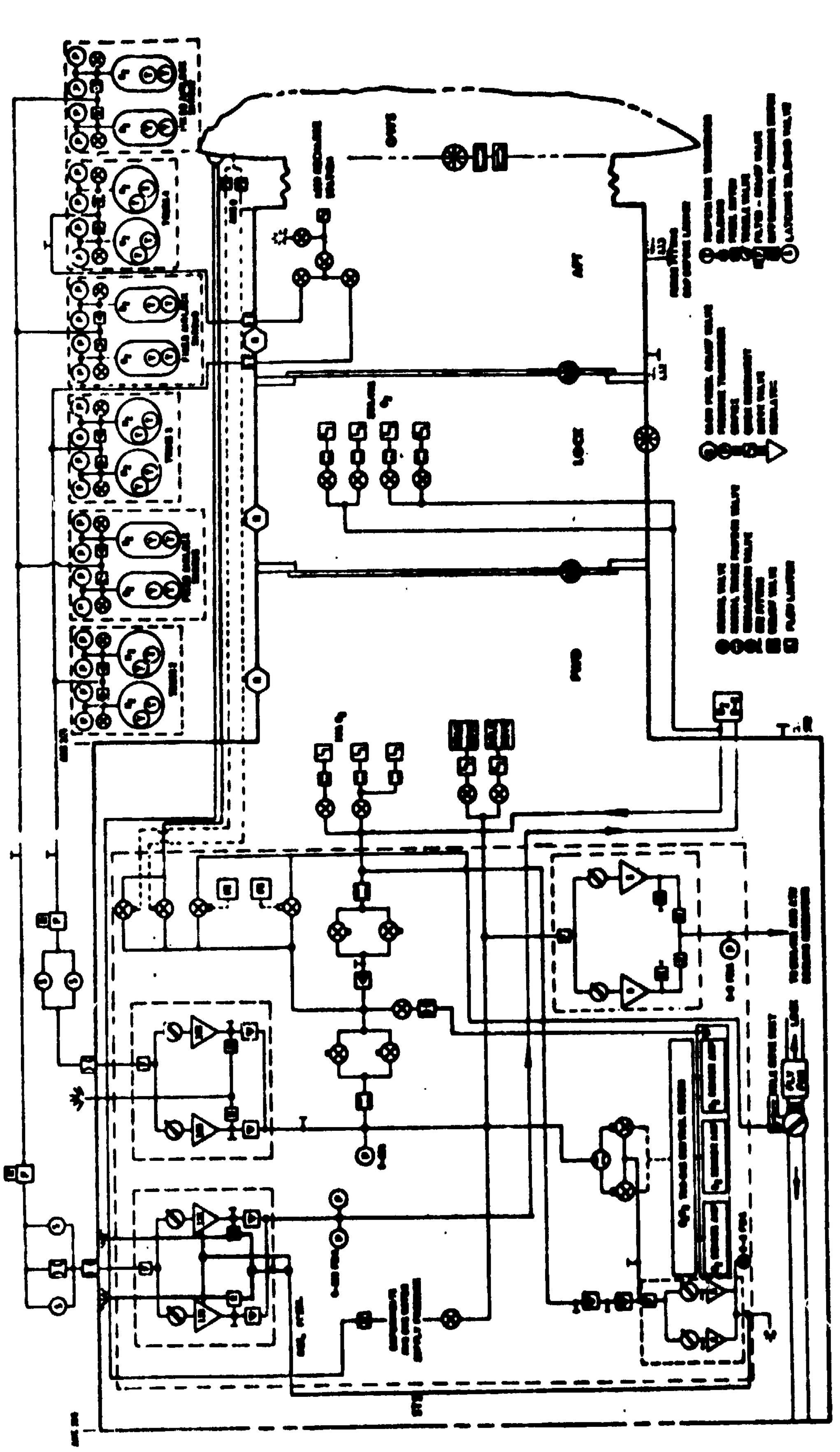


Figure VA-6. Oxygen/Nitrogen System Schematic

(1) Oxygen tanks. Tanks were required to store a minimum of 5,985 lb gaseous oxygen at 3,000 psig to support an estimated Skylab mission requirement of 4,046 lb. The cylindrical tanks with eliptical heads are made of a 2-in. thick fiber glass winding over stainless steel liners. A fill valve assembly, consisting of the fill valve, two pressure transducers and a tank isolation check valve is attached to each tank at the fill-end dome plug. The oxygen tank was developed and qualified for Skylab.

At lift-off the total load in oxygen tanks was 6,112 lb. The quantity of oxygen was sufficient to support the three manned missions including the duration extension of approximately 23 percent. During the Skylab misssion, at high beta angles, oxygen tank #4 temperatures were off-scale high according to telemetry data. Calculations indicated that the maximum temperatures of this tank reached approximately 225 °F. A similar tank had been tested at 275 °F with 50,000 cycles (0 - 3000 psi) showing a maximum strength reduction of about 15 percent. Since Skylab testing was accomplished with pressure at 4,500 psig, it was predicted that potential strength reduction was not sufficient to cause tank rupture. Demonstrated oxygen tank capability throughout the remainder of the Skylab mission verified this prediction. The oxygen tanks and associated tubing exhibited no detectable leakage during the Skylab mission.

(2) Filter/relief valve assembly. One component, of the same type, is in each application to filter both oxygen and nitrogen. The filter, developed specifically for Skylab, is required to remove 100 percent of particles in excess of 10 micron size. Required flow through the filter at 70 ±20 °F and 450 psig inlet pressure is 2.5 lb/min oxygen with a maximum pressure drop of 10 psid.

Estimated operating requirements were 5,938 hr and 25 impulse cycles. The unit was subjected to static fatigue with oxygen at 4,500 psig for 8 months, and 1,000 relief valve crack and reseat cycles at 0 to 150 percent flow at 70 °F. Additionally, a successful filtration flow fatigue test was conducted.

No measurements were provided to determine if the assemblies were in the filtering or gas bypass modes on-orbit. Gas supply at an acceptable level was maintained throughout the Skylab mission.

(3) Oxygen pressure regulator assembly. One oxygen pressure regulator assembly, containing two parallel circuits for redundancy is installed in the airlock. At temperatures from -10 to 160 °F the assembly flows 0.002 to 0.38 lb/min oxygen at inlet pressure of 300 to 3,000 psia and 0.002 to 0.52 lb/min at 435 to 3,000 psia inlet pressure. Both circuits are required to regulate outlet pressure at 120 ±10 psig. The assembly has a single inlet with a 10 micron filter and a single outlet. Between these two ports the paralleled circuits each contain an upstream toggle shutoff valve, a regulator and a relief valve isolated downstream by a check valve. A test port in each circuit is used to verify relief and check valve operation.

The oxygen regulator assembly was developed specifically for Skylab. In-line modifications included: changing internal springs because of stress corrosion (from maraging steel Bellville springs to 17-7PH springs); changing seals to Precision Rubber silicone compound 11207 because of failures in the similar nitrogen regulator which allowed internal leakage past the LS-53 O-ring; changing the Viton "A" relief valve set material to Silastic 675 for improved low temperature compatability and, regulator relief valve and reference ports were manifolded together and provided with an extended inlet line for remote sensing of cabin atmosphere to preclude formation of frost from cabin atmosphere when oxygen flow temperatures were low.

The 120 psig oxygen regulator functioned within its specified limits. Flowing pressure was 126 psia (12 psig) and lockup was 133 to 140 psia depending on how hard the regulator was flowing when lockup occurred.

(4) Cabin pressure regulator. The O_2N_2 supply cabin pressure regulator is required to maintain the atmospheric pressure at 4.8 to 5.2 psia. The flow rate requirement is 1.15 $\pm$0.15 lb/hr at 70 °F with inlet pressures from 100 to 215 psig. Nominal inlet operating pressure is 120 psig with 215 psig being maximum. Outlet lockup pressure is 5.3 psig.

One cabin pressure regulator is used in Skylab. It has two independently operating parallel circuits, with each circuit containing a toggle shut-off valve, a test port, and a pressure regulating valve. Both circuits are contained in one stainless steel housing and have a common inlet port with a 10 micron filter, a common orifice, and a common outlet port. Regulated flow capacity of either path is 1.0 to 1.3 lb/hr of gaseous oxygen or gaseous nitrogen.

The oxygen/nitrogen supply cabin pressure regulator (Figure VA-7) was developed for Skylab. Successful qualification tests were run to ensure compliance with requirements specified. They included vibration to 7.0 g rms, acceleration to 7.2 g, altitude to 10^{-5} psia, humidity to 95 percent with temperature cycling, temperature extremes to 160 and -20 °F, 5 percent salt fog for 48 hr, 100 percent oxygen at 6.2 psia for 40 hr with 2 hr at 160 °F, proof pressure to 430 psig, burst pressure to 860 psig, and leakage tests conducted at 215 psig inlet pressure. Internal leakage was measured with the regulator locked up and maximum allowable was 3.5 x 10^{-4} lb/hr oxygen. External leakage was measured with the solenoid valve open and maximum allowable was 3.15 x 10^{-5} lb/hr oxygen. Estimated operating requirements were 3,610 hr, 675 pressure cycles, and 80 shutoff valve cycles. Demonstrated capability was 10,000 full flow/reseat cycles and 1,000 shutoff close/open cycles on each of two sides. Also, the regulator was included in the 8-month endurance test.

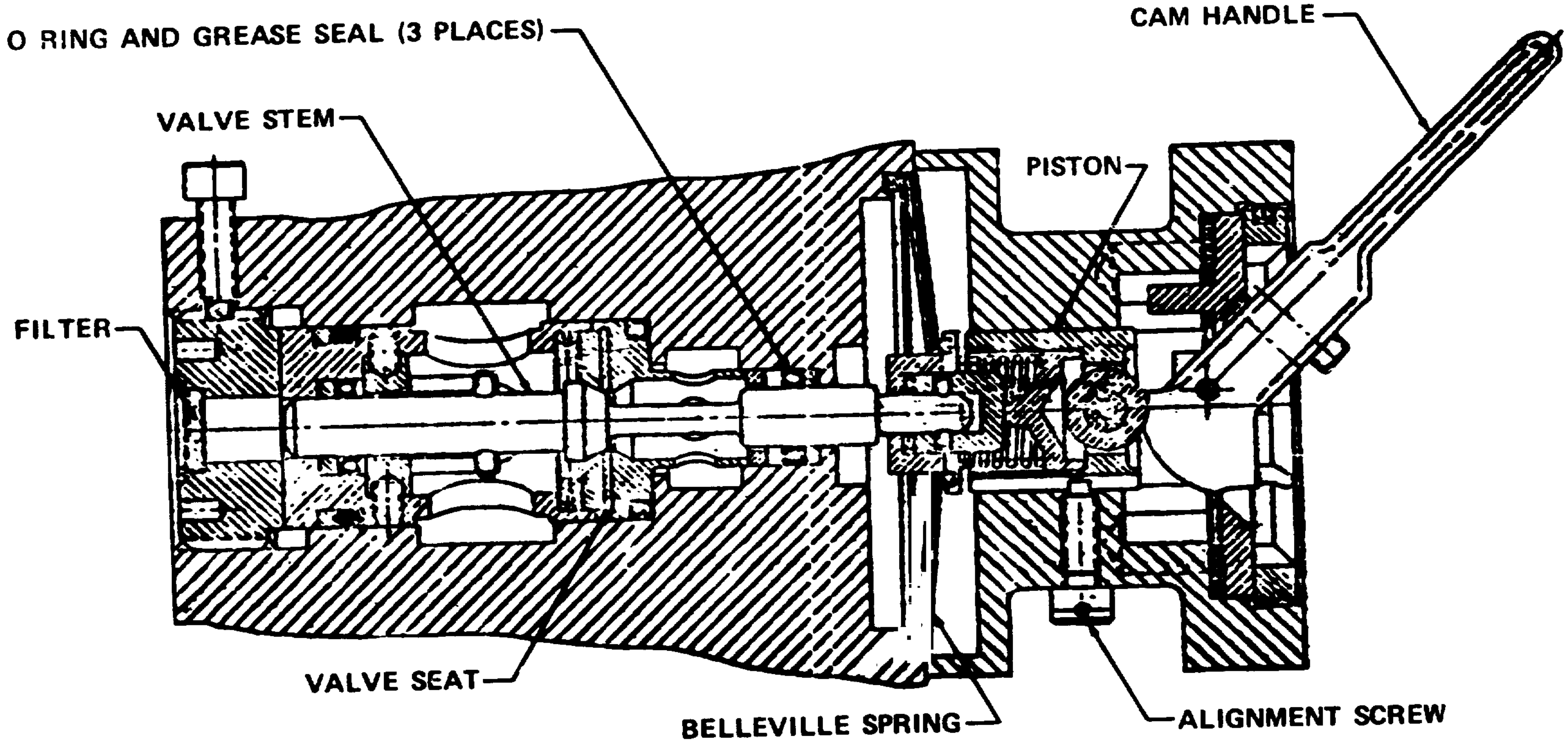


Figure VA-7. Cabin Pressure Regulator

Pressures and partial pressures of atmosphere, oxygen, and nitrogen were maintained at nominal values throughout the manned missions, except during extravehicular activities and operations with astronaut maneuvering experiments which caused pressures higher than the required 5.2 psia. Prior to operations with the astronaut maneuvering experiments, hand operated nitrogen valves were closed to provide a slightly oxygen-rich atmosphere since the experiments released nitrogen from their thrusters. The highest cabin pressure recorded caused by operation of these experiments was 5.8 psia during the third manned phase. After this experiment run, when total pressure had been reduced to near nominal, a crewman operated the manually controlled oxygen valve to increase oxygen content. When the atmosphere was not being pertubated, the cabin pressure regulator maintained pressure at approximately 5.05 psia.

(5) Nitrogen storage tanks. Required nitrogen stowage tank operating pressure is 3,000 psia at -20 to 160 °F with ambient external pressure at 10^{-10} psia. The volume of each of the six tanks is 19.2 ft^3 at zero psig and 70 °F. The tanks are required to support an estimated mission nitrogen usage of 978 lb. Each spherical tank is an assembly made from two 20-in. inside radius titanium hemispheres joined at the equator by TIG welding and strengthened by a backup ring.

The nitrogen tank assemblies (Figure VA-8) are similar to tanks used for the Saturn program. Qualification by similarity included vibration to 7.0 g rms and acceleration to 9.5 g. Skylab qualification included temperature extremes to 160 and -20 °F at 3,000 psi, proof pressure at 5,000 psig, burst pressure at 6,660 psig, and leakage at the maximum rate of 0.0021 x 10^{-3} scc/second helium at 3,000 psi. Operating requirements were 6,658 hr. Demonstrated capability included 240 days at 3,000 psi with temperature at 160 °F and 100 cycles from 0 to 3,000 psig.

Tank operation was normal except that the combination of high loading prior to lift-off, high beta angles during the first manned phase, and no usage of the M509 experiment caused pressures to exceed established limits on tanks #1 and #2, but did not exceed tank design capability. The nitrogen tanks and associated tubing exhibited no detectable leakage for the duration of the Skylab mission.

(6) Nitrogen pressure regulator. One unit is required in Skylab to reduce the stored gaseous nitrogen from 3,000 to 150 psig. The unit flows 0.38 lb/min at 70 °F and 3,000 psia inlet pressure.

Nitrogen regulator assembly design was based on a similar design used in Apollo. Two in-line changes were incorporated: internal springs were changed because of stress corrosion of maraging steel Bellville springs which were replaced with 17-7PH springs and internal seals were changed from LS-53 O-rings to Parker compound S-383-7 because of excessive leakage. Skylab qualification tests included

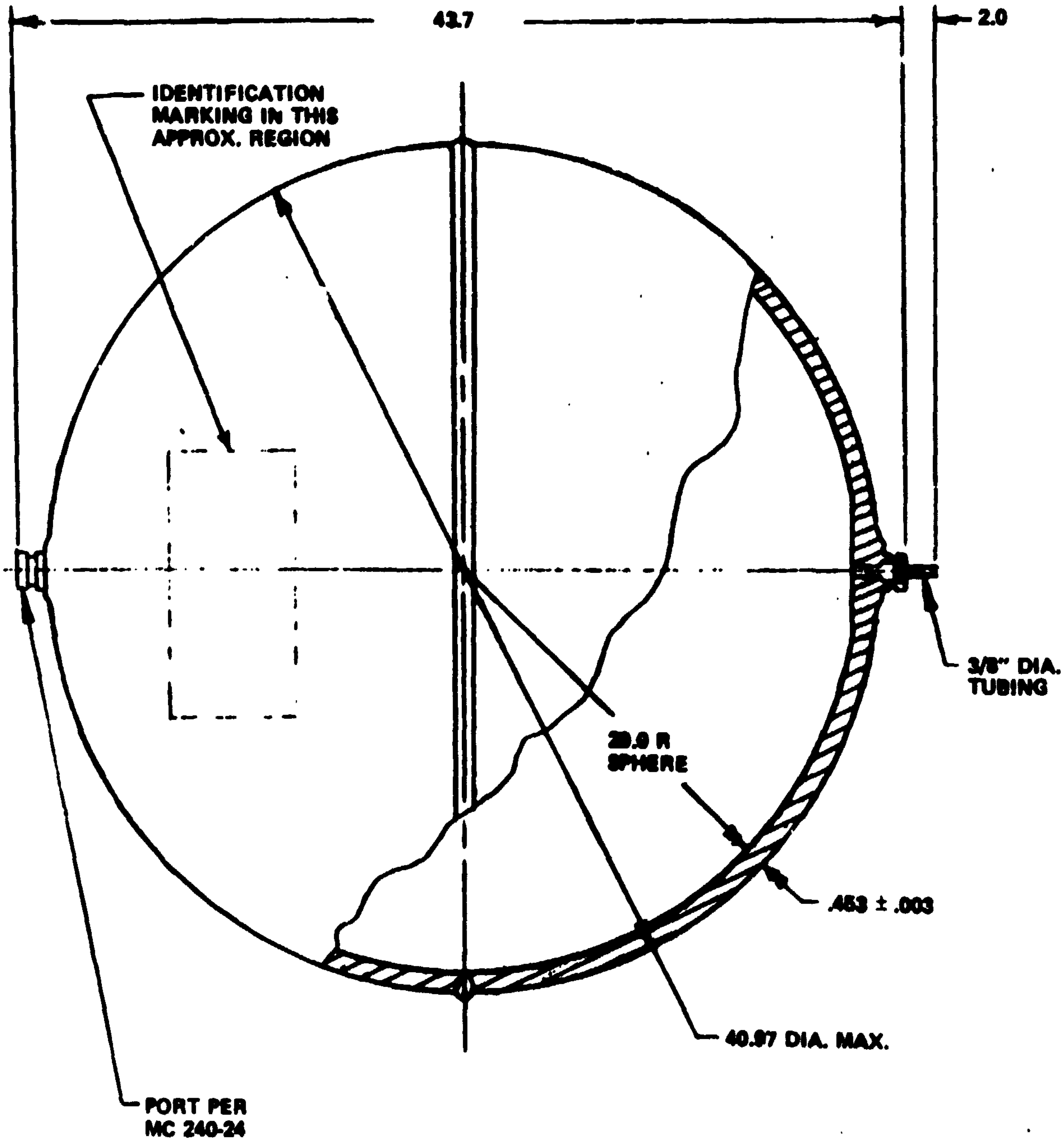


Figure VA-8. Nitrogen Storage Tank

vibration to 7.0 g rms, acceleration to 7.4 g, altitude to 10^{-8} psia, temperature extremes to 160 and -20 °F, and burst pressure to 12,000 psig for the regulator and toggle valve and to 840 psig for the check valve. Operating requirements were estimated to be 5,938 hr, 675 regulating cycles, and 80 shutoff cycles. Demonstrated capability was 10,000 regulator cycles, and 100 relief valve cycles plus performance during the 8-month endurance test. During acceptance, the regulators and check valve were proof tested to 6,000 and 450 psig, respectively, and leaked checked. At lockup, internal maximum leakage was 3.27×10^{-4} lb/hr nitgrogen. Check valve leakage was less than 3.15×10^{-5} lb/hr oxygen at 210 psig. Shutoff leakage was less than 3.15×10^{-5} lb/hr oxygen with delta pressure of 3,000 psig.

The nitrogen pressure regulator exhibited anomalous behavior by allowing regulated pressure to drift downward. It had dropped from 160 psia on the 5th day of the first manned phase to 140 psia on the 21st day of the first manned phase. Two days later, the crew verified that both nitrogen regulator toggle valves were open and the on-board meter was reading 140 psia. Telemetry also read approximately 140 psia. No further decrease was seen during the first manned phase. When the molecular sieve was deactivated on the last day of the first manned phase, the outlet pressure began to increase and was 175 psia 3 days later, at which time the lines downstream of the regulator vented to 4 psia.

At activation by the second Skylab crew, the nitrogen regulator outlet was 158 psia. Twenty-days later, the pressure had dropped to 150 psia. The pressure contintued to drift downward until it was 141 psia on the 28th day of the second manned phase. On this day, during extravehicular activities, the pressure increased to 145 psia. The reason for this is unknown. The pressure decreased to 141 psia the following day.

Since the terminating of nitrogen regulator flow had somehow restored the outlet pressure at the end of the first manned mission, it was decided to close one regular toggle valve and leave it closed for 5 days in an attempt to increase the outlet pressure. Five days later, on the 34th day of the second manned phase, the regulator outlet pressure was 140 psia. At that time, toggle valve A was opened and toggle valve B was closed. The regulator outlet pressure immediately increased to 155.5 psia. Five days later, pressure had decreased to 151 psia. Toggle valve A was closed and valve B opened. The regulator outlet pressure fell immediately to 148 psia and in 5 more days had decreased to 145 psia. Toggle valve B was closed and toggle valve A was opened and the pressure increased to 155.5 psia. It was decided that rather than switch toggle valves every 5 days, they would be switched when the nitrogen regulator outlet pressure approached a zero flow condition. By deactivation the pressure had decreased to 146 psia. When the molecular sieve supply was closed, the regulator outlet pressure increased to 165 psia.

At deactivation by the second Skylab crew, both toggles A and B were opened per procedure. The supply solenoids were closed and the regulator pressure slowly drifted down as the system leaked during storage. The toggle valve configuration was not changed during activation by the third Skylab crew. On the seventh day of the third manned phase, toggle valve A was closed in 26 days. The pressure drifted only from 155 psia to 148 psia. At that time the crew inadvertently closed the open toggle valve B. When they reconfigured the system , they opened A and the pressure went up to 160 psia. During the next 18 days the pressure had drifted to 150 psia and by the end of the mission, the pressure was still 150 psia.

In addition to the above described onboard troubleshooting, ground tests were run to try to duplicate regulator characteristics. These included low demand and moisture tests, but the symptoms were not reproduced. The reason for the drift in outlet regulator pressure has not been established. The oxygen regulator is mechanically very similar, but did not display these drift characteristics.

(7) Water tank pressure regulator. One regulator of this type is required in the system to reduce gaseous nitrogen pressure from 150 psia to 5 $\pm$0.2 psia at a flow of 0.005 to 0.05 lb/hr for control of pressure in the telescope mount control and display coolant water tank and the two suit cooling loop water tanks. Inlet pressure range for operation is 80 to 210 psia and lockup was required at 5.5 psia outlet maximum.

The maximum acceptable leakage rate was 2.92 x 10^{-5} lb/hr nitrogen, both internally for the shutoff condition and for external leakage, with pressure at 210 psid. In the lockup condition, maximum allowable leakage was 3.24 x 10^{-5} lb/hr nitrogen. Estimated operating requirements were 5,938 hr and 1,200 cycles. Skylab capabilities demonstrated included 100 full cycles and 10,000 automatic cycles plus operation as one of the units included in the 8-month endurance test.

The water tank pressure regulator operated properly throughout the mission. During launch the relief portion of the regulator vented the airlock water tanks from sea level ambient to 6.0 psia and locked up. This was well within the stated requirements. The regulator maintained tank pressures of 5.2 to 5.5 psia throughout the Skylab mission. This was also well within the stated requirements.

b. Thermal control system. Passive thermal control systems are incorporated into all Skylab modules. The requirement for active thermal control for the habitable Skylab volume and some externally mounted components is provided by the airlock. The system is required to reject heat through a radiator. In addition to the radiator, system design required pumps, valves, heat exchangers, thermal capacitors, cold plates, filters, reservoirs, switches, sensors, orifices, flowmeters, and lines to control coolant circulation and heat transfer. Functional relationship of the system is as shown in Figure VA-9. System design criteria required two redundant coolant loops with operational modes selectable: by the crew, and by automatic switching enabled/disabled by ground command.

(1) Pump assemblies. The requirement to provide 230 lb/hr minimum coolant (Coolanol) flow at 120 °F maximum and 50 psid for the Skylab mission duration was satisfied by installing two pump assemblies in each coolant loop. The nominal operating mode is one pump in each loop during both manned and unmanned mission phases. The backup manned mode operates two pumps in the same loop. The secondary system operated with two pumps during the period when the primary system was shut down because of leakage. Two pumps were also run in the primary loop late in the third manned phase to compensate for high temperature during high beta angles.

No serious pump problems arose during the development, or during ground verification tests of the coolant systems. All flight pumps were satisfactorily operated with one and two pumps per loop during ground checkout operations.

Qualification testing for Skylab consisted of pump usage in conjunction with other hardware operated during the 8-month endurance test. Estimated mission operating time for the Skylab pumps was 5,938 hr.

A discrepancy occurred prior to launch for the first Skylab crew when an automatic switchover from primary to secondary loop occurred twice for no detectable reason. When the primary loop was later operated successfully using the backup automatic switchover sensors, it was concluded that a faulty sensor was the cause of the discrepancy. Pump inlet pressure, pressure rise (delta pressure), and outlet temperature were monitored throughout the mission.

On day six of the first unmanned period, while the primary inverter #1 (pump A) was running, the coolant loop switched automatically from the primary to the secondary loop. Switchover measurements were well above the automatic switchover limits. About 5 hr later, primary inverter #1 (pump A) was again selected and both auto switchover groups were enabled, but after 19 min of operation the coolant loop again switched from primary to secondary. Telemetry indicated that auto switchover group #1 sensor circuitry initiated the first switchover, but it could not be determined which group initiated the second switchover. The primary inverter #1 (pump A) was commanded "on" a few days later with only switchover group #2 enabled and performed normally. The primary coolant loop was operated several times later in the mission with group #2 sensor circuitry without problems.

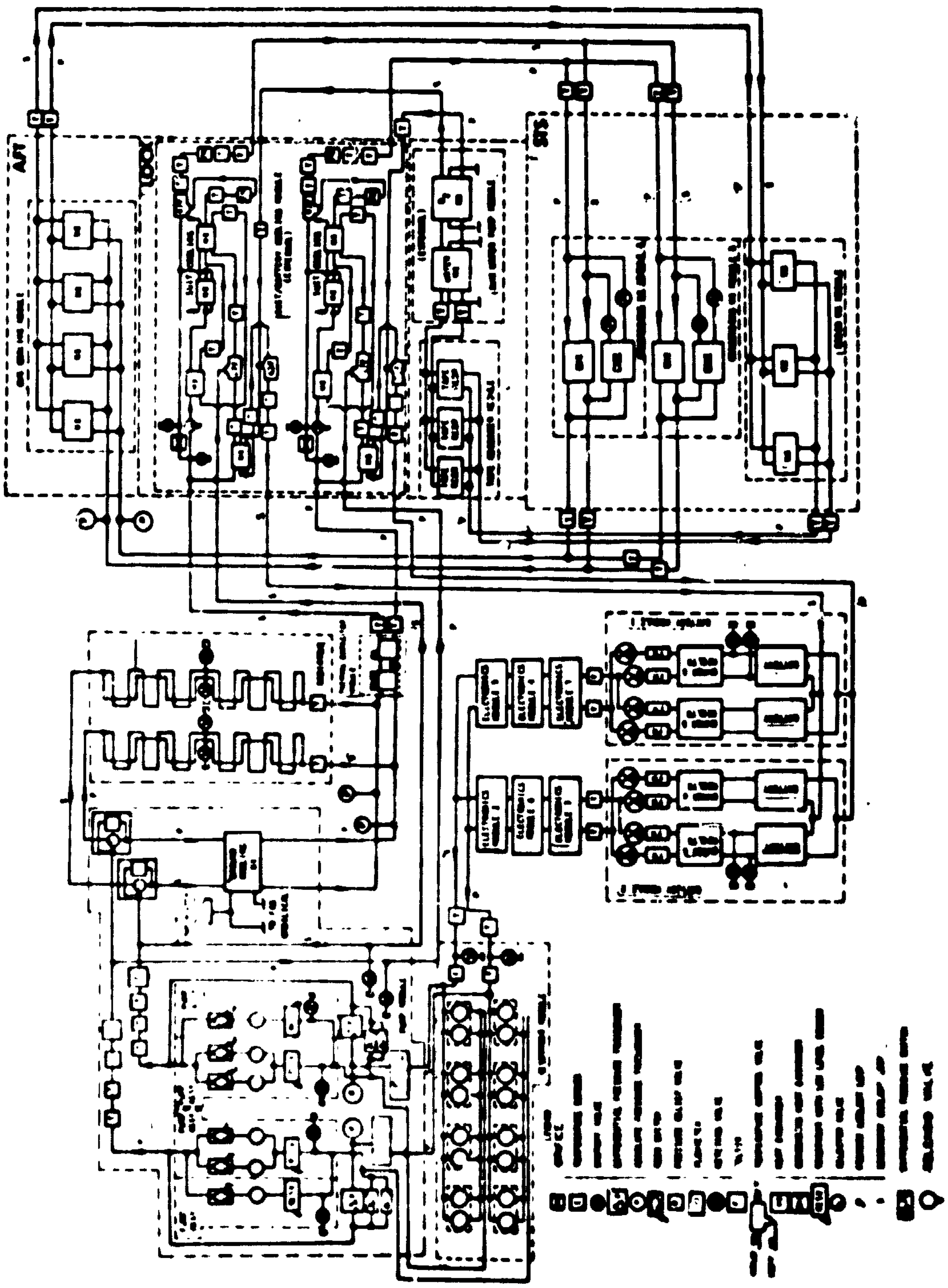


Figure VA-9. Thermal Control System

On day five of the first manned phase, while operating the secondary coolant loop pump A and inverter #1, the inverter circuit breaker opened. Available data indicated normal operation at the time. Following the third manned phase, a recheck of pump A and inverter #1 proved the problem was in the inverter electronics, not the pump.

(2) Temperature control valve (47 °F). The valves were required to control their outlet temperatures to 47 $\pm$2 °F at 240 lb/hr flow and 50 psia outlet pressure. Relaxation to $\pm$3 °F was allowable when flow was 460 lb/hr with outlet pressure of 110 psia.

The 47 °F temperature control valve (Figure VA-10) has two inlet ports, one for hot fluid and one for cold, and one outlet port. It uses a spring opposed thermal actuator to position a flow regulating spool, which varies the relative size of the hot and coid inlet is filled with Dow Corning DC200 working fluid and is a sealed unit with a stainless steel bellows.

The airlock installation incorporates four valves: two each of part number 61V830062-1 and two each of part number 61V830062-3. Dash numbers reflect differences in the allowable internal leakages and pressure drops.

During qualification, valves were subjected to high level random vibration, acceleration to 7.4 g, burst test to 920 psig, vacuum at 10^{-8} psia, internal and external leakage, and functional capability. High and low temperature extremes for testing were 120 and -65 °F. Developmental testing included simulation of the temperatures, pressures, and duration of the expected flight. Because of Skylab operational estimates and 150,000 partial cycles, with 200 of these cycles at temperatures from 120 to 0 °F and return to 120 °F. Each flight valve was proof pressure tested to 460 psig.

During the first extravehicular activity of the first manned phase, the 61V830062-1 valves (thermal control valve B) in each coolant loop stuck in a position that provided colder flow than desired. The primary loop valve stuck in a colder position than did the secondary loop valve and the first extravehicular activity was conducted using the secondary coolant loop and suit umbilical system #2 for both crewmen.

During the period when two crewmen were conducting extravehicular activities, the third crewman in the airlock structural transition section performed the following:

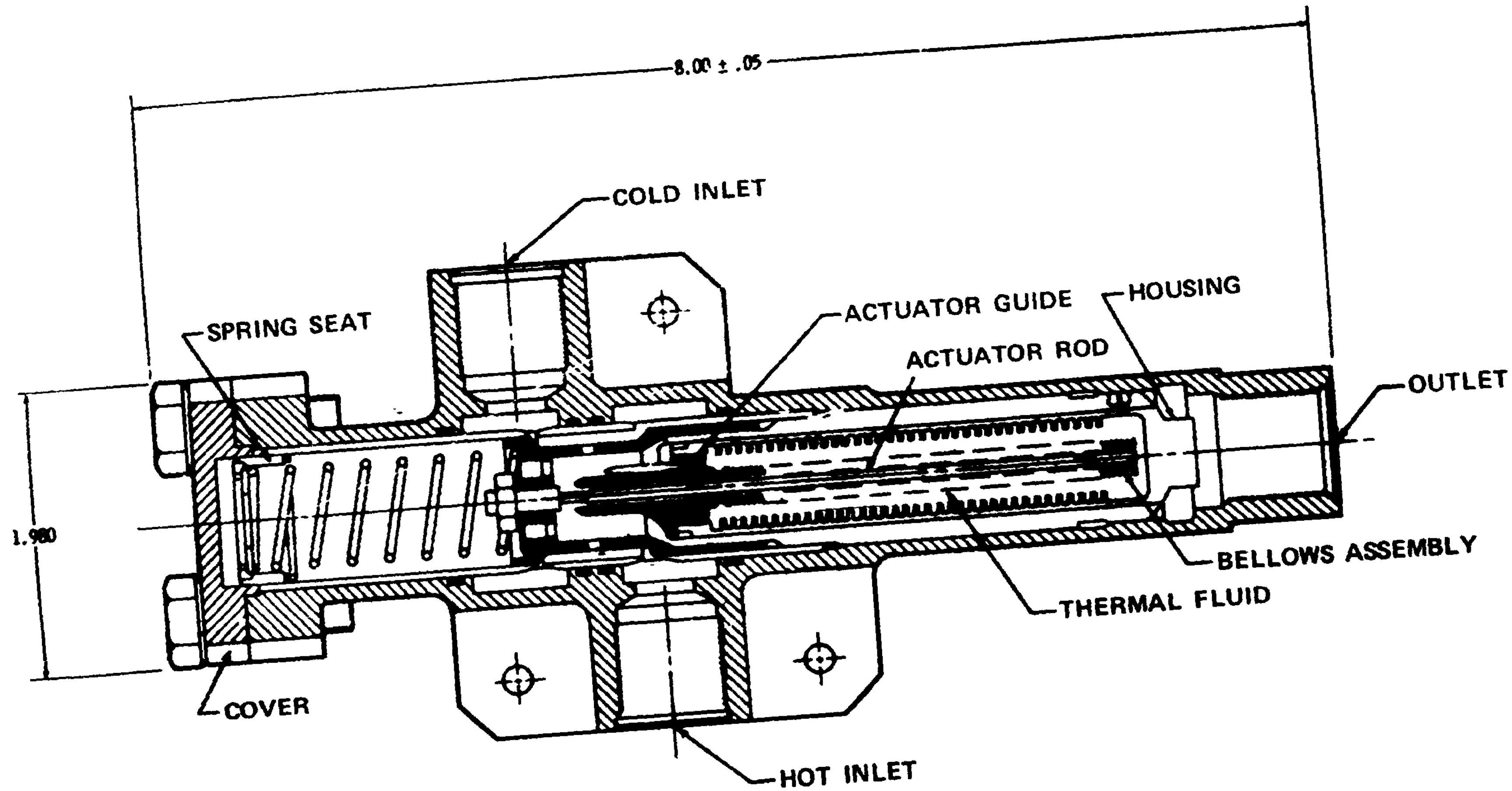


Figure VA-10. Temperature Control Valve (47°F)

- Turned both pumps on in suit umbilical system #1 with no flow recorded indicating frozen condition.

- Turned on primary coolant loop inverter #1 (pump A) with bypass valve in "EVA" position. Temperature control valve outlet went off scale low at 0 °F. Inverter and pump turned off.

Following extravehicular activities, after suit umbilical system #2 was turned off, the secondary coolant loop thermal control valve moved toward an interim position to maintain a 47 °F outlet temperature, but it again stuck. Without the heat load from the suits, valve outlet temperature dropped to 35 °F, causing a caution alarm. The secondary coolant loop was turned off and the primary turned on. The molecular sieve fan became noisy, probably caused by frozen condensate. The coolant loops were switched from primary to secondary and another caution and warning alarm occurred. Various alternate operating modes were tried with no success. The telescope mount control and display panel was powered up to approximately 170 W to add heat via its cooling loop to the secondary coolant system. Also, two liquid cooled garments were attached to suit umbilical system #2 and placed by the warmest workshop water tank. This combination led to a stabilized 40 °F temperature at the secondary coolant loop thermal control valve outlet.

Warming of the thermal control valve sensor cartridge causes it to expand, providing a positive movement. Cooling of the sensor cartridge causes it to contract and the opposite movement is provided by a spring. The sensor cartridge in the nonoperating primary loop was allowed to warm, providing sufficient force to free the thermal control valve. About 48 hr after completion of the extravehicular activity, when the primary system was operated with two pumps, the thermal control valve outlet temperature returned to 47 °F. Operation with one pump was also satisfactory.

Four days after the extravehicular activity, the second coolant loop thermal control valve outlet temperature was still 40 °F. Pumps were turned off for about 22 hr and when again turned on, the thermal control valve modulated, maintaining an outlet temperature of 47 °F.

Bypass valves were in the "BY-PASS" position when the thermal control valve cartridges were warmed and they remained in the "BY-PASS" position throughout the remainder of the Skylab mission.

As a result of the sticking thermal control valves in both coolant loops, a heater and controller were built and launched with the second crew to provide the capability to add heat to the suit cooling loop and thereby add heat at the hot inlet to the thermal control valves, should either of them again stick. The heater and controller were never used.

After both the primary and secondary coolant loop thermal control valves were unstuck, both maintained temperatures within the control band of 47 ±2 °F. Review of performance data indicated that the secondary loop valve was somewhat sluggish. As a result, subsequent extravehicular activities were conducted with all crewmen on suit cooling system #1 with the primary coolant loop until its loss during the second manned phase because of leakage. A decision was made to use the secondary loop rather than have the second Skylab crew perform their second extravehicular with only oxygen cooling. During this period of operations, the secondary coolant loop thermal control valve outlet temperature dropped from 47.3 °F to 41.7 °F and stayed at approximately that temperature for the remainder of the second manned phase.

The third extravehicular activity by the second Skylab crew was accomplished with oxygen cooling since the primary loop did not have sufficient coolant and the thermal control valve in the secondary loop might stick in a less desirable position if liquid suit cooling was used again during extravehicular activities.

The outlet temperature of the secondary coolant loop thermal control valve dropped to approximately 40 °F during the third unmanned period when the loads on the system were very low. No attempt was made to free the valve since it was stuck in an acceptable position, and based on past experience when the loop was cycled off and on, in an attempt to unstick the valve, the valve outlet actually moved to a colder position than before. Had this occurred, the heater may have been required to maintain acceptable temperatures during the third manned phase if the planned primary coolant loop reservicing was not successful.

The secondary loop was operated with the thermal control valve stuck until the earth observation pass on mission day 58 of the third manned phase when the higher radiator outlet temperature increased the valve inlet temperature. The hot leg flow was decreased to 114 lb/hr during the pass, but returned to the 40 °F position following the pass. During the earth observation pass on mission day 60 of the third manned phase, the valve decreased the hot flow to 0 lb/hr, but again returned to the original position. During the earth observation pass on mission day 64 of the third manned phase, the valve came unstuck and began to modulate following the pass. It continued to modulate until the end of the mission.

After reservicing the primary coolant loop during the third manned phase, all extravehicular activities were accomplished using the primary coolant loop and suit cooling system #1.

(3) Temperature control valve (40 °F). The two temperature control valves (one in each loop) are used to maintain a nominal 40 °F coolant temperature to the airlock batteries by controlling the mix

of cold fluid from the radiator via a heat exchanger and hot fluid from the outlet of the cooling system. The design flow rate requirement was 500 lb/hr at 40 °F with temperatures to 120 °F. Allowed pressure drop was 1.22 psid.

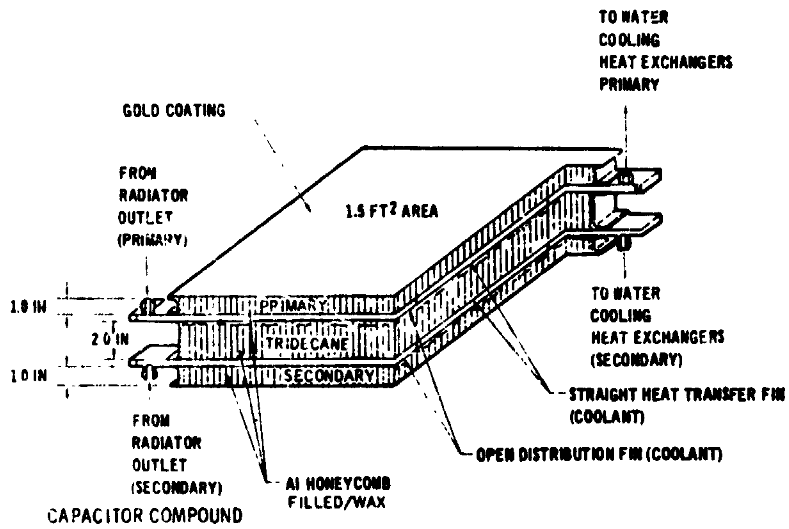
The 40 °F temperature control valve has two inlet ports, one for hot fluid and one for cold, and one discharge port. It uses a spring-opposed thermal control element (wax mixture with a high coefficient of expansion) to position a flow-regulating sleeve, which proportionately opens and closes the hot and cold inlet ports to provide a fluid mix of the desired temperature. "Flight operation was satisfactory."

(4) Thermal capacitor. This unit (Figure VA-11) is a phase change heat sink installed downstream of the radiator to supplement radiator performance. The unit consists of primary and secondary coolant loop cold plates sandwiched between honeycomb-type chambers containing 19.6 lb of paraffin (Tridecane). Melting of the Tridecane occurs at 22.35 °F with a heat of fusion of 66.5 Btu/lb. A structural transition occurs at -0.7 °F with an associated heat absorption of 17.9 Btu/lb. The capacitor is able to store heat by melting while the vehicle is on the hot side of the orbit and reject heat (freeze) on the cold side. Flow of 220 lb/hr minimum coolant at 75 °F, with a maximum pressure drop of 2.1 psid was required. Cold plate operating pressure was 140 psig and wax chamber operating pressure was 40 psig.

Two problems were encountered during Skylab testing. The capacitor supplied for use in the workshop refrigeration system developed a crack during bench test at MDAC-W; it was determined to be the result of more rapid temperature swings than the airlock qualification provided. Also, Napco foam insulation bulged and debonded from the capacitor when exposed to vacuum. The foam specimens tested continually failed when exposed to vacuum. The rapid temperature swing problem required redesign and requalification and the foam problem was overcome by replacing it with multilayers of fiber glass sheets.

Qualification tests included high-level vibration, altitude to 10^{-6} psia, temperature extremes of 160 and -140 °F, burst pressure to 350 psig, 200 complete freezing and thawing cycles, 2,400 partial freezing and thawing cycles, and temperature control performance. The thermal capacitor melted only during the high beta angle conditions that occurred during the third manned phase. Storage capability aided thermal control at this time and performance was as designed.

(5) Coolant filters. The coolant filter assembly was used in the airlock cooling system to filter all solid contaminants larger than 100 microns out of the coolant fluid. Two of these double



CAPACITOR COMPOUND

MATERIAL	TRIDECANE
WEIGHT	~20 LB
SPECIFIC HEAT	0.4
MELTING POINT	22.2° F
HEAT OF FUSION	66.5 Btu/LB
STRUCTURAL TRANSITION AT -6° F REQUIRES	18.6 Btu/LB

Figure VA-11. Thermal Capacitor

element filters were used, one at the inlet of the coolant reservoirs, and one at the outlet of the coolant pumps. The flow capacity requirement was 366 lb/hr at 230 psi and 80 °F with an allowed pressure drop of 1 psi.

Each filter contains two independent parallel flow path filter systems located side-by-side within one housing. Each filter system consists of a removable, irreversible filter element, a mechanical shutoff to allow ground replacement of filters, a nonadjustable relief valve to permit fluid to bypass the filter in the event of filter clogging, and a bleed valve to facilitate purging of the filter system of air. The housing assembly is made of aluminum alloy, the filters are stainless steel, and the seals are neoprene.

Estimated operational requirements were 5,938 hr and five cycles. Time requirements were satisfied by the 8-month endurance test. During Skylab qualification, a filter relief valve was successfully cycled 200 times. Each flight type unit was subjected to leakage and proof tests at acceptance. No leakage was allowed for a period of 15 min at 230 psig gaseous nitrogen. Proof pressure was 460 psig. System operation including pressures and pressure differentials indicate adequate operation of filter assemblies.

(6) Bypass valve. One latching solenoid valve (Figure VA-12) allows partial bypass of coolant around suit cooling heat exchangers in each cooling loop. Valve flow rate requirement was 450 lb/hr at 70 °F with 120 psia and the allowable pressure drop was 5 psid. The bypass valve, operated from the intravehicular activity panel, is a three-way, two position, latching solenoid valve. It has limit switches to indicate its position.

Maximum acceptable leakages were: internal, 0.21 x 10^{-5} lb/hr coolant; and external, 1 x 10^{-7} scc/sec helium with temperatures from 0 to 120 °F and the external body subjected to 10^{-6} torr.

The bypass valve in each coolant loop was changed from "BY-PASS" to "EVA" in orbit only once; at the beginning of the first extravehicular activity. When the 47 °F temperature control valves stuck, the bypass valves were switched back to the bypass mode. Adequate astronaut cooling was demonstrated in the "BY-PASS position and since there was some speculation that particulate matter from the heat exchangers may have caused the temperature control valve to stick, it was decided to leave the bypass valve in the "BY-PASS" position.

(7) Radiator bypass and relief valve. A pilot solenoid controlled, hydraulically actuated bypass valve was used to direct the coolant flow either through the ground cooling heat exchanger or through the space radiator. With power applied to the solenoid,

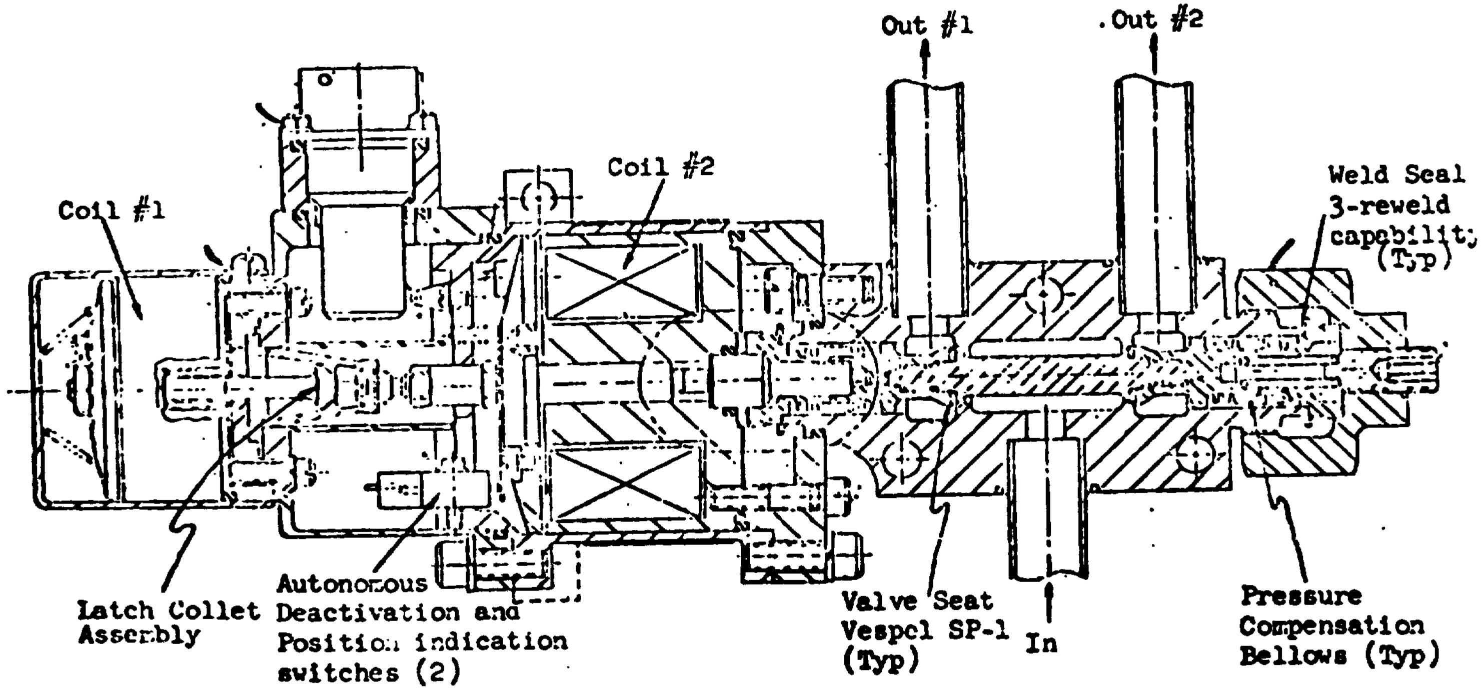


Figure VA-12. By-pass Valve

flow was through the ground cooling heat exchanger. When power was removed, the valve cycled to radiator flow. With coolant at 120 °F and 230 psig, the valve was required to permit 366 lb/hr flow with a maximum pressure drop of 2.5 psi through either flow path. A relief valve was required to bypass the radiator through the ground cooling heat exchanger path at a differential pressure of 220^{+10}_{-5} psid in the event of radiator freezing or other blockage.

The radiator bypass and relief valve was qualified for airlock use primarily by Gemini tests and usage. Estimated use was 5,982 hr and 251 cycles. This component was included in the 8-month endurance test to satisfy operating time requirements. It also passed a 500 cycle life test. Gemini qualification included vibration to 8 g, acceleration to 7.25 g, altitude at 5×10^{-4} psia, humidity to 95 percent with temperature cycles, high and low temperature extremes of 160 and -60 °F, leakage not to exceed 3.15×10^{-5} lb/hr gaseous oxygen at inlet pressure of 230 psig, proof pressure to 460 psig and burst pressure to 690 psig. Each valve was acceptance tested for leakage to the same requirements as the qualification leak test. The flight coolant systems were operated during ground checkout operations with the bypass valves in the ground cooling position with no problems or anomalies. At 10 min before launch, the primary bypass valve was switched to radiator flow. The secondary system was not operating at launch, so the secondary valve was already in the radiator position. Pump delta pressure telemetry indicated the valve did cycle. From system flow/delta pressure characteristics and cooling performance it has been determined that the relief valve was never cracked nor was internal leakage excessive. Thus it can be concluded that the radiator bypass relief valve performed as designed throughout the mission.

(8) Radiator. The radiator assembly consisting of 11 panels was required to reject heat to space and provide meteoroid protection for the structural transition section.

The 11 radiator panels (Figure VA-13) have a surface area of 432 ft^2. Four panels are mounted to the structural transition section and seven panels are mounted to the docking adapter. Magnesium extrusions, each with one coolant passage, were seam welded to the eleven magnesium skin panels. Extrusions were attached to provide two coolant loops. Fiber glass stringers were bolted to the panels and provide structural connection to the pressure vessels. Aluminum tubing provides coolant line connection to the radiators.

As a matter of interest, it is pointed out that following a period of pre-installation storage, it was discovered that some radiator skins had developed holes through the 0.050-in. thick (airlock) and 0.032-in. thick (docking adapter) skins. The problem was attributed to reaction of humidity with cleaning agent residue. Some skins were replaced and improved cleaning, passivating and rinsing techniques eliminated further occurrence of the problem.

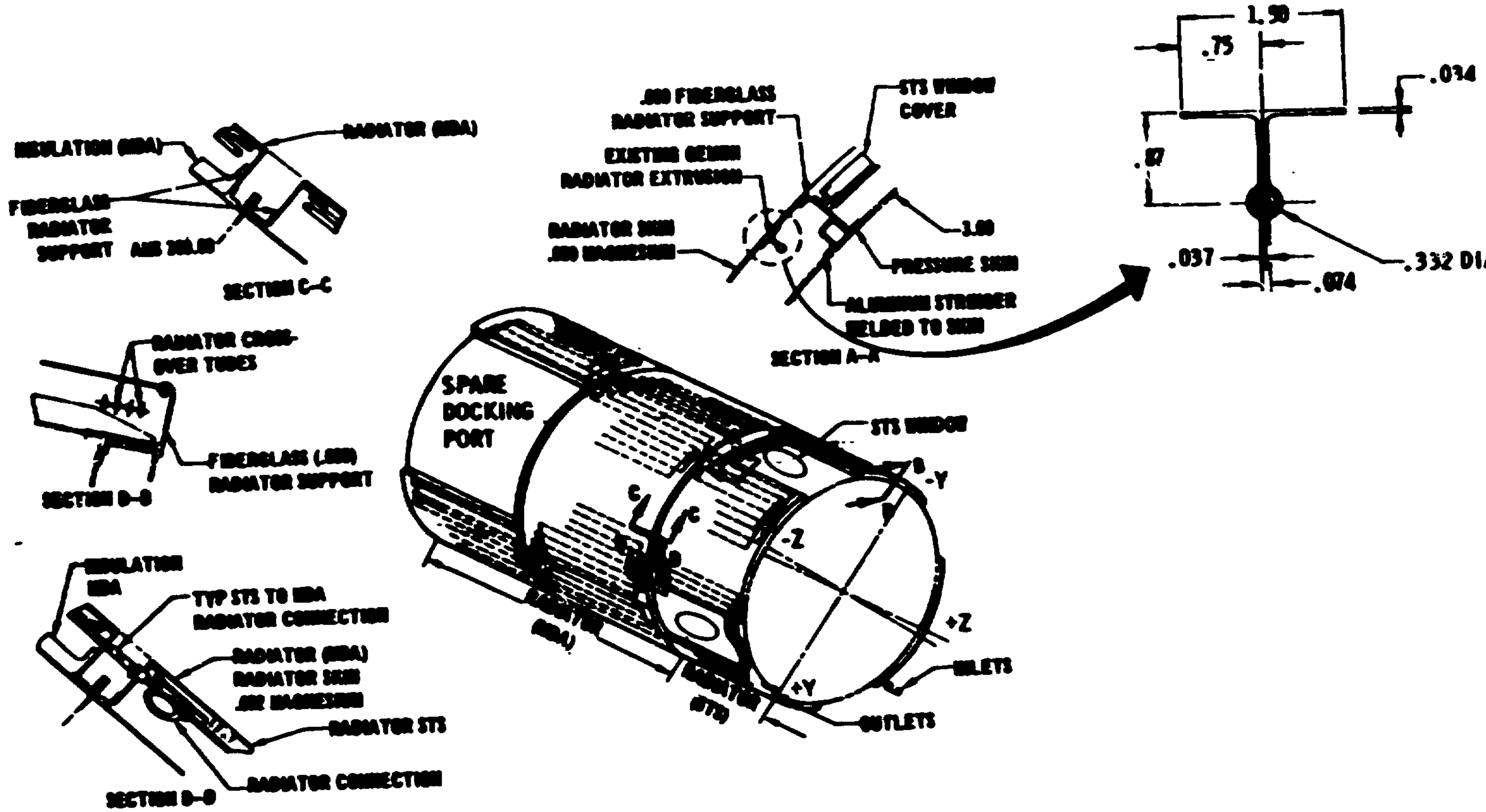


Figure VA-13. Radiator

Radiators were qualified to high temperature of 500 °F under 136 percent of design load, low temperature to -180 °F at 235 psig, vibration to 7.0 g rms, altitude to 1.74 x 10 psia with temperatures from -65 to 200 °F at pressure to 300 psig, tested to proof pressure of 460 psig and burst pressure of 920 psig, and leak tested at 920 psig without leakage. The radiators performed satisfactorily from a structural/mechanical point of view based on the absence of negative crew reports and instrumented data.

(9) Coolant system reservicing equipment. The subassemblies defined in Figure VA-14 were used to replenish coolant (Coolanol 15) lost through leakage that began to develop in the primary coolant loop on day 22 of the first manned phase. A decline in system mass of approximately 0.08 to 0.12 lb/day was indicated. Following day 13 of the second unmanned period, the average primary loop temperature began to stabilize while inlet pressure continued to decrease, a further indication of loss of fluid. At 0710 GMT on the ninth day of the second manned phase, the primary coolant loop reservoir "LOW" indication light illuminated on ground consoles and was confirmed by crew observations of onboard instrumentation. On the 27th day of the second manned phase at 1830 GMT, the primary coolant loop was shut down, after the pump inlet pressure reached a low point of 5.8 psia, to prevent pump cavitation and resulting pump damage.

During the latter portions of the second unmanned period, the secondary loop also began to lose mass. After the second manned phase began, the loss of fluid effectively stopped, but during the third unmanned period, the leak resumed. It was noted that the secondary loop "LOW" light came on momentarily on the last day of the third manned phase, but sufficient fluid remained for mission completion.

The second Skylab crew attempted to ascertain the location of the coolant leak. Panels were removed and insulation unwrapped from suspect lines. Wrapped lines were visually inspected for bulging, color changes, and wetness, but no evidence of leakage was found. During extravehicular activities, the crew inspected the accessible exterior areas, especially the radiators, for evidence of coolant leakage but none was found.

The method designed for reservicing the coolant system consisted of pressurizing a coolant supply tank with 35 psig gaseous nitrogen, thus forcing coolant into the primary loop through a saddle valve that was attached to and pierced an internally accessible coolant line. The reservicing hardware included 3 saddle valves, a preserviced 3-ft servicing hose, a tank module assembly, a 60-ft servicing hose, a quick disconnect adapter, and a 1-ft leak check hose. The tank module assembly was launched with 42 lb of coolant and nitrogen pressurized in a volume of 180 in.; the coolant and nitrogen were separated by a flexible teflon bladder. The tank was maintained at a positive pressure prior to and during launch by initially aerating the coolant to a dissolved gas content of 340 ppm by weight and pressurizing the tank to 26.7 psia for launch.

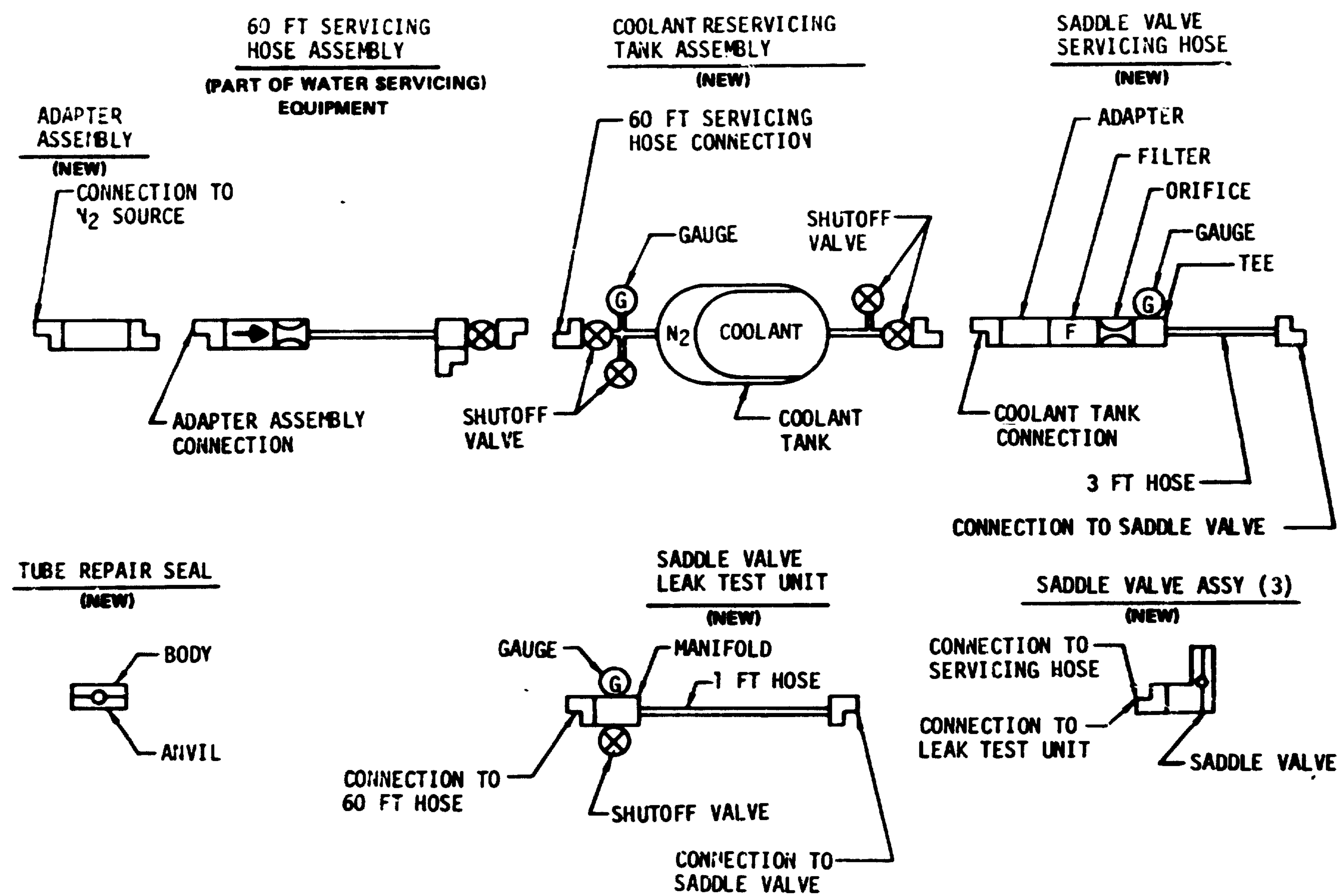


Figure VA-14. Coolant Reservicing Equipment

Coolant servicing was accomplished by: (1) piercing the primary coolant line by turning the saddle valve stem until the stem bottomed on saddle valve body, then retracting the stem to stop, (2) opening the coolant supply valve on the reservicing tank to establish flow into the primary loop, (3) closing the coolant supply valve on the tank module after the primary loop was serviced to the desired level (as indicated by the pressure gage on the 3-ft servicing hose), (4) turning the saddle valve stem until the stem bottomed on saddle valve body, and (5) disconnecting the service hoses.

Piece parts of the coolant reservicing equipment were primarily qualified by similarity to like equipment used in other airlock applications. The tank was adapted from a command module fuel tank. The saddle valve was developed by NASA. The total assembly was integrated, tested, and verified at MDAC-E. Verification included successful ground reservicing of the Skylab ECS/TCS test unit and the backup airlock coolant systems.

In-flight servicing began on the fourth day of the third manned phase at 2100 GMT, with removal of primary coolant line insulation at the cabin heat exchanger module. The servicing procedure progressed smoothly through attachment of the saddle valve (without penetration) and pressurization of the leak test hose with the 35 psi GN_2 from the workshop panel 500. The purpose of the leak check procedure was to verify that the saddle valve was not leaking prior to penetration. The leak check procedure involved pressurizing the saddle valve and leak check hose to a pressure greater that 30 psig, closing the supply valve in the 60-ft water servicing hose, and monitoring for 30 min. If the pressure decay was less than 2 psi, the servicing was to proceed. However, the leak test hose gage indicated an initial pressure of 33 psi and 35 min later 30.5 psi. After an additional 20 min, the pressure was down to 25 psi. Thus, a leak was indicated in the saddle valve or the leak check hose. To determine the location of leak, the crew was instructed to disconnect the leak check hose from the saddle valve and repressurize the leak check hose. The leak check hose alone showed a pressure drop of 2 psi in 20 min and 2.5 psi in 27 min, thus indicating the leak was in the leak check hose. The crew was then instructed to disconnect the leak check hose from the servicing hose and to connect the coolant servicing tank and the coolant servicing hose. The coolant valves were then opened to supply coolant to the saddle valve under pressure prior to piercing the coolant line. No coolant leakage was observed; the primary coolant line was then pierced and the servicing proceeded. On the fifth day of the third manned phase, at 0042 GMT, telemetry indicated a primary loop coolant pressure of 27.2 or 22.2 psig. Servicing was completed with no leakage at the saddle valve. Pump B was activated and obtained 65 psid across the pump. The crew also activated pump C for a two-pump operation for 5 min. All telemetry data indicated normal readings.

At 0349 GMT, the pilot confirmed coolant servicing was complete with saddle valve cover in place and all servicing hoses stowed. The coolant tank was placed on the mass measuring device and it was determined that 7.7 lb of coolant had been added to the primary coolant system.

The reservicing of the primary loop permitted return to the two loop operation of the coolant system during the periods of high beta angles and extravehicular activity (high heat load periods).

Conclusions: Thermal control valves were designed to close tolerances. It is recommended that future applications of this type use the greatest possible tolerances commensurate with effective operation. Also, future applications should provi filters upstream of moving parts in fluid systems to minimize potential contamination.

The exact cause of the primary and secondary coolant loop leakage may never be established. The is evidence that the leak could both be inside the pressure shell and outside the pressure shell. During the period following deactivation of the loop, the pressure in the loop fell below (2.5 to 3.7 psia) and remained below cabin pressure. This would indicate that the leak in the primary loop was external to the cabin. The premise that the leak was internal to the cabin is enhanced by finding a trace of coolanol constituents in the returned $PPCO_2$ cartridge. If a leak were interna. to the cabin, the most likely location would be under the molecular sieve "A" cover, based upon analysis of the returned cartridges.

The benefits obtained by the successful reservicing of the primary coolant loop were to restore two-loop operation and redundancy in the system. The value of two-loop operation was illustrated during the period of high beta angles, day 62 through day 70 of the third manned phase. The internal temperature would have been excessive without two-loop operation.

The use of backup hardware and the ECS/TCS Skylab test unit for simulations and testing proved invaluable during the development of the reservicing kit.

c. Suit cooling system. The dual suit cooling systems consist of coolant reservoirs, pumps and motors, heat exchangers, an EVA/IVA liquid gas separator, check valves, relief valves, and connecting coolant lines. Functional relationships are shown in Figure VA-15. The system is required to be capable of transferring 2,000 Btu/hour from each of the two water loops to the airlock coolant system. Pressure drop from components can not exceed 12 psid at a water coolant flow rate of 250 lb/hr per loop. The coolant flow rate at the suit umbilical is required to be 200 lb/hr minimum per loop. Cooling is regulated by adjusting the flow rate of temperature controlled water through each liquid cooled garment using a flow diverter valve in the pressure control unit.

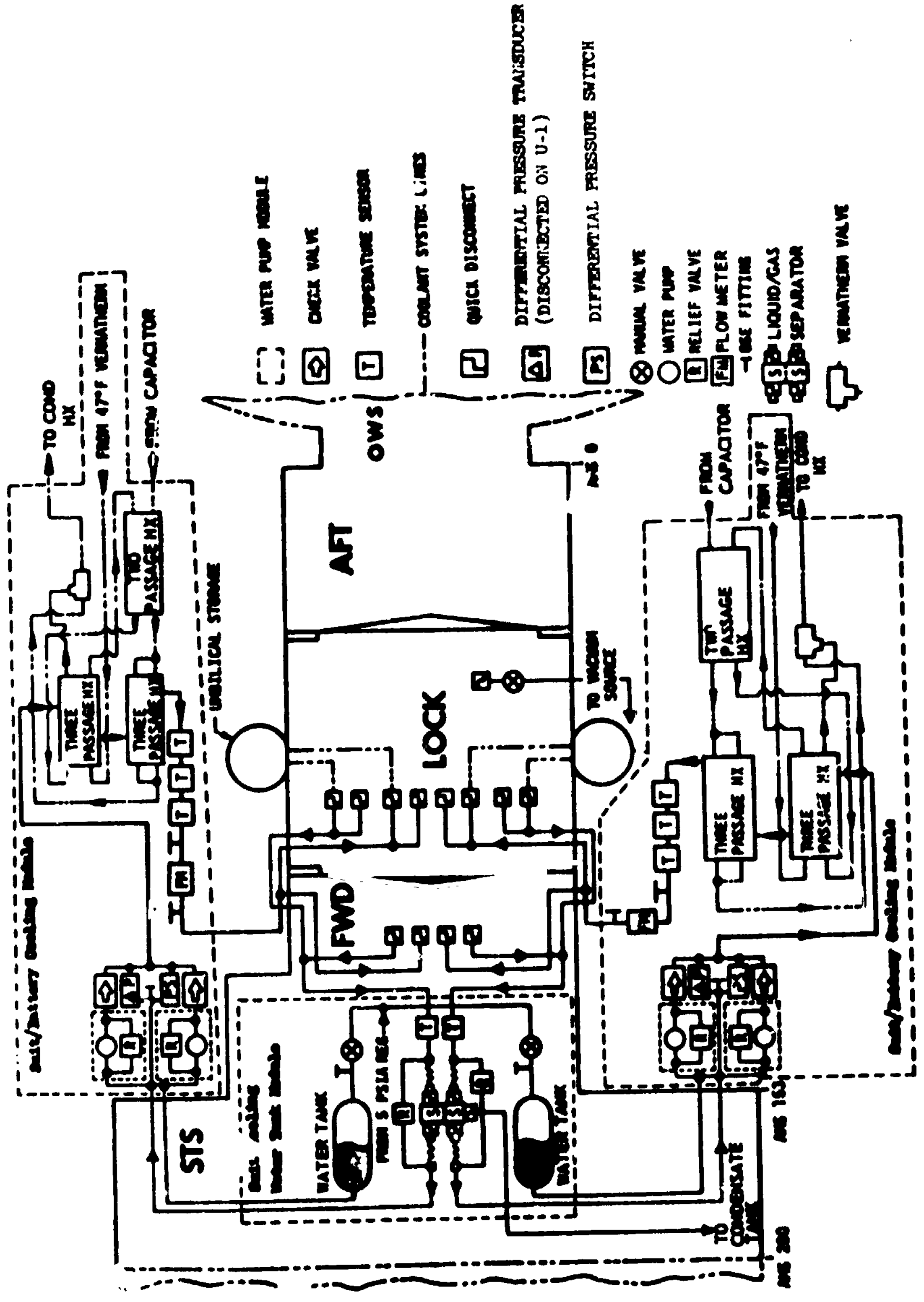


Figure VA-15. Suit Cooling System

(1) Water pump. Each suit cooling system contains two pumps in parallel for redundancy. Flow characteristics ranged from 220 to 280 lb/hr with a minimum delta pressure of 11.0 psid, with an input voltage of 22 to 30 Vdc. Inlet pressure is 5.0 to 23.1 psig.

The water pump is a positive-displacement, rotary vane, electrically powered pump assembly consisting of five subassemblies: (1) pump, (2) relief valve, (3) ac electric motor, (4) dc to ac inverter, and (5) outer housing that encloses the entire assembly. Pumps can continue to operate with the outlet line blocked because the internal relief valve allows flow from the outlet back to the inlet side of the pump when the outlet pressure builds up to the relief valve cracking pressure. Most structural parts are made of corrosion-resistant steel and the bearings are carbon journals. The entire unit is hermetically sealed by welding. The motor stator and inverter are separated from the rotor and pump and are sealed in an inert atmosphere.

On the ground water pumps in the suit cooling system experienced failure to start after a long off period. The problem was traced to a nickel phosphate precipitate. Corrosion inhibitors were changed to chromates and pump clearances were increased, resulting in subsequent normal operation.

The modified water pump assembly was partially qualified by similarity to a previously qualified unit used on a classified Air Force research vehicle. Qualification by similarity included shock to 100 g while not operating and to 15 g operating, acceleration to 7.8 g, altitude to 200,000 ft, humidity at 95 percent minimum with temperature cycling, and temperature extremes to 160 ± 5 °F and -35 °F. Airlock qualification included vibration to 9.6 g rms and burst pressure at 144 psig. Operating requirements were established at 3,610 active hours and 2,328 storage hours plus 20 operating cycles. Demonstrated capability included 2,500 hr operating time and 100 cycles. This item was also included in the 8-month endurance test. Acceptance testing of flight type articles included proof pressure to 72 psig and leakage not to exceed 1.5×10^{-5} lb/hr water at operating pressure. Operation during the Skylab mission was adequate.

(2) EVA/IVA liquid gas separator. These units provide a means of removing free gas from the suit cooling systems and also act as a particulate filter. The flow requirement is 200 to 350 lb/hr of coolant with 20 sec/min gas with an allowed pressure drop of 1.25 psid. Gas removal efficiency was 95 percent of 20 sec/min influent free gas at normal coolant flow while the gas discharge pressure was 0.5 to 6.2 psi below inlet water pressure.

The EVA/IVA liquid gas separator contains parallel hydrophobic and hydrophillic surfaces. Coolant water entering the separator passes between the parallel surfaces and is separated from any free gas present by the selective flow characteristics of the two surfaces. The hydrophobic surface permits only gas to pass to a gas collection manifold and ultimately on to the water condensate system. The hydrophillic surface essentially permits only water to pass through. The stainless steel screen construction of the hydrophyllic surface also serves the function of a particulate filter. The EVA/IVA liquid gas separator contains two independent coolant flow paths, one for each EVA/IVA coolant system, and one overboard gas manifold that is common to the two coolant flow paths.

The liquid/gas separator assembly was developed for Skylab. Qualification tests included vibration to 6.0 g rms, shock to 180 g peak between 500 and 2,000 Hz, altitude to 1.93 x 10^{-8} psia, humidity to 100 percent with temperature cycling, temperature extremes to 120 and -140 °F, burst pressure to 350 psig and leakage tests. Allowable leakages were measured for three conditions: at 14 psig liquid to gas discharge pressure, maximum leakage per coolant loop was 0.001 scc/m. At pressures from 9.3 to 140 psig, case leakage maximum was 2.98 x 10^{-5} lb/hr air. At 9.2 psig, gas discharge pressure, check valve maximum leakage was 2.98 x 10^{-5} lb/hr air. Operation during the Skylab mission was adequate.

(3) Water tank. Five water tanks are used for water storage for the various systems within the airlock, with one tank being required as a reservoir for each suit cooling loop. The capacity of each tank is 16 lb water at 5.5 psia at 115 °F. The external pressure rating is 15 psid. Maximum internal pressure rating is 6 psid.

The tank is a cylindrical vessel made of epoxy-polyurethane copolymer material. A flexible Viton diaphragm divides the vessel into a water chamber and a gas chamber. The gas chamber is pressurized by nitrogen from a 5 psia (nominal) regulator for tanks used in the various water cooling systems.

At KSC, it was discovered that a Viton bladder had split in the tank in the telescope mount control and display cooling loop. This discrepancy was believed to be the result of physical changes in the Viton material caused by long-term exposure to water and additives. Bladders in all tanks were replaced with new units and no additional problems occurred.

Skylab operating estimates were 3,[illegible]10 hr and 150 cycles. Gemini qualification demonstrated 3,949 hr operation. As a part of the hardware installed for the 8-month endurance test, the tank demonstrated its capability for the Skylab mission. At acceptance, flight tanks were subjected to proof pressure tests at 28.5 psig and leak tested at maximum operating pressure. Maximum allowable leakage was 3.15 x 10^{-5} lb/hr gaseous oxygen. Oribtal operation was without incident.

d. Telescope mount control and display console/EREP coolant system. The telescope mount control and display console/EREP coolant system is used for temperature control of the telescope mount control and display panel and earth observation equipment. The cooling system, Figure VA-16, consists of a telescope mount tank module located in the airlock structural transition section and a telescope mount water pump module located exterior to the structural transition section. The telescope mount tank module contains a water tank, filter, and filter bypass relief valve. The telescope mount water pump module contains three parallel plumbed positive displacement rotary vane water pumps and a ground cooling type heat exchanger interfacing with the MMS-602 coolant loops. Each pump has an integral bypass relief valve, a differential pressure transducer for telemetry, and an outlet check valve. A flow transducer upstream of the pump module inlet provides telemetry measurement of system flow. The single water loop is capable of removing 1,335 Btu/hr from the docking adapter load and 102 Btu/hr from the operating pump. The system delivers water to the docking adapter at a temperature between 40 and 75 °F at a flow rate of 220 lb/hr minimum. The maximum water delivery pressure is limited to 37.2 psia by a relief valve. Location of the heat exchanger in the airlock coolant loop is downstream of the tape recorder cold plates to avoid impacting atmosphere conditioning and suit cooling module performance.

The requirement to remove 98 percent of contaminants 10 microns or larger and all particles above 25 microns from this system is accomplished with a filter assembly (Figure VA-17) consisting of a head, bowl, and filter element. The unit has particulate capacity of 1 g/gal/min. The head provides inlet and outlet ports in straight-through alignment, three nonsymmetrical mounting bosses, and a threaded opening for attaching the cylindrical bowl. The bowl holds the filter element in place in the head. Water enters the inlet port on the head and flows into the bowl. The water passes from the periphery of the bowl through the filter element and center support core to the outer port. The filter element is made of sintered stainless steel mesh. All other parts are stainless steel except for a Viton O-ring between the bowl and the head.

The water filter was partially qualified by similarity to porous filters used by The Boeing Company. The airlock filter is rated at 60 psi water using Viton A O-rings. Qualification by similarity was limited to a low temperature at -65 °F. Skylab qualification included a fluid compatibility test in which the filter was submerged for 90 days in water that was nominally between 60 and 80 °F. A part of this test increased the temperature to 160 °F for 72 hr. A particulate capacity test, vibration to 7.0 g rms and burst pressure to 240 psia at 70 ±10 °F were also conducted. Estimated operating requirements were 3,610 hr. Capability for the Skylab mission was demonstrated in the 8-month endurance test. Acceptance testing of flight type articles included

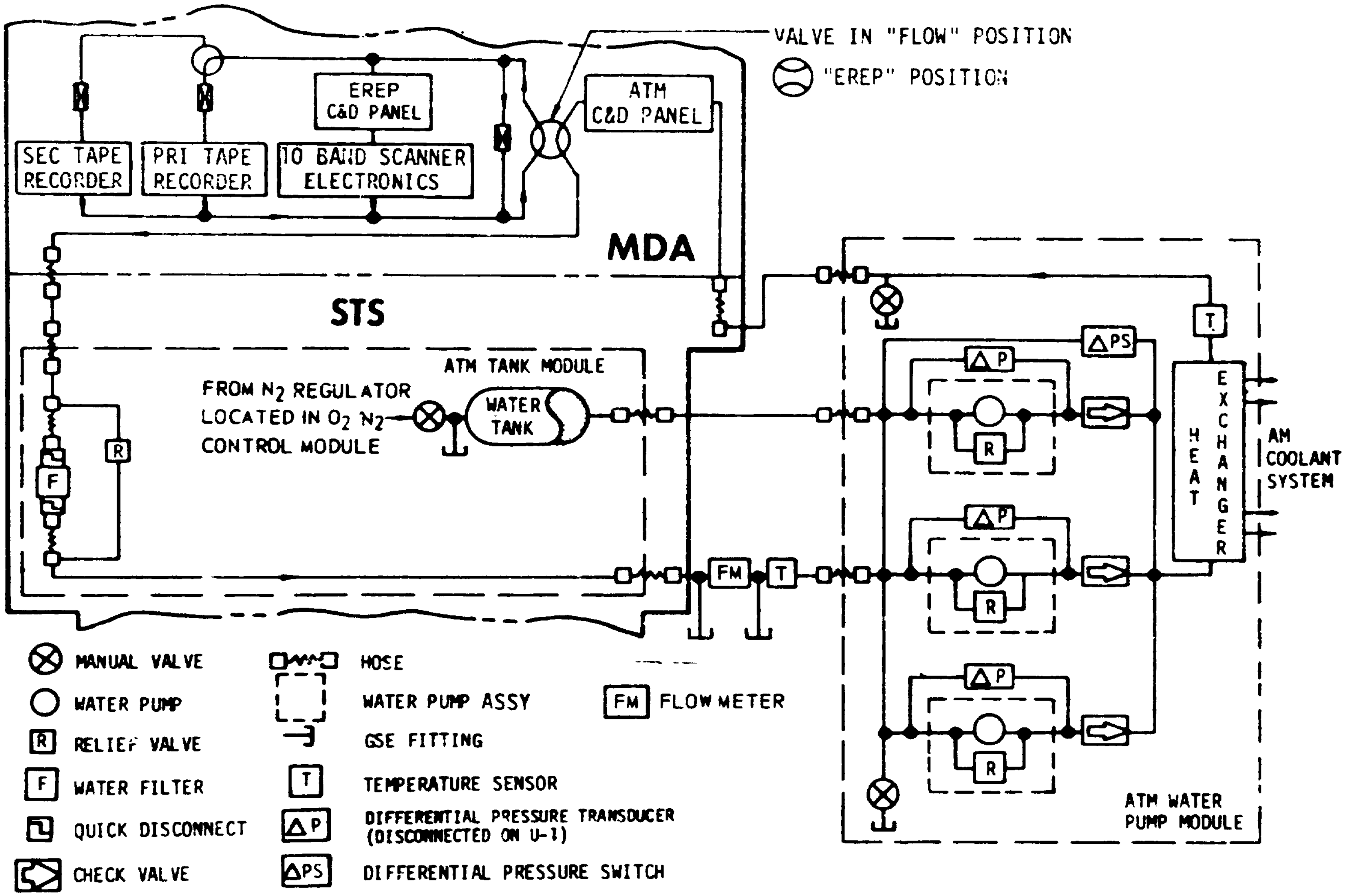


Figure VA-16. Telescope Mount Control and Display Coolant System

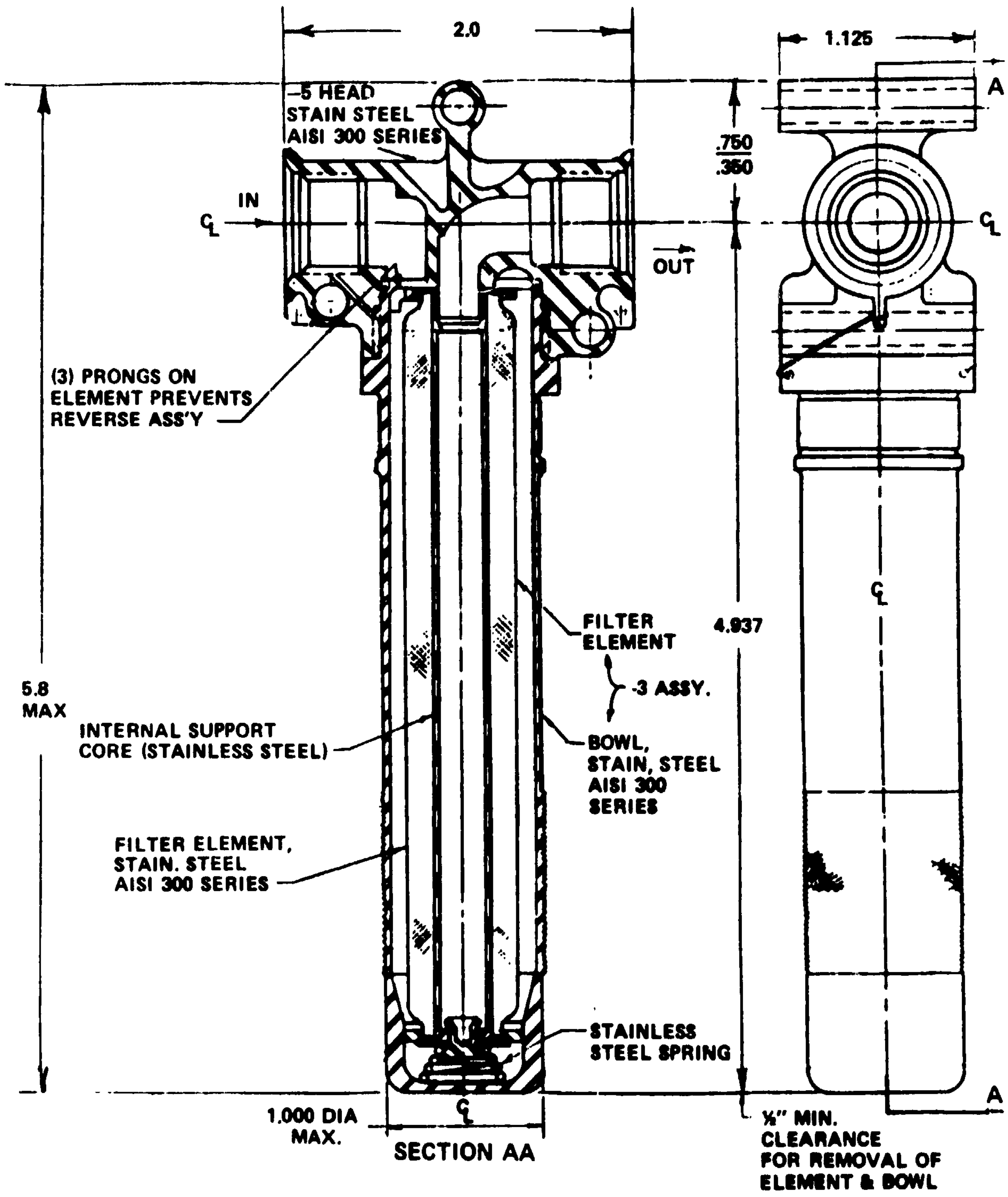


Figure VA-17. Water Filter Assembly

proof pressure to 120 ± 5 psig and leakage not to exceed 1.15×10^{-5} lb/hr water at 60 psig and 70 °F.

Shortly before launch, it was discovered that one of the pump differential pressure transducers had an internal electrical short of a type that could cause loss of a major segment of telemetry data. Since there was insufficient time to do a failure analysis and corrective action or to procure a new qualified unit, it was decided to cut the leads to all transducers of this type in both the telescope mount control and display cooling loop and the suit cooling loop, and fly without those measurements.

During all three manned phases of the Skylab mission, flow oscillations and/or dropouts were indicated for all three pumps in the telescope mount control and display water loops. At times, the anomalies were accompanied by gurgling and/or high pressure relief sounds. Various checks using the different pumps were made, including filter changes, without conclusive results. Since a bearing failure was suspected in pump A, the water filter was returned for analysis at the end of the second manned phase. Results showed residue to be 0.2688 g. This was not significantly different from the residue found in the two filters returned following the first manned phase which had 0.4502 and 0.1557 g of residue. The major elements of the residue following both missions were nearly the same: 8.9 percent aluminum, 16.2 percent potassium, and 43.7 percent phosphate as PO_2.

Ground tests with a liquid/gas separator installed in the backup airlock water loop showed that gas could be removed from the water should it be present in the system. As a result, the third Skylab crew installed the flight spare liquid/gas separator in place of the filter. During this installation, the crew noticed considerable gas in the liquid, at the quick disconnects, in the barrel of the filter and in the folds of the filter cartridge. Pump C was run for 15 min, the flow rate increased and stabilized. Pump B was also run with the gas separator in the loop and its flow also increased and stablized. It was concluded that the gas in the system and possibly some contamination had caused the flow problems in the loop. The gas separator was then removed and the filter reinstalled.

The system operated normally for approximately 15 days with only occasional flow oscillations and a gradual decrease in the continuous flow. It then started having significant flow dropouts that continued periodically until the liquid/gas separator was installed again. The flow rates increased and the flow dropouts stopped. Flow dropouts were observed again prior to the end of the mission. However, the frequency did not warrant reinstallation of the separator.

Pump A was turned on about 3 days before mission termination. The flow appeared to be normal with no dropouts so it was concluded that the pump had not failed mechanically.

e. Condensate system. The condensate system was required to remove, collect, and dispose of condensate throughout the Skylab mission. The system required active and backup (spare) condensate tank modules and a condensate dump system. The condensate tank module has quick disconnect lines to the condensate collection system and the condensate dump system. The condensate tank module has quick disconnect lines to the condensate collection system and the condensate dump system. The module contains a condensate tank, water fill/dump selector valve, tank pressure/evacuate selector valve, and a tank delta pressure transducer. Tank pressure for dumping is supplied from cabin atmosphere when the selector valve is in the PRESSURE position.

The condensate dump system consists of lines to a quick disconnect nipple in the aft compartment for use in transfer of condensate to a larger volume collection tank in the workshop and a backup overboard dump system. The overboard dump system consists of two solenoid shutoff valves, two dump nozzle heaters, control panel lights, and two manual three-position switches. The operational system shown in Figure VA-18 was launched dry.

The backup condensate module, stowed in the workshop for launch, contains a water/sterox solution which is used to prewet the condensing heat exchanger water separator plates.

During normal operations, negative pressure within the condensate tank is sufficiently low to allow moisture condensed in the heat exchangers to be drawn through the heat exchanger water separator plates into the condensate tank. Condensate is transferred into the workshop storage tanks by cabin ambient pressure. When the tank pressure increases to approximately 0.8 psi below cabin pressure, the collected condensate and gas is dumped into the waste tank. During extravehicular activity the flex hose to the workshop holding tank is disconnected to allow closure of the workshop hatch. Then only the condensate module in the airlock collects condensate water.

All the components of this system, with the exception of the manually controlled selector valves, are similar to items discussed in previous systems.

Two identical manually controlled selector valves, one for water fill/dump and the other for pressure/vacuum, are used in the airlock condensate system. With water at 70 °F and 50 psig at the inlet, valves are required to permit 0.5 lb/min flow with a maximum pressure drop of 2.0 in. of water. Maximum leakage requirements are 3.15×10^{-5} lb/hr gaseous oxygen at 6 psid and 70 °F. Operating torque to reposition the selector valve is required to be 20 in.-lb maximum.

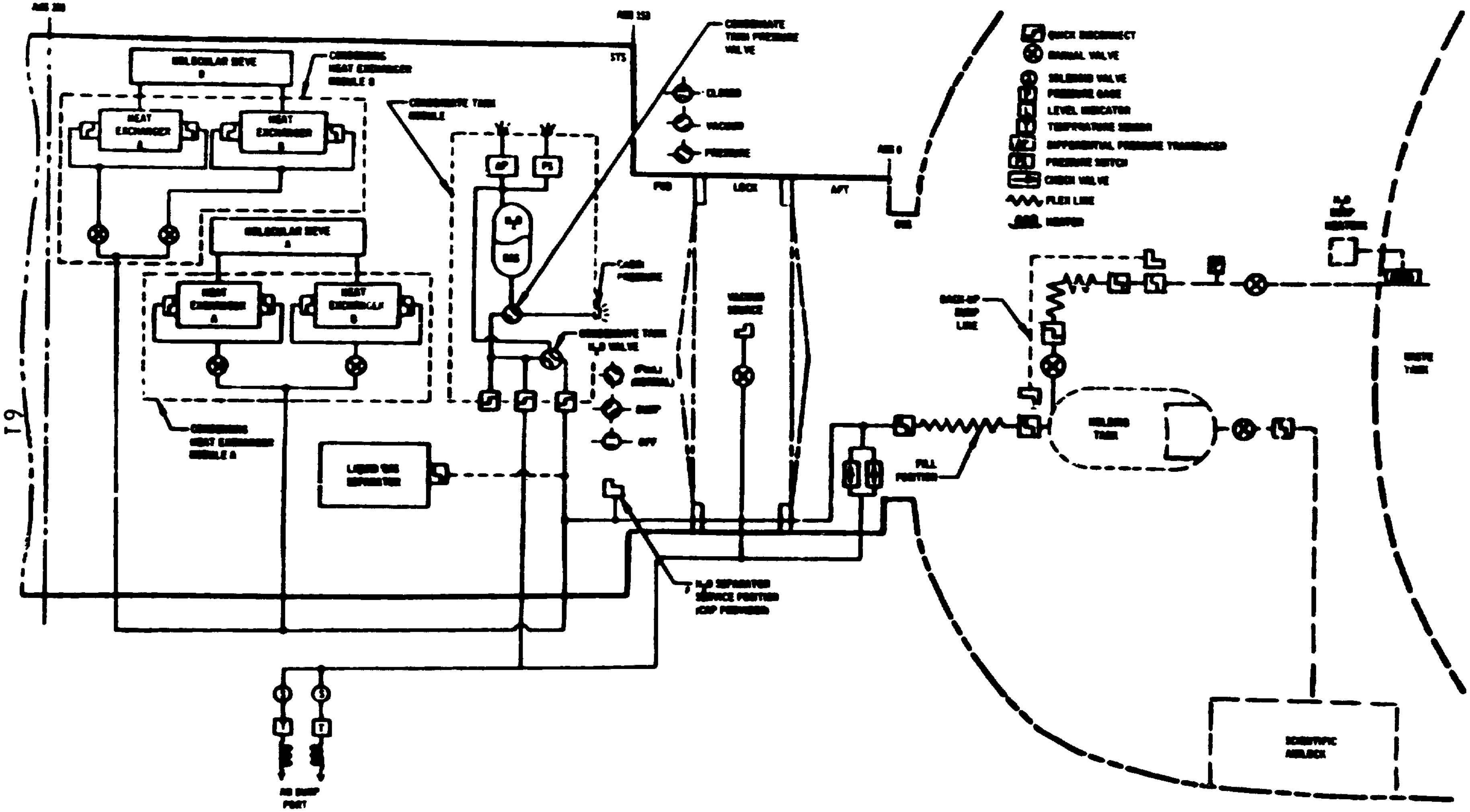


Figure VA-18. Condensate System

The water dump valve is a three-way position selector valve that contains two inlet flow ports and a common outlet flow passage. The valve consists of a hollow rotating shaft sealed at one end with two offset ports drilled 120° apart using O-rings for sealing against leakage. A flat portion is machined on the valve stem for indexing and mating with a control handle. A shaft rotation of 120° is required to select flow from one port or the other. A midstroke off position blocks flow from either inlet.

The valve was qualified for Gemini and flew on flights 2 through 12. Gemini qualification included burst pressure to 200 psig. Because of Skylab operational estimates of 3,610 hr and 190 cycles, the valve was qualified to a demonstrated life of 3,966 hr and 1,000 cycles. At acceptance, each valve was tested for leakage (3.15×10^{-5} lb/hr O_2 max) and proof pressure tested to 100 psig. One unit was subjected to a 90-day corrosion test to support Skylab requirements.

During the Skylab mission the condensate removal capability of the condensate system performed to specification. However, leaks in the system caused many more dumps than had been planned. During the second manned phase, when the leak persisted for 32 days, the system was dumped each day. The system performed other functions successully, such as servicing/deservicing of life support umbilicals/pressure control units, servicing heat exchanger separator plates, and removing water from the command and service module water tank.

During the first manned phase, when the workshop holding tank was disconnected from the condensate system for extravehicular activities, the system delta pressure dropped rapidly. Since the pressure did not decrease rapidly with the holding tank connected, it was concluded that a leak existed on the gas side of the airlock system.

During the second manned phase, water separator plate servicing, the condensate delta pressure again began to decrease. The leak was believed to be in one of the gas side quick disconnects. However, the leak stopped and did not appear again until late in the third manned phase (mission day 80) when the liquid gas separator quick disconnect was disconnected for extravehicular activities. After attempts to stop the leak with universal sealant, a cap launched on the third manned phase specifically for this purpose was installed on the hose side of the liquid/ gas separator gas connector. No further leakage was observed.

On mission day 34, during the second manned phase, the waste tank management system failed to completely dump the workshop condensate holding tank. The dump probe was considered to have frozen. A 35 psi hot water dump cleared the probe with the bus #2 heater on. However, a try at dumping the condensate tank failed. Another 35 psi hot water dump using bus #1 heater cleared the line. Subsequent holding tank dumps were successfully performed but were slower than they should have been. On mission day 36, a holding tank dump was unsuccessful. The water dump probe assembly was replaced. No further problems were encountered.

f. Quick disconnects. Quick disconnects are used in H_2O, CO_2, GN_2, and GO_2 systems and are available in either 1/8-, 1/4-, or 3/8-in. size. Their primary use is to provide a quick means of removing or connecting an environmental control system line. The 1/8-in. quick disconnects are used either in the H_2O system or for CO_2 sensing and are made of 303 stainless steel with Buna N seals for the H_2O use, and 316 stainless steel with silicone compound 11715 seals for CO_2 sensing. The 3/8-in. quick disconnects are used in the H_2O system and are made of 316 stainless steel with Buna N O-rings. The 1/4-in. quick disconnects are used in the GO_2 and GN_2 systems and are made of 316 stainless steel with batch tested Viton A O-rings for the GO_2 and silicone Parker compound 11715 for the GN_2.

The quick disconnects assembly consists of a valved nipple assembly, a valved coupler assembly, and a pressure cap assembly. Mating fittings are held together by a spring-loaded ball-lock mechanism. When disconnected, valves in the nipple and the coupler are held in closed position by both spring pressure and system pressure.

The quick disconnects were developed for use on the Gemini program, but were never used. Skylab qualification included vibration, humidity, temperature, and bust pressure. Life test was 1,000 connect/disconnect cycles in addition to the 8-month endurance test. During initial receipt of components, recleaning was done that later caused leaks because of the loss of lubricant on O-rings. O-rings were relubricated, stopping all leaks. Also, during early testing, some quick disconnects experienced negative pressure leaks that were traceable to cut or damaged O-rings or contamination on O-rings.

During the process of changing the filter in the telescope mount control and display water loop, the first Skylab crew reported a slight spillage of water (approximately 2 to 4 oz) from one of the water quick disconnects. The crew said during the post-flight debriefing that the internal plunger of the quick disconnect did not close fully when it was disconnected. It was reconnected and disconnected again with no apparent leakage. It is believed that the leakage was a momentary malfunction that was cleared by the connect/disconnect procedure used by the crew.

B. Fixed Airlock Shroud

The fixed airlock shroud is required to provide continuity of the external environmental protection from the payload shroud to the instrument unit. Structurally the fixed airlock shroud supports the combined airlock docking adapter, the payload shroud, the telescope mount deployment assembly, and six high pressure oxygen bottles. Concentrated loads generated at these attachments are distributed by the fixed airlock shroud to reduce peaking loads at the instrument unit interface. A ground umbilical was required to provide electrical connections for telescope mount prelaunch checkout, payload shroud purge gas, docking adapter insulation purge gas, telescope mount purge gas, and ground cooling of the airlock coolant system. The fixed airlock shroud was designed to a safety factor of 2.0 times the predicted loads for all unmanned phases of the mission and 3.0 times the predicted loads for all manned phases to preclude requirements for static test.

The fixed airlock shroud is a ring-stiffened, thick-skinned cylinder approximately 80 in. in height, 260 in. in diameter, and configured as shown in Figure VB-1. Intercostals distribute concentrated loads introducted by the deployment assembly, airlock, and oxygen support points. Two doors are on the fixed airlock shroud: one for access to the fixed airlock shroud interior and the airlock extravehicular activity hatch during ground operations, and one for access to ground umbilical connectors. Four antennas, two deployable discones, and two UHF antennas are mounted on the fixed airlock shroud. The fixed airlock shroud structure also contains extravehicular activity support equipment: Ingress/egress handrails, work platforms, film cassette tree supports, film transfer boom also called Tubular Extendible Element (TEE), a TEE hook stowage box, and lights.

Ground verification of the static load capability was conducted by analyses to verify the design safety factors. The vibroacoustic test at JSC verified the capability of the fixed airlock shroud to withstand the launch and boost vibration and acoustic noise levels. Figure VE-6 presents the results of the JSC tests. The umbilical separation and automatic umbilical door closure were verified by testing at MSFC using a flight-type umbilical and door and a swing arm simulator.

The first crew reported in crew debriefing that the fixed airlock shroud structure and all shroud mounted equipment appeared in excellent condition with no evidence of buckling or breaking. Thus, it is obvious that the fixed airlock shroud performed as expected and the practice of designing such a configuration to a factor of two times predicted loads without test verification is acceptable when weight is not critical. Photographs of the first unmanned phase lift-off show that the fixed airlock shroud umbilical door had closed properly.

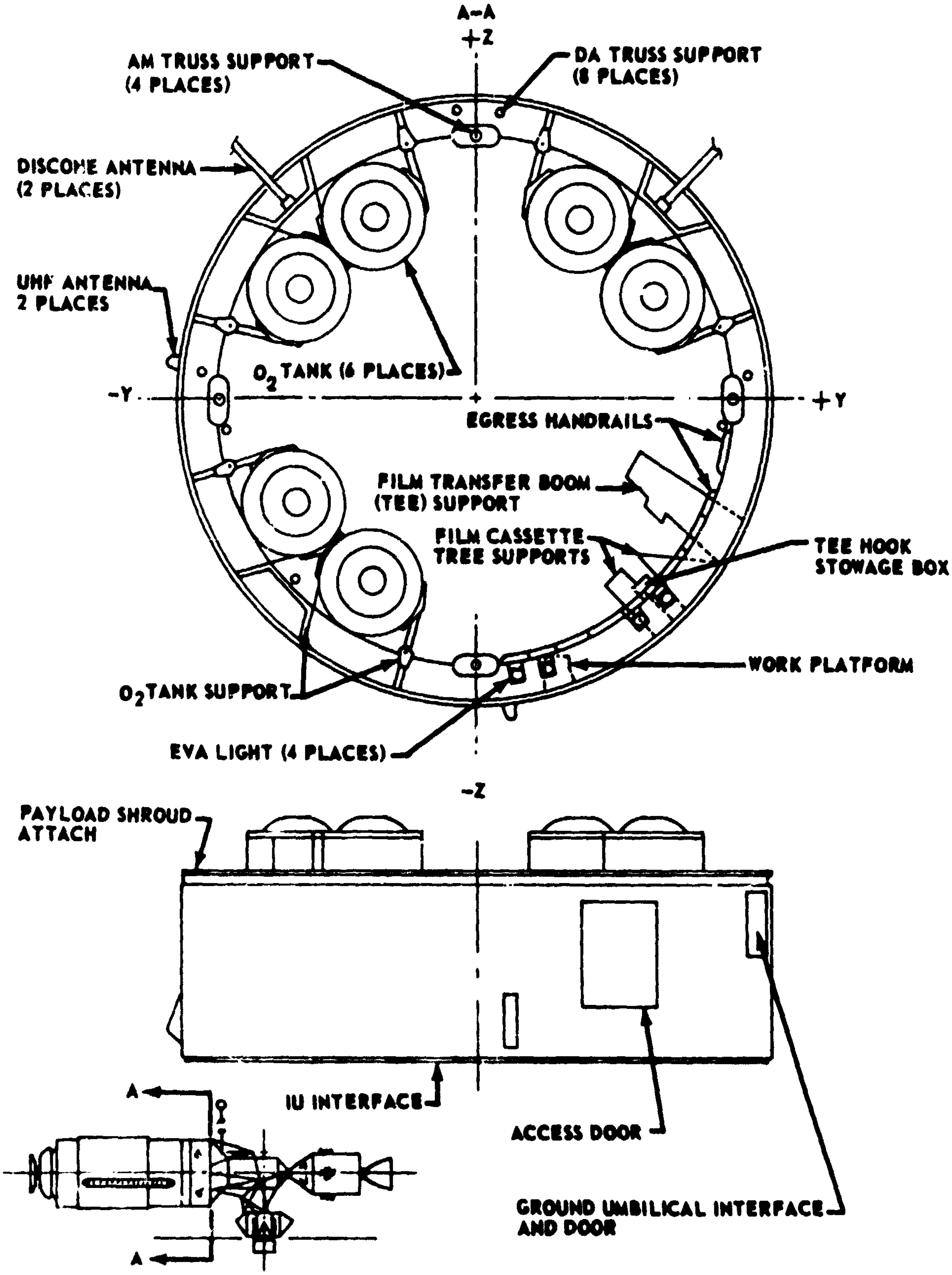


Figure VB–1. Fixed Airlock Shroud (FAS)

C. Telescope Mount

1. General Description. The telescope mount structural/mechanical system provides for the housing of the solar astronomy experiments with their support equipment, including their pointing control and thermal control systems. It also provides for the mounting of Saturn Workshop attitude control equipment and the telescope mount electrical power system.

The structural/mechanical system, Figure VC1-1, consists of the telescope mount rack, experiment canister, rack/canister interface hardware, the solar array deployment mechanism, aperature doors, film retrieval doors, thermal control system hardware, ordnance systems, and extravehicular activity hardware. The telescope mount assembly is attached to the deployment assembly through rigidizing attach fittings at the rack base. This interface did not provide load carrying capability during the prelaunch and launch phases. For these phases, load transfer into the vehicle structure was provided through the telescope mount/payload shroud interface.

2 Telescope Mount Structural Systems. Figure VC2-1 shows the basic telescope mount structural system.

The rack is the main structural assembly of the telescope mount and consists of octagonally shaped upper, lower, and solar array support rings. These rings are each comprised of an inner and outer cap connected by webs and web stiffeners. A longitudinal load path between rings is provided by vertical beams at each corner of the octagonal support rings. Outrigger fittings at each octagonal corner connect to the outrigger tubular members that form four pyramid shaped trusses, and the entire telescope mount is supported in the launch configuration by interfacing with the payload shroud/telescope mount support fittings where the truss members converge.

The rack was originally designed as an open truss structure. As the program evolved and as telescope requirements changed, the truss arrangement became difficult to adapt to new requirements. Primarily the difficulties fell into two categories: (1) insufficient mounting space for black boxes (quarter panels were added to three bays of the rack and additional panels to the upper ring surface), and (2) thermal requirements imposed severe restrictions to avoid radiation to other components. The adaptability of the rack structure to provide thermal shields or to accommodate shifting of components to more favorable thermal zones was quite limited. The structure was modified to incorporate nonstructural thermal panels in addition to the original truss members.

Some early difficulty was also experienced in providing a structure that was adaptable to continually changing equipment locations.

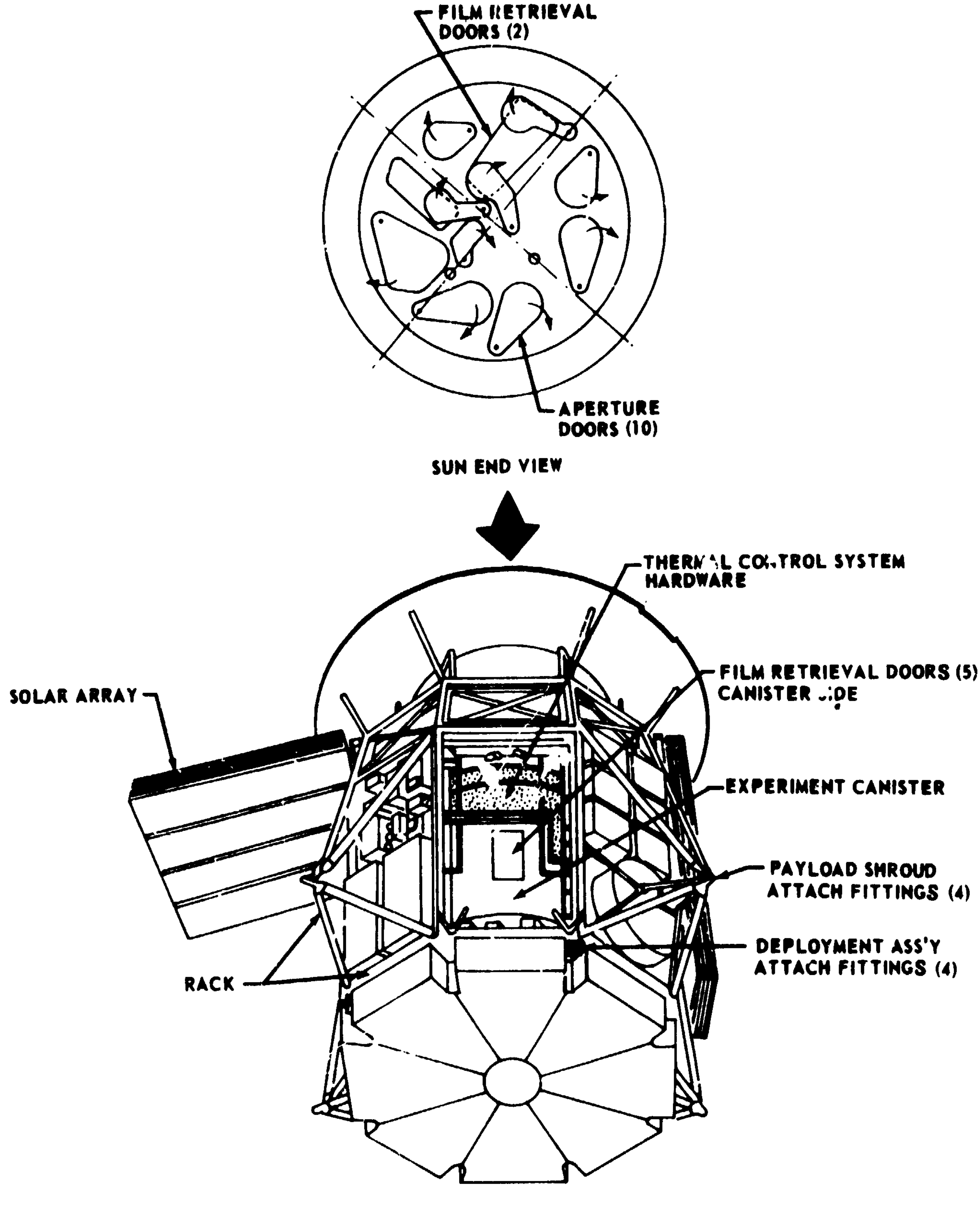


Figure VC1-1. Telescope Mount Assembly

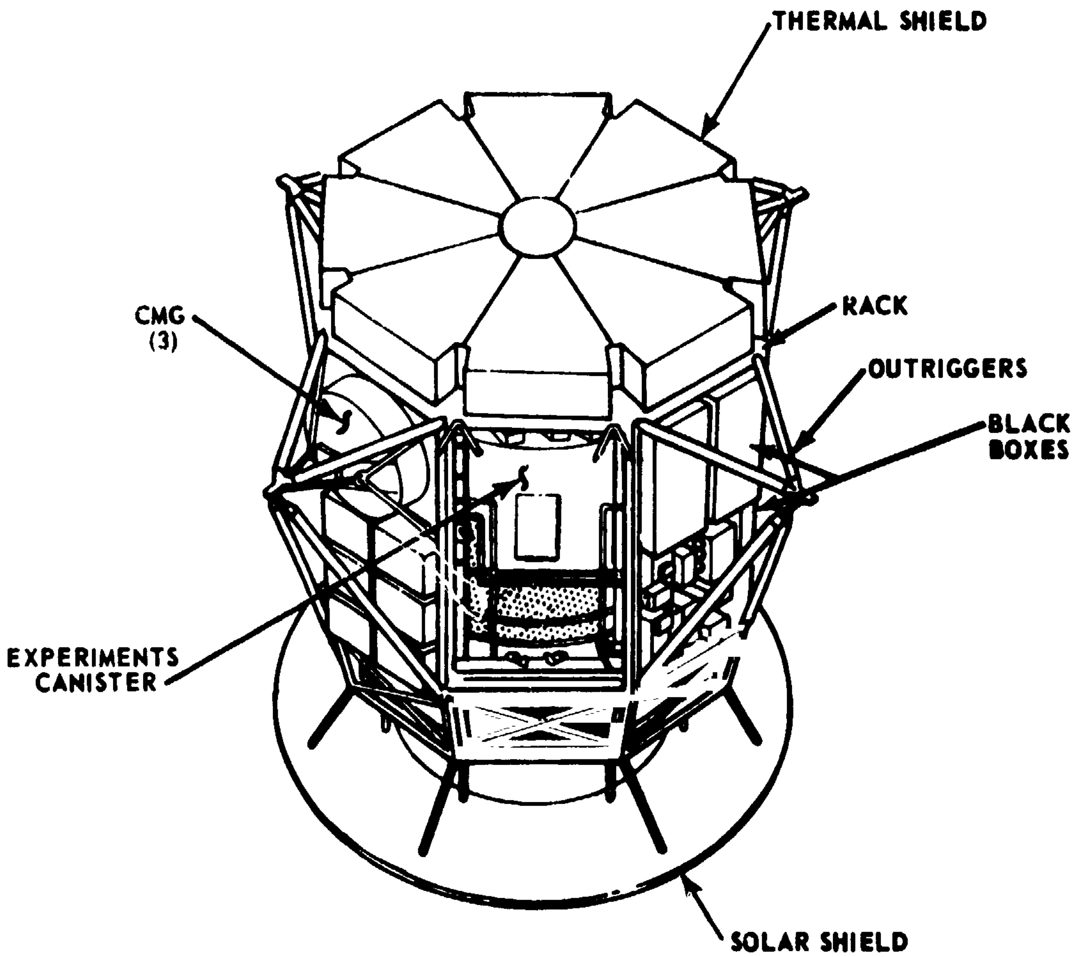


Figure VC2-1. Telescope Mount Structural Systems General Arrangement (Without Solar Array)

The original stiffened sheet metal design proved to be inflexible; consequently, a honeycomb panel design was adapted. This allowed relocation of panel inserts, thus accommodating new equipment mounting locations, without redesign or scrapping of existing hardware.

One of the eight bays of the rack structure is left open to permit access, both ground and extravehicular activity, to the experiment package. Originally, it had been planned to maintain the diagonal truss in this bay, but design it for extravehicular activity removal. However, studies and analysis proved the rack strength to be satisfactory for all flight conditions without the diagonal strut. Consequently, the design incorporated only those provisions needed for ground handling conditions and the diagonal strut was then removed in the VAB after telescope mount stacking.

During the vibration testing of a full telescope mount assembly, dynamic cross coupling of the rack structure was discovered (a longintudinal excitation produced a lateral response). As this was very late in the program, a satisfactory hardware solution could not be found. Consequently, the only remaining course of action was to reduce the forcing function. This was accomplished by changing the S-IC engine cutoff sequence from 1-4 to 1-2-2. The rack structure was designed to an ultimate safety factor of 1.25 and a yield safety factor of 1.1. The rack was statically tested to seven flight load conditions and four transportation conditions. Table VC2-1 shows these test conditions and the design loads acting at the c.g. of the rack and experiment package.

The experiment canister consists of a spar with a girth ring in the center, a ring on each end, and two canister halves to enclose the experiments (telescopes) mounted on the spar (Figure VC2-2).

The spar is a cruciform structure made up of three 1.0-in. thick aluminum plates. It serves as the structural support and optical bench for the telescope mount experiments. Two inch diameter holes are drilled throughout the plates to minimize weight.

No flight instrumentation was available on the telescope mount basic structure to enable direct comparison with design values. From other flight performance data such as vehicle acceleration, wind velocity, and booster engine gimbal angles on the launch vehicle, the load during flight at the rack and experiment package c.g. were calculated for two conditions. These calculated values are compared to the design values in the following tabulation.

TRANSPORTATION G FACTORS

CASE	TITLE	$\ddot{X}$	$\ddot{Y}$	$\ddot{Z}$	$\ddot{\theta}Y$	$\ddot{\theta}Z$
1	HIGH ANGLE OF ATTACK	-3.95	0.05	-0.24	0.0028	0.0004
2	HIGH ANGLE OF ATTACK	1.82	0.04	-0.17	0.00485	0.00116
3	STEADY STATE FLOOR VIB.	-1.0	-1.08	1.08	-0.0158	-0.0158
4	STEADY STATE FLOOR VIB.	-1.0	1.07	1.07	-0.0158	0.0158

FLIGHT G FACTORS

CASE	TITLE		$\ddot{X}$	$\ddot{Y}$	$\ddot{Z}$	$\ddot{\theta}Y$	$\ddot{\theta}Z$
1 (A)	PITCH PLANE LIFT-OFF	RACK	2.2	0.784	0.32	0.0039	-0.0116
	95% GROUND WIND	CAN.	4.68	0.684	0.281	0.0083	-0.0279
2 (A)	YAW PLANE LIFT-OFF	RACK	2.2	-0.308	0.813	0.0101	0.0045
	95% GROUND WIND	CAN.	4.68	-0.266	0.703	0.0234	0.0106
2 (B)	YAW PLANE LIFT-OFF	RACK	2.2	0.0	0.869	0.01105	0.0
	95% GROUND WIND	CAN.	4.68	0.0	0.751	0.0257	0.0
2 (C)	YAW PLANE LIFT-OFF	RACK	2.2	-0.614	0.614	0.0078	0.0078
	95% GROUND WIND	CAN.	4.68	-0.531	0.531	0.01816	0.01816
3 (A)	MAX Q ALPHA	RACK	1.96	0.0	0.353	0.00127	0.0
	MAX Q ALPHA	CAN.	1.96	0.0	0.349	0.00108	0.0
3 (B)	MAX Q ALPHA	RACK	1.96	-0.25	0.25	0.0009	0.0009
	MAX Q ALPHA	CAN	1.96	-0.247	0.247	0.00076	0.00076
4 (A)	MAXIMUM ACCELERATION	RACK	4.7	0.0	0.05	0.00005	0.0

PANEL COMPONENT G FACTORS

FLIGHT DIRECTION	5 G'S
NORMAL TO PANEL	10 G'S

Table VC2-1. Telescope Mount Structural Test Loads

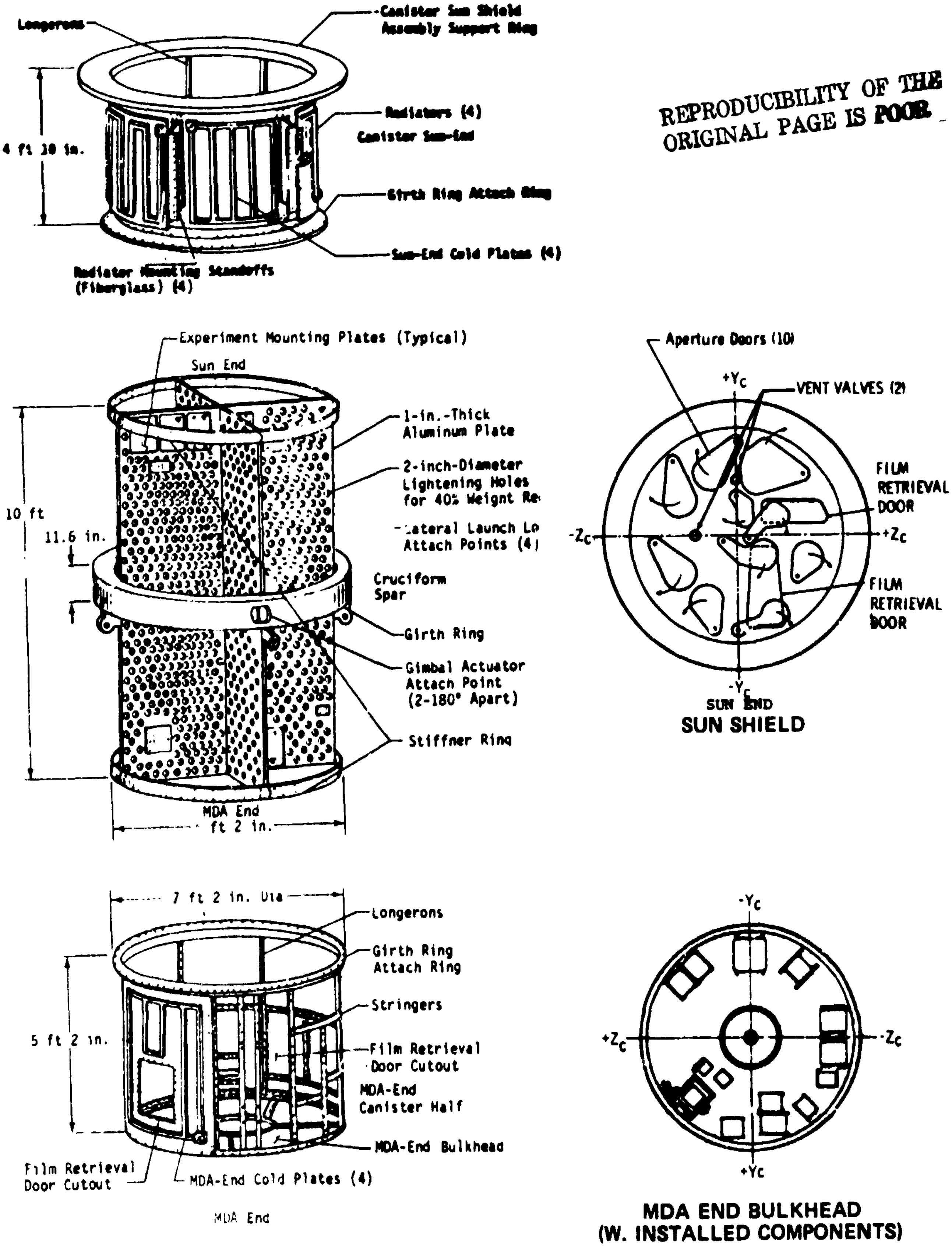


Figure VC2-2. Experiment Package Structure

	Rack Axis			Exp. Package Axis		
	X	Y	Z	X	Y	Z
Design "G" Factors	4.7	0.0	0.05	4.68	0.684	0.28
Flight Value at S-IC Center Eng. Cutoff	4.51	-0.008	0.019	4.51	0.008	0.20
Flight Value at S-IC Outboard Eng. Cutoff	4.49	-0.003	0.012	4.49	0.0006	0.01

It can be seen that the loads experienced in flight were less than the design and test loads.

Two flapper type vent valves are installed at two of the four sun shield openings which are provided for the installation of the zero g fixture during telescope mount ground handling and checkout. The location of the valves and their configuration are shown in Figure VC2-3. The purpose of the valves was to prevent a delta pressure buildup inside the canister to exceed 0.5 psi and also to provide contamination protection for the telescope mount experiments. A test program qualified the valve for flight and also served for determining the final adjustment of the counter balance for the required crack pressure. Although flight measurements to monitor valve operation were not provided, no problems were experienced with any experiment canister equipment, indicating that the pressure profile was kept within design limits.

Rack/experiment package interface hardware, consists of the launch locks, experiment pointing control system (EPCS) hardware, and the GN_2 purge disconnect system hardware.

Four lateral launch locks and one torsional launch lock prevented rigid body motion of the telescope mount spar/canister during ground handling, transportation, and launch. Each of the lateral launch locks consists of a longitudinal strut and a lateral strut that form a truss with the telescope mount rack on which they are mounted. Also, there are four snubber assemblies, one between each lateral strut and the gimbal ring. The function of the snubber assembly is to prevent, during launch, the vibration of the gimbal ring that is not locked by the four lateral launch locks. The torsional launch lock prevents roll of the spar/canister about the longitudinal axis. Refer to Figure VC2-4 for hardware arrangement.

The four later launch locks and the torsional launch lock were designed to be released by pyrotechnially initiated pin pullers after orbital insertion. The pins retract into the pin puller housing, allowing the lateral launch lock struts to rotate against the side of the canister, and the torsional launch lock pin to be help captive by a spring loaded cover. Refer to Figure VC2-4 for detail.

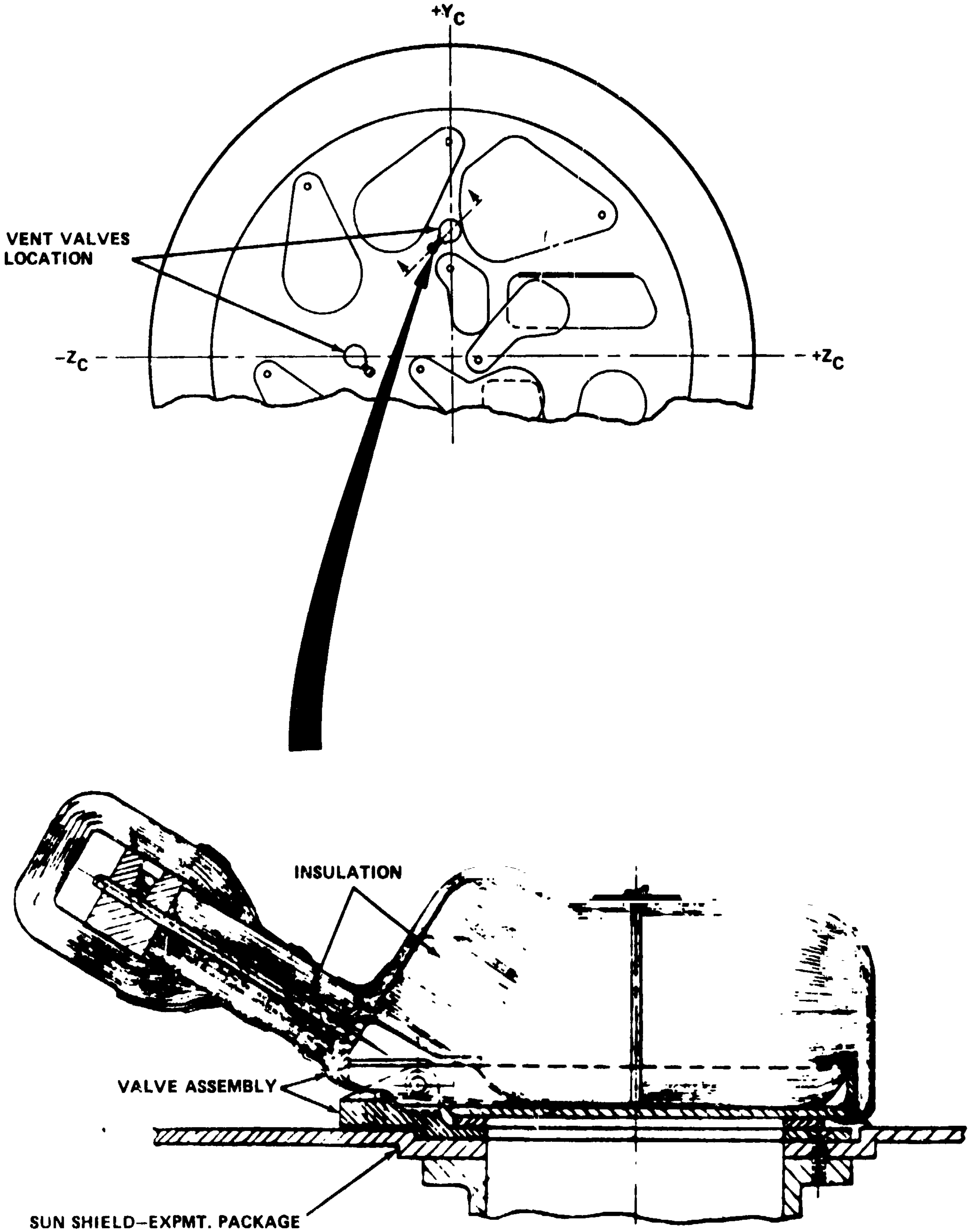


Figure VC2-3 Experiment Canister Vent Valve

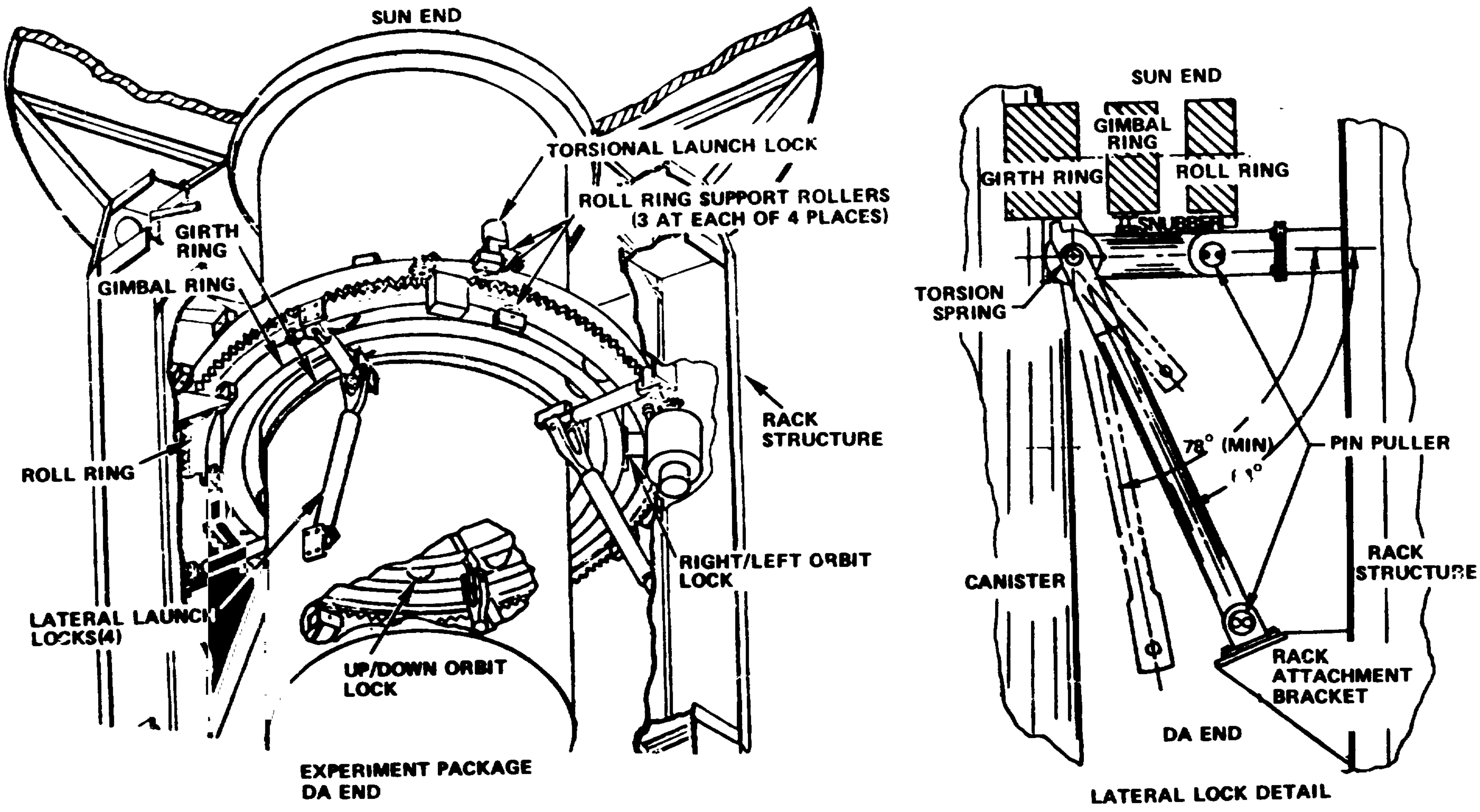


Figure VC2-4. Launch Locks and Experiment Pointing Control Hardware

The telescope mount launch lock release ordnance system shown in Figure VC2-5, was completely redundant and consists of 2 EBW detonators, 2 CDF manifolds, 2 CDF tees, 21 CDF assemblies, and 18 CDF pressure catridges. A schematic of these components is shown in Figure VC2-6. An electrical signal was transmitted from an EBW firing unit that initiated the EBW detonators and started the propagation train. The propagation was transferred through the CDF manifolds, CDF tees, and CDF assemblies to the CDF pressure cartidges that developed the pressure to retract the nine pin puller pins, thereby, unlocking the canister from the rack.

The EBW firing unit charge and trigger voltage measurement data are not available for evaluation since telescope mount telemetry was not activated at the time of this event. Also, there was no instrumentation to indicate positively that the pin puller pins had retracted or that the locks had released. Therefore, the first positive confirmation that this system had functioned properly was by actual canister rotation during initial activation.

To prevent contamination, the telescope mount experiment canister was purged on the pad with gaseous nitrogen (GN_2). A retractable fitting (disconnect coupling) carries this GN_2 from the telescope mount rack to the canister. The ground support equipment line is disconnected at the fixed airlock shroud prior to launch, but the retractable fitting between the rack and canister is not. The rack to canister fitting was released when the crew rotated the canister the first time in orbit. A guide rail and roller prevents the spring loaded arm from retracting before the male fitting is completely disengaged. To properly disengage the purge fitting, the canister must rotate at lowest speed in a clockwise direction (looking toward the sun end) (Figure VC2-7).

The mechanical system components (male and female disconnect coupling with lead-in tubing component) conformed to the following qualification and performance requirements: Operating pressure of 0 to 75 psig, leakage not to exceed 50 scims, disconnect force not to exceed 15 lb, proof pressure of 113 psig, and burst pressure of 188 psig.

Development and qualification tests were performed according to S&E-ASTN-TMU (73-36) with vibration levels specified in memorandum S&E-ASTN-TMV (73-26). Also, the retract mechanism was operated successfully 10 times under vacuum conditions after subjection to the launch environment. Initial canister roll and subsequent operation confirms that the GN_2 purge disconnect retracted also folded out of the way against the rack.

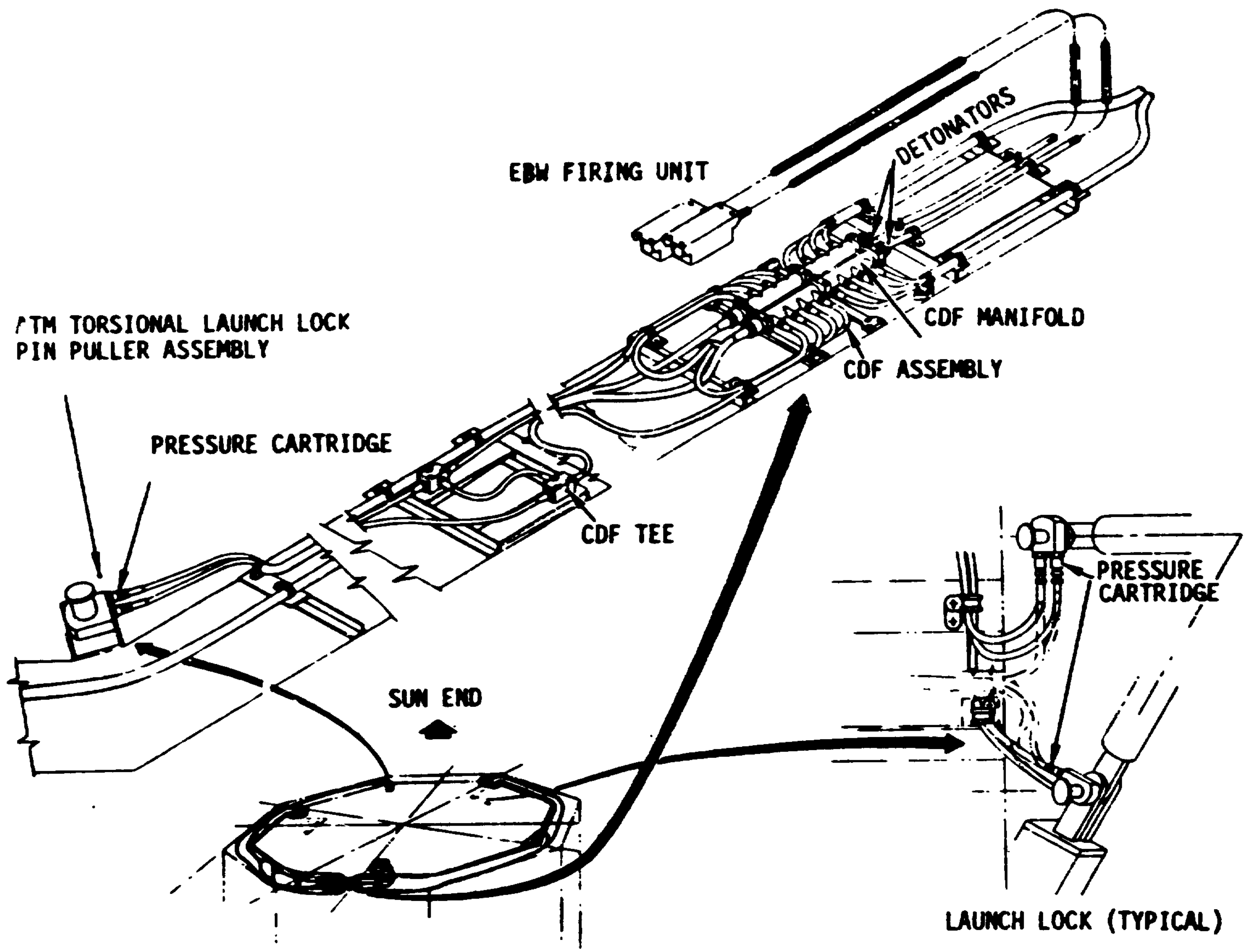
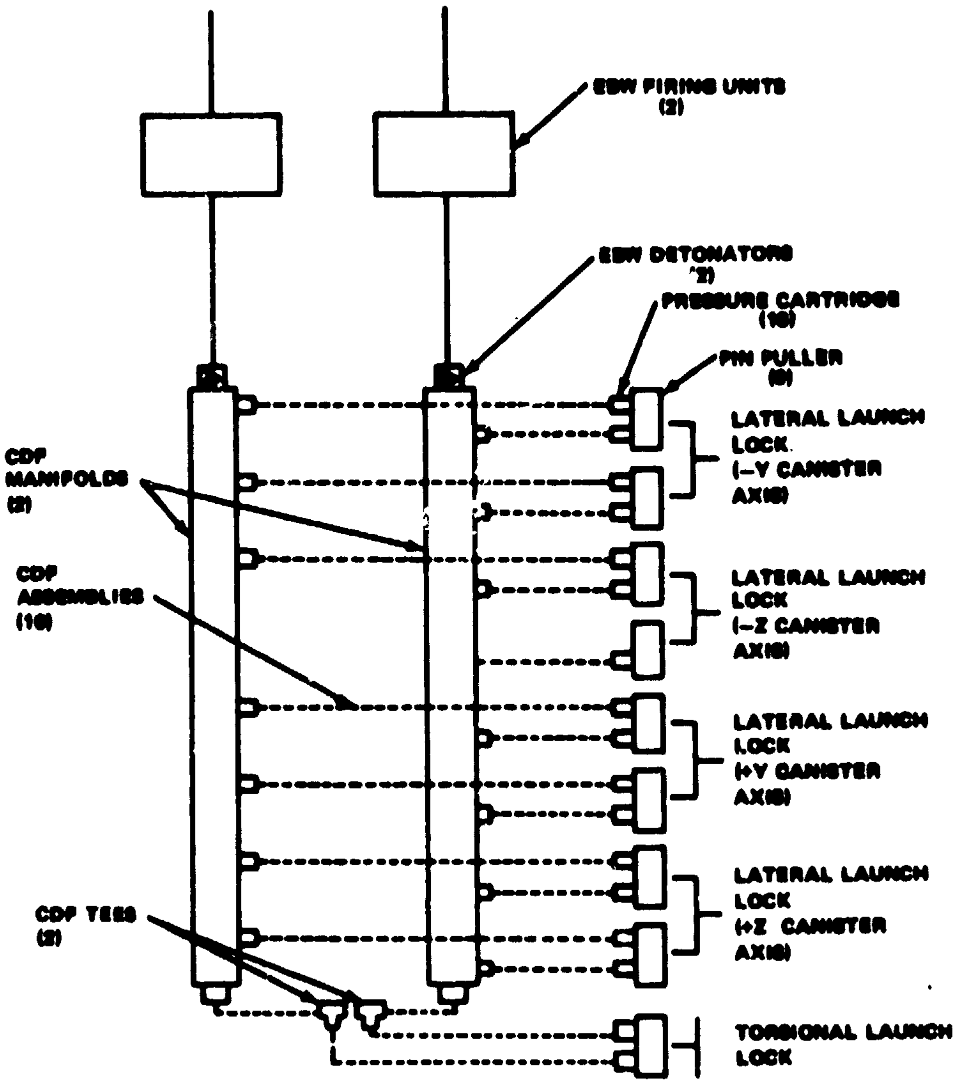
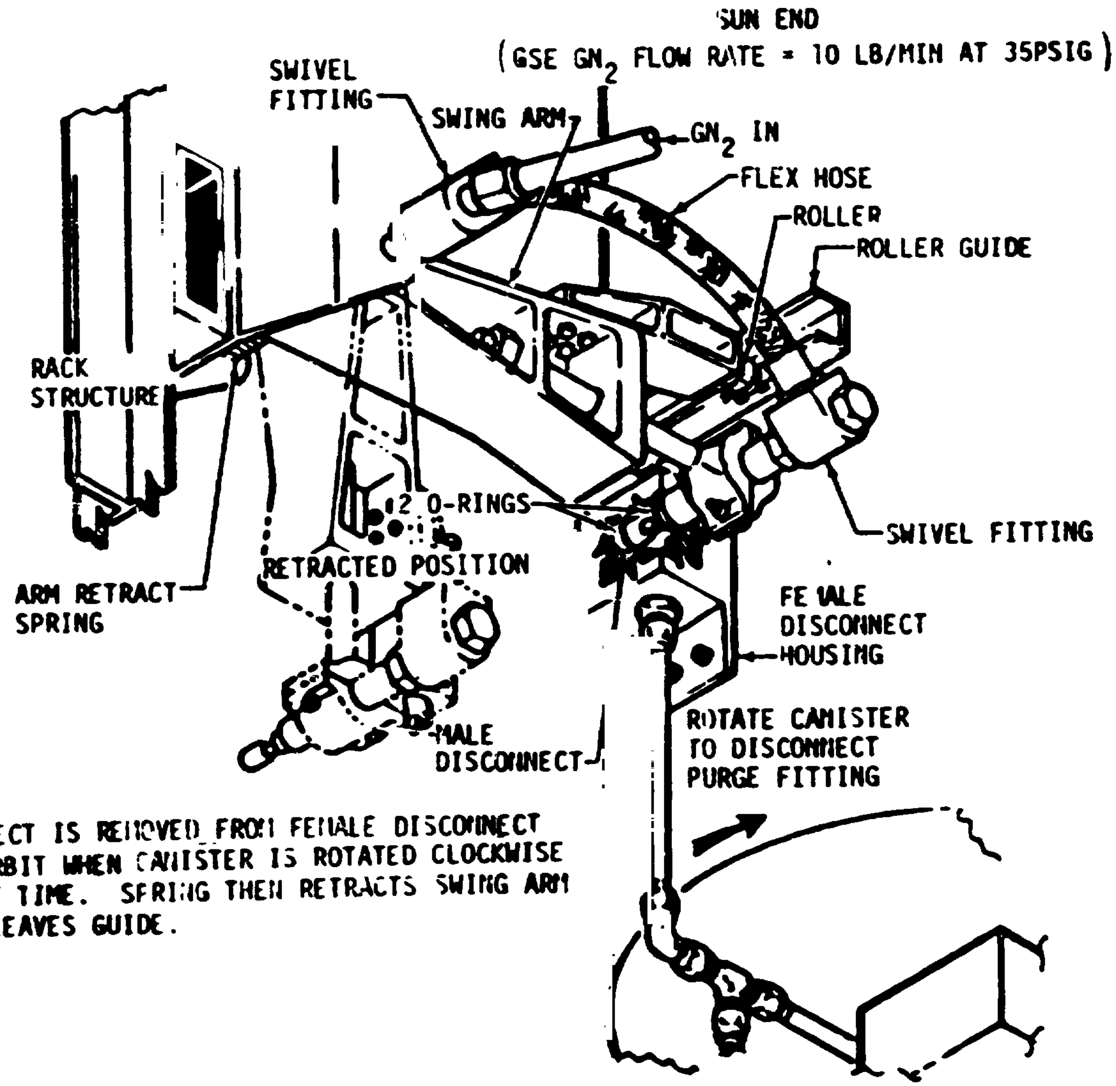


Figure VC2-5 Telescope Mount Launch Lock Release Mechanism



VC2-6 Telescope Mount Launch Lock Release Ordnance



NOTE: MALE DISCONNECT IS REMOVED FROM FEMALE DISCONNECT HOUSING IN ORBIT WHEN CANISTER IS ROTATED CLOCKWISE FOR THE FIRST TIME. SPRING THEN RETRACTS SWING ARM WHEN ROLLER LEAVES GUIDE.

Figure VC2-7. GN_2 Purge Disconnect Arrangement

The telescope mount solar array panels are stacked in a vertical orientation at launch configuration. These panel stacks are supported by cinching to a mounting structure. The telescope mount solar panel cinching devices are released by means of pyrotechnic actuators. Ordnance in the decinching mechanism (Figure VC2-8) consists of 2 EBW detonators, 2 CDF manifolds, 16 CDF assemblies, and 16 CDF pressure cartridges.

The pyrotechnics portion of the decinching mechanism consists of two redundant ordnance systems with primary initiation by a signal from the automatic instrumentation unit system and a secondary initiation provision by a signal from the airlock digital command system. These signals are used to trigger the EBW firing units. The firing unit initiates the EBW detonator that is installed in a CDF manifold. The explosive force is propagated through the CDF manifold and CDF assemblies to CDF pressure cartridges. The output charge of the CDF assemblies to CDF pressure cartridge actuates the solar power thruster assembly (Figure VC2-9).

The EBW firing unit charge command initiating solar array deployment was sequenced for 17:54:48.7 and the trigger command at 17:54:52.3. Data confirming this sequence are not available for evaluation, however, based on actual function of the telescope mount solar array, there can be no question as to the proper function of this system. Fly around photographs show the fully deployed solar array.

3. Aperture and Film Retrieval Doors.

a. Film retrieval doors. The manually operated film retrieval doors are shown in Figure VC1-1 and VC3-1. These doors provide access for film cassette exchange.

During the entire Skylab mission, the S082A film retrieval door was opened and closed seven times and the S082B door five times. Operation of S082 door was without incident. The S082B door exhibited stickiness on two occasions, however, the film exchanges were accomplished successfully.

The film retrieval doors accessible from the center work station are shown in Figure VC3-1. These doors are manually operated and the canister must be rolled to gain access to each of the doors. The doors were qualified for 320 cycles of closure and relock with an operating force of 10 lb or less.

The center work station film retrieval doors operated as designed with no anomaly. On day 26 of the first manned phase, the crew indicated that the S054 film retrieval door would not lock mechanically but was being held closed by the magnetic secondary latching system. However,

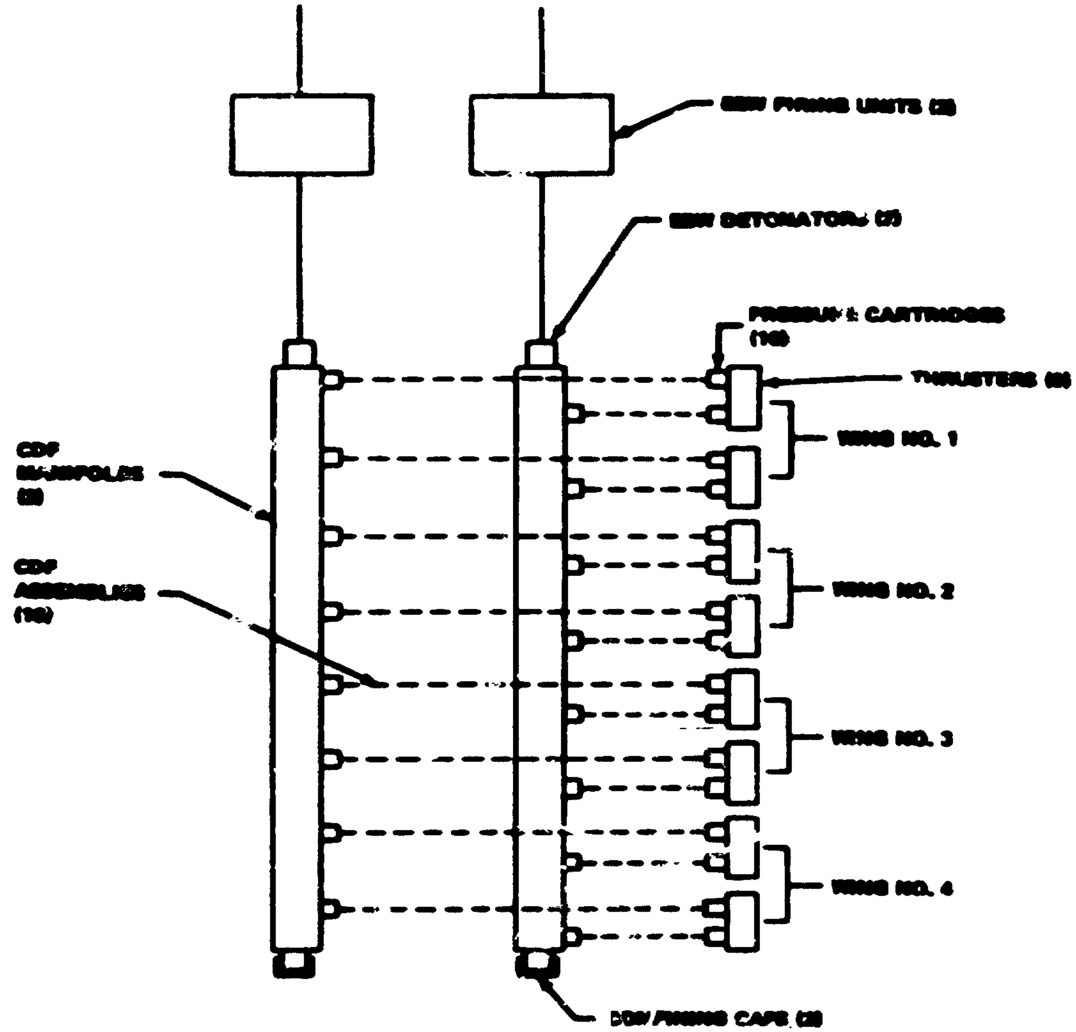


Figure VC2-8. Telescope Mount Hardware SAS Pacinch Ordnance

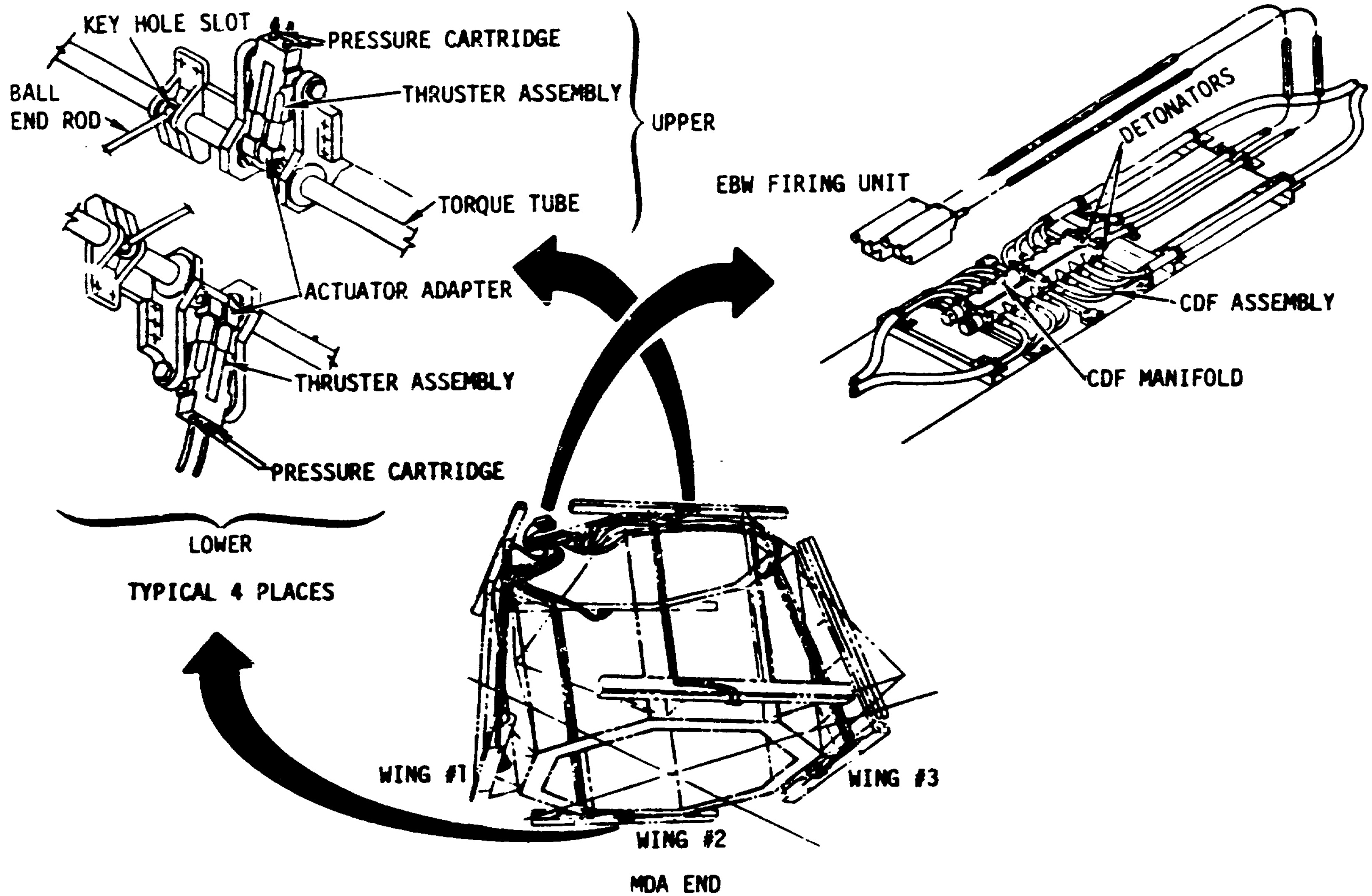


Figure VC2-9 Telescope Mount Solar Panel Decinching Mechanism

C-2

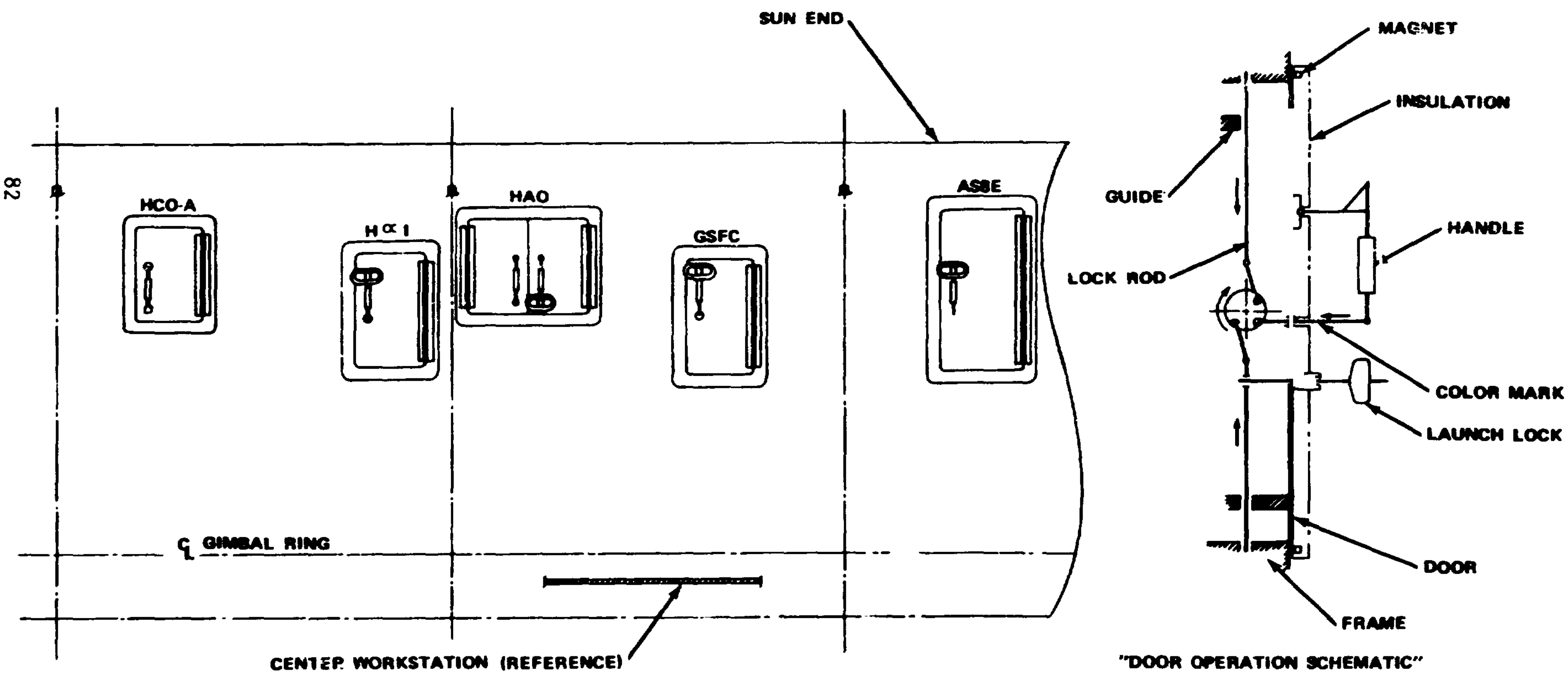


Figure VC3-1. Telescope Mount Film Retrieval Doors the second crew reported the S054 door was mechanically locked and operated normally. During the Skylab mission, the doors were used a total of six times each.

b. Aperature doors. Eight telescopes and the five sun sensors are installed inside the telescope mount experiment canister. The optics are aligned for observation through apertures in the canister sun shield bulkhead. These apertures are covered by movable doors that aid thermal and contamination control for the complete experiment canister. Doors are operated by electromechanical systems mounted on the inside of the sun shield bulkhead. Door arrangement and identification are shown in Figure VC3-2.

Design criteria for the 10 aperture doors are service life of 5,000 cycles, capability for automatic operation, and capability to manually open and disable in the event of motor mechanism failure.

An aperture door drive mechanism is shown in Figure VC3-3. Each aperture door consists of a fiber glass shell filled with aluminized mylar with a shaft attached to one corner and a tapered latch opposite the shaft. The tapered door latch fits into a "U" shaped ramp latch stop attached to the sun shield to retain the door during ascent.

A bulb-shaped seal of silicone rubber, covered with a low frictional cloth called "NOMEX," was attached to the perimeter of the door face to seal the aperture at the sun shield ramp. A release pin and clutch permits manual disabling and disabling latches restrain the door in the open position.

The door drive mechanism consists of two redundant 28 Vdc torque motors, a spindle with an Acme thread, a carriage, a motor lever, a mounting bracket, and limit switches (Figure VC3-3). The torque motor rotates the spindle to move the carriage and the lever. The lever rotates the door shaft to swing the door open and closed. Open and closed positions are controlled by limit switches backed up by spindle hard stops.

Initial operation of the aperture doors began during the first manned phase (mission day 3) after the telescope mount was activated. Operations occurred without incident until mission day 9 when the S054 door malfunctioned. A door by door review of flight performance follows:

(1) S054 aperture door. A review of the astronaut communications during the first manned phase showed that malfunction procedures were accomplished and that the door was stuck in the closed position. The crew then opened the door with both motors and a decision was made to temporarily leave the door open.

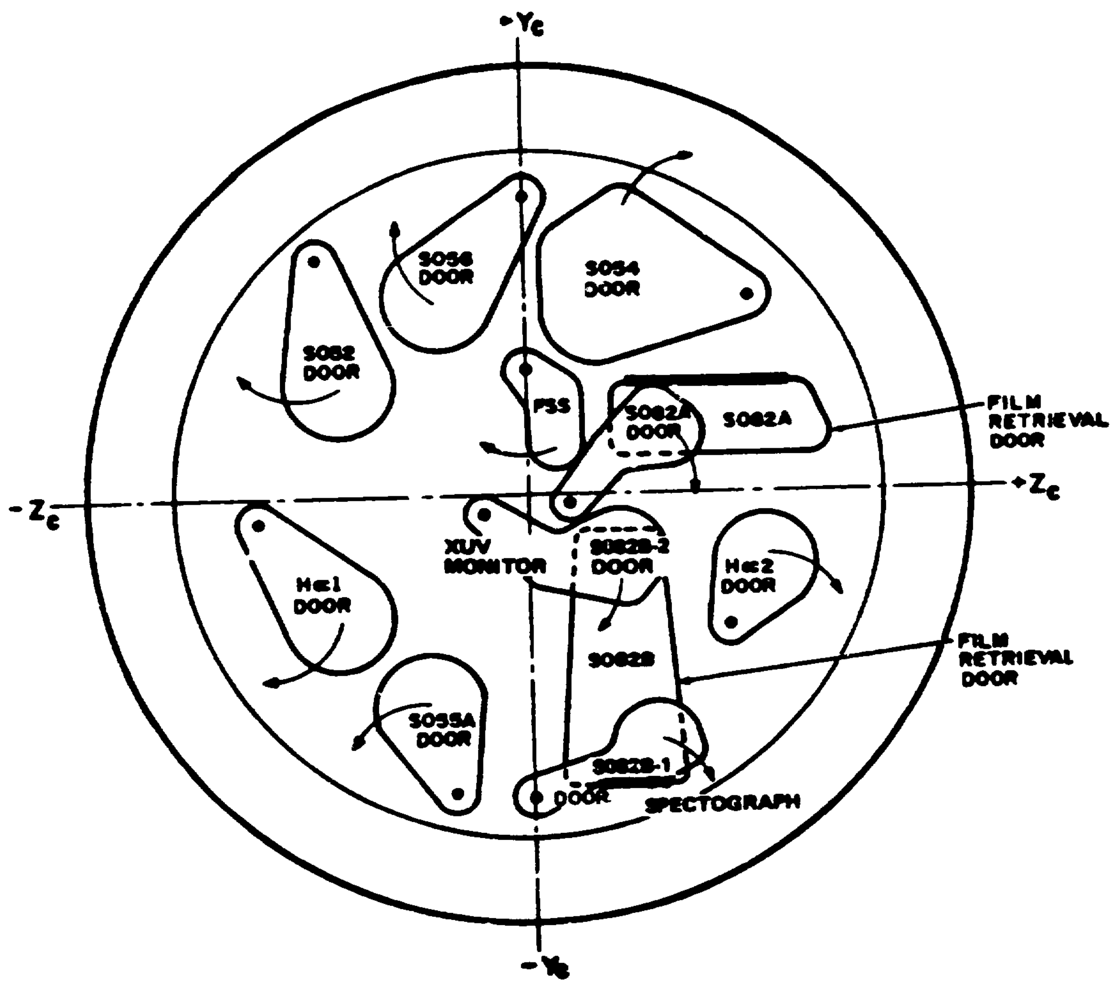


Figure VC3-2. Telescope Mount Aperture Doors

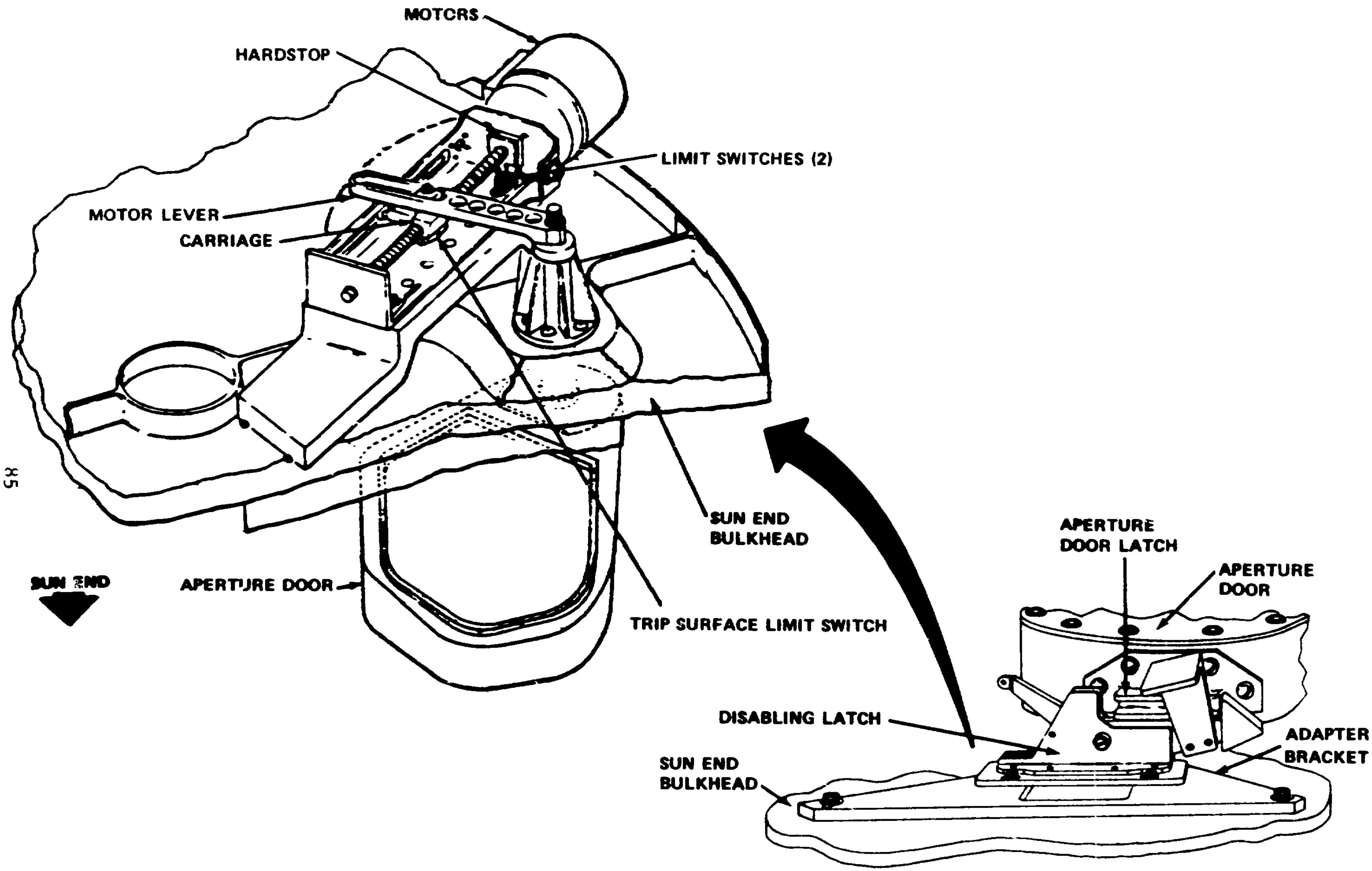


Figure VC3-3. Typical Aperture Door Drive Mechanism Schematic

To establish the operational condition of the door, a troubleshooting procedure was written. However, after several reviews, the decision was made to latch the door open. The crewman found the door closed on the first extravehicular activity and he disabled and pinned it in the open position on mission day 15.

(2) S055A aperture door. A door open/close malfunction on the S-055A door, similar to that on the S-054 door, was observed during the first manned phase, mission day 20. This continued intermittently through the remainder of the first manned phase and again during the second manned phase. Some confusion existed between the crew and ground personnel in the interpretation of the electrical logic and results of malfunction procedures. However, it was determined that the primary motor was capable of opening and closing the door. Consequently, door operation was resumed on the single motor after each malfunction procedure was accomplished. A special troubleshooting test was run on the S055A door while making a pass over Vanguard during the second unmanned period (5 days after completion of the first manned phase). Three timed opening and closing cycles were made using first the primary and then the secondary motor. The result showed normal operation on either motor, i.e., 9 sec to open/close.

Later, malfunctions increased in frequency and a decision was made to resort to two-motor operation on day 3 of the second manned phase. Because of these malfunctions, the ramp latch was removed on mission day 10 of the second manned phase. Single motor operation was then resumed without further S055A door malfunctions throughout the remainder of the Skylab mission.

(3) S056 aperture door. Failure to open malfunction indications occurred on the S056 door during the first manned phase on mission days 24 and 25. However, normal operation was resumed following the malfunction procedures.

During initial telescope mount operation for the second manned phase, malfunctions re-occurred on the S056 door with failures to open recorded on mission days 11 and 12. Malfunction procedures enabled operations with two motors until the ramp latch was removed during extravehicular activities on mission day 28. The S056 door was then operated by a single motor throughout the remainder of the Skylab mission,

(4) S052 aperature door. The S052 door was the fourth to experience a malfunction. The same type malfunction occurred at two different times during the second manned phase, on mission day 8 and again on mission day 28. The S052 door telemetry indicated "OPEN", but analysis later proved the door to be closed. On mission day 8, the malfunction occurred when the solar inertial mode was commanded, and on mission day 28 when S052 standby power was commanded. The final analysis theorized that the malfunctions were caused electrically. The S052 door operated normally throughout the remainder of the Skylab mission.

(5) S082A aperture door. On mission day 23 of the second manned phase, the S082A door was switched to two-motor operation after opening/closing malfunctions were evident. On mission day 28, the S082A ramp latch was removed during extravehicular activities and one-motor operation was then resumed. Other malfunctions appeared during the third manned phase on mission days 19, 20, 24, and 25. Subsequently, two-motor operation was used until the door was latched open on mission day 40 during extravehicular activities.

(6) H alpha 2 aperture door. The H alpha 2 door malfunction occurred initially on mission day 43 of the second manned phase when it failed to close, followed by the same type failure on mission days 52 and 56. Data showed that the time required to open and close the door subsequent to mission day 43 was normal, 9 sec. However, a new problem appeared during malfunction procedures and it became evident from the absence of open or closed indications that the primary motor circuit had failed. The door was latched open during the first extravehicular activity of the third manned phase (mission day seven).

(7) S082B aperture door. The seventh and final aperture door to malfunction was the S082B door, half way through the third manned phase. The malfunction was similar to the other doors. While operating on one motor, failure to open and close was observed. Rather than a special extravehicular activity to remove the ramp latch, a decision was made to inhibit the door in the open position since no other experiment was electrically tied in to the S082B-2. The electrical inhibiting was performed on mission day 45.

The corrective actions described above were supported by analyses and tests as outlined in the following. Cycle times for each door were recorded throughout the mission. Spot checks were made from time to time to determine degradation in opening/closing times for the doors. However, no trends in the data allowed prediction of door failure. This is illustrated by the opening and closing times shown in Table VC3-1. Also, operating time with dual motors ran approximately 2 sec less than with single motors.

The prototype telescope mount operating in the sun-end-down orientation was used to investigate malfunctions of the S055A aperture door.

A 100-cycle test was run with both motors operating. The times were consistently 10 sec to open and 10 sec to close.

After the 100-cycle test was run with both motors, single motor operation was attempted with first the primary and then the secondary motor. Five cycles were run on the primary motor, and it took 13 to 14 sec to open. However, the time to close increased from 21 sec on the first cycle to 43 sec on the fifth cycle. The first cycle on the secondary motor

APERTURE DOORS CYCLING TIMES – OPEN/CLOSE [SEC]										
EXPMT. / DOY	FSS	S052	S054	S055A	S056	H α 1	H α 2	S082A	S082B	XUV
153	10.0 / 11.0	8.3 / 7.7	13.0 / 12.0	8.0 / 9.0	12.0 / 12.0	11.0 / 11.0	9.0 / 9.0	6.0 / 7.0	9.0 / 9.0	10.0 / 10.0
262	11.0 / 11.0	7.3 / 7.4	PINNED DOY 158	8.0 / 9.0	12.0 / 12.0	11.0 / 11.0	9.0 / 9.0	6.0 / 5.0	9.0 / 10.0	10.0 / 11.0
397	10.0 / 11.0	7.3 / 7.0		9.0 / 9.0	12.0 / 12.0	11.0 / 11.0	PINNED DOY 326	PINNED DOY 359	INHIBITED CROSS - TIED W. XUV DOY 364	INHIBITED CROSS - TIED W. S082B =

Table VC3-1. Door Cycling

took 13 sec to open but it reached the ramp latch on the closed cycle and consequently did not close since the limit switches could not then be triggered.

At this point, the test reverted to a two-motor operation and the time to open and close the door was again 10 sec. The test was resumed after approximately 90 min and six cycles were run on the primary motor; door opening took 13 to 14 sec with closing time increasing from 20 sec on the first cycle to 52 sec on the fifth cycle. The door did not close on the sixth cycle. On these six cycles the time required to reach the ramp latch was 13 to 14 sec.

The secondary motor opened the door in 13 sec but would not close it although the ramp latch was reached in 15 sec. In an attempt to alleviate the problem noted above, a lubricant, Apiezon L, was applied to the ramp latch. Five cycles were then run on each motor. The primary motor opened the door in 13 sec and the closing time was a constant 15 sec. The time required to reach the latch was 12 to 12.5 sec.

A final test was made by removing the ramp latch. In a five-cycle test, the secondary motor opened the door in 13 to 14 sec and closed it in 14 sec, and the primary motor opened the door in 13 sec and closed it in 14 sec. Five cycles were also run with both motors. The door opening and closing times were a constant 10 sec.

In summary, these tests showed: (1) A failure mode with one-motor operation in a 1 g environment (sun-end-down) caused by dry film lubricant wear; (2) Continued operation could be expected with two motors.

Because of the malfunctions encountered during the Skylab mission, three aperture door ramp latches were removed, one (S055A) during a second manned phase extravehicular activity and two (S056 and S082A) during a third manned phase extravehicular activity. All latches were stowed in the workshop and subsequently returned to MSFC for inspection and analyses. Considerable galling on the ramp surfaces was evident and could have prevented the aperture doors from opening and closing on one motor. The returned ramps were found to be properly coated with a MLR-2 lubricant. However, the material was verified to be 7075 aluminum and hardness tests indicated that it was probably in the annealed condition.

As a result of the foregoing, three items are noted for future aperature door design considerations:

1. Hard anodize and coat aluminum working surfaces with a lubricant which does not require baking to eliminate annealing of the coated aluminum.

2. Do not design devices that have a launch/ascent environment protection function which must also be used operationally, but serve no operational function.

3. When reviewing generalized timing data for trends, be alert to the possibility that the problem phase may be so minute that it is not recongnizable within the times recorded.

4. Thermal Control System Hardware. The telescope mount thermal control system consists of an active and passive system that provides a controlled temperature environment for the performance of the experiments and supporting equipment. The active system provides thermal control for the canister experiment package by means of a cold plate/radiator/heater network. The entire canister is insulated with a multilayer aluminized Mylar shroud that isolates the experiments and canister from the external environment and rack-mounted equipment. The spar is thermally insulated to provide minimum spar temperature transients and gradients, and to maintain the required alignment between the telescope mount telescopes and the fine sun sensor. The rack-mounted support equipment is thermally controlled by passive means. This is achieved with insulation, low conductance mounts, and thermal coatings. Also, several telescope mount experiments and rack-mounted equipment contain individual, thermostatically controlled heaters to provide for internal temperature control.

The active thermal control system (Figures VC4-1 and VC4-2) is a fluid (80 percent methanol/20 percent water) coolant loop that rejects heat from the solar experiments. The heat from the experiments (500 W max.) is radiated to the walls (panels) of the telescope mount canister. The coolant flow to the upper and lower canister half is equal.

The division of the flow is obtained by adjustment of two flow path restrictor valves. The valves were adjusted and locked in position during checkout of the telescope mount thermal control system. The heat collected by the canister panels is transferred to the coolant, and the coolant is pumped to radiators for heat rejection to space. The temperature control system consists of a modulating flow control valve (mixing valve, electronic controller, heater, temperature sensor, and radiator bypass). The modulating flow control valve (MFCV) mixes coolant from the radiator and bypass to provide canister inlet temperature. The heater is located in the bypass line and is activated when the temperature falls below 47.7 °F.

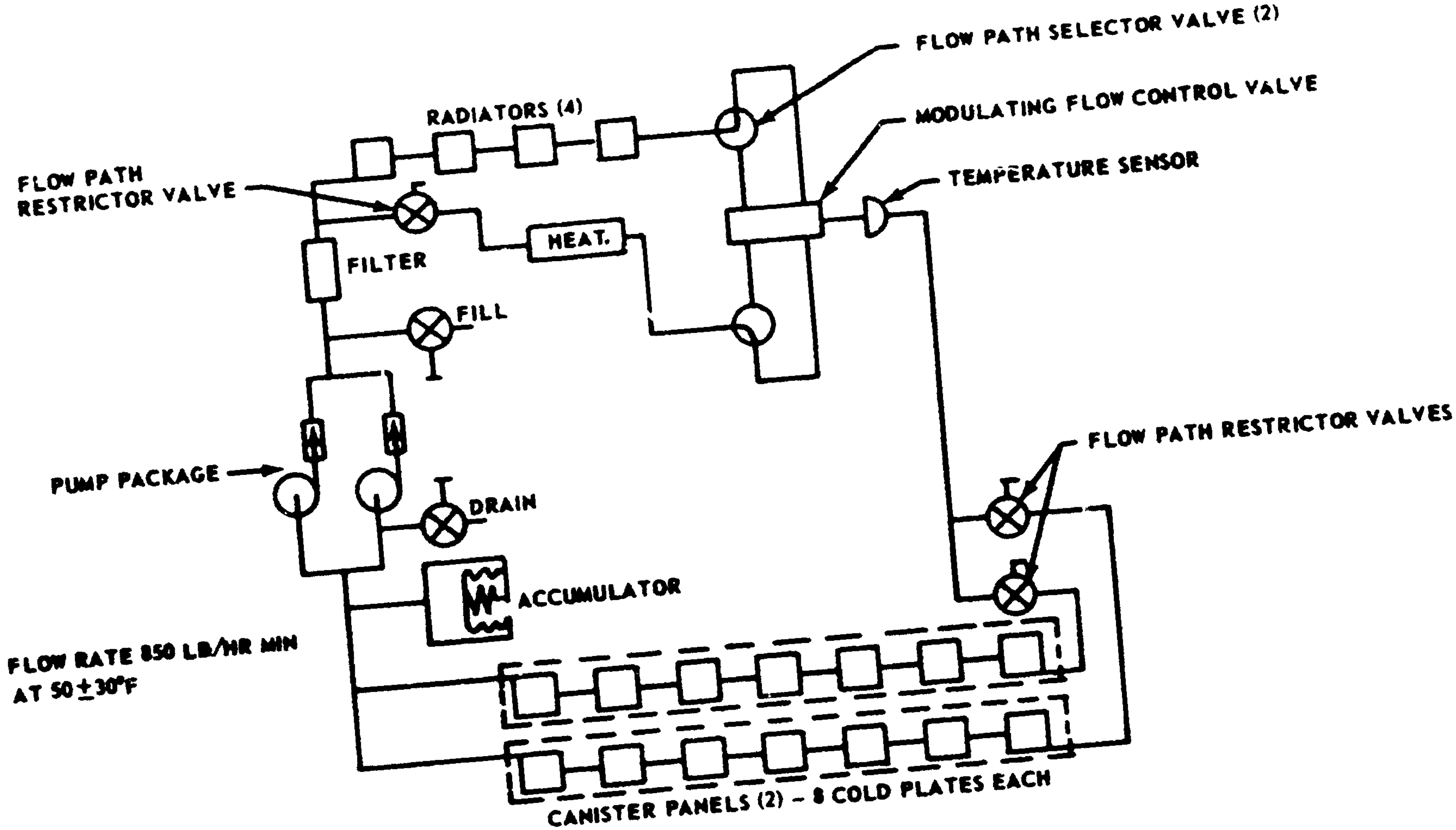


Figure VC4-1. TM Active Thermal Control System Fluid Schematic

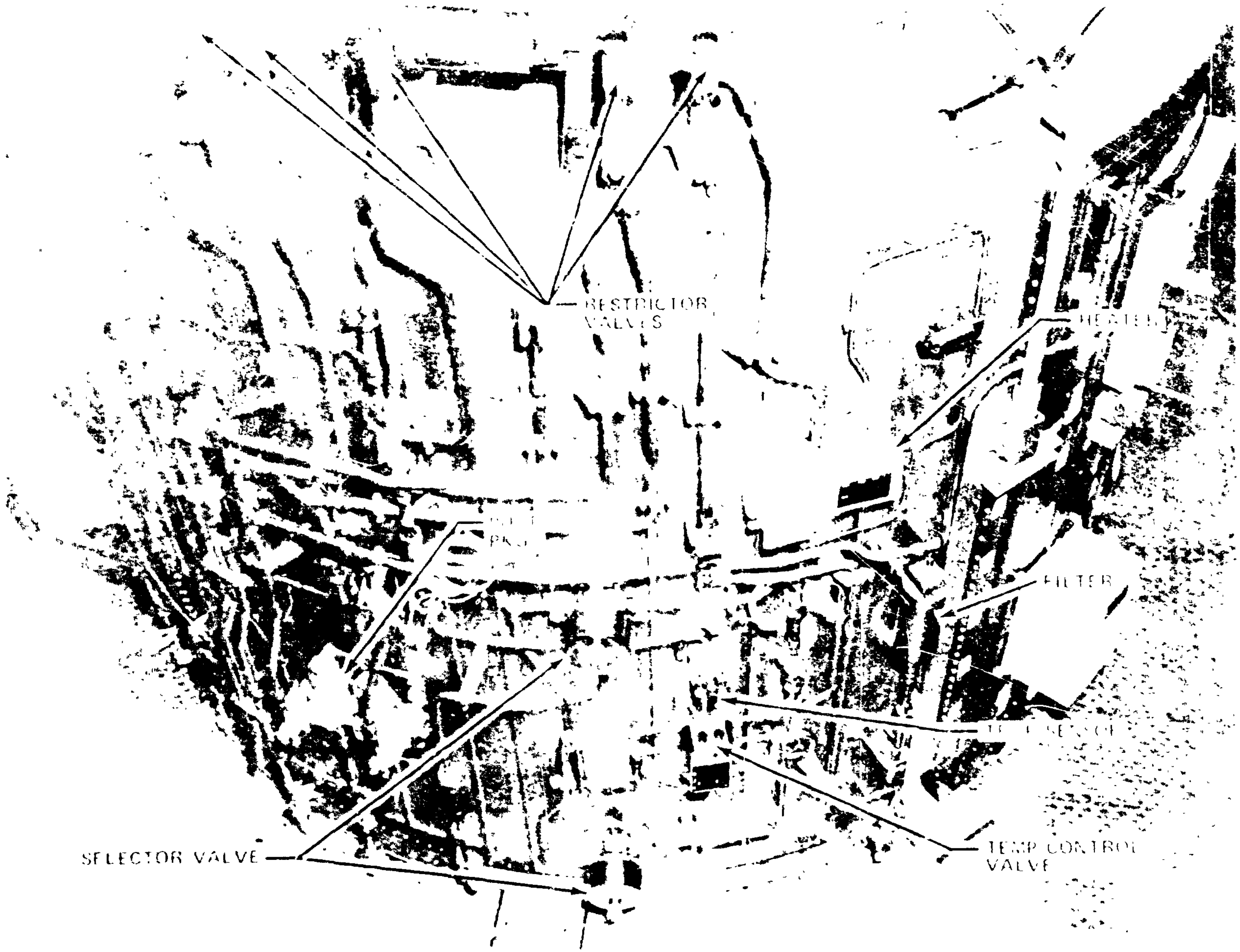


Figure VC4 2 TCS Installation (Insulation Removed)

The thermal control system coolant accumulator provides a positive pressure at the pump inlet, allows for thermal expansion/contraction, and provides makeup fluid for leakage. Fill and drain valves are the same configuration as the flow path restrictor valves. A second flow path restrictor valve is installed in the bypass line upstream of the heater to allow for equalization of flow resistance in the radiator and radiator bypass portion of the system.

Two flow path selector valves are upstream of the MFCV to select either the primary or secondary MFCV.

During system development, two significant problems were encountered: coolant pump bearing swell and seizure, and temperature control instability.

a. The coolant pump bearing material (Fiberite; TFE-- a fluorocarbon resin) was selected to eliminate corrosion between the bearing and the coolant medium. Long term exposure to the coolant resulted in bearing swell and finally seizure. The design fix utilized the same bearing material. However, the bearing assembly was modified to allow for determined swell rates and long life capability.

b. The temperature control instability was the result of:

1. High response rate of the modulating flow control valve.

2. Modulating flow control valve flow orientation with respect to the valve flapper.

3. Electronic controller response rate.

4. Inadequate fluid mixing at the temperature sensor.

The effect of the above described conditions was valve instability and inadequate temperature control.

The system was modified by relocating components such that fluid pressure opposed the valve flapper movement, making the system more dynamically stable. The controller gain was decreased and fluid mixing was improved to prevent temperature stratification at the temperature sensor.

The pump (Figure VC4-3) was qualified for a flow rate of 850 lb/hr minimum of 80 percent methanol/20 percent water at 31 psid and life of 6,672 hr.

During the Skylab mission, the flow rate varied, as a function of MFCV position, between 325 and 975 lb/hr. The pressure rise of the pump varied between 26.5 and 27.75 psi.

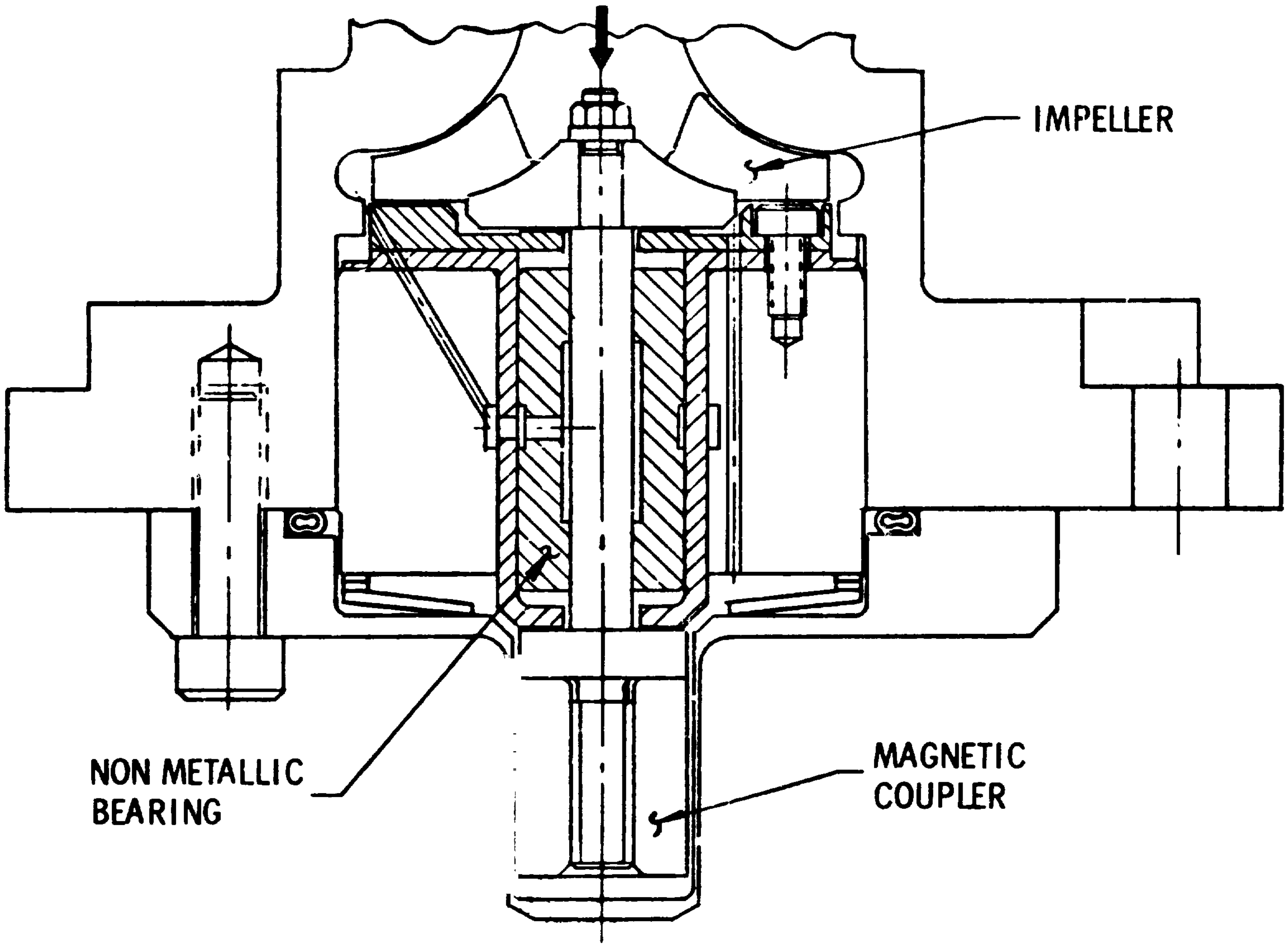


Figure VC4-3. Pump Assembly

The primary pump operated for 6,154 hr of flight time and 305 hr prior to flight with no problems.

The secondary pump operated for 2 hr in flight and for 216 hr prior to flight, after being dormant for approximately 275 days. The flow rate was comparable to the flow rate experienced during preflight checkout of 875 lb/hr.

The accumulator was qualified to maintain pump inlet pressure above 7 psia, to provide fluid for leakge, and to provide for thermal expansion and contraction between bulk coolant temperatures of -65 to 100 °F.

During the Skylab mission, the pump inlet pressure was always above 12 psia with no detectable leakage.

The main flow control valve was qualified to fully open to radiator flow at an outlet valve temperature of 51 °F, to fully open to heater flow at an outlet valve temperature of 49 °F, and to maintain a valve outlet temperature of 50 ± 1 °F. During thermal vacuum testing, the valve set point appeared to be 49.7 °F.

During the Skylab mission, two excursions were noted during the early part of the first manned phase, and the temperature control point appeared to have shifted downward by about 0.36 °F from the control point noted during the ground thermal vacuum testing. Also, the temperature control point did not appear as stable as that noted during the thermal vacuum testing. One explanation for the temperature excursion is that contamination in the coolant prevented the valve from fully closing as the radiator fluid temperature decreased. As the temperature of the radiator warmed up, allowing the valve to open, the contamination washed through the valve and was trapped in the filter. After several days, the system was cleaned by the filter. The reason for the long cleaning time is that the radiator flow was very small. The small flow prevented a good thorough flush of the radiator.

For future design it is recommended that filters be installed at the inlet of valves with close operational tolerances. Breadboard the system as early as possible to detect instabilities. Locate valves and sensors at mixing points to reduce system lags.

5. Extravehicular Activity Hardware. A crew translation hardware system is installed on the telescope mount external structure to provide support for extravehicular activity. The system consists of handrails, foot restraints, and life support umbilical clamps. Three extravehicular activity work stations are provided at the rack structure: the center work station located at the open rack bay 45° between the telescope mount coordinate axes +Z and Y; the transfer work station located at the edge of the solar shield; and the sun end work station located on the outer surface of the solar shield. Figure VC5-1 depicts the arrangement of this hardware.

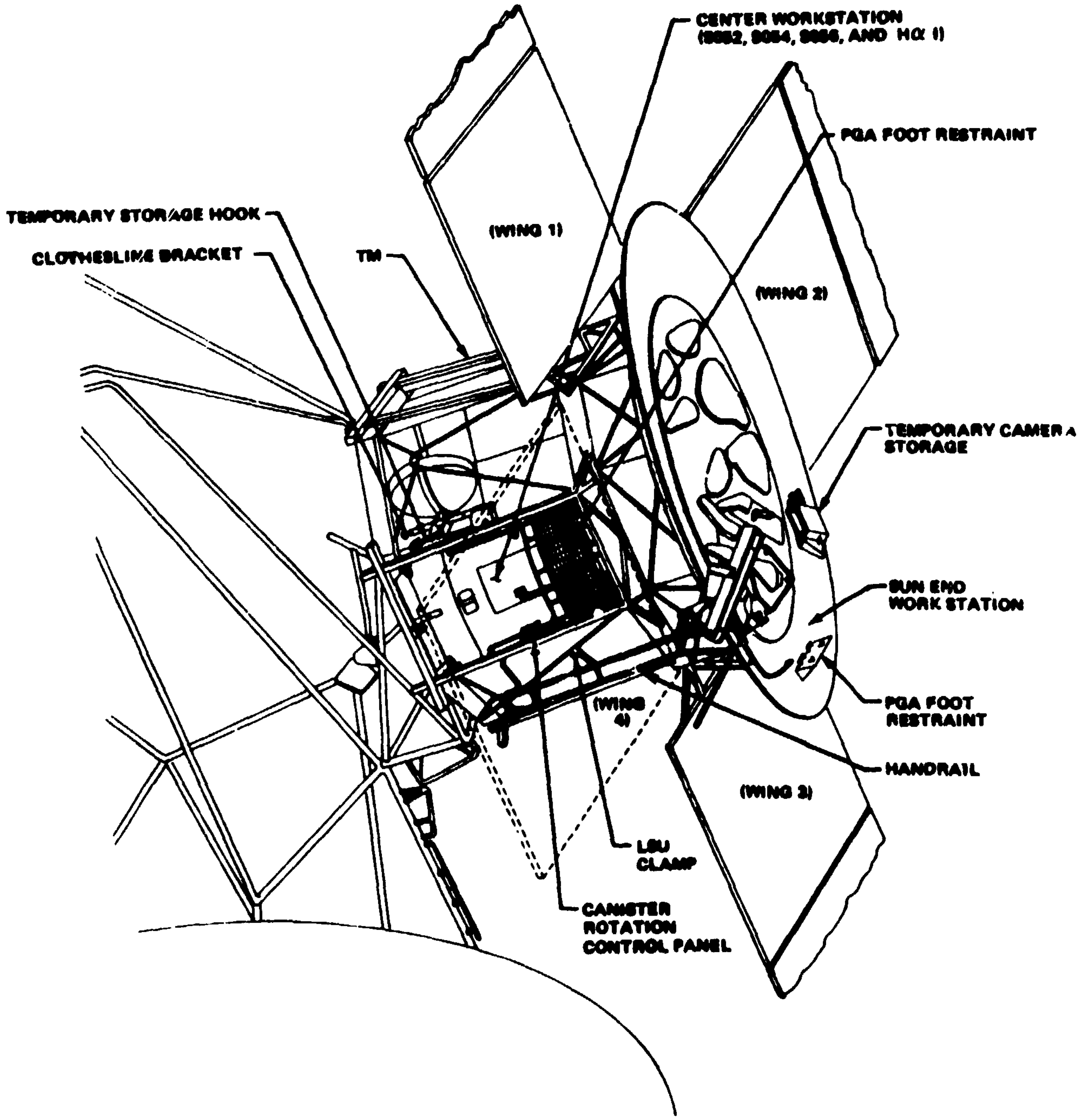


Figure VC5-1. Extra Vehicular Activity Support Hardware

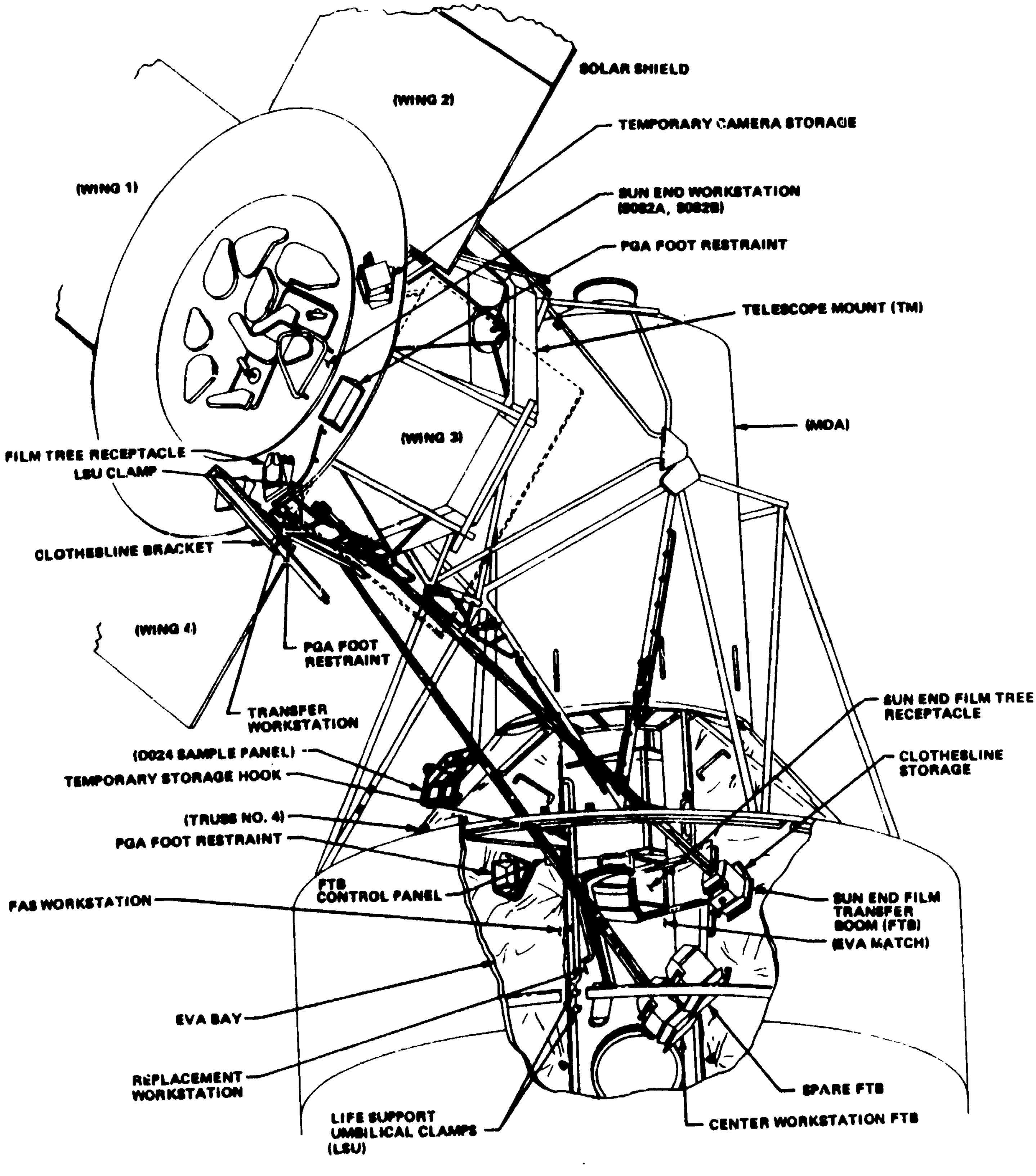


Figure VC5-2 Extra Vehicular Activity Support Hardware

Transfer of experiment film cassettes and cameras is accomplished by: camera trees and tree receptacles, film transfer booms, and a clothesline system.

a. Telescope mount handrails. The basic requirement for the extravehicluar activity film retrieval handrails was to provide a route for the crewman from the airlock hatch to the center work station and to the sun end work station on the telescope mount.

The telescope mount handrails consist of a single rail adjacent to the outrigger, dual rails from the outrigger to the sun end between the solar array panels, and a single handrail from the outrigger to the center work station. Figure VC5-1 shows the location of the telescope mount handrails in the extravehicular activity path.

A design criterion for the hand rails was that they be designed for an ultimate load of 600 lb at any location. With this load and the long span between supports, steel was required for the handrails.

Additional constraints on the rails, supports, and fasteners were that they have no sharp edges or burrs. Protruding bolts and nuts are avoided where possible, and where necessary the nuts are covered with RTV-140 to ensure no burrs. Also, the handrail routing is configured to provide clearance to the payload shroud in the launch configuration.

The handrails were installed on the telescope mount vibroacoustic test article and no problems were found. The strength of the rails was verified by analysis and utilized in neutral buoyancy testing.

The functional adequacy of the handrails was verified by neutral buoyancy training of the crewmen

The shape of the cross section of the handrails is satisfactory and the overall design was adequate based on crew comment. It is recommended for future applications that a 200-lb design limit load be used for handrails commensurate with airlock handrail design criteria, which was found to be satisfactory.

b. Extravehicular activity foot restraints. To accommodate hand tasks, extravehicular activity-type foot restraints are provided at various Saturn workshop exterior locations and a portable restraint was provided in the workshop.

The telescope mount foot restraints are designed for a 100-lb concentrated load at the heel clips and an 1800 in.-lb torsional load. The plate and heel and toe clips are made from 7075-T73 aluminum.

The structural integrity of the foot restraints and their installation was verified by analysis.

The foot restraints were installed on the vibroacoutic test article and no problems were found.

A set of "boot gages" was provided by JSC that represents maximum and minimum tolerance of boot heel clips. These were tested in a qualification test foot restraint at ambient, low, and high temperatures, and no problems were found.

The functional adequacy of the foot restraints was verified by neutral buoyancy training of the crewmen.

Foot restraints satisfactorily performed their function during use by the Skylab flight crews.

c. Umbilical clamps. To aid in managing the umbilical during extravehicular activities, a requirement to provide a clamp for the umbilical adjacent to the center work station and between the transfer work station and the sun end work station was established. Also, two of these clamps are located in the fixed airlock shroud area for management of both umbilicals used by extravehicular activity.

The clamps were designed with a fixed jaw and an over center, spring-loaded, movable jaw that could be opened for inserting the umbilical. The movable jaw clamped over the fixed jaw after inserting the umbilical. The jaws are lined with a 0.13-in. layer of mosite rubber to provide a friction surface and to protect the umbilical. The linkage for the over center spring-loaded movable jaw is critical to ensure that the jaw will remain open when set in the over center position, but still close positively when the umbilical is inserted.

The structural integrity of the umbilical clamps was verified by analysis.

A qualification test unit (flight design) was functionally cycled 150 times, about 5 times the expected flight use, using a section of simulated flight umbilical. Also, the qualification test unit was subjected to the high and low temperature extremes expected and cycled 12 times with no problems.

During neutral buoyancy training, it was determined that sun end film retrieval could be accomplished without using the clamp. This clamp, however, was left on the telescope mount for potential unanticipated requirements. It is questionable whether the clamp at the telescope mount center work station was a necessity and some crewmen did not use it as stated in crew debriefings.

Clamps used by crewmen performed satisfactorily in retaining inserted umbilicals.

d. Center and sun end work station trees and tree receptacles. The "trees" were frames that provided for attaching the film cameras to expedite their handling during camera retrieval and replacement on the telescope mount. One "tree" was provided that held the cameras used at the center work station and another that held the cameras used at the sun end. The replacement cameras were loaded on the trees inside the airlock before extravehicular activities. The loaded trees were then passed by hand from one crewman in the airlock to the second crewman in the fixed airlock shroud area. This second crewman installed them in a receptacle. For the center work station, the cameras were removed from the trees and sent to the work station on the film transfer boom or on the clothesline. For the sun end work station, the complete tree was sent to the work station, installed in a receptacle, and the cameras were removed from the tree.

The tree receptacles were simple surface plates with contoured guides for centering the trees and an opening for accepting the latch on the trees. The structural integrity of these was verified by analysis. These were functionally tested with the trees to ensure proper installation, fit, and latching.

The trees were stowed in the docking adapter for launch. The sun end tree used the same type spring-loaded finger latches for stowing as were used to attach the sun end cameras to the tree. The center work station tree was stowed using four quick release fasteners. Both trees were subjected to vibration tests in their stowed configuration and no problems were found. The loads criteria for trees were that they take a load of 100 lb applied at any place on the tree when the tree was in its receptacle. The structural integrity of the trees was verified by analysis.

In addition to vibration testing of the trees in the stowed position, functional tests were made on the qualification test trees. The trees were latched and locked to the receptacles in ambient, high, and low temperature extremes. The cameras were installed and removed from the trees. In all the functional tests, the operations were repeated 5 times for each scheduled operational use.

Both trees were used successfully during the Skylab mission. The sun end work station cassette tree was used seven times and the center work station tree was used six times. The tree to receptacle positioning is a design that may be useful for future applications.

The sun end work station cassette tree with the S082 A&B camera canisters is shown in Figure VC5-3.

The center work station tree, with four cameras installed, is shown in Figure VC5-4. These are the four cameras that were retrieved and replaced in the center work station.

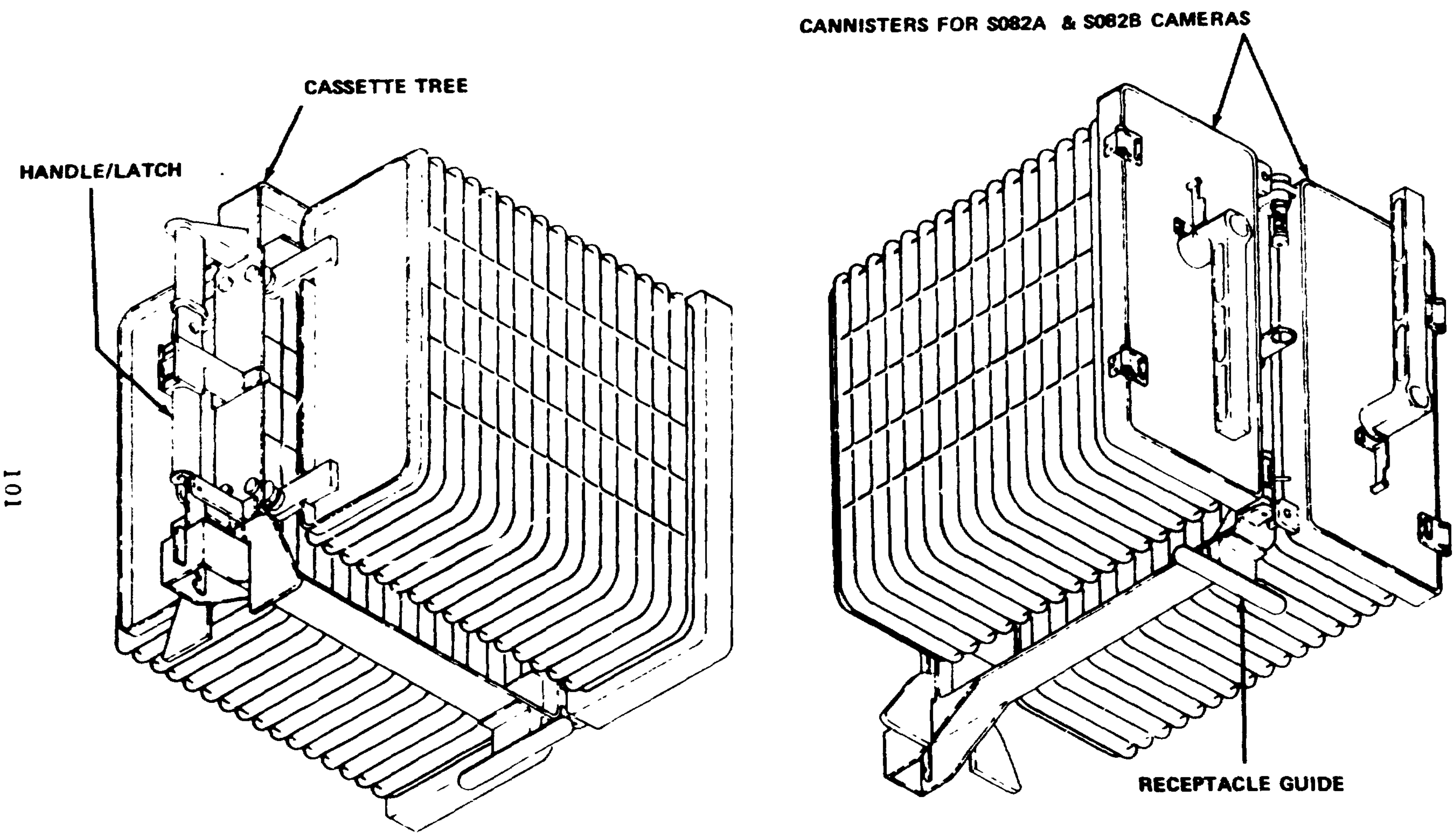


Figure VC5-3. Sun End Workstation Cassette Tree with S082A and S082B Camera Cannisters

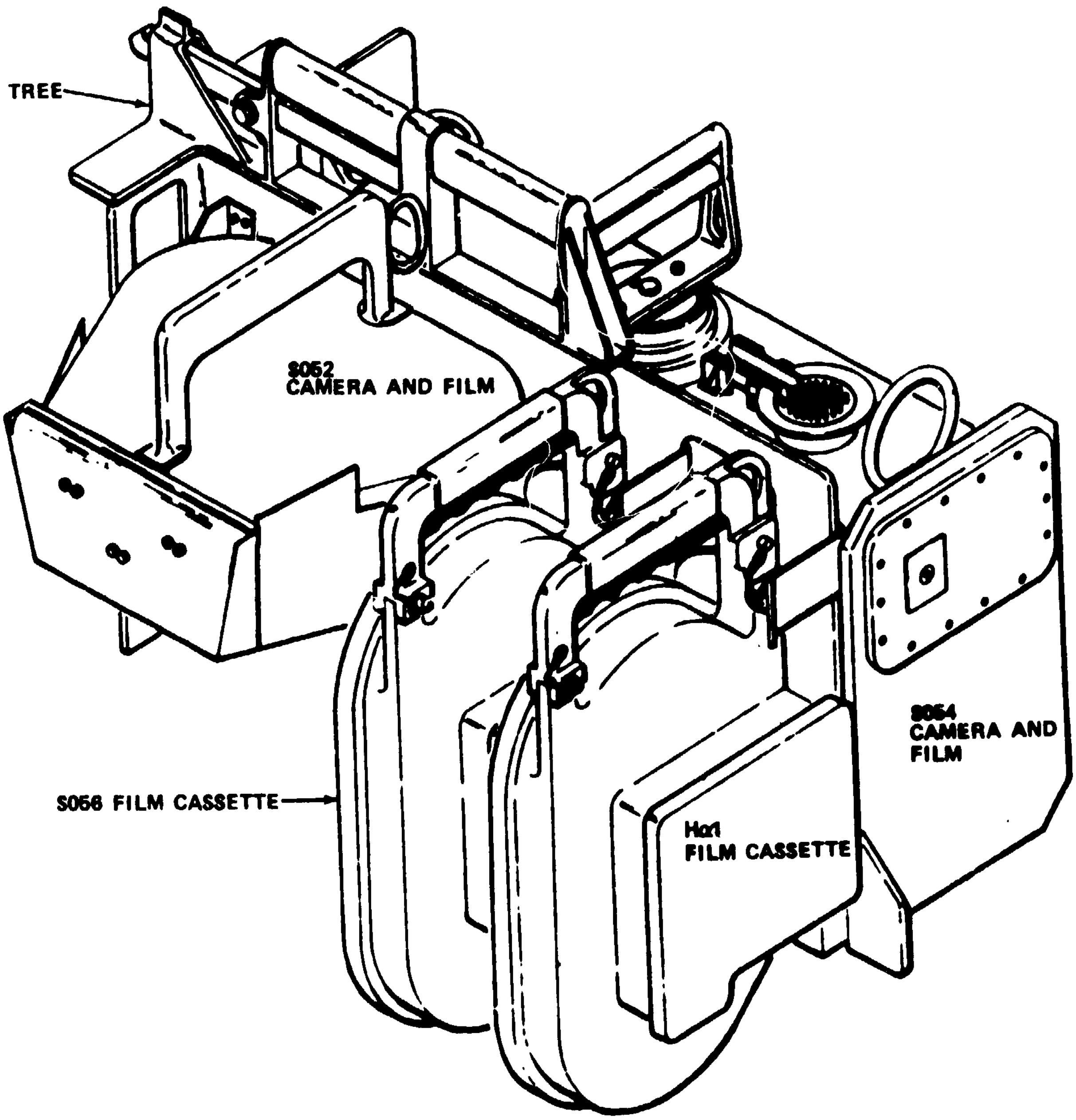


Figure VC5-4 Center Workstation Tree (loaded)

e. Camera transfer equipment. Two basic methods of transporting the film cameras between the airlock and the telescope mount were developed. The primary method was the film transfer boom that was developed and tested by McDonnell Douglas and the Fairchild-Hiller Company. The backup method was referred to as the "clothesline" where the cameras were transported by attaching them to hooks on a rope for movement to the required work area.

The film transfer boom uses a motor to extend two preformed sheet metal rolls through guides to form a semirigid element when extended. By reversing the motor, the element is rewound on the rolls to retract.

A boom hook was developed for attachment to the tip of the extendable element. The size required for this boom hook, to attach the cameras and to be capable of one handed operation by the suited crewmen, dictated that the boom hook be stowed separately from the film transfer boom and installed on the boom tip by the crewman. A stowage container developed for the boom hooks is installed in the fixed airlock shroud area within convenient reach of the crewman. Hook installation on the tip of the film tranfer boom uses a modified fluid-line-type quick disconnect, omitting the usual seals.

The film transfer boom was subjected to the vibration tests that were expected during the boost flight, the temperature extremes expected during the mission, and the tip loads expected during operation.

The boom hooks and their stowage container were subjected to the vibration environment expected during operation. The functional procedure of installing the quick disconnect on the tip and the functional operation of the hook latching, locking, unlatching, and unlocking from the camera handles were verified. The functional operation of the film transfer boom and boom hooks with cameras attached was verified in neutral buoyancy training with the suited crewmen.

This primary method of camera transport was used with no problems on all the extravehicular activities except the final one where the backup clothesline method was used to evaluate that system. The film transfer boom assembly is shown in Figure VC5-5.

The clothesline system provides a continuous loop of cord with an attach hook at each end and two hooks spliced into the cord for camera attachment. A separate looped cord is provided for the center work station and for the sun end work station. The cords are packaged in containers attached to the film transfer boom mounting structure in the fixed airlock shroud area.

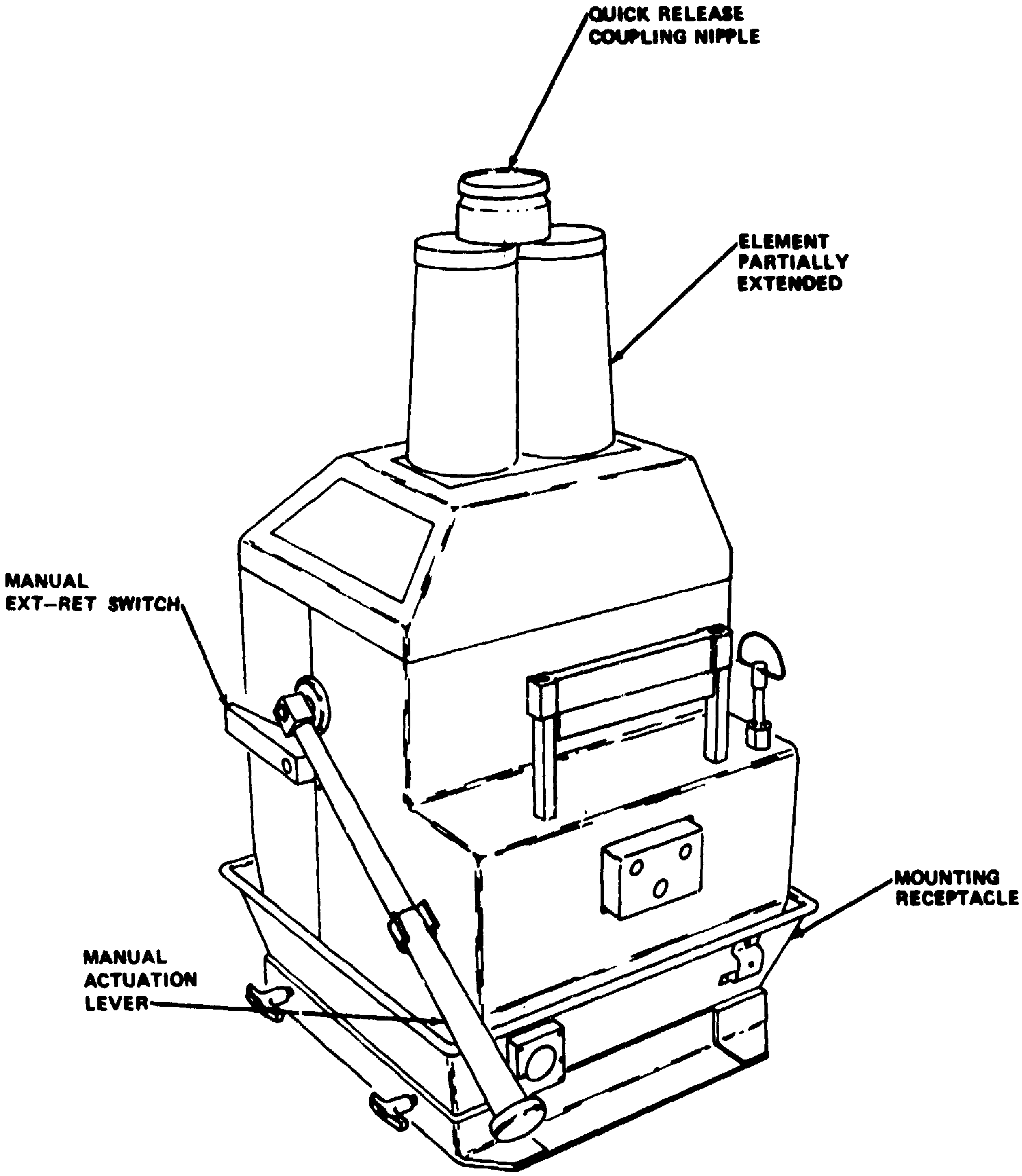


Figure VC5-5. Film Transfer Boom

On the telescope mount, a clothesline "pole" is installed adjacent to the center work station and one adjacent to the transfer work station at the sun end. The optimum location for the pole tip, where the clothesline hooks attach at the work stations, was determined in early neutral buoyancy simulation. To keep the clotheslines and camera transfers out of the work area, the pole tips were required to be at a considerable distance from the work station. This necessitated installing the poles in a stowed configuration that was inside the payload shroud envelope and then deploying the poles to their required operational configuration. The center work station pole is hinged and deployed at variable angles with a spring loaded detent. The sun end pole was secured in the launch configuration with a quick release pin and latched in the deployed position with a spring-loaded detent and a spring latch.

The poles were designed for a maximum load of 300 lb applied at the tip after deployment. This load is not realized during clothesline operation, but a crewman could conceivably apply equivalent dynamic load during deployment or in emergency hand hold use.

The center work station pole was vibration tested on the vibroacoustic test article and subsequently functionally tested by deploying it 5 times the maximum anticipated flight use at ambient, high, and low temperature extremes.

The sun end closthesline pole assembly was vibration tested on a test fixture and functionally tested by deploying it at ambient, high, and low temperature extremes.

The clothesline cord material was PBI cord. The splices were made using PBI thread with a coating of Silastic RTV-140 over the splice to prevent fraying. A section of the cord containing a splice joint was tested for strength after fabrication and then was subjected to the equivalent of 2,000 hr of sun exposure in a vacuum and retested. The average load at failure for three specimens on the initial test was 427 lb and at the end of exposure the average load at failure for three specimens was 352 lb. Based on these tests, the material was considered satisfactory.

The structural integrity of the hooks used with the clothesline was verified by analysis. Qualification test hooks were subjected to cyclic testing at ambient, high, and low temperature extremes. The number of cycles was at least five times their maximum expected use with no failures; therefore, the hooks were considered qualified.

From the crew debriefing, it is evident that the decision to use the film transfer boom as the primary system was the right one. The cameras were successfully retrieved using the clothesline system, but the crewman found that the cords could easily be tangled and thus additional time was required to straighten them.

To summarize, the functional aspect of the clothesline system worked well. Its presence did not compromise the use of the primary system, but it was not as efficient a system as the film transfer boom. This is substantiated by the following comments made by the third Skylab crew during debriefing on February 22, 1974:

CARR - "Boom operation - The booms just worked like champs. We found them to be superior to the clothesline operation because you didn't have the tangle, the intertwining problem, that you had with the clothesline. I think our modes of operation were the right way to go. The boom is the prime mode and the clothesline is the backup mode if the boom fails. The clothesline mode is a good mode of operation. It's quite usable but it take more time and it's a little more trouble".

GIBSON - "Talking about clotheslines gets into what we encountered during the last EVA. That was the amount of clutter we had in the FAS work station in the way of clotheslines. We had two clotheslines out in the stem. We had all the ATM flim which was stowed back there which we had retrieved. We had S020 out and T025, and a DAC out there and or a Nikon, and two people up in that area working. I found it really did get crowded. We were able to get it all sorted out. I believe that's a higher level of mechanical and geometric complexity than you should put into an EVA. Also, that's when I got the rope from the clothesline hooked into my PCU".

f. Temporary stowage components. Film retrieval aids for temporary stowage of the cameras at the work stations consisted of a container at the sun end work station where the S082 camera could be stowed during replacement. This container was a simple aluminum box, which allowed about 1-in. clearance around the camera envelope, and plastic snubbers to keep the camera contained. This design was verified by analysis. At the fixed airlock shroud and center work stations, permanently mounted temporary stowage hooks were provided for holding the film cameras temporarily during replacement operations. These were a similar design to the clothesline hooks and the design was verified by analysis and functional testing at ambient, high, and low temperature extremes.

Both the temporary stowage container and the temporary stowage hooks were used during extravehicular activities. Their use during this complex activity aided Skylab crewmen.

D. Telescope Mount Deployment Assembly

The telescope mount deployment assembly is required to rigidly support the telescope mount in orbit in a position rotated 90° from the launch axis. It was flexibly attached to the telescope mount during launch and boost through floating joints at the telescope mount interface, such that the lateral and axial loads generated by the telescope mount were reacted solely by the payload shroud. Provisions to rigidize the floating joint immediately upon shroud jettison and prior to deployment of the telescope mount were made.

1. Deployment Assembly Structure. The deployment assembly structure consists of two aluminum tubular truss assemblies joined together by two trunnion joints (Figure VD-1) that allowed the upper truss assembly to rotate through 90° to deploy the telescope mount. The lower truss assembly is attached to the fixed airlock shroud at 10 places through rod ends that allowed adjustment for precise telecope mount alignment The upper truss assembly incorporates four rigidizing mechanisms attaching to the telescope mount (Figure VD-2). The rigidizing mechanisms were capable of 2.0 in. deflection in all directions while the telescope mount was supported by the payload shroud. Following payload shroud separation, springs in each rigidizing mechanism retracted and rigidly affixed the telescope mount to the deployment assembly. An overcenter spring/lever mechanism locked the rigidizing mechanisms in the retracted position. The deployment assembly structure is required to have a design safety factor of 3.0 times the maximum predicted loads to preclude extensive ground testing. The stiffness of the deployment assembly structure in the deployed and latched position is required to provide a minimum natural frequency of 0.6 Hz.

Ground verification of the static load capability was conducted by analyses to verify the design safety factors. A complete qualification program was conducted on the rigidizing mechanism to verify performance at high (160 °F) and low (-70 °F) temperatures and after exposure to humidity and the predicted flight vibration levels. A deployment qualification test was run on a complete flight-type assembly with the longitudinal axis in a horizontal position. The unit was assembled and aligned in one building then transported to another building with the ground support and transport equipment. It was then set up vertically and six deployment tests run to verify that alignment was not affected by handling and transporting. The unit was then set up in a horizontal position with the mass/inertia characteristics of the telescope mount simulated by the weight of the upper truss assembly and telescope mount simulator, supported by a cable and pulley arrangement, and a full deployment test was satisfactorily completed. The first production flight-type

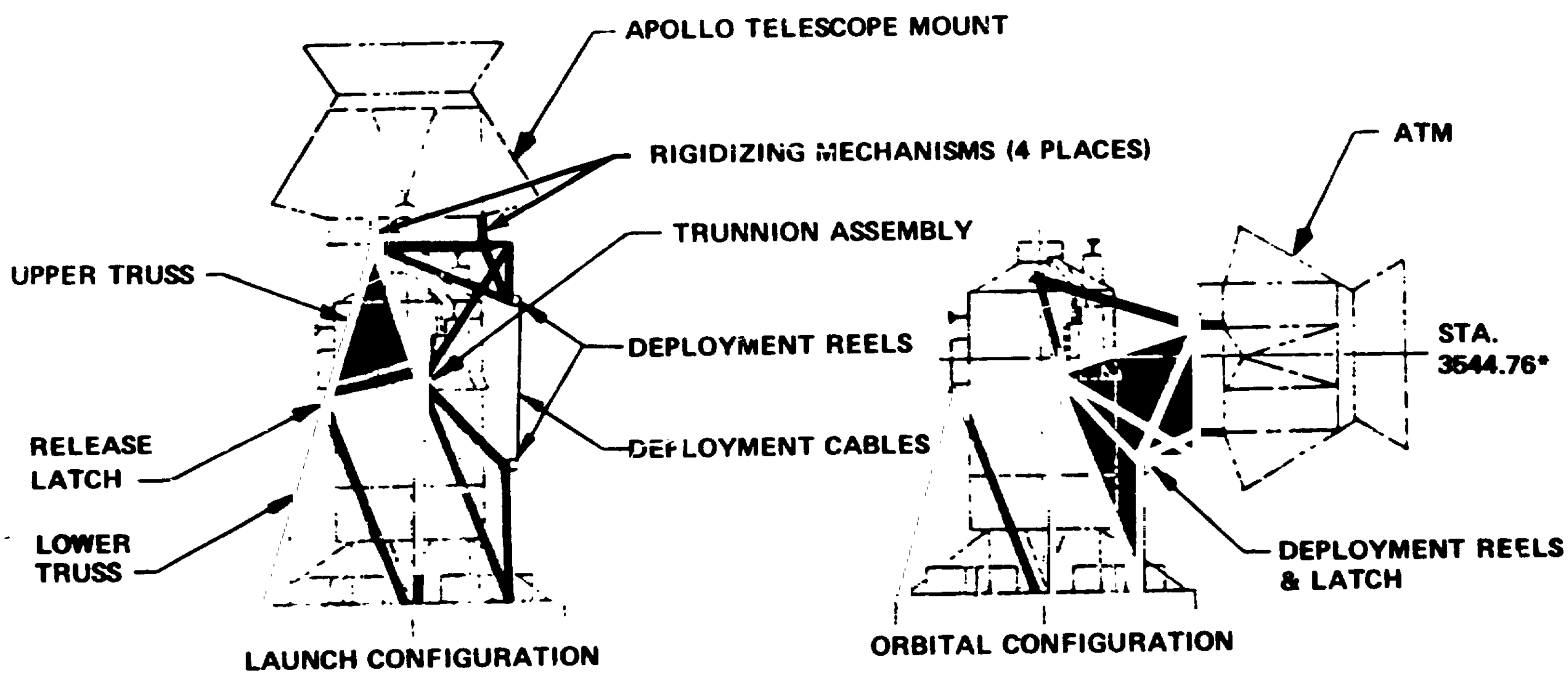


Figure VD-1. Telescope Mount Deployment Assembly General Arrangement

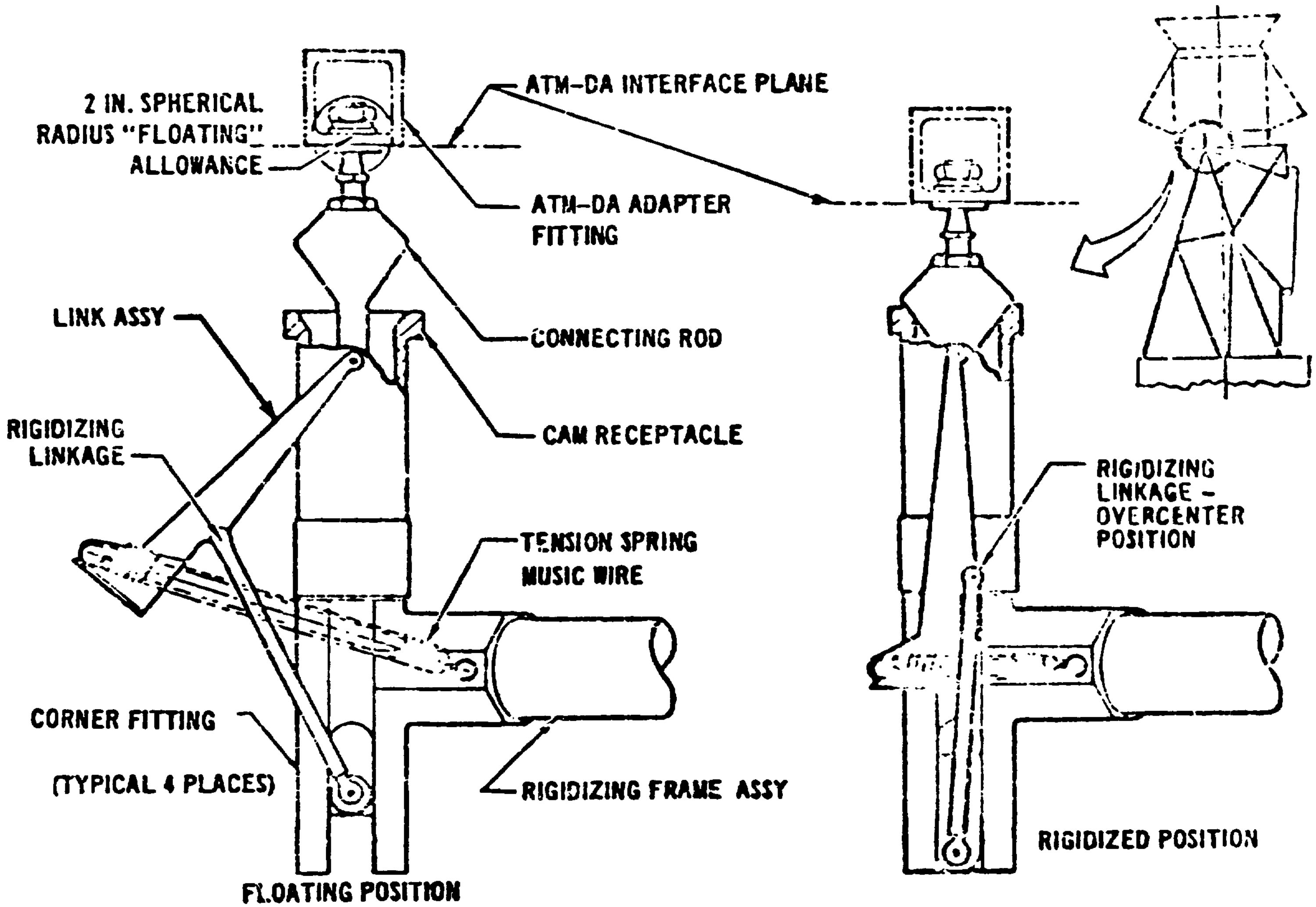


Figure VD-2. Rigidizing Mechanism—Telescope Mount deployment assembly was subjected to the vibroacoustics tests at JSC in both the launch and in-orbit deployed positions. In the deployed position tests, the weight of the attached telescope mount was supported externally.

Although there was no instrumentation onboard to measure deployment assembly loads or deflections during activation, it is obvious that the deployment assembly structure performed satisfactorily since numerous photographs depict it and the telescope mount in excellent condition and there were no reported telescope mount alignment problems. Figure VD-3 is a photograph made by the first crew, which shows two of the rigidizing mechanisms fully retracted (rigidized position). The other two have been observed on other photographs.

2. Deployment Assembly Ordnance Release System. The telescope mount deployment assembly was required to provide additional truss members to stabilize the upper truss from overturning loads in the launch position. This was accomplished by pinning two stabilization struts of the upper deployment assembly to the lower deployment assembly truss with pyrotechnically actuated release latches (pin pullers) to release the upper deployment assembly after shroud separation prior to deployment. Each strut is pinned with two pin pullers either of which could release the strut (Figure VD-4). Pressure to operate the pin pullers was supplied by pressure cartridges threaded into the pin puller housing. The pressure cartridges were actuated by confined detonating fuse assemblies redundantly interconnected as shown in Figure VD-5. Each confined detonating fuse was initiated by exploding bridgewire detonators installed in confined detonating fuse manifolds. The detonators were initiated by exploding bridgewire firing units with one firing unit charged from airlock power bus #1 and the other charged from bus #2 to continue the redundancy. Both exploding bridgewire firing units received charge commands from the instrument unit/workshop switch selector and redundant trigger commands spaced 0.2 sec apart to complete the system redundancy. An alternate means of initiation was available using the airlock digital command system but was not needed.

Initiation of the pressure cartridges by confined detonating fuse lines and operating of the pin pullers with and without applied loads was demonstrated in development tests using a steel tube prototype deployment assembly unit with flight-type mechanisms. Successful release of the strut with only one pin retracting (simulating a failure of the second pin) was also achieved with the prototype. Vendor qualification tests on the pin puller and cartridge demonstrated satisfactory operation at high (160 °F) and low (-65 °F) temperatures under worst case side load after having been subjected to predicted flight vibration levels. Ground and flight verification of environmental compatibility and operational capability of the firing units, the exploding bridgewire

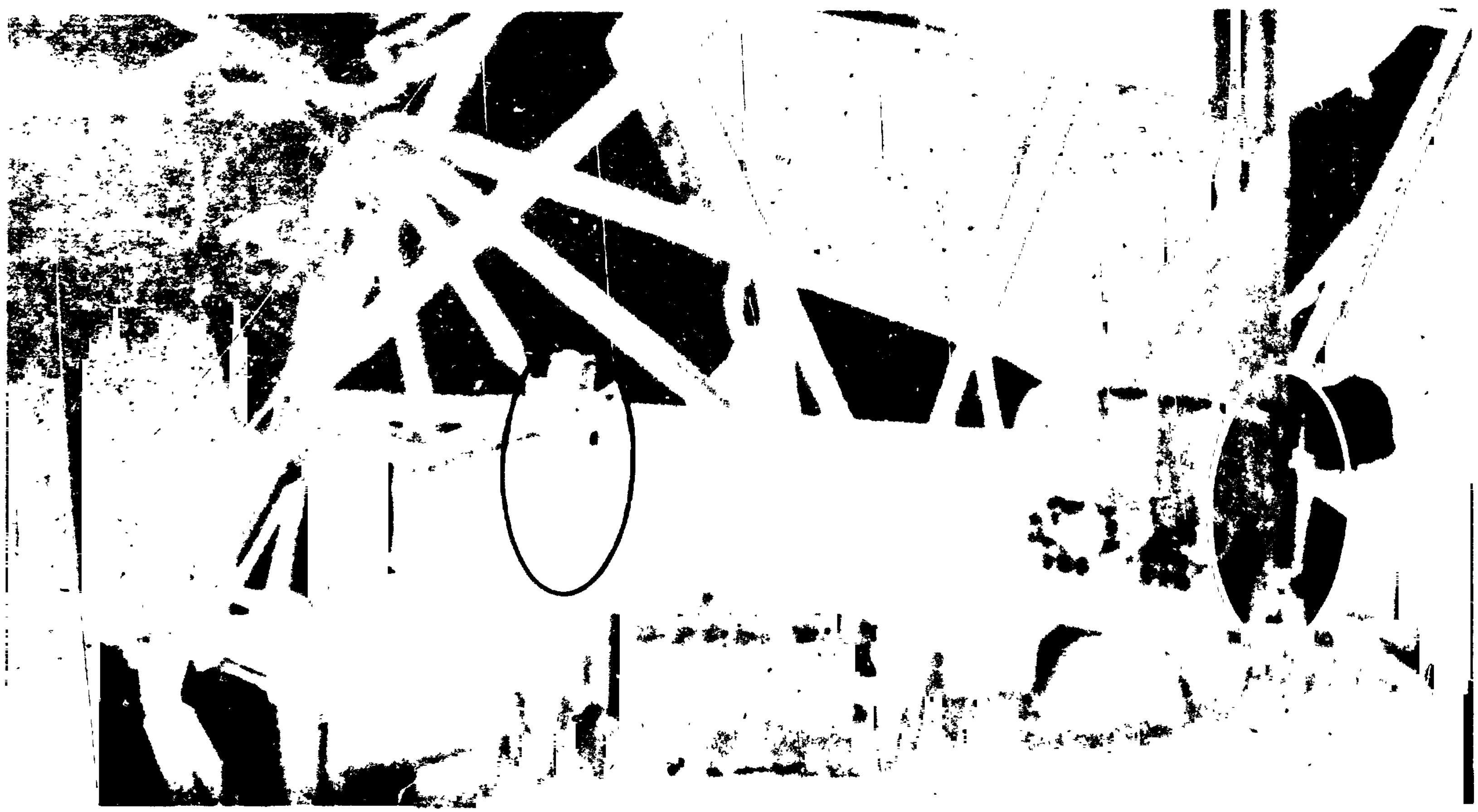


Figure VD 3 Rigidizing Mechanisms in Rigidized Position

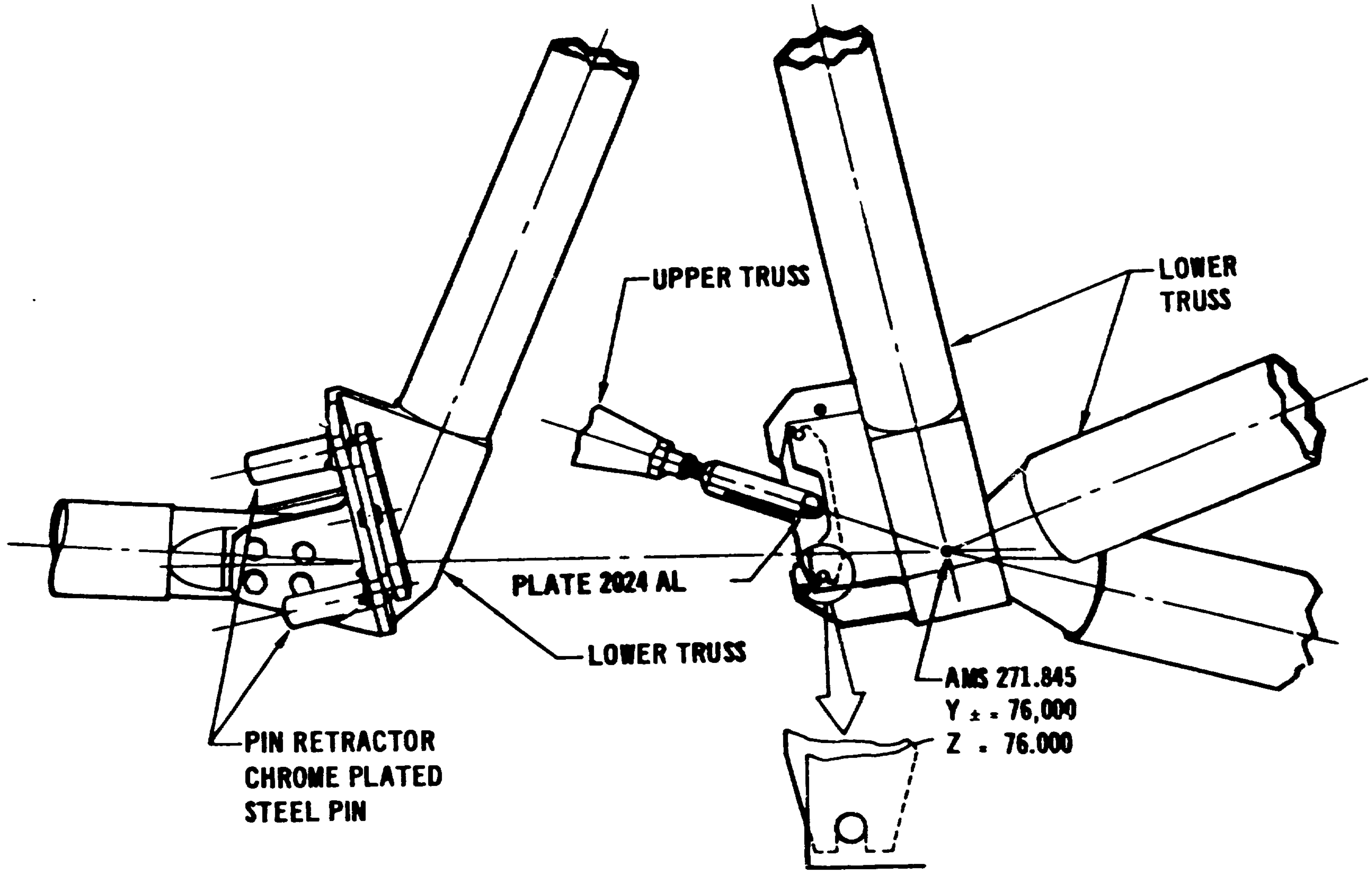


Figure VD-4. Deployment Assembly Release Mechanism

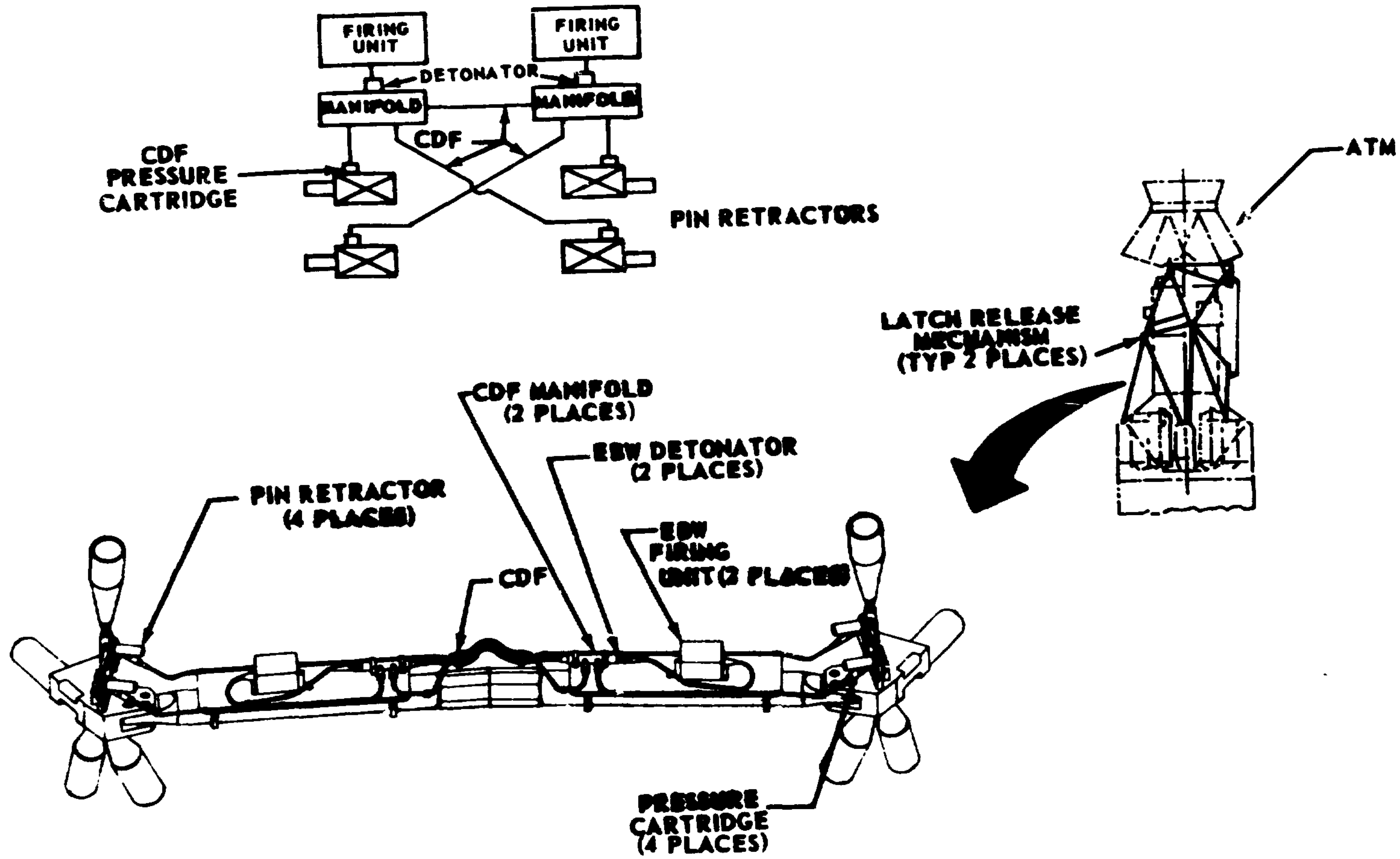


Figure VD-5. TM Deployment Release Mechanism detonators, and the confined detonating fuse had been previously conducted on the Saturn IB and Saturn V booster programs.

The airlock telescope mount deployment assembly truss release latch exploding bridgewire firing unit charge and trigger voltages were monitored during flight. Evaluation of the pin puller exploding bridgewire firing unit voltages indicated a normal charge and trigger of both the primary and secondary units. Compressed computer printout data show that both exploding bridgewire firing units were in the fully charged condition by 134:17:46:36.79. The trigger command was given at 134:17:46:37.1 and the storage capacitor in both firing units had discharged by 134:17:46:37.59. A plot of the pin puller exploding bridgewire firing unit data is shown in Figure VD-6.

3. Deployment System. The telescope mount deployment system was required to rotate the upper truss assembly 90 $\pm 1°$ from the launch axis within 10 min, using two completely redundant systems. Each system was to be capable of total deployment regardless of the point of failure of the other system. Upon reaching the deployed position, a latch mechanism was required to lock the upper and lower truss assemblies together to prevent any possibility of reverse rotation. To achieve these requirements, the deployment system was designed to coil two redundant cables onto four spools on two independent motor driven reel assemblies (Figure VD-7). Each reel assembly incorporated a rachet locking mechanism to prevent reverse rotation. One reel assembly was powered from airlock bus #1 and the other from bus #2 to continue the system redundancy. An electrical inhibit would have prevented voltage from being applied to the motors if the latch release firing unit charge and trigger commands had not been sent by the instrument unit/workshop switch selector. An overriding backup command was available from the airlock digital command system but was not used. Two redundant RW (reverse wind) negator springs mounted on the trunnion joints retarded rotation to provide damping and control of the rotation rate. The two trunnion joints that provided the deployment rotation axis are spherical monoball bearings mounted on the lower truss assembly (Figure VD-8). Bearing redundancy was achieved by ensuring that a frozen bearing would turn on the bolt or in the housing of the mount. A spring latch mechanism was used to retain the deployment assembly in the deployed position (Figure VD-9). As the deployment assembly approached the deployed position, the latch hook contacted the lower truss and was cammed into a retracted position against a spring force. After the nose of the hook passed over the tip of the lower truss cam, the spring forced the hook to rotate around the cam, latching the upper and lower truss assemblies together. Ratchet teeth on the inside of the hook engaged a notch in the cam surface, locking the hook in the latched position. Redundant switches on the latch mechanism initiated a time delay relay that turned off the deployment motors after they had continued to run for 16 sec to ensure that the upper truss and latch mechanism was pulled up tight. These same switches provided telemetry signals to indicate deployment assembly latch-up

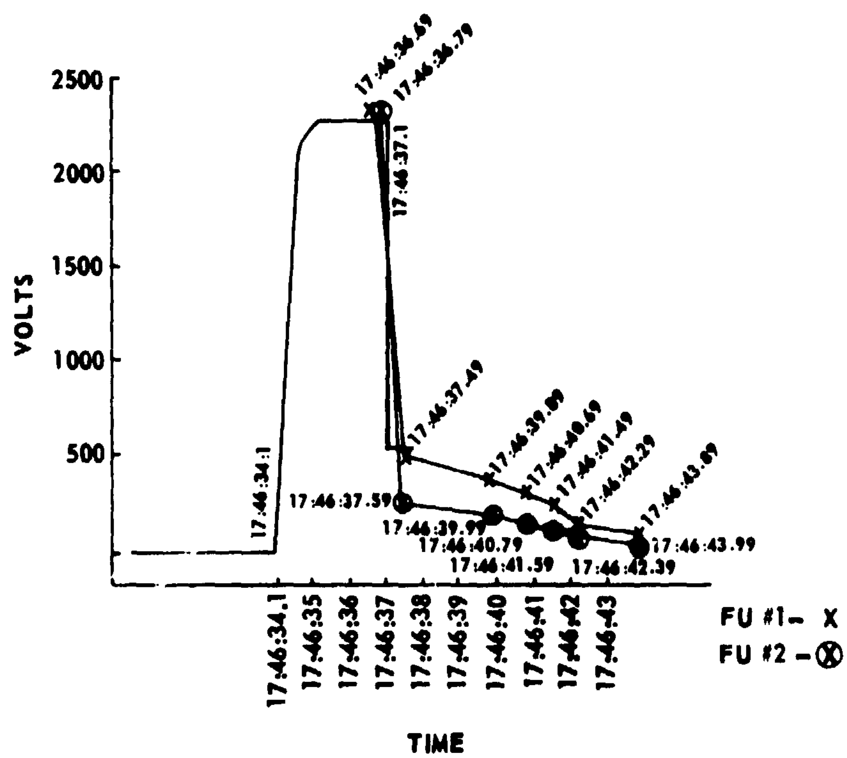


Figure VD-6. Telescope Mount/Deployment Assembly Truss Release EBW Firing Unit Voltage Data

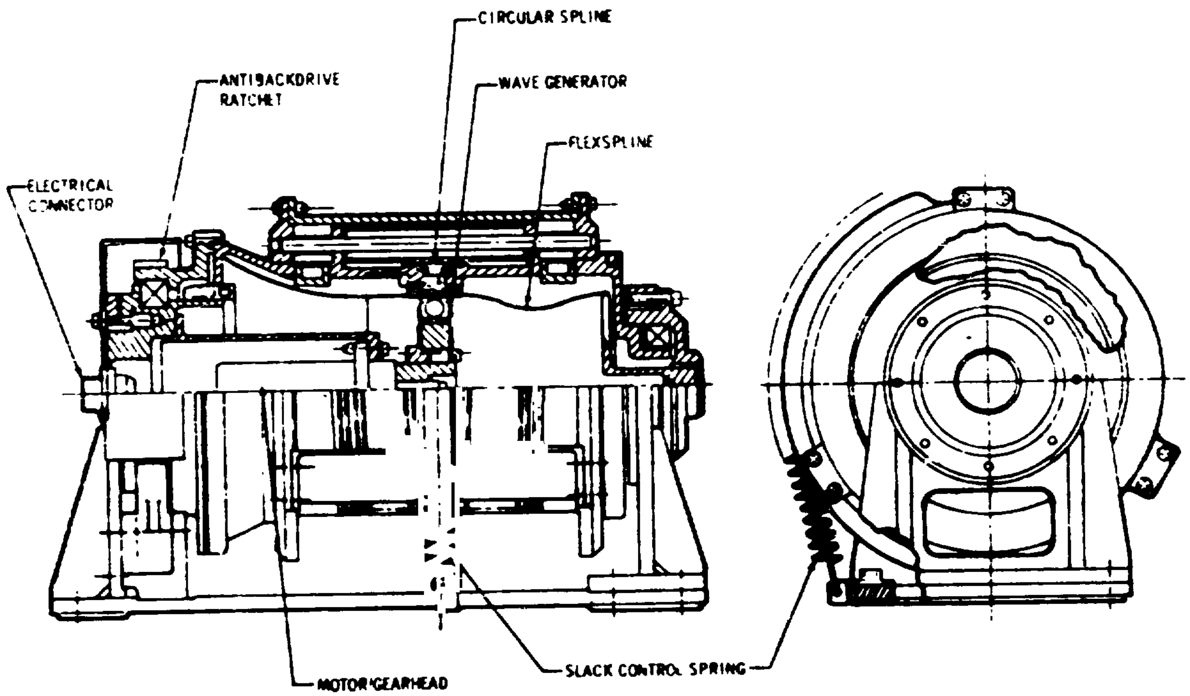


Figure VD-7 Deployment Reel Assembly

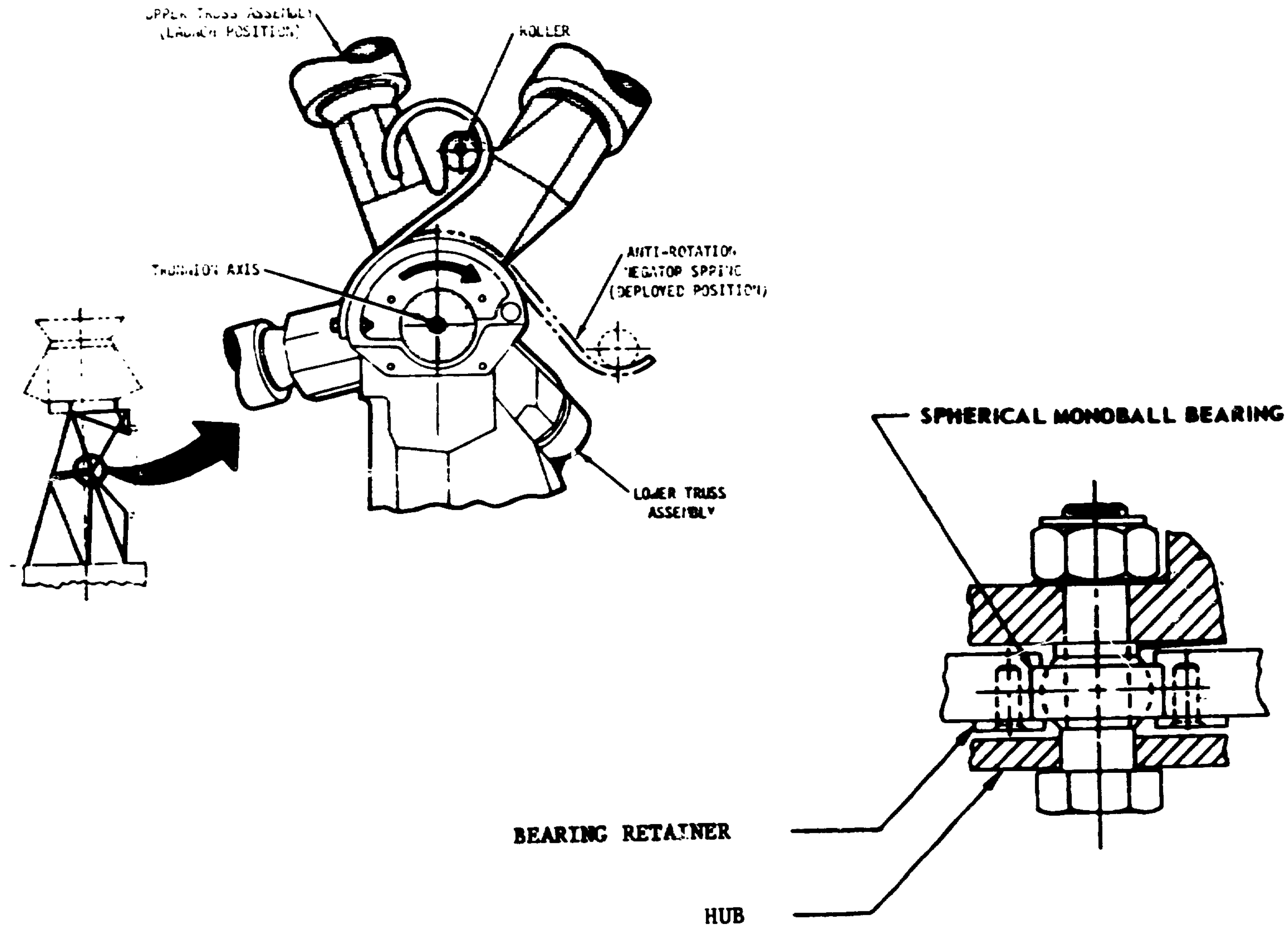


Figure VD-8, Trunnion Joints–Deployment Assembly

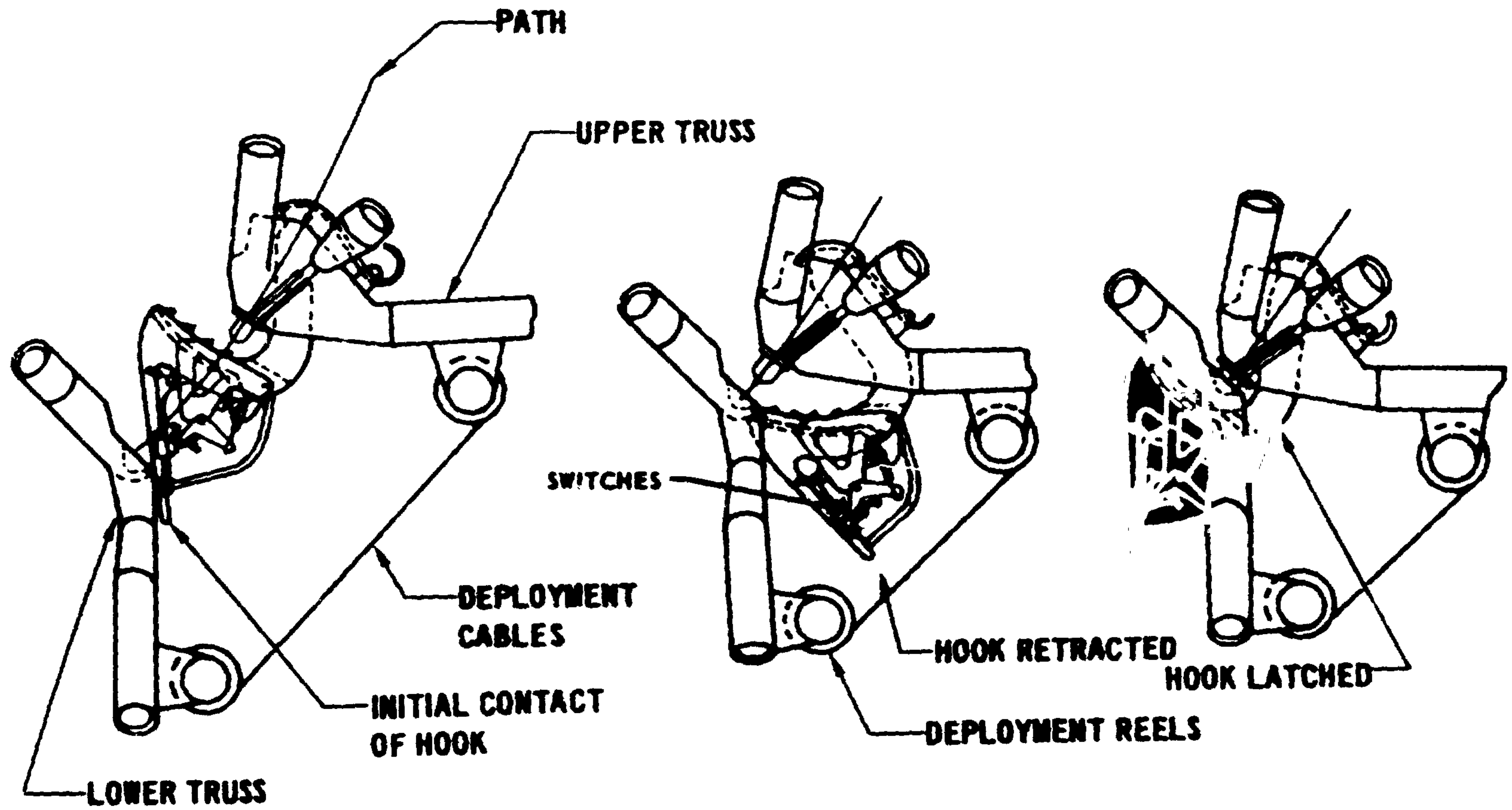


Figure VD-9. Telescope Mount Deployment System Latching Mechanism and also removed the inhibit from the telescope mount solar array deployment system.

The ground verification program on the deployment system was quite extensive. Developmental tests with prototype structure and flight-type mechanisms verified design criteria of trunnion spring and bearing torques, cable loads and reel assembly torque capability, latching mechanism geometry, and redundancy of deployment system components. Thirty-four complete deployment cycles were run with the assembly in a horizontal position. The mass of the upper truss assembly and telescope mount simulator was supported on a cable pulley arrangement with counterbalance to simulate zero g in the 1 g environment. Individual component qualification tests on the trunnion assembly, deployment reel assembly, and latching mechanism demonstrated the capability to operate satisfactorily at altitiude under maximum loads at high (160 °F) and low (-65 °F) temperatures after withstanding predicted flight vibration levels. A deployment assembly qualification test was run on the first production assembly, built for the vibroacoustics test, in the horizontal test setup using simulated telescope mount mass and geometry. All mechanisms performed correctly and the time to deploy with both reel assemblies operating was 3 min 35.1 sec, which was well within the time allowed. A frozen bearing test measured the torque required to rotate the bearing around the trunnion bolt (3100 in.-lb). This was well below 40,500 in.-lb min. stall torque of the cable reel assembly. A complete deployment assembly was included in the vibroacoustic test and tests were conducted with the deployment assembly in the launch position inside the shroud, and with the shroud removed and the deployment assembly in the deployed position. There were no anomalies with any of the deployment assembly equipment. After analyzing the vibroacoustics data, it was determined that the deployment reel assembly had been subjected to higher vibration levels than it had been qualified for. Consequently, that unit was subjected to a complete acceptance test to determine that no damage or performance degradation had occurred. The flight unit was assembled vertically on the flight fixed airlock shroud at the contractor's facility and then deployment functional tests were run with each deployment reel assembly, and with both reel assemblies operating. The mass of the upper truss and telescope mount simulator was counterbalance with a pulley and cable arrangement. Essentially the same test was repeated with the same setup at KSC prior to final stacking.

Real time strip charts recorded at MSFC showed that the deployment system performed flawlessly after launch and payload shroud separation. Telemetry discrete signals indicating that voltage had been applied to both deployment reel drive motors were received at 134:17:46:47.1 GMT, 10 sec after the deployment assembly release latch firing unit trigger command, providing further verification that firing unit commands had been sent. Both switches on the deployment assembly latch mechanism operated at 134:17:49:58.1, indicating that the upper and lower truss assemblies were latched together. At 134:17:50:12.7, the voltage was removed from the drive motors.

E. Payload Shroud

1. Basic Requirements and Configuration Selection. The Payload shroud (PS) provided an aerodynamic fairing for The Saturn Workshop (SWS), structural support for the telescope mount, an environmental shield with purge capability (to maintain positive internal pressure for protection of enclosed modules), and a noncontaminating separation and jettison system. From a variety of proposed configuration concepts, two were selected for detailed separation capability evaluation: over-the-nose and segmented. The over-the-nose concept was to be jettisoned axially using thrusters. The basic configuration for this concept is shown in figure VE-1. The segmented concept contained four 90° segments to be pyrotechnically severed and jettisoned laterally as shown in figure VE-2. Both configurations were determined to be technically feasible. The primary reason for selecting the segmented configuration was programmatic, based upon cost and schedule. Also, it had one potential advantage deserving mention for possible future application, but which was never used; separation of the PS during ascent if required for performance.

2. Payload Shroud Structures. The maximum weight allowable for the payload shroud was 26,024 lb; weight at lift-off was 25,473 lb. Its major structural assembly configuration is shown in Figure VE-3.

Skins on both nose cone sections were 0.250 in.-thick 2024-T351 aluminum sheets with internal rings formed from 7075-T6 aluminum. Cylinder skins were 2024-T351 aluminum, 0.375-in. thick on the lower one-third and 0.313-in. thick on the upper two-thirds. The cylinder frames were formed from 7075-T73 aluminum I-beam extrusions and were spaced approximately 23 in. apart.

Telescope mount loads were supported at 90° intervals on the forward end of the cylindrical section. A support link, secured between fittings by semi-cylindrical slots, provided for attachment of the telescope mount outrigger fittings through eccentric bushings with 0.125-in. radial adjustment. This installation is shown in Figure VE-4. During payload shroud jettison, movement of the support fittings at 45° to the axes of the shroud released the telescope mount. The support link was designed to remain attached to the telescope mount during subsequent orbital operations. Figure VE-5 verifies the design by showing the ring still attached during orbit.

a. Axial loads and bending moment. Except for a factor of safety of 1.25 in the rebound direction for the telescope mount attach bolts, the payload shroud was designed to a factor of safety of 3.0 to eliminate testing to ultimate load. The payload shroud was to support a telescope mount weighing 20,000 lb min. and 25,000 lb max., and was to be capable of withstanding flight, wind, handling, and separation system ultimate loading without failure. These requirements dictated that

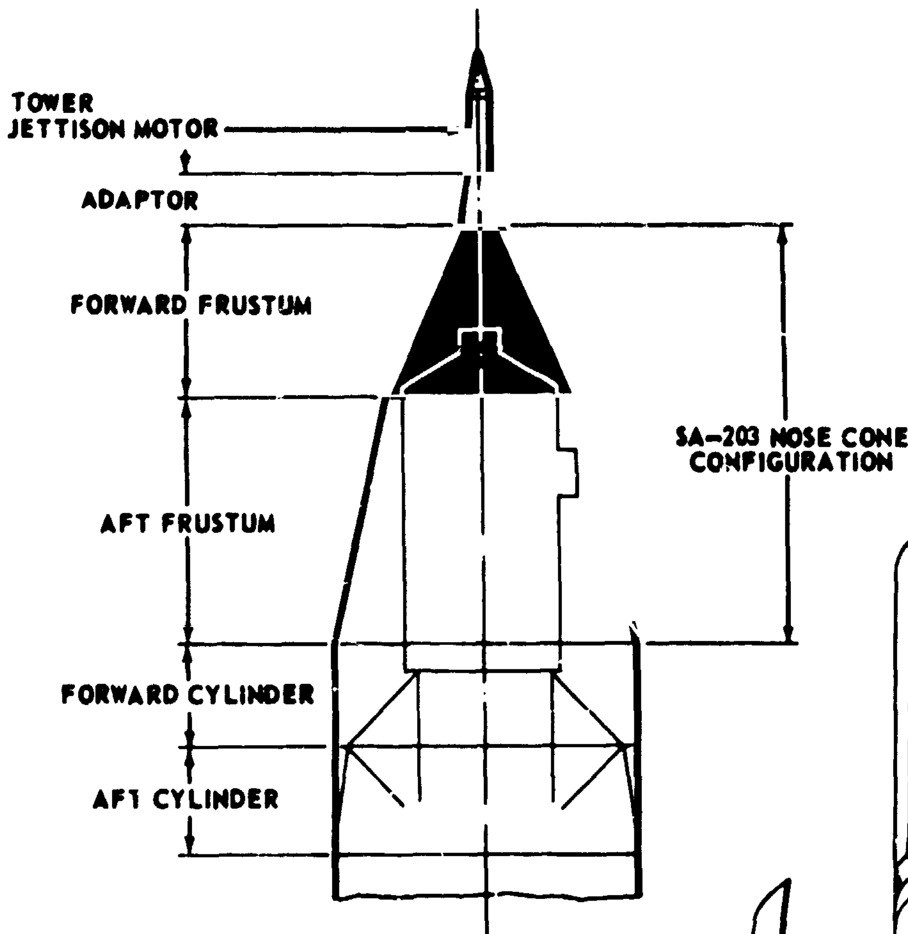


Figure VE-1. Over Nose Jettison

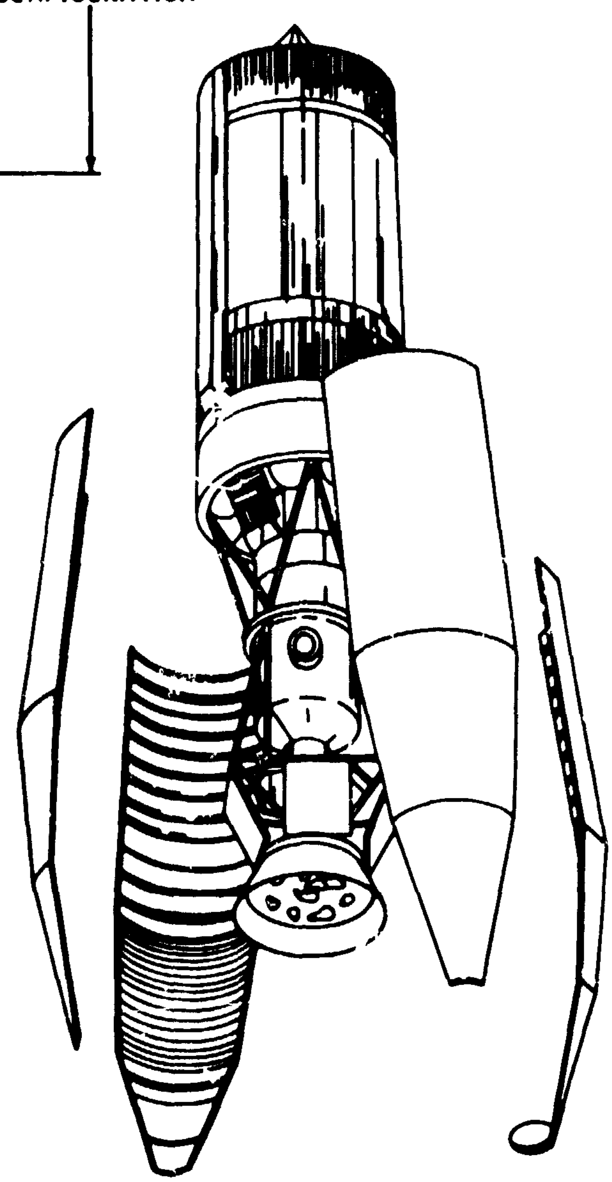


Figure VE-2. Split Shell Concept (Selected Configuration)

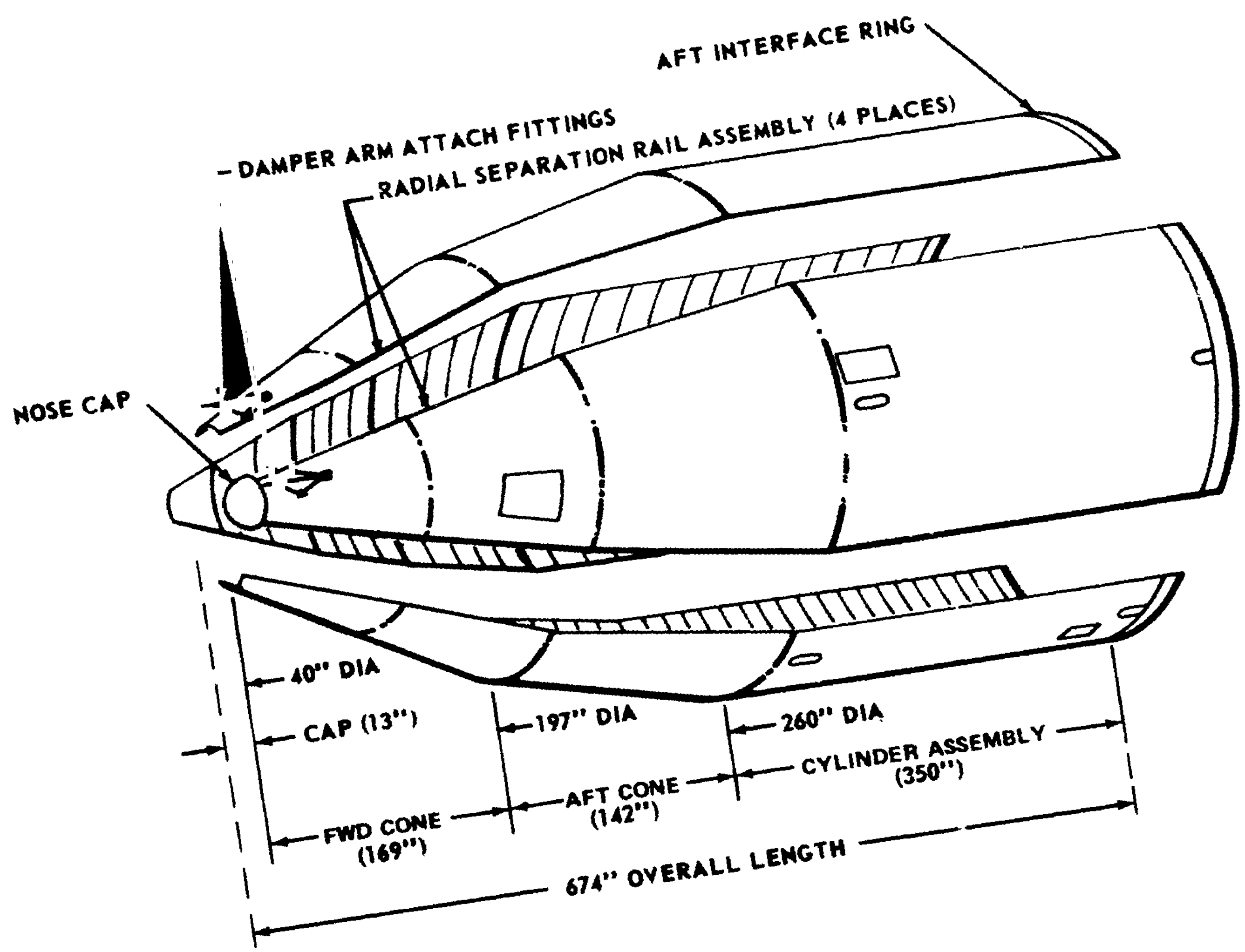


Figure VE-3, Payload Shroud Structural Arrangement

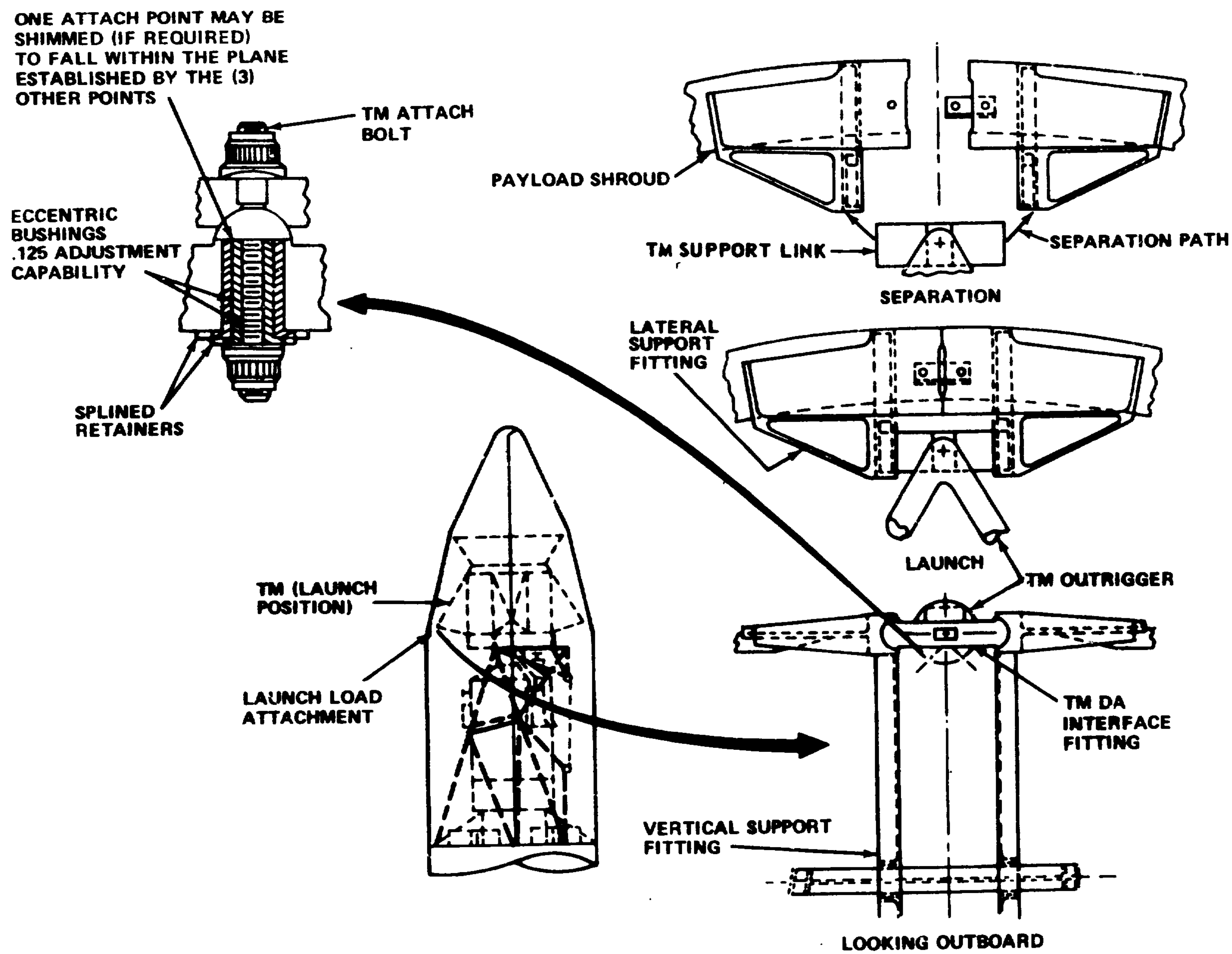


Figure VE-4. Hardware Attaching TM to Payload Shroud

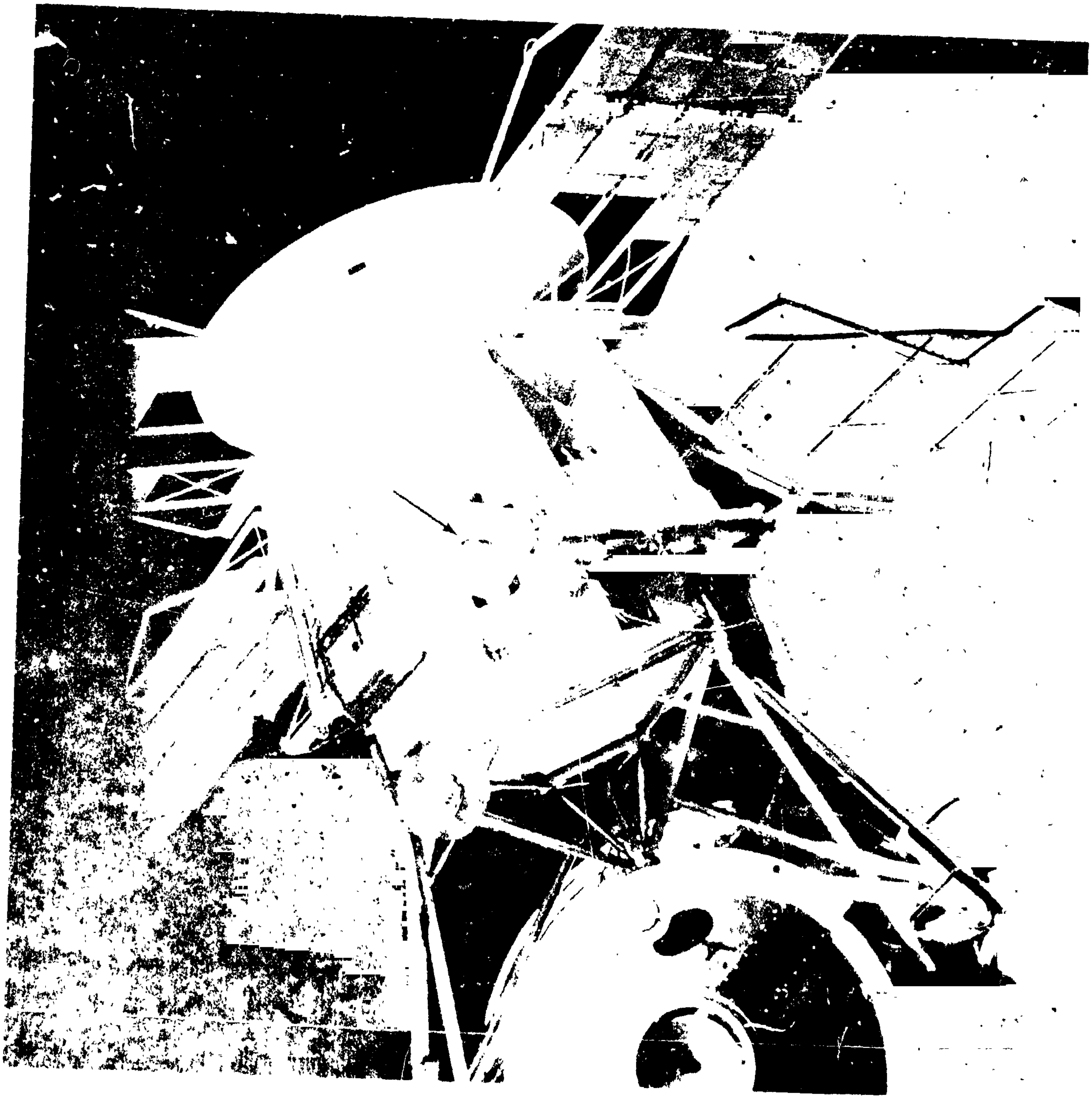


Figure VE-5. Telescope Mount Support Link structural integrity analysis be accomplished in lieu of testing. Structural analyses did verify the adequacy of the payload shroud to meet all stress combinations defined for the useful life expectancy, including the launch site. Stress measurements and calculations showed the realized factors of safety for the payload shroud to be 5.77 relative to bending moment, and 3.24 in respect to axial compression loads.

b. Acoustical data. Vibroacoustical testing of the payload shroud was accomplished at the JSC. Modified input sound pressure levels and acoustic criteria were used for the acoustical simulations in the enclosed test area at the JSC. Test criteria and measured test results for both the lift-off and boundary layer conditions are shown in Figure VE-6. Payload shroud attenuation was adequate under the test conditions and provided better attenuation than the specification required during flight (Figure VE-7).

3. Natural Environments Design. Natural environments such as atmospheric temperature, humidity, particulate matter, rain, ground winds, in-flight winds, radiation and meteoroids were considered in the design of the payload shroud. On the ground, the payload shroud protected the internal equipment from contamination by dust, rain, and wind. As shown in Figure VE-8, the payload shroud contained a purge duct that interfaced to the purge duct in the fixed airlock shroud. The combined ducts provided class 100 air to a diffuser located 13 in. below the vehicle tip. The diffused air maintained temperature, humidity, and cleanliness for the entire payload shroud. Operation with conditioned air began with the stacking of the telescope mount and continued until 30 min prior to launch vehicle cyrogenic loading, when a nitrogen purge was initiated through the ducts. The purge duct system was designed to maintain a flow rate of 50 $\pm$10 lb/min at a temperature of 63 $\pm$5 °F with maximum duct internal pressure of 1.5 psid. Evidence of hardware compliance with the design is reflected in two Kennedy Space Center measurements: (1) measurement CRT FR-09 shows the flow rates measured at the facility outlet to be 66 to 68 lb/min during countdown demonstration test and 68 to 70 lb/min during launch (2) measurement 2C139, recorded on the umbilical arm, shows the pressure to be 1.2 to 1.3 psig during test and 1.3 psig at launch. The Kennedy Space Center Skylab 1 Post-Launch Report, RCS76-0000-00048, states that all specifications for flow rates, temperatures, and pressures were met, and that flow/pressure was satisfactory during the air to GN_2 changeover. There were no problems.

Protection to prevent wind-driven or falling rain and runoff from penetrating the interior of the payload shroud, while in a fully assembled condition, at the KSC launch site was a requirement. During a rain storm at KSC the payload shroud leaked, apparently in the nose cap to nose cone interface area. However, pinpointing the leak source was not definite.

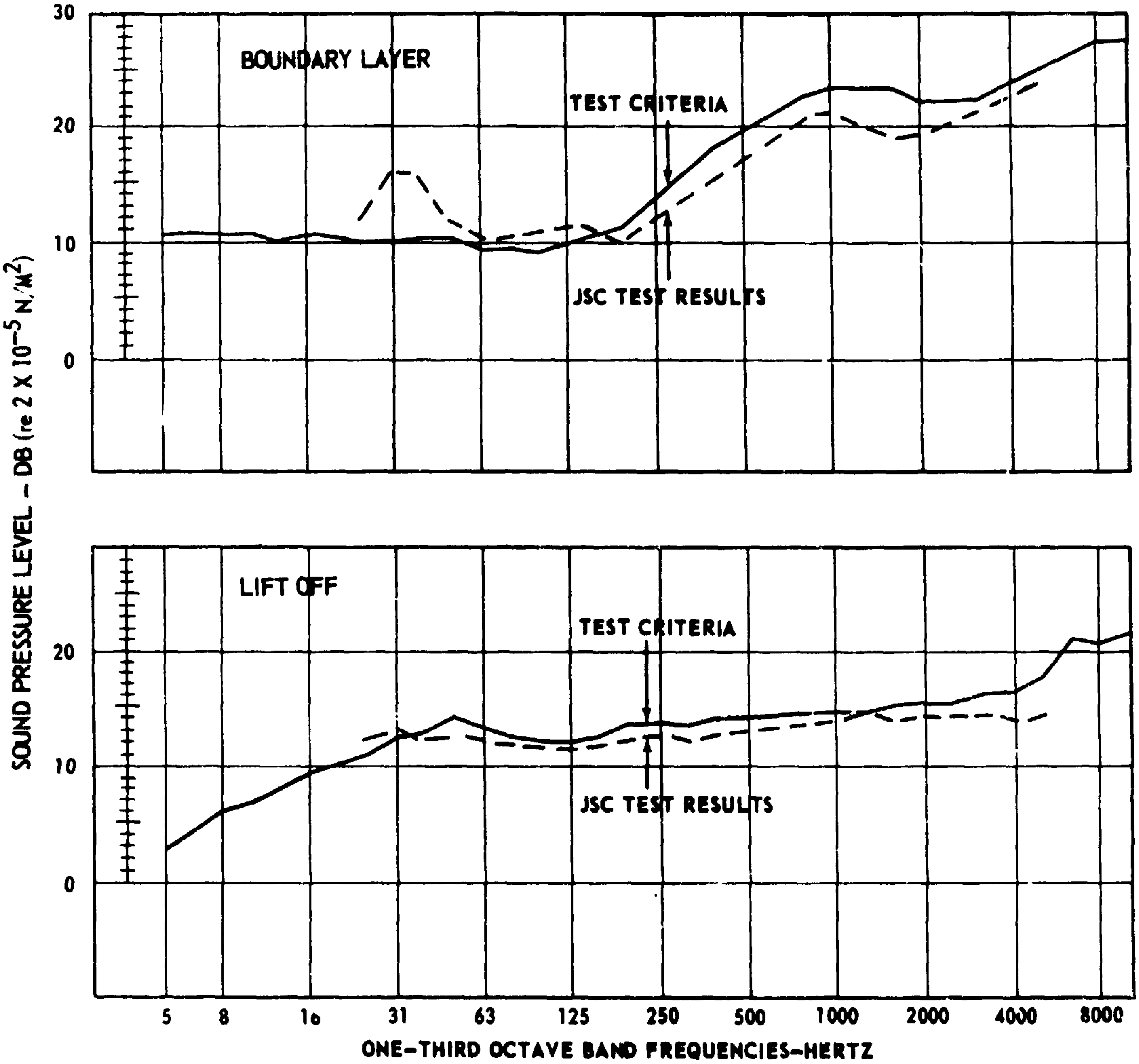
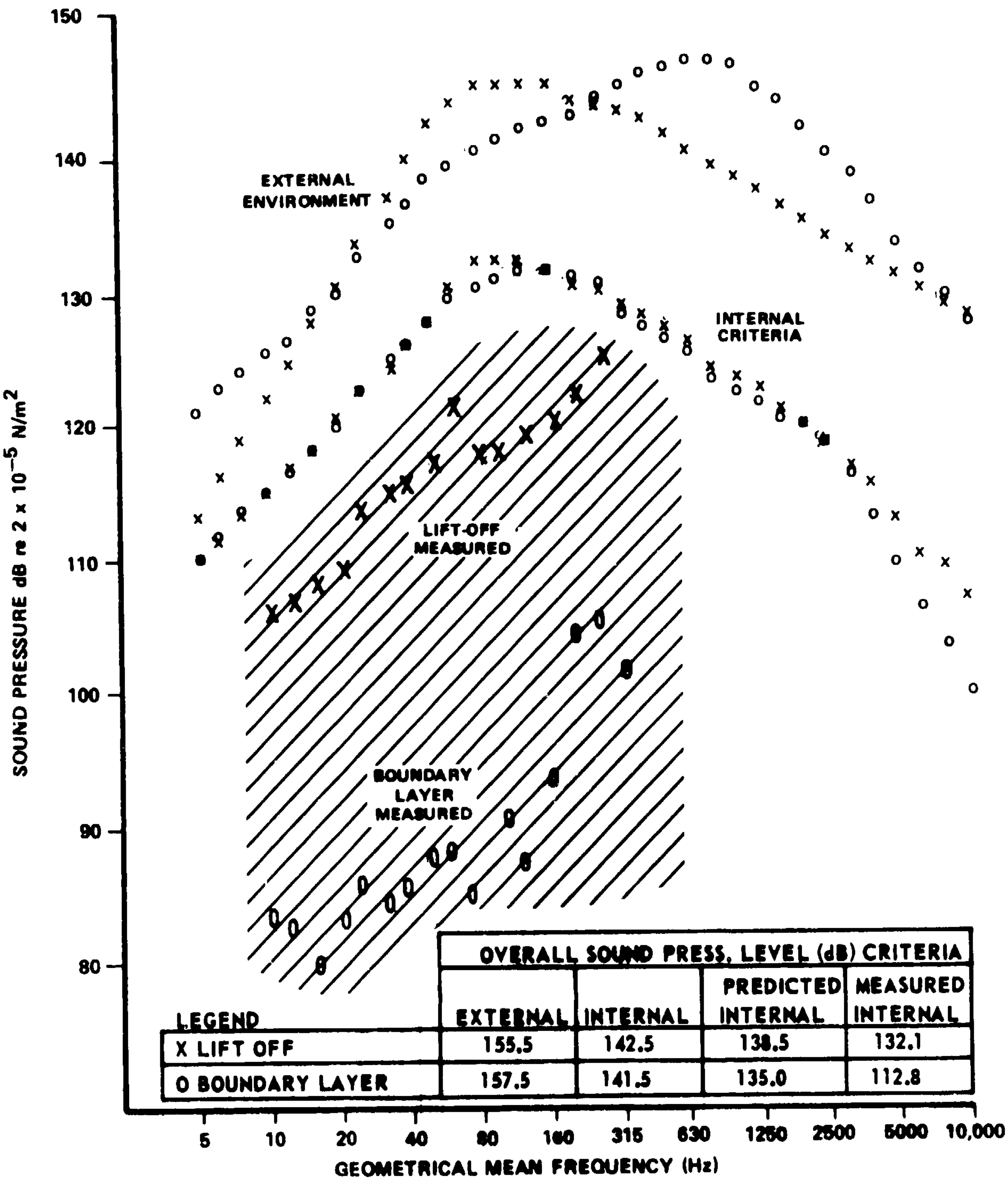


Figure VE-6. Payload Shroud Noise Reduction



LEGEND	OVERALL SOUND PRESS. LEVEL (dB) CRITERIA			
	EXTERNAL	INTERNAL	PREDICTED INTERNAL	MEASURED INTERNAL
X LIFT OFF	155.5	142.5	138.5	132.1
O BOUNDARY LAYER	157.5	141.5	135.0	112.8

Figure VE-7. Payload Shroud Acoustical Data

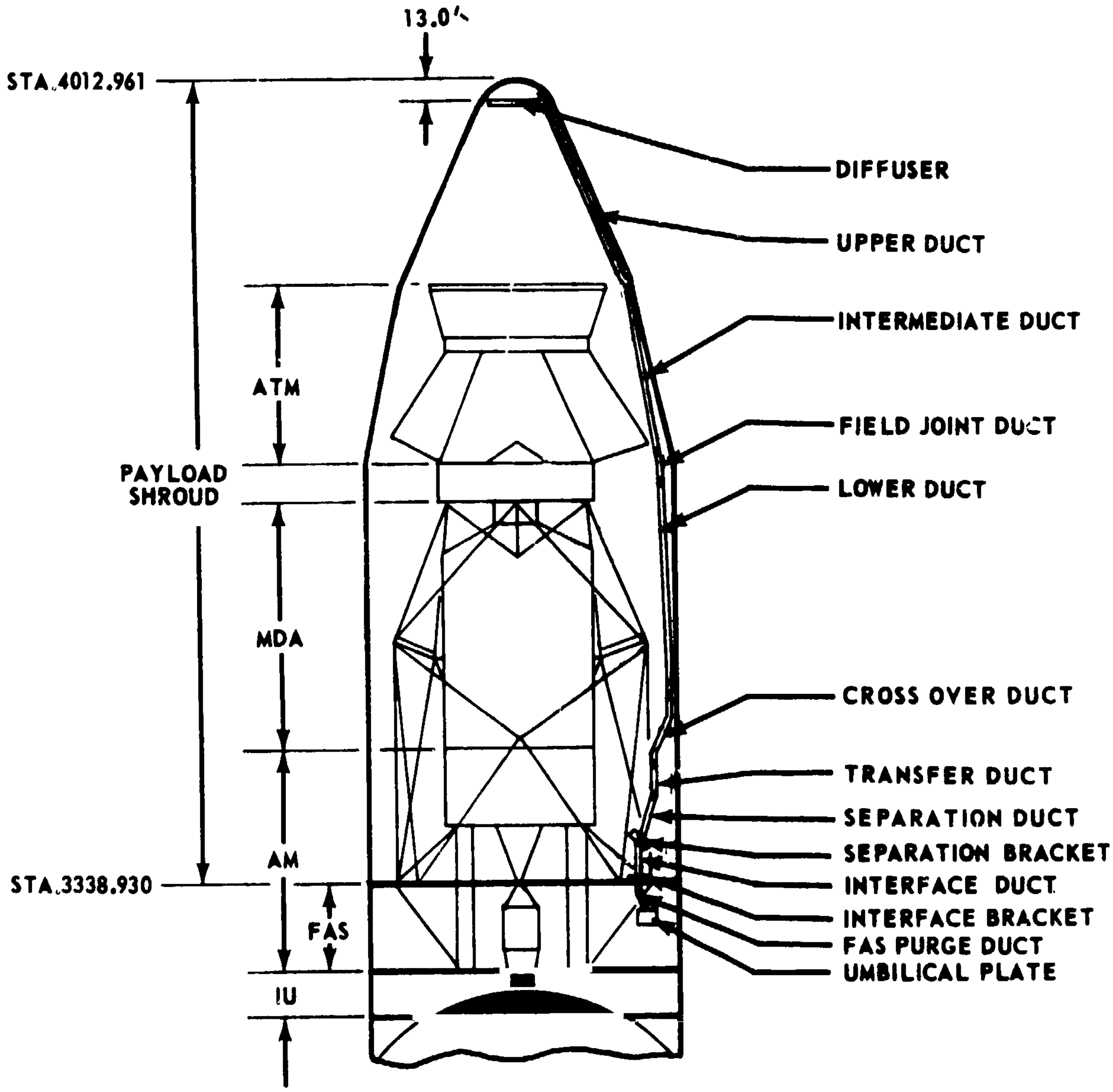


Figure VE- 8. Payload Shroud/Fixed Airlock Shroud Purge Duct Routing investigation indicated no sealant around the cap to cone interface joint and the application of sealant was not recorded in the inspection records, although it was an engineering drawing requirement. Engineering drawings, however, did not specify sealant application around the cap to cone fittings. The proposed fix, shown in Figure VE-9 and implemented at the launch pad, applied STM598-02 sealant around the cap to cone interface joint as well as the fittings. The payload shroud continued to leak, but at a reduced rate, during subsequent tests and another rain storm. However, since this leakage was considered to be of insufficiant magnitude to require rework, no further corrective action was taken.

Payload shroud leaks may have been prevented by completeness and clarity of engineering drawings and quality control attention to detail. However, it was pointed out that there was potential leakage inherent in the payload shroud design because movement between the nose cap and conical sections could not be eliminated because of attachment of the damper arm to the payload shroud. Future complex designs/configurations with rigorous requirements should not attempt to provide 100 percent rain protection, unless complete and conclusive rain tests are required under all simulated conditions anticipated.

4. Separation System. The payload shroud separation system incorporated two noncontaminating longitudinal thrusting joint assemblies, and eight discrete latches, four at the base and four at the upper ring of the payload shroud cylinder. Eight backup or redundant discrete latches were also in the system. At separation the thrusting joints imparted a radial velocity to each payload shroud quarter segment so that the jettisoned shroud would not interfere with the functioning of the orbital assembly. Each shroud segment further incorporated lanyard operated electrical disconnect assemblies. Continuity through the disconnects was monitored by airlock telemetry to record actual shroud separation. To reduce the probability of recontact of the shroud segments with the orbital assembly, procedures were established whereby the vehicle was maneuvered to a nose down gravity gradient position, and oriented so that the four shroud segments could be jettisoned in a plane not parallel to the orbital plane.

a. Discrete latches (latch actuator). The discrete latch system provided continuity of payload shroud ring strength across the separation planes by providing a path for concentrated loads at the lower ring and at the cone/cylinder junction ring with a tie link and clevis arrangement. Additionally, it provided a means of releasing the four payload shroud quad sections (Figure VE-10). The tie link was released from the clevises when either of the two latch actuator pins were pulled from the link by the pyrotechnically actuated latch actuators.

The four latch actuators in each quadrant were actuated by a linear explosive assembly contained within closed manifold tubing. The

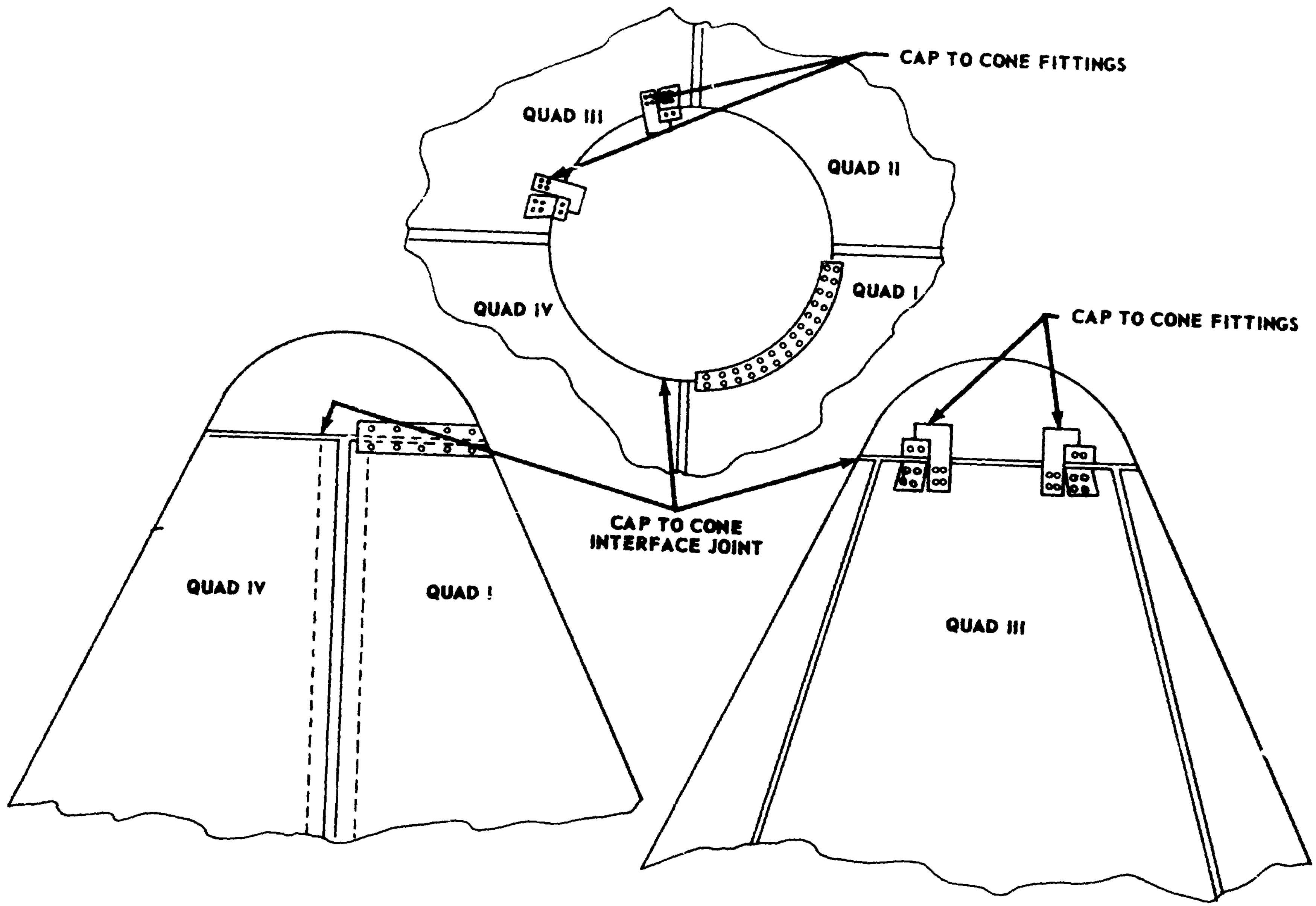


Figure VI-9. Payload Shroud Nose Cap Installation

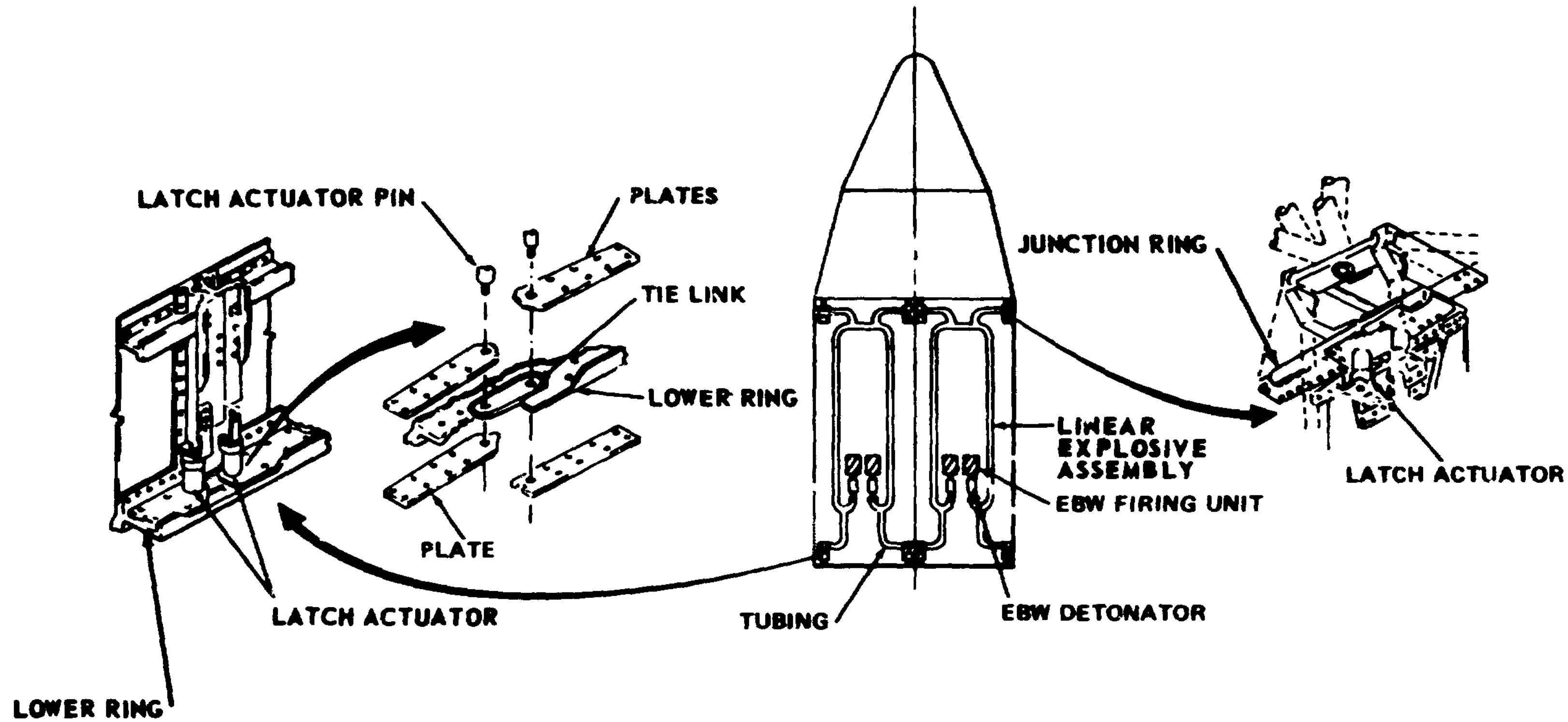


Figure VE-10. Payload Shroud Discrete Latch linear explosive assembly was detonated by exploding bridgewire detonators at each end of the manifold to provide redundancy. The detonators were initiated by exploding bridgewire firing units, with one firing unit for each assembly, powered from airlock electrical bus #1 and the other from bus #2. Each exploding bridgewire firing unit was programmed to receive redundant charge and fire commands, spaced 0.2 sec apart, from the instrument unit switch selector. An alternate means of initiation was available from the airlock digital command system, but was not needed.

Operation of the discrete latch system was demonstrated through numerous ground tests. The exploding bridgewire detonator confinement and propagation tests were conducted on 26 test specimens, representative of flight hardware, to demonstrate proper confinement of the explosion products and the propagation of detonation under conditions more adverse that encountered in-flight. Verification tests at MDAC-E on four full-scale flight-type latch actuators, representing one-fourth of the vehicle system, demonstrated operation of the latch actuators when exposed to conditions of temperature (-20 to 160 °F), loading (0 to 333 percent of nominal side load), and explosive charge (50 to 150 percent of nominal) after exposure to anticipated flight vibration levels. In three full-scale altitude chamber separation system tests run at Plum Brook, Ohio, the latch actuators (16) also operated within the conditions stated above. One quadrant (quad IV) was then equipped with a full complement of ordnance components to be fired subsequent to the vibroacoustics test to be run at the JSC. It was successfully fired (one detonator only) after the conclusion of the vibration and acoustic tests. The other detonator, not connected to the power circuit, was sympathetically fired by the linear explosive.

The payload shroud discrete latch exploding bridgewire firing unit charge and trigger voltages were monitored during flight to verify sequences. The programmed sequence was: (1) charge command given, (2) 5.02 sec later the trigger or fire command given, and (3) 5 sec later unit reset. Compressed computer printout data of these measurements indicate that all eight of the latch firing units were in a charged condition by 134:17:43:14.01 and that all eight units had triggered by 134:17:43:14:81. While it has not been possible to reconcile programmed charge and trigger times with the times displayed for actual voltage points in the computer printout, a plot of the firing unit charge, trigger, and reset voltage is normal when compared with plots of the ground tests for like firing units. A typical plot of the firing unit charge and trigger voltage, constructed from a real time strip chart recording with times established by computer printout data, is shown in Figure VE-11.

b. Thrusting joint. The two noncontaminating thrusting joint assemblies, located in the longitudinal separation planes between the payload shroud quadrants, consisted of a linear cylinder and piston riveted together to form a rail assembly. These joints provided the force

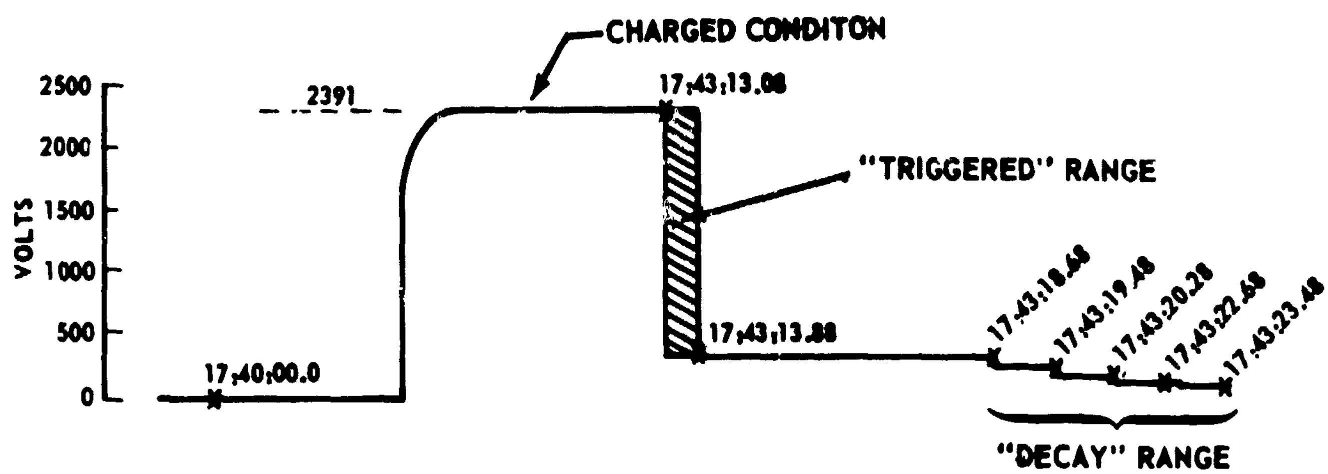


Figure VE-11. Typical Payload Shroud Pin Puller Firing Unit Data necessary to separate the payload shroud quad sections at the required trajectory (45° to orbital plane). The cylinder was attached to one quadrant and the piston was attached to the adjacent quadrant. Inside the cylinder rail was a collapsible tubular bellows with a multiple orificed attenuator tube containing a linear explosive assembly. Gases generated by the detonation of the linear explosive assembly pressurized the bellows and forced the piston and cylinder rails apart, shearing the rivets and imparting a velocity to the shroud quadrants (Figure VE-12). Two opposite shroud segments, quads II and IV, contained the linear explosive assemblies. The linear explosive assembly was detonated by exploding bridgewire detonators, one located at each end of each linear explosive assembly, thus providing redundancy. The detonators were initiated by exploding bridgewire firing units with the firing unit at one end of the explosive train powered from airlock electrical bus #1 and the unit at the other end powered from bus #2. Each exploding bridgewire firing unit was programmed to receive redundant charge and fire commands, spaced 0.2 sec apart, from the instrument unit switch selector. An alternate means of initiation was available from the airlock digital command system, but was not used.

Operation of the thrusting joint system was demonstrated by numerous ground tests which showed that a nominal explosive charge had the ability to shear twice the number of rivets contained in the flight hardware. Other tests demonstrated the ability of the system to contain 129 percent of the nominal explosive charge without release of the contaminating explosive products.

Three full scale separation systems development/verification tests, denoted as Plum Brook (PB) firings #1, #2, and #3, were conducted in the Plum Brook Space Power Facility vacuum chamber, Sandusky, Ohio. The shroud separated in all three tests. All testing was performed with the same payload shroud; between firings it was refurbished by replacement of functional parts and the incorporation of certain design changes. Inspection after PB #1 separation test revealed three slits in the tubular bellows, several dislodged sheared rivets, a fractured ring frame in the forward cone, and a fractured cone/cylinder attach angle. These problems were corrected through redesign and rework. Test PB #2 was conducted and separation was again accomplished. Inspection revealed that the four transfer tubes containing the linear explosive assemblies had ruptured. Failure investigation indicated the problem was due to scoring of the interior of the tubes caused by interacting detonation shock waves of adjacent strands of explosive, gas pressure then ruptured the tubes along the scored surface. This was corrected by redesign and rework incorporating a polyethylene liner inside the tubes and using thicker walled tubes. The third separation test (PB #3) was accomplished with no anamolies. At this point, the separation systems test was considered successfully concluded. Quadrant IV was then refurbished with new thrusting joint ordnance equipment for the vibroacoustic test at JSC.

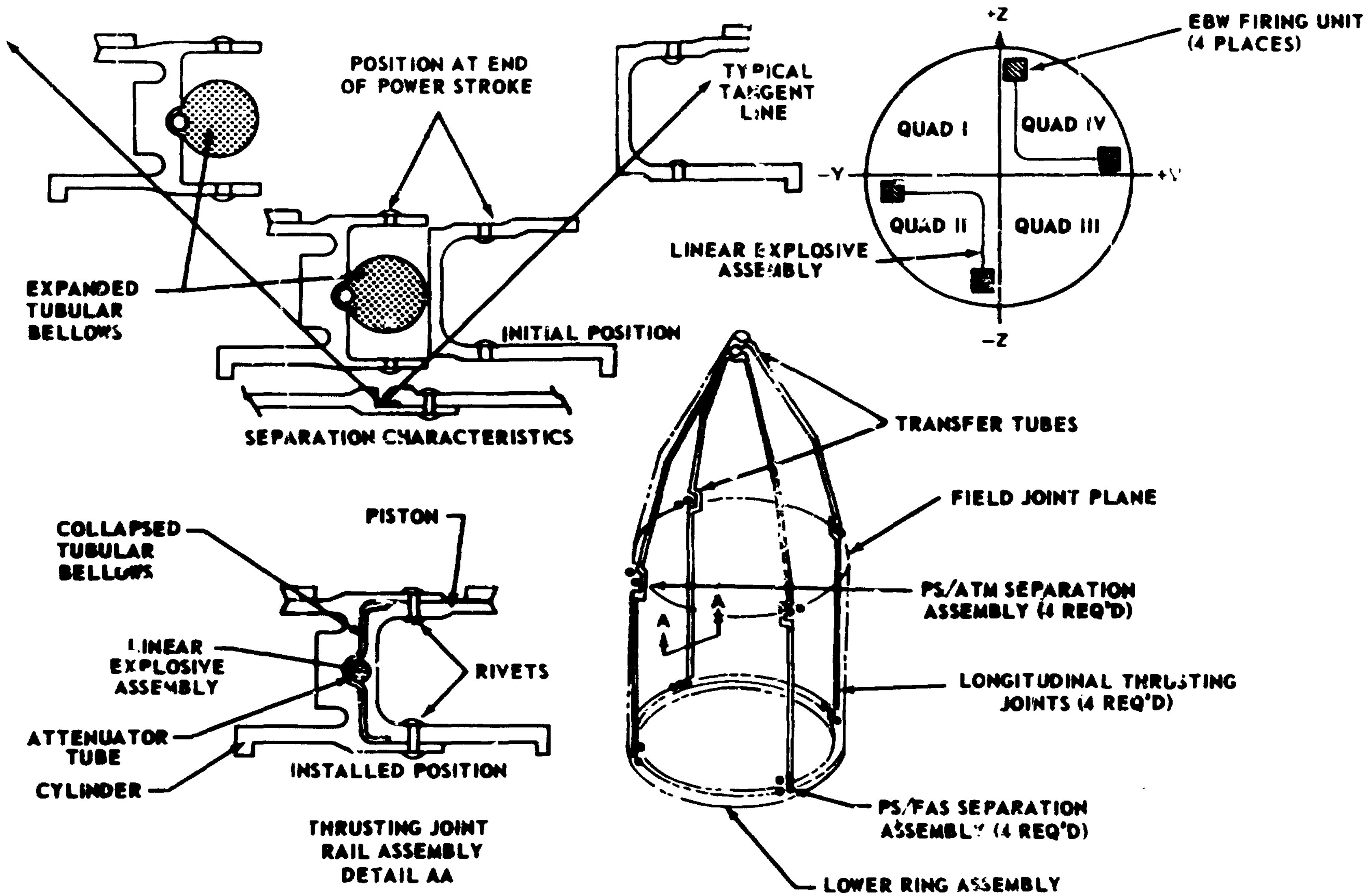


Figure VI-12. Payload Shroud Thrusting Joint

No problems were encountered during or after this test. The thrusting linear explosive assembly was removed from the shroud after the vibroacoustic test and subjected to detonation velocity tests. The detonation velocity was above the 18,000 ft/sec minimum allowed, indicating that the explosive assembly was not affected by the vibration or the acoustic environments. A rail assembly for the vibroacoustic test shroud was refurbished at Plum Brook with new equipment and subjected to a 10-month aging/vibration test to demonstrate that the thrusting joint system was not affected by shelf life/storage and exposure during vibroacoustic testing. The aging portion of the test was terminated at 10 months and the test was considered successfully concluded.

One major anomaly occurred in the payload shroud linear explosive assembly that was discovered during a KSC inspection prior to launch. The payload shroud separation system used long (97 and 122 ft) linear explosive assemblies (Primaline) to develop the pressure for operation of the discrete latches and for inflation of the bellows in the thrusting joint. During the KSC inspection it was discovered that shrinkage of the Primaline had occurred with an average shortening of 3.9 in. per 100 ft over a 3-month period. This was 1.4 in. less than the minimum installable length, therefore, a serious installation problem was created. However, since MSFC had no previous component (Primaline) experience and shrinkage had not been a problem up to this point, it was not a design consideration. An investigation revealed that the shrinkage was induced by both time and temperature and that it was accelerated by high ambient temperatures (e.g., 120 °F transport and storage). To correct the problem a new Primaline assembly was fabricated and additional design requirements imposed to control shrinkage, which included thermal conditioning of all new Primaline at 125 $\pm$5 °F for 2 wk to preshrink it. Also, an 80 °F maximum temperature during manufacturing and 50 °F maximum temperature during shipment and storage were imposed. Observance of these constraints resolved the shrinkage problem; shrinkage was estimated to be approximately 0.5 in. per 100 over a 3-month period. This was well within the 2.5 in. required for a minimum installable length.

Evaluation of the flight payload shroud thrusting joint exploding bridgewire firing unit voltages and command times indicated a normal function of all four firing units in this system, as compared with plots of ground test firings of like units. This is further confirmed by the time indicated for lanyard disconnect. Note that at the time of disconnect the telemetry and power circuits were opened and reset of the firing unit was not possible. A typical payload shroud thrusting joint charge and trigger plot is shown in Figure VE-13. The time and data points for this plot were taken from compressed computer printout data. This operation was also verified by real time strip charts recorded at MSFC.

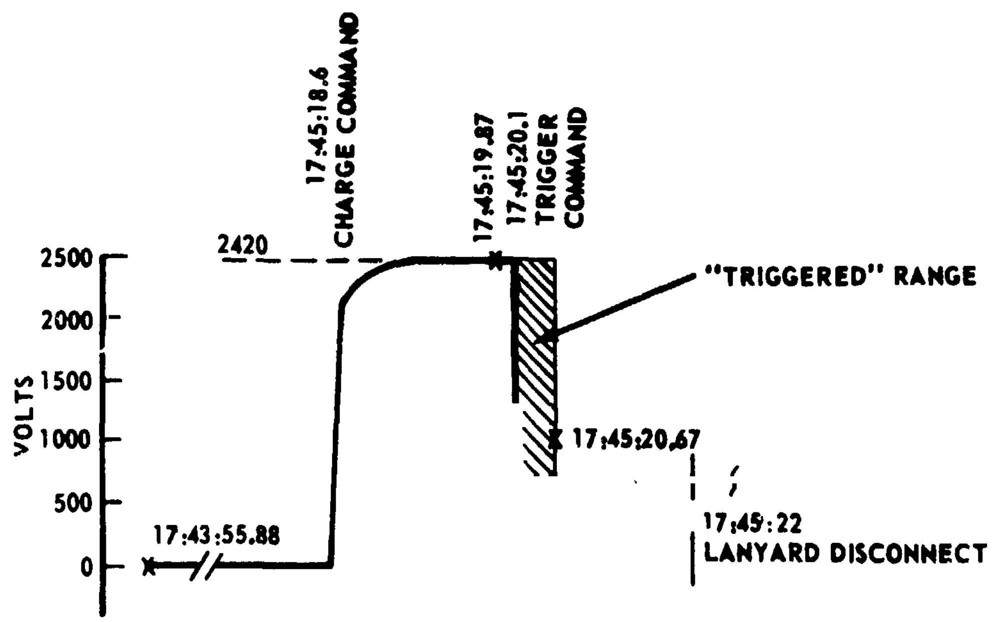


Figure VE–13. Typical Payload Shroud Thrusting Joint EBW Firing Unit

MSFC has identified four objects from the NORAD tracking data to be the four payload shroud segments. Using orbital mechanics to extrapolate backwards from the NORAD reported orbits, the direction and minimum velocity at which the objects departed Skylab has been determined to be (Table VE-1):

TABLE VE-1. SEPARATION DATA

Satellite no.	Minimum in Orbital Plane	Separation
6637	14 ft/sec Retrograde	19.8 ft/sec
6638	12 ft/sec Posigrade	17.0 ft/sec
6643	11 ft/sec Retrograde	15.6 ft/sec
6651	13 ft/sec Posigrade	18.4 ft/sec

Separation values in Table VE-1 compare very favorably with results of the three separation tests conducted at Plum Brook, Ohio, where measured segment velocities varied from 16 to 19 ft/sec, thus indicating the shroud jettisoning was nominal.

c. Contamination. Verification that products of detonation (explosion) were contained by the flight unit was not possible since the shroud disconnected from the orbital assembly upon separation. However, ground tests did demonstrate this capability on the test units, and no indication of major separation contamination was reported. Data from the Quartz Crystal Microbalancer (contamination measurements) did indicate a major reduction of contamination on the crystal surfaces immediately upon achieving orbit (Figure VE-14). Thus, it can be concluded that no detectable contaminants were released by the payload separation system.

5. Conclusions and Lessons Learned. Specifications and criteria governing payload shroud design and performance were adequate. Payload shroud performance other than leakage on the launch pad because of rain was satisfactory. Future designs of this type should ensure that shroud/facility interfaces preclude movement at the attaching joint to prevent internal contamination by rain or particulate matter. Sealants and their application should be more closely monitored, and verified prior to exposure of hardware to uncontrollable environments. Primaline shrinkage, as a factor of temperatures and time (reference paragraph 4b, this section) is an important lesson to be applied to future designs incorporating long lengths of this material.

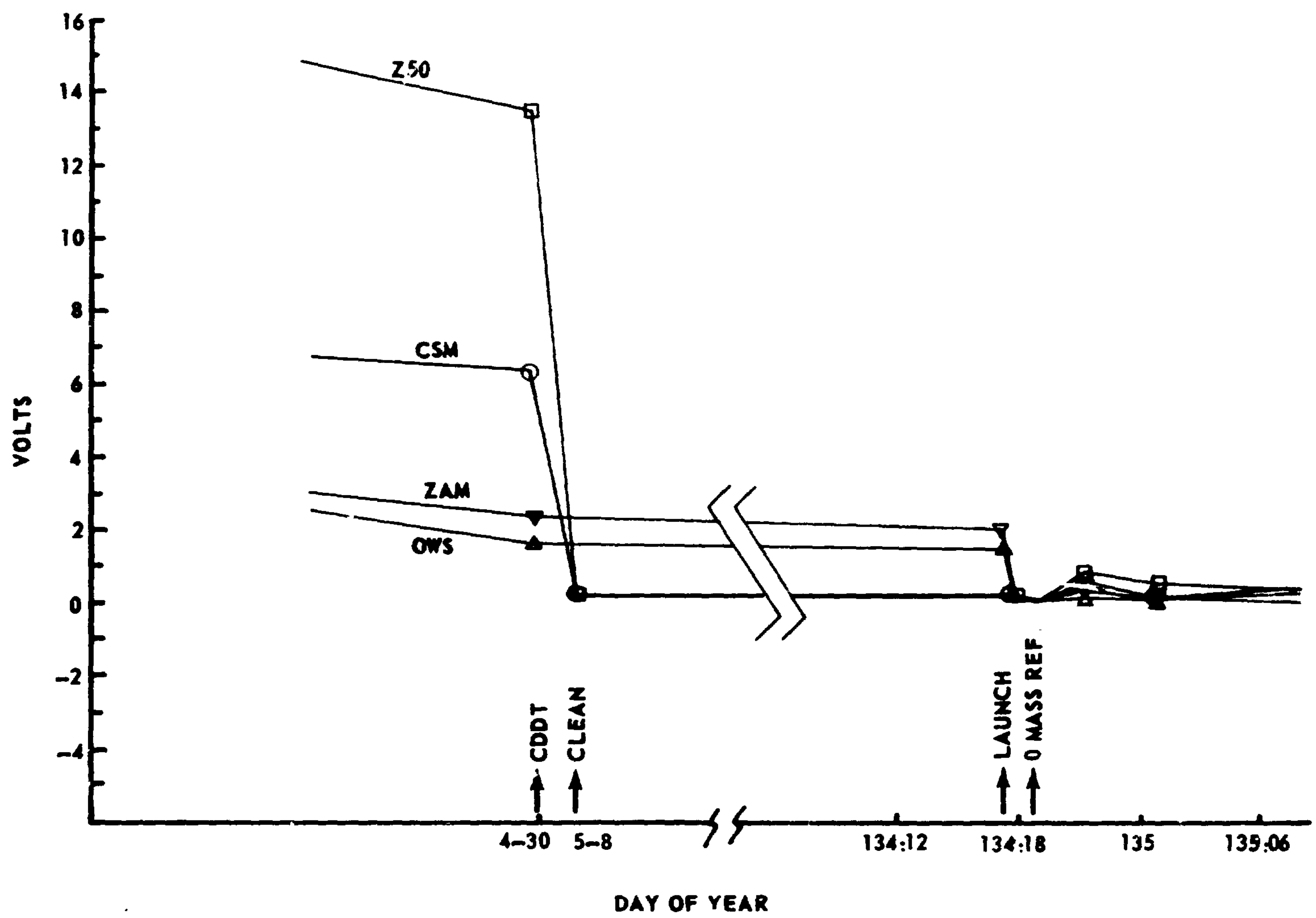


Figure VE-14. EREP QCM Time line Prior to and During Orbital Insertion

F. Docking Adapter

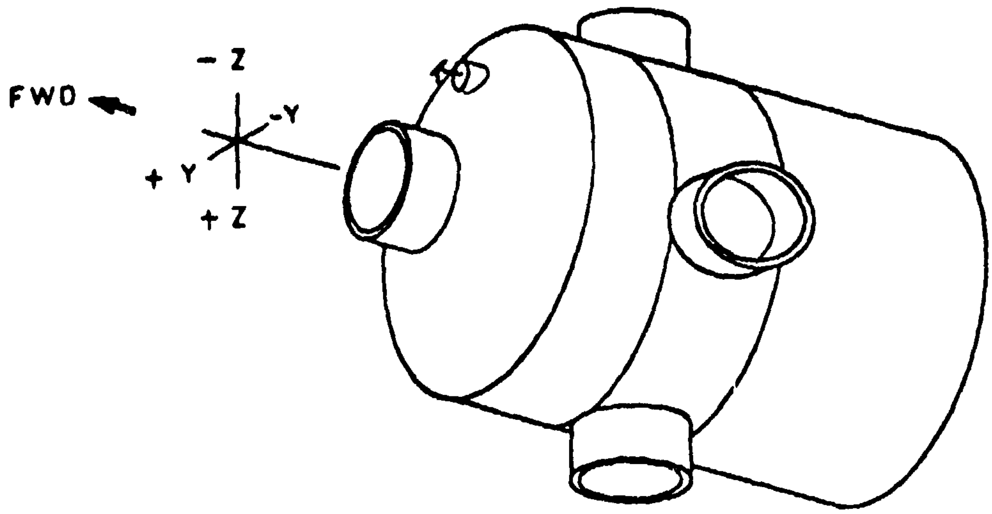
1. <u>Basic Requirements and Configuration Evolution.</u> The docking adapter structure is required to provide: A pressurized passageway between the airlock and the docked command and service module; two docking interfaces; stowage for hardware and experiments, support for experiments and control consoles; and, prelaunch purge and ascent and orbital venting of the airlock and docking adapter.

Initially, under the wet workshop concept, the docking adapter was was designed with five docking ports for the command and service modules, the lunar module/telescope mount, and resupply modules. A scientific airlock and various experiments were to be permanently installed and gridwork panels were to provide launch mountings for equipment to be transferred to the spent S-IVB stage by the crew on-orbit. After the stowed equipment was transferred to the workshop, the docking adapter would become a work area. During design evolution, three docking ports were eliminated, but the wet workshop shell was retained and extensive internal changes were made when the dry workshop concept was adopted. External views of both configurations are provided in Figure VF-1.

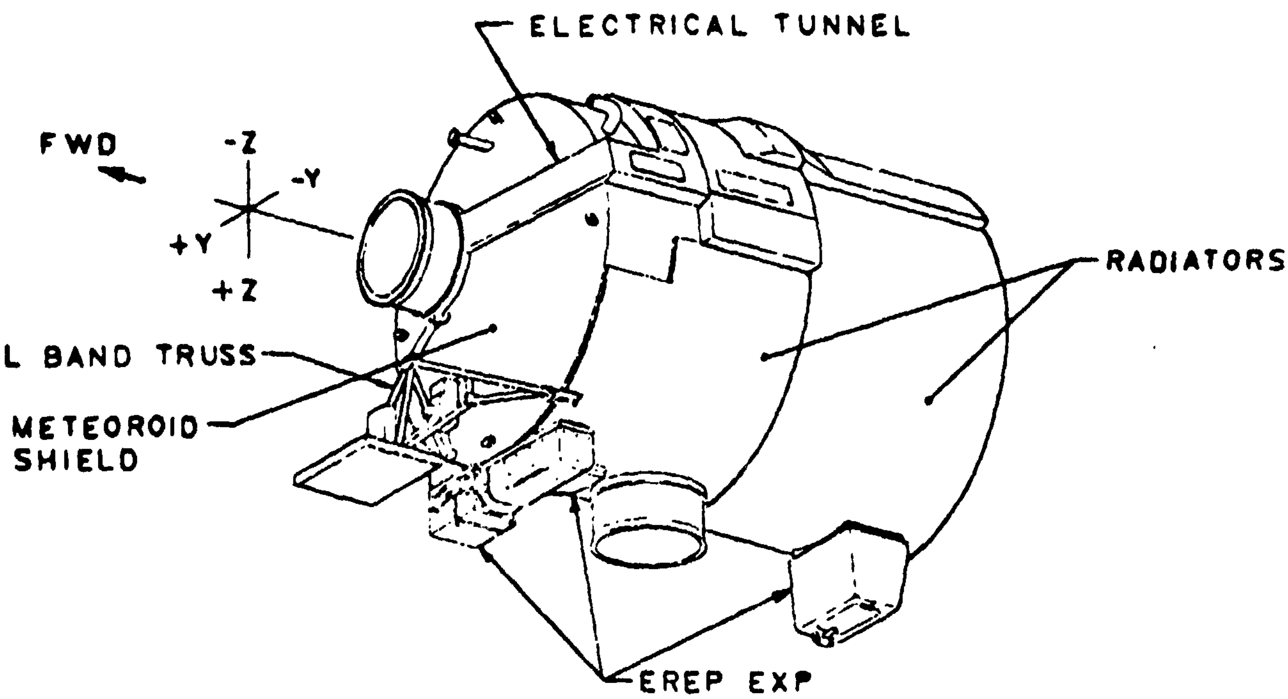
The dry workshop allowed experiments and equipment previously planned for launch in the docking adapter to be permanently installed in the workshop. The docking adapter became available for additional permanent installations including the telescope mount film vaults, the materials processing facility, the S009 experiment, a viewing window, and various containers. The scientific airlock was relocated to the workshop. Figure VF-2 shows how structural ring frames and longeron splice plates were added internally to the docking adapter to take the high local loads resulting from changed requirements.

The addition of earth resources/observation experiments required installation of three experiments on the earth-facing side of the docking adapter. Experiments S191 and S192 are installed with portions internal and sensors protruding from cutouts in the docking adapter skin. The S194 L-band antenna, the proton spectrometer, and the telescope mount inverter/lighting control assembly are mounted on a truss at the cone end of the docking adapter. The earth-viewing window was upgraded in optical quality for use with the S190 experiment. An external view of these installations is provided in Figure VF-1.

During installation, checkout, and test phases, other requirements were identified, including tools for in-flight maintenance and potential contingencies. A three drawer tool box for standard tools and a container for hatch contingency tools were designed. The standard tool container is mounted internally and the contingency tool container is mounted externally on the axial hatch.



FIVE DOCKING PORTS



DA-TWO DOCKING PORTS
(AS FLOWN)

Figure VF-1. External Views of Docking Adapter Configurations

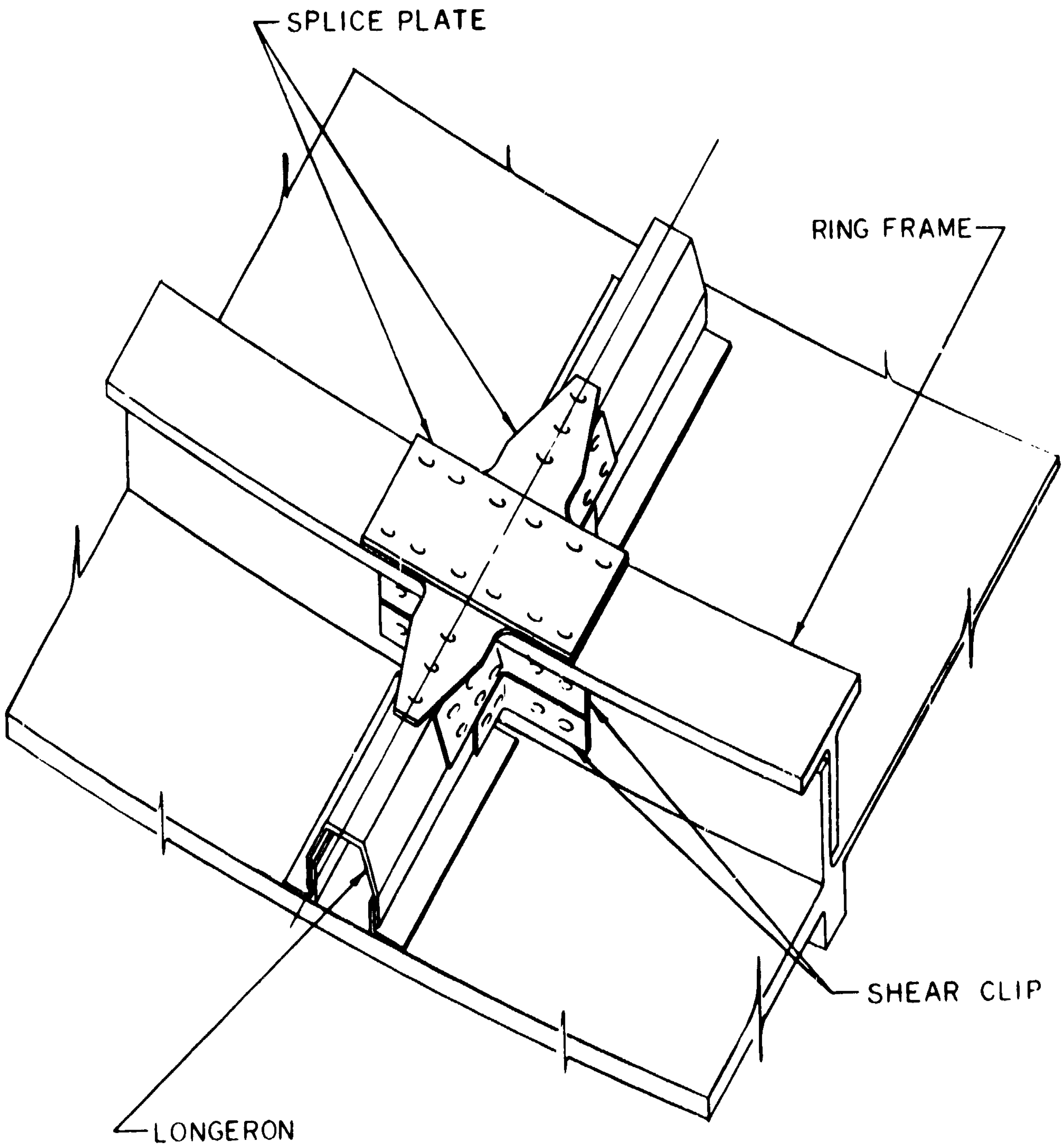


Figure VF-2. Docking Adapter Structural Ring Frames and Longeron Splice Plates

2. <u>Structures</u>. The docking adapter shell is a 10-ft diameter 17.3-ft long, semi-monocoque pressure vessel. It consists of a cylinder 13.6-ft long, a 120° included angle cone 2.1-ft long, an axial docking port on the forward end of the cone, and a radial docking port attached to the cylinder on the +Z axis. Four pickup points are provided in the upper ring frame to satisfy handling and transportation requirements. Each of the pickup points is capable of supporting a limit load of $\pm$7,626 lb radial shear, +2,094 lb tangential shear, and $\pm$19,400 lb tension simultaneously applied. Transportation and handling loads were never more critical than flight loads, except at the lifting points.

The docking adapter shell is required to withstand loads from handling and transportation, prelaunch, launch, ascent, orbital operations, and docking. It is also required to provide structural support for internal and external installations. The docking adapter was designed to withstand hot and cold temperature extremes and the vibration, shock, and acoustic levels specified in IN-ASTN-AD-70-1 for Saturn V vehicle; except for the deviations contained in the contract end item specification for the L-Band truss, delta pressure gage, vent valve sealing device, fan mufflers and shroud, M518, M512 foot restraint, and the telescope mount control and display foot restraint. Design vibration and acoustic levels for these items were derived after analysis of data from the vibroacoustic test at the JSC. The data showed the environment to be less severe than stipulated in IN-ASTN-AD-70-1, and compatible with the previous qualification environment for the delta pressure gage. Based on the reduced criteria, it was possible to show an analytical factor of safety of 3.0 for all these items, thus eliminating the requirement for testing.

a. **Pressure** vessel. The docking adapter was structurally designed to withstand a positive limit pressure differential of 6.2 psi and a negative (collapsing) limit pressure differential of 0.50 psi. Relief pressure was 6.2 psig differential as controlled by pressure relief valves in other Skylab modules. The docking adapter was designed to be compatible with a minimum internal pressure of 0.5 psia during periods of orbital storage. This requirement was an important consideration in materials selection and the design of equipment to be installed internally. Maximum allowable leakage rates were based on an operational pressure of 5.0 psia at 70 °F. The atmosphere was composed of oxygen at 3.5 psia and the balance of the operational pressure made up of nitrogen. Leakage of the docking adapter was required not to exceed the following criteria:

Shell (includes the window, vent frame attachment, electrical umbilical frame feed through attachment and longeron attachment)	1.26 lb/day

Installation penetrations (Includes vent plate assembly attachment, umbilical connectors, umbilical feed through attachment, telescope mount connectors, two hatches, miscellaneous fasteners, window cover mechanism penetrations, M512 vacuum vent penetrations)	0.94 lb/day
Total	2.20

Leak tests were conducted with docking port hatches installed and a tooling plate at the docking adapter/airlock interface. Measured leakage was 1.10 lb/day. Also, leakage of the combined docking adapter/airlock flight article was checked prior to launch at a pressure differential of 5.0 psi. The measured leak rate was 975 sccm. The allowable leak rate was 3,540 sccm.

The docking adapter was required to withstand tests for proof pressure at 1.5 times the limit pressure of 6.2 psid, yield pressure at 1.1 times proof pressure, and burst pressure at 2.0 times limit pressure. The burst pressure test was conducted at MSFC on the static test article during static tests. The flight article was subjected to a proof pressure test at 9.3 psid and leak tests at 5.0 psid. Compressed on-orbit pressure history of the docking adapter, as provided in Figure VF-3, is typical for the manned portion of the Skylab mission after the second day of the first manned phase and shows that the maximum allowable differential of 6.2 psi was never reached. However, pressure increased each time the astronaut maneuvering experiments were conducted. Maximum pressure of 5.8 psia was reached during M509 operation on the third manned phase.

Leakage of the docking port was checked by the crew using pressure decay rate before each activation and was found to be within acceptable limits. Leakage of the cluster throughout the mission was less than the allowable rate, indicating that the docking adapter leakage was also satisfactory.

b. Loads. Design of the docking adapter used Apollo docking data as a basis in determining the requirements and the design criteria necessary to dock the command and service module and to transfer the crew to another vehicle. The docking adapter was designed to accept loads resulting from axial or radial docking. The docking adapter specification weight was 14,050 lb; lift-off weight was 13,650 lb. The docking ports and pressure vessel were designed to the most severe condition of the following limit loads, either separately or in combination, for each condition.

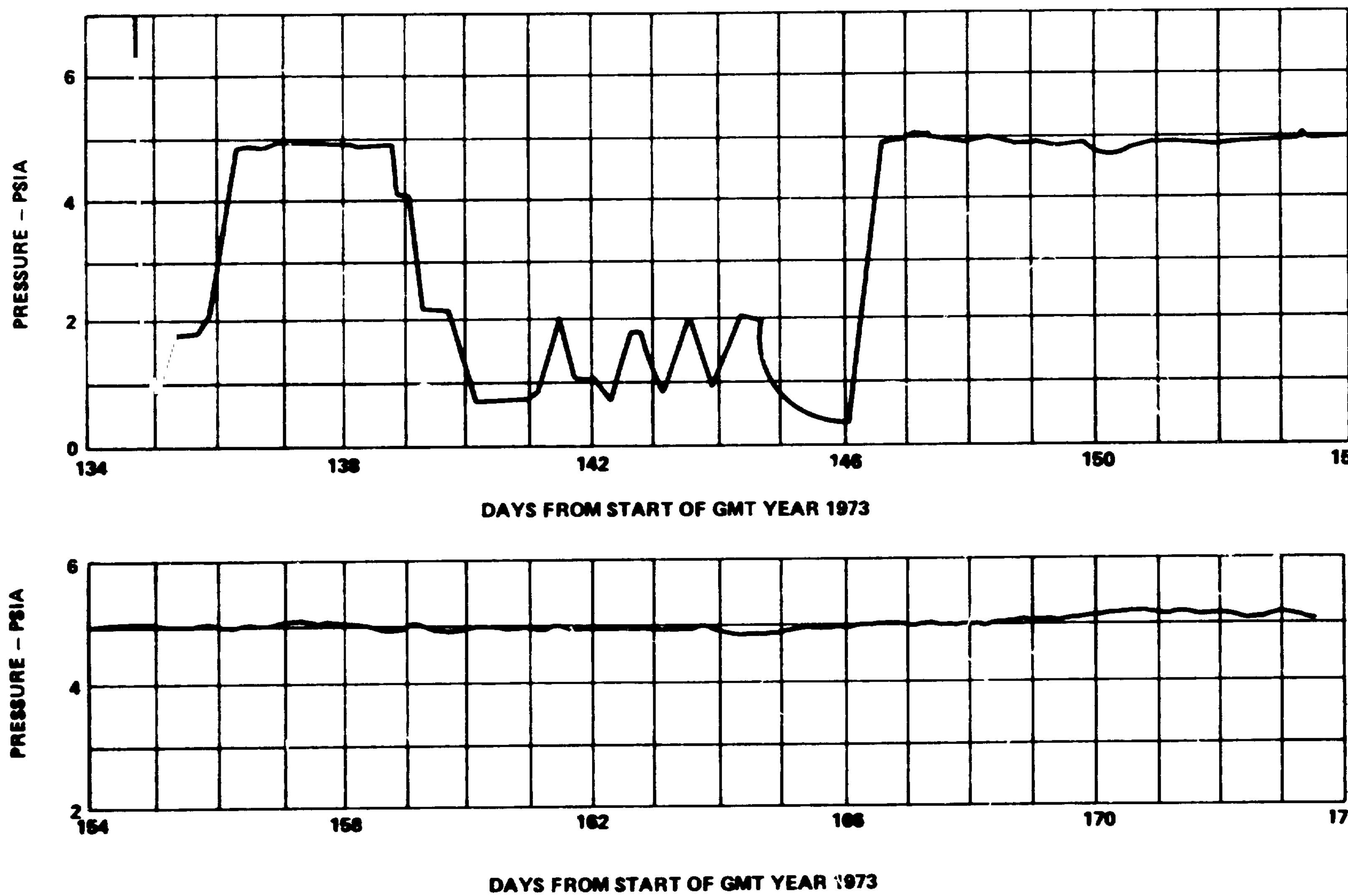


Figure VF-3. Docking Adapter Internal Absolute Pressure (Typical of Mission)

Latched Interface Loads

Bending moment*	±600,000 in.-lb (both ports)
Axial load	±14,610 lb
Shear*	±4,400 lb
Torsion	±150,000 in.-lb
Latch load (each latch)	±12,000 lb

*Shear and bending moment can act in any plane.

The following factors of safety were used in designing the docking adapter structure:

Common Structure

Manned Vehicle:
Yield factor of safety =1.10
Ultimate factor of safety =1.40

Unmanned Vehicle:
Yield factor of safety =1.10
Ultimate factor of safety =1.25

Support Brackets

As a design guide to eliminate testing, all brackets to support new equipment, film vaults, and experimental packages shall not yield at 3.0 times limit load, nor fail at 3.0 times limit load.

The analysis of the shell for the worst case loading condition (lift-off) was handled as follows: The highest stresses in the structure were at or near heaviest loaded components. Components in adjacent bays were assumed to act simultaneously for random vibration loads. These random loads were combined using a root-sum-squate (rss) technique. The resulting member loads were then added to the loads caused by vehicle dynamics and steady state acceleration. Structural integrity was demonstrated analytically for the docking adapter shell subjected to the combination of loads.

The highest loads for most of the packages would occur during the Saturn V launch and ascent. The loads were due to steady state acceleration, vehicle dynamics, and random vibration. The worst time-consistent combination of these loads occurred shortly after lift-off and was governed primarily by random vibration. Loads data used in the analysis were composed of the following:

- Steady State Acceleration - The Saturn V accelerates from approximately 1.2 g's at lift-off to 4.7 g's shortly before staging (IN-ASTN-AD-70-2).

- Vehicle Dynamics - Vehicle dynamics are low frequency excitations caused by flight transients. The maximum transient would be encountered at SIC-SII separation, and was predicted to be 2.3 g's maximum in the docking adapter.

- Random- The maximum random vibration environment occurs at lift-off and maximum dynamic pressure. It is due to acoustic excitation of the structure and is defined in IN-ASTN-AD-70-1. This document split the docking adapter into zones, which are regions between ring frames and subzones, and account for mass attenuation effects or type of structural mounting (skin, longeron, etc).

The minimum margin of safety for launch loads determined by analysis was 0.19, as shown in the docking adapter strength analysis report.

Several successful static tests were performed to verify structural integrity. Structural testing was done on a static test article, which consisted of a docking adapter shell with docking ports, windows, and infrared spectrometer fitting. Testing was conducted at MSFC.

The objectives were to verify structural integrity of the structure for docking loads and loads imposed on local structure by equipment and experiment packages, to determine deflections and stresses of the critical loads conditions, and to verify analytical methods.

Nine separate conditions were tested. Six conditions simulated worst case pressure and docking/latching loads. Three conditions were tested to verify structural integrity for local loading conditions. The local loads were derived from the worst case, static equivalent combinations of steady state acceleration, random vibration, and vehicle dynamics. A factor of safety of 1.4 was applied to design limit shear, moment, and axial loads, and 2.0 to pressure loads.

c. Acoustical data. Following completion of static tests at MSFC, the docking adapter static test article was returned to Denver where it was updated to the flight configuration prior to the vibroacoustics test at the JSC. Flight type hardware or mass simulators, both internally and externally mounted, were installed for this test. Modified input sound pressure levels and acoustic criteria were used for the acoustical simulations in the enclosed test area. No measurements, internal to the docking adapter, were provided during launch, boost, and insertion. However, external criteria and flight results, which are the equivalent to payload shroud internal criteria and flight results, are plotted in Figure VE-7 (payload shroud section).

An internal noise spectrum design goal for the docking adapter was based on the requirement that the summation of cluster individual sound pressure levels at any given time on-orbit would be less than the values shown in Figure VF-4. During orbital operations, flight crews using portable equipment measured internal sound pressure levels and results are shown in Table VF-1.

Quantitive recordings of the overall sound pressure level within the docking adapter indicate that it complied with the specified requirement for the acoustic environment. The rate gyro six pack is the only noise producing installation that exceeded the design sound pressure levels in some of the frequency ranges. However, it was lower in ambient noise level than the design specification.

The rate gyro six pack was launched with the command and service module for the second manned phase, to provide a contingency installation as backup for the basic rate gyro system. Rate gyro module failures had occurred in the basic system during the first manned phase. This consideration would tend to qualify the criticality of this induced noise environment. Likewise the sound pressure level measurements recorded by the commander were afforded some degree of qualification in his accompanying comments on the dump tape:

> "Remember that these were taken in the environment with other equipment running. And so sometimes you're not getting a pure sound level on this except for pointing the instrument at it."

A qualitative assessment establishes that the acoustic environment is within comfortable limits based on the subjective evaluations reported by the second Skylab crew.

d. Vibration data. Vibration pickups were not provided in the docking adapter. Operational capability of equipment shows that the docking adapter and its internal and external installations maintained structural, mechanical, and operational integrity. No anomalies occurred that can be attributed to excessive vibration.

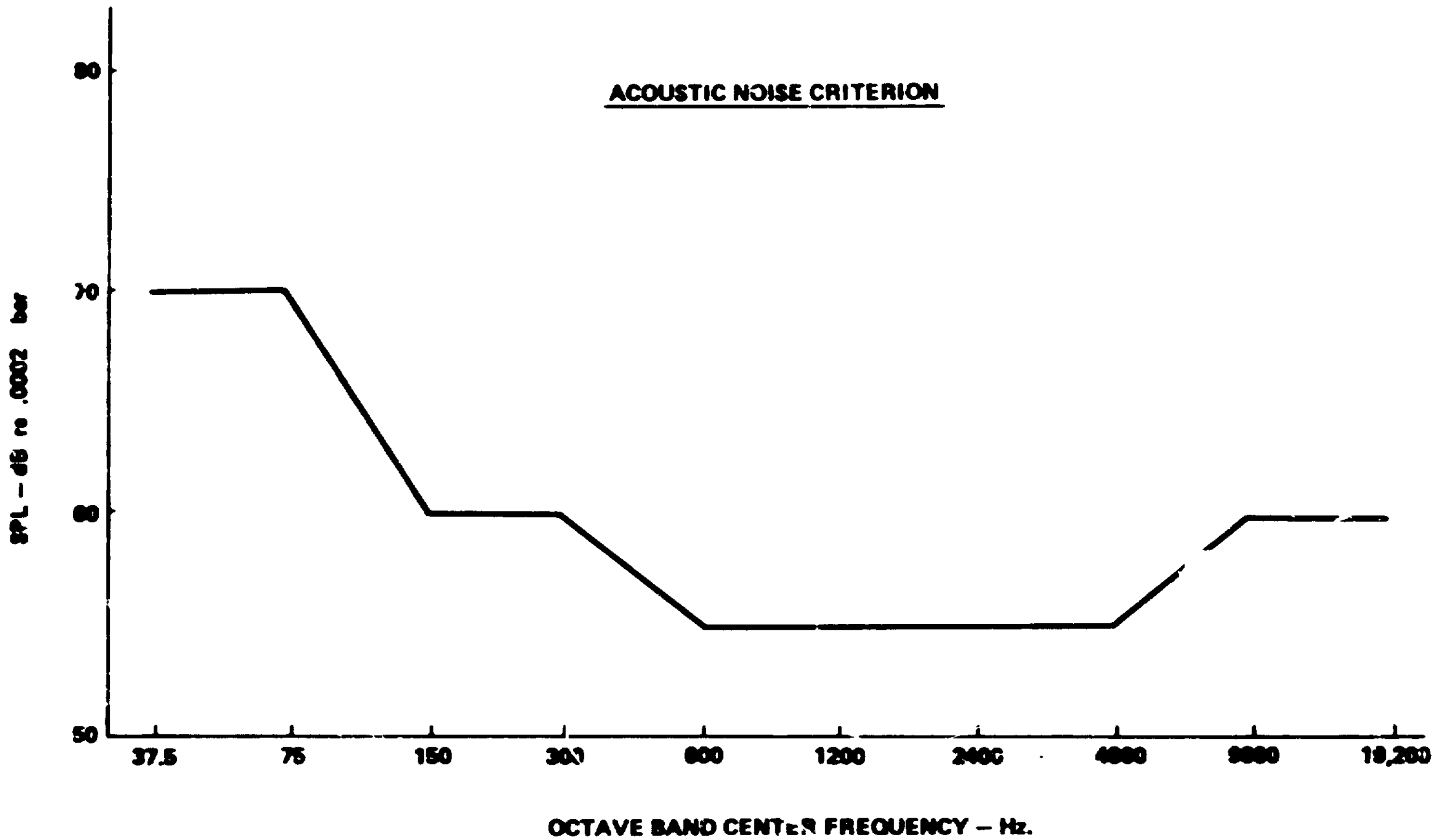


Figure VF-4. Sound Pressure Levels

LOC	AMB NOISE LEVEL	(125Hz)	(250Hz)	(500Hz)	(1KHz)	(2KHz)	(4KHz)	(8KHz)	(16KHz)
1. DA	˙1.5	61	56	55	52.5	51.5	46 5	44.5	34
2. RATE GYROS	72	61	59.5	67	60.5	57.5	50	49.5	42
3. DA AREA FAN NO. 1	66	58	56	55	54	51	45	41	29
4. DA AREA FAN NO. 2	67	60	57	58	55	50	48	45	34
5. CSM FAN	67	61	56	56	52	49	43	41	28

NOTE: Measurement 1 DA was made by the PLT on DOY 249 (MD41) at M512 Experiment crew station with sound level meter microphone pointed toward the STS (-X direction).

Measurements 2 through 5 were made by the CDR on DOY 251 (MD43) at the locations indicated with the fans on high speed setting.

All values are in decibels.

Table VF-1. Third Manned Phase M487 Noise Measurements

3. Natural Environments Design. Docking adapter requirements for maintaining habitable volume in the space environment, including radiation protection, are provided by structural design. Appropriate protection against particulate matter, excessive humidity, rain, ground winds, flight winds, and facility gases (induced through the airlock and vented from the docking adapter vent valve) are primarily a function of the payload shroud, the fixed airlock shroud, and the KSC facility.

Meteoroid protection for the pressure shell is provided by 0.050-in. thick aluminum panels on the cone and 0.020-in. thick aluminum panels on a portion of the barrel. The meteoroid panels are supported 3 in. from the pressure skin on fiber glass standoffs. Figure VF-5 shows a typical meteoroid panel installation.

Approximately 75 percent of the cylindrical portion of the docking adapter is protected against meteoroids by thermal radiators. These are constructed of 0.030-in. magnesium with coolant tubes attached, and are finished with a special reflective white paint. The radiators are bolted to 3-in.-high fiber glass standoffs attached to the docking adapter. The radiators are functionally a part of the airlock environmental control system.

The docking adapter is required to withstand meteoroid impact that could result in pressure loss or loss of functional capability when subjected to the meteoroid flux model defined in NASA TM-X-53798. Design requirements for the meteoroid panels are:

- Withstand prelaunch, launch, and ascent environment.
- Meet 0.995 probability of no pressure shell penetration.
- Meet 0.995 probability of no electrical wiring penetration.

Acoustic loading on the radiators and meteoroid panel was derived from analysis of the as-flown panels subjected to the environments defined in IN-ASTN-AD-70-2. A static equivalent acoustic pressure of 0.09 psi was combined with launch accelerations (4.7 g's longitudinal, or 2.3 g's longitudinal combined with 1.8 g's lateral) to determine design loads for the panels and supports. All structures were designed with a factor of safety of 3.0.

Panel acoustic vibration tests were conducted on the thinner (0.020 in.) meteoroid panel. High velocity pellet meteoroid penetration tests were run on all structural configurations and exposed wire harness configurations used on the docking adapter. All tests were successful in meeting their requirements.

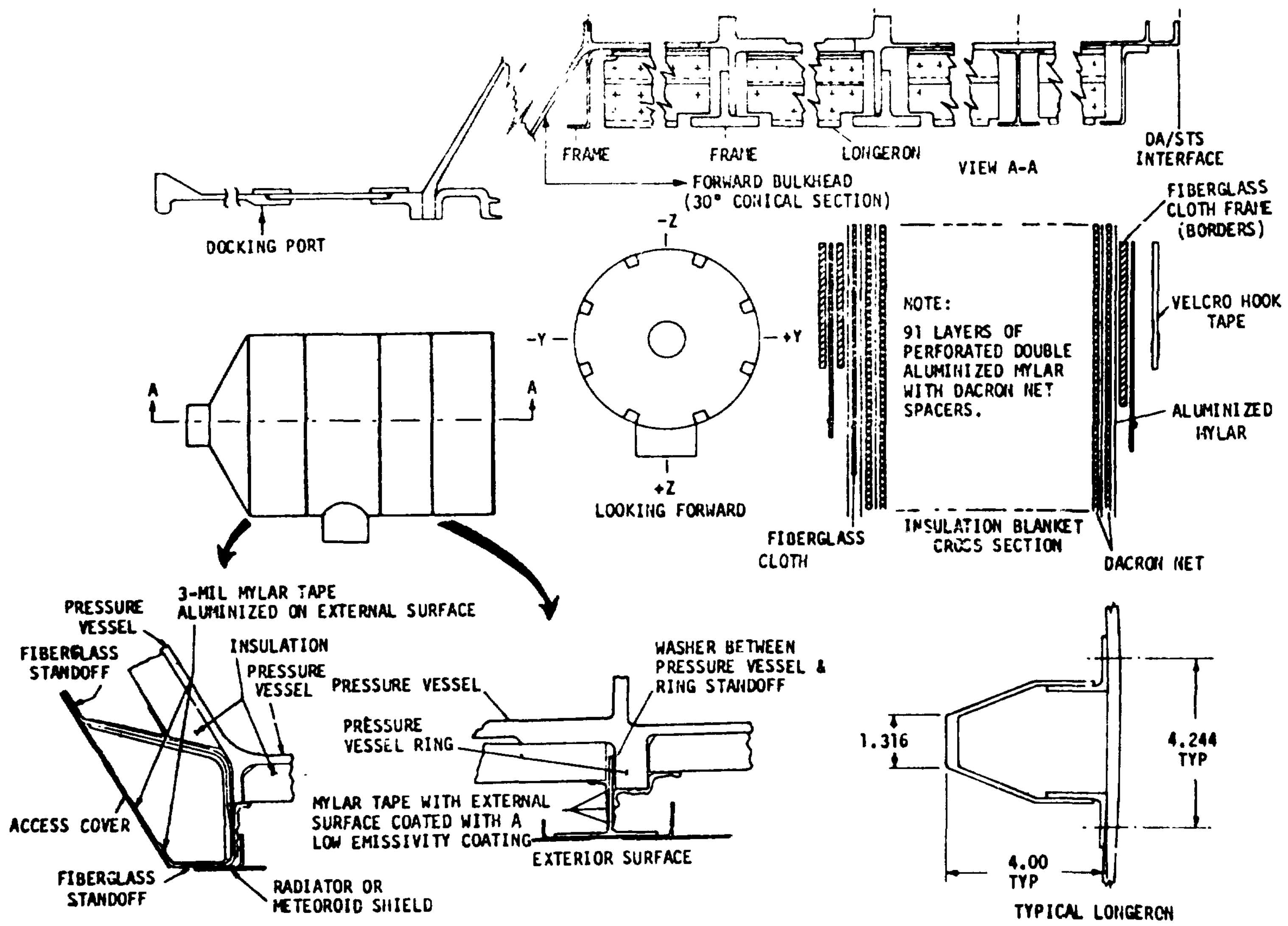


Figure VF-5. Docking Adapter Basic Shell

The radiators and meteoroid panels survived the launch and boost environment without damage. No penetrations by meteoroids were observed. The crew reported a change in the white paint on the cone to a golden color. But there was no noticeable temperature change in docking adapter as a result.

The requirements to provide breathable atmosphere at class 100 level during operations at KSC and to permit purging of the airlock and docking adapter during prelaunch were provided by the airlock hardware. Venting at KSC, during ascent, and on-orbit was a function successfully accomplished by the docking adapter.

Structural design is required to limit internal radiation to 0.6 rad/day. On-orbit radiation measurements were taken in the workshop. Extrapolation to the docking adapter using a computer model shows radiation to be 0.10 rad/day average dose at the center of the docking adapter.

4. Mechanical Components. Mechanical components were designed and/or selected to withstand the natural and induced environments stipulated for the basic docking adapter structure. Although the mechanical components were functionally tested with the docking adapter and during integrated tests, the more meaningful data for evaluation purposes were often recorded during their qualification. As a function of these tests, many operating mechanisms were required to meet specific life-cycle and/or operating time requirements; therefore, such data are provided where pertinent.

a. Windows. The docking adapter contains four windows that are integral parts of the pressure vessel. The windows provided sensing ports in the structural shell for the S190, S191, and S192 earth resources/ observation experiments.

The S190 window is a single pane of borosilicate crown glass (BK-7) 1.6-in. thick by 18 in. by 23 in., mounted and sealed in an aluminum frame and installed directly above the radial docking port. The frame was designed to take all flight loads except pressure. The window was designed to take only pressure loads and is supported b. a spring system so that vehicle distortions do not induce other flight ds into the glass. The S190 window installation is shown in Figure -6.

The function of the S190 window is to maintain structural and pressure-leakage integrity of the docking adapter and to admit visible and infrared light to the S190 multispectral camera with a minimum of optical degradation.

Structurally, the S190 window was designed to maintain structural and pressure leakage integrity by withstanding:

- Vibroacoustic loads
- Shock loads

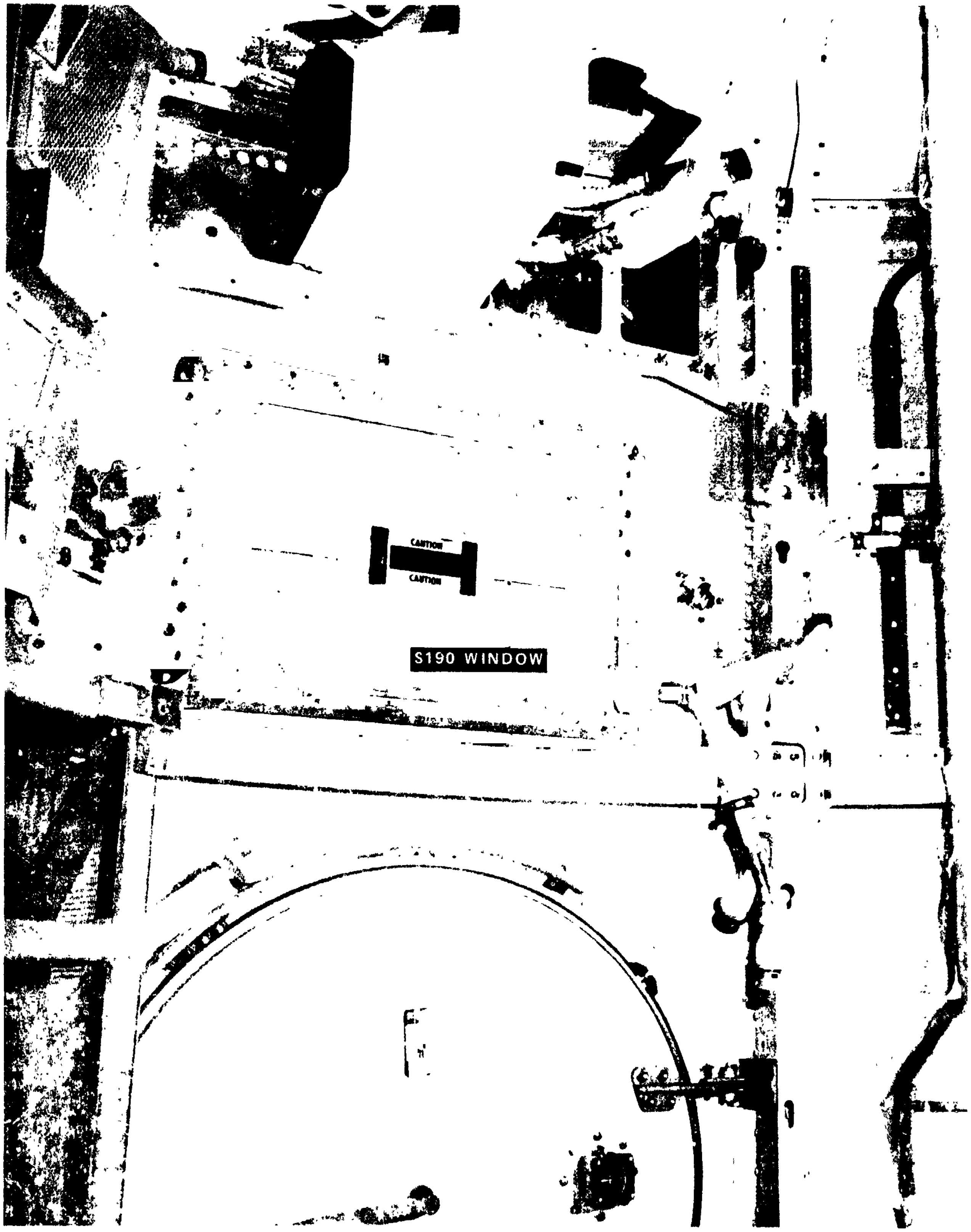


Figure VF-6. S190 Window Installation

- Pressure (6.2 psig limit pressure)
- Temperature extremes (-40 to 160 °F)
- Van Allen Belt radiation
- Micrometeoroid impact
- Accidental impacts from the crew

The window was designed to the following structural factor of safety requirements:

Proof pressure = 2.00 times limit pressure
Burst pressure = 3.00 times limit pressure

As a sensing port for the S190 multispectral camera, the window was designed to comply with the following specific requirements:

- Wavefront distortion
- Transmissibility
- Reflectance
- Glare
- Contamination control
- Moisture condensation prevention
- Crew protection from ultraviolet radiation.

The S190 window was subjected to an extensive development and qualification program to prove its ability to meet those requirements.

Tests of the window included development, qualification, acceptance and specimen testing:

- Successful development, qualification, and acceptance testing of the prototype and flight window included flaw-screening, vibration, shock, proof pressure, seal leakage, impact and thermal stress. Optical testing included wavefront distortion, transmissibility, reflectance, and glare.
- Full-size specimens--A full-size window pane was successful. tested to 124 psid (safety factor of 20). A second specimen successfully withstood 18 psid after being scored with a glass

cutter to make a shallow 18 in. long scratch. After a 0.35-in. deep flaw 1-in. long was made in the center of the other (unscratched) surface, using a thermal stress cycling technique, the specimen broke at 24 psid. Both tests were conducted with the flawed surfaces in tension.

- Sample specimens--100 6-in. diameter specimens of BK-7 glass were tested to determine the degradation caused by coatings, buss bars, humidity, temperature extremes, Van Allen Belt radiation, and 8-month vacuum exposure. The specimens were arranged in groups of 10 for statistical evaluation of individual environments. No degradation in glass strength was caused by any of the environments.

- Bar specimens--25 bars of BK-7 glass were tested to obtain data about critical stress versus crack size and crack growth rate for long-term loading effects.

- Block specimens--12 pieces of BK-7 glass (1.6-in. thick) did not static discharge or crack after being subjected to varying amounts of electron radiation, up to an amount corresponding to more than 200 yrs in orbit.

Each window pane was flaw-screened by pressure testing to 30 psid. Window assemblies were proof pressure tested to 14.7 psid.

Three smaller windows in the docking adapter were used by the S191 and S192 experiments. The S191 window was 4 in. in diameter by 0.48-in. thick. Its function was to act as a viewing port for crewmen when pointing the viewfinder t acker system at the selected target. The two S192 windows were 3 in. in diameter by 0.25-in. thick. One was made of germanium and the other of fused silica (Infrasil). These windows transmitted selected wavelengths of radiation to the S192 experiment internal scanner.

An extensive test program was conducted on the S191 and S192 windows and window material. These tests include development, qualification, acceptance, and specimen testing:

- Successful development qualification and acceptance testing of prototype and flight windows included flaw-screening, vibration, shock, proof pressure, and seal leakage. Optical testing included transmittance, reflectance, wavefront distortion, and surface quality.

- Full-size specimens (25 germanium test specimens and 25 of Infrasil) were tested to determine whether degradation was caused by coatings, humidity, temperature extremes, Van Allen Belt radiation, and 8-month vacuum exposure. None of the environments caused a degradation in strength.

- Bar specimens--Two bars of germanium and two of Infrasil were tested to obtain data about fracture toughness and crack growth rates under sustained loads. Similar data were obtained for BK-7 glass in connection with S190 window testing.

Each window was flow-screened by pressure testing to 30 psid for the S192 windows and to 45 psid for the S191 windows.

Careful monitoring of window temperatures and atmosphere dewpoint was carried on throughout the mission, bacause of the sensitivity of the performance of the window to condensation. Figure VF-7, showing dewpoints and window temperatures for the first manned phase, shows that condensation conditions did not occur. This conclusion is supported by crew observations.

All four windows apparently operated normally throughout the mission. No problem was reported by any of the three crews. No distortion of data due to the windows was observed.

b. Covers. External and internal protective covers for the S190 window are provided (Figure VF-8) to protect the surfaces of the window from micrometeoroid impacts, contamination, and internal impacts. The external window cover is mechanically operated from within the docking adapter. It was opened for all earth resources/observation passes, hand-held photography, and viewing. Insulation is installed on the external window cover to minimize heat loss. The safety shield (internal cover) is transparent and removable from the inside. It was designed to provide sealing redundancy for the S190 window and to withstand pressure, impact, and vibroacoustic loads. The shield was removed only for earth resources/observation operations.

The external cover is a curved fiber glass honeycomb panel 1-in. thick by approximately 20 by 31 in. It contains metal fittings, integrally bonded to the panel, for hinge attachment and latch engagement. Multilayer insulation is installed in a fiber glass pan that is attached to the internal surface of the cover. The cover (including the pan) is painted black for thermal control and to minimize reflected light on the window. A resilient foam seal around the cover edge closes the gap between the cover and the meteoroid shield to prevent dust and other contaminants from reaching the window. The honeycomb panel has several vent holes to relieve internal pressure during boost.

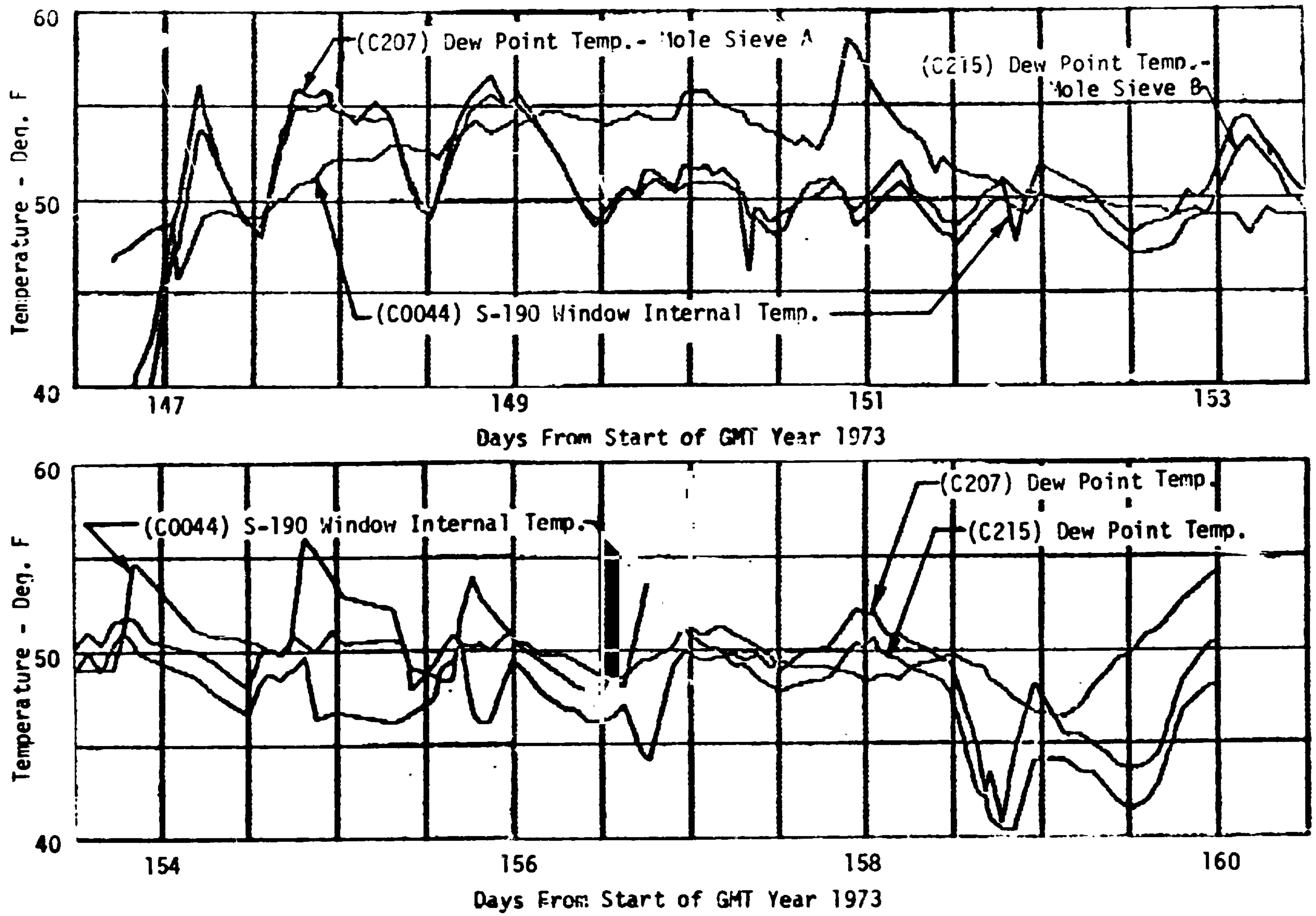


Figure VF-7. Dew Point and S190 Window Temperatures

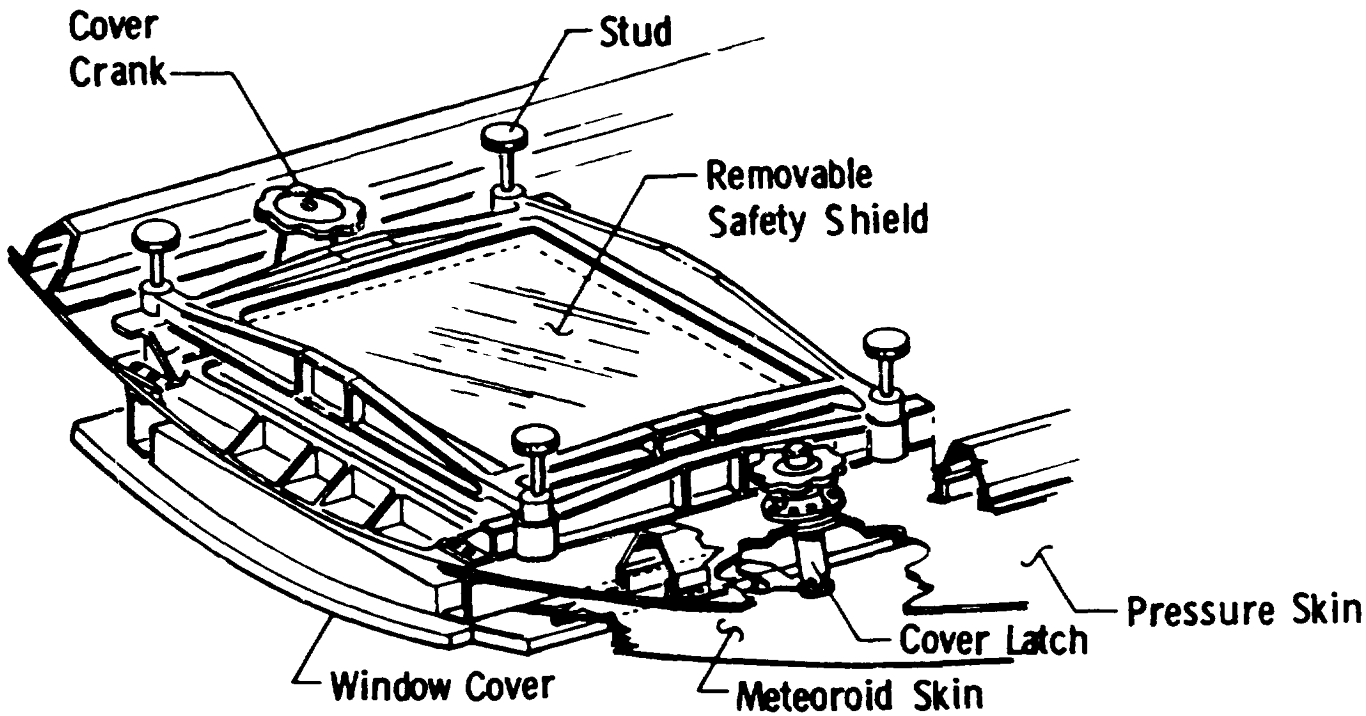


Figure VF-8. Protective Covers for S190 Window

The cover was designed to meet the following criteria:

- Provide meteoroid protection for the window equivalent to the adajacent structure.
- Minimize heat loss through the window.
- Provide contamination protection for the window.
- Provide venting for the cavity between the cover and window to accommodate pressure changes during transportation, purging, and the ascent portion of the flight.
- Withstand all imposed loads.
- Meet outgassing criteria.

Another protection the cover provided for the window was against the space radiation environment.

Two mechanical devices, an actuator and a latch, are installed on opposite sides of the S190 window to hold the external cover in place during boost and storage periods, and to enable the crew to open the cover when the window is to be used. The latch is operated by turning the latch handle counterclockwise approximately seven turns. This operation moves an external latching arm outboard, freeing the edge of the cover so it can be opened. The cover is then opened by turning the actuator handle clockwise. This actuator handle is connected to the cover hinge through a gear set that moves the cover through one-half the angle of the handle. Rotating the handle 270° moves the cover to an angle of 135° from the window, which removes the cover from the field of view of the S190 camera. The actuator handle has a cam that operates a warning-light microswitch to indicate to the crew that the external cover is closed. The actuator and latch mechanisms were designed for ease of operation and to provide comfortable touch temperatures for the crew. They were also required to withstand all launch and operational environments without functional degradation. A mechanical design was specified to assure reliability. Maximum design loads were 133 in.-lb torque on the latch and actuator knobs. Actual operating torques were 13 in.-lb or less as measured in preflight checkout at KSC. Specifications required window covers to be designed for a minimum of 300 operating cycles. The S190 window cover mechanism was qualification tested for 500 cycles and operated in orbit for 100 cycles without any problems.

The safety shield is a removable internal cover for the S190 window. It is positioned against the inside of the window frame and hand fastened there by a crewman whenever the S190 experiment was rotated into its stowage position back away from the window. The safety shield consists of a high-strength glass panel (Corning Chemcor 0315), 0.290-in. thick, mounted and sealed in an aluminum frame. The frame includes an O-ring on its mounting surface to provide sealing redundancy for the S190 window. The function of the safety shield is to protect the S190 window from possible damage resulting from the impact of loose objects within the docking adapter and to act as a redundant pressure seal in case the S190 window had failed. The safety shield pressure seal was launched in its stowage position on the aft end of film vault #4.

Tests were conducted to verify the capability of the window cover mechanisms to withstand the design environments and operating conditions for the duration of the mission. The tests included measurements of operating torque and leak rate before and after exposure to vibration and acoustic excitation. A 30-day vacuum test demonstrated that the mechanism would function properly without excessive leakage under orbital conditions. A development external cover was included in the mechanism tests of vibration, acoustics, and operating cycles. Although the sole purpose of the cover in these tests was as a mass simulator, the cover successfully withstood all test environments.

Acceptance and qualification testing of the safety shield included impact resistance, vibration, proof pressure, and leakage of both the glass seal and O-ring redundant seal.

Acceptance tests on the safety shield glass panels included both structural and optical tests. Structural integrity was demonstrated by pressure testing and thermal-shock testing to 540 °F to screen the panels for hidden flaws. Optical clarity was assured by testing for high transmittance and absence of distortion.

The external window cover and the safety shield successfully performed all required functions during the Skylab mission. No adverse comments were reported by any of the three crews.

c. Docking ports. The docking adapter has two docking ports. The axial docking port, which is the primary docking port, is located at the forward end of the docking adapter and centered about the X axis. The radial (secondary) docking port is located 103 in. forward of the docking adapter/airlock interface on the +Z axis.

Both the axial and the radial docking ports have standard Apollo drogues and docking interfaces to permit docking of the command module. The axial port is equipped to transfer electrical power, communications, and conditioned air between the docking adapter and the command and service module following docking.

Docking aids for visual orientation and alignment are provided to facilitate docking. The docking aids were designed to permit docking operations to be accomplished independent of flight control data. The docking targets are of the Lunar Module type. The external surface of the forward cone of the docking adapter is provided with a white circular stripe at its outer diameter to aid docking. Four exterior lights are installed for visual orientation.

No tests were required to verify structural integrity of the docking hardware, since the existing Apollo docking system was used. A docking test was conducted at KSC using the flight docking adapter and the command and service module used for the first manned phase to verify the docking system. Objectives of this test were to verify docking capability at the axial port, to check the docking interface leak rate, to verify alignment of the axial docking target, to verify fit of the air interchange duct and the electrical bonding between the command and service module and the docking adapter. All tests were successful. The measured leak rate of the docking interface was 0.009 lb/day, compared with an allowable rate of 3.14 lb/day.

A soft docking was done on the first manned phase. At this time, the command and service module probe captured the docking adapter drogue but the probe retraction mechanism was not activated since a standup extravehicular activity was planned. The probe was then released and the standup extravehicular activity maneuver was performed. Following this, hard docking was attempted. Several unsuccessful attempts to engage the drogue and probe were made. The command and service module probe was then corrected by the crew during extravehicular activity to permit retraction without capture latch engagement. Docking was achieved by applying reaction control system thrust until the docking latches engaged. No data are available on actual docking rates, but successful docking was demonstrated by the automatic engagement of all 12 latches and the absence of leaks in the docking tunnel. The crew reported that alignment accuracy was 1.5°, as measured by an index scale that was permanently installed in the docking port.

The command and service module successfully docked to the docking adapter on the initial attempt by the second manned crew. Docking conditions were nominal. The docking adapter internal pressure was 5 psia at the time of docking. After entry into the docking adapter, the crew noticed that command and service module docking latches #1 and #10 were loose on the docking adapter docking ring. The remaining latches were adequate to take all expected loads and to prevent leakage in the docking tunnel; therefore, no corrective action was necessary. Alignment accuracy of docking was [illegible]°, as measured by the index scale in the docking port.

Three attempts were required to achieve a hard dock by the third manned crew. All 12 latches engaged automatically. Alignment accuracy was 0.7 °F, as measured by the index scale in the docking port.

d. Hatches. The docking adapter had two circular, inward-opening hatches, one at each docking port. The hatches are 32 in. in diameter and 1.2-in. thick. Each hatch is held in the closed position by six overcenter latches. The latches are controlled by linkages attached to a central shaft. Handles are attached to the shaft on both sides of the hatch, thereby allowing opening and closing from either side. The handles are restrained in a launch lock which is locked from the outside only, but could be unlocked from either side. The edge of the hatch is a lip that depressed a silicone rubber seal in the docking adapter shell docking port ring to achieve a pressure tight closure. The amount of seal indentation is limited by six mechanical stops to prevent overstressing of the seal. The hatch is shown in Figure VF-9.

Requirements of the hatch are: (1) provide a pressure tight closure (2) be operated easily by the crew, and (3) withstand all handling and operation environments without functional degradation.

The hatch handle temperatures are required to be maintained between 105 and 35 °F during docking and all manned operations. Minimum force required to activate the hatch handles is 2 lb. The maximum force required to actuate the latching handle and pressure equalization valve is not to exceed 25 lb. The maximum force required to open the hatch is not to exceed 45 lb. The stowage provisions for the hatch in the open position were designed so that a force of 5 to 25 lb on a hatch would be sufficient to latch and unlatch the hatch. The hatches are also required to be removable and interchangeable.

Component qualification tests of the hatch were used to demonstrate the adequacy of the design for launch and orbital conditions. Qualification tests verified the final hatch design for pressure, vibration, pyrotechnic shock and ultimate handle forces. Maximum pressure in these tests were 12.4 psig. The ultimate handle force tested was 210 lb.

The flight hatches were tested for leakage before installation and for handle operating forces after installation. Maximum leakage measured on a flight hatch was 0.046 lb/day, compared with an allowable rate of 0.52 lb/day. Handle operating forces were 6 lb for opening, closing, and unstowing the hatch, well within the 5 to 25 lb allowed.

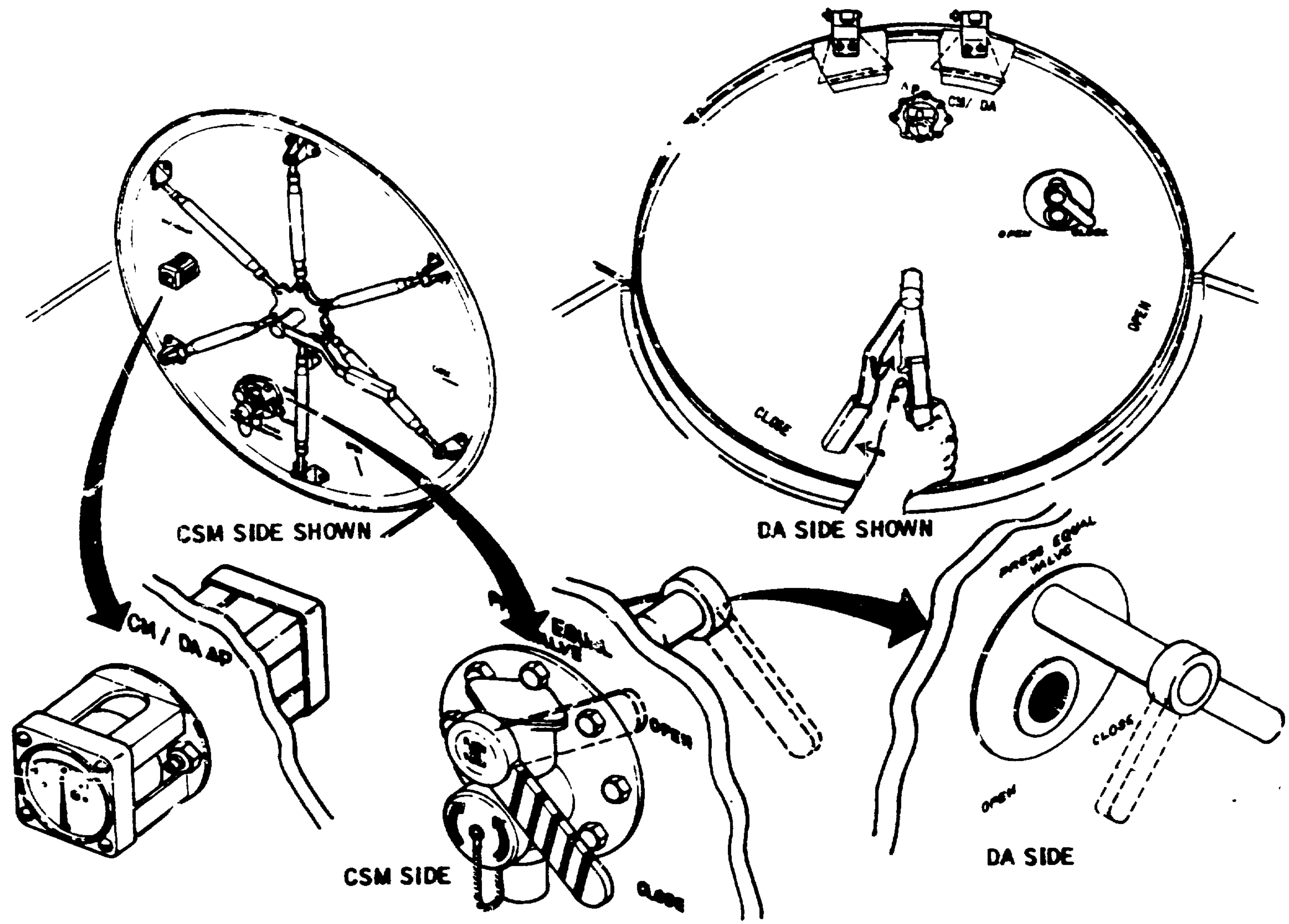


Figure VF-9. Docking Adapter Pressure Hatch Details

The hatch seal was tested separately to ensure integrity under long life thermal vacuum conditions. Tests were conducted on sections of hatch seals to determine the ability of the seal to withstand long periods of compression under operational conditions and to determine whether sticking of the seal to the hatch could become a problem. Test specimens were 4-in. long cut from a production seal. The specimens were compressed in a fixture resembling a seal retaining ring, and placed in a vacuum chamber with controlled temperature. Periodically, specimens were removed from the chamber, pulled to measure sticking force, and examined for damage. The test program was successfully concluded after 8-months exposure of the final seal configuration with no damage and with acceptably small sticking forces. Another test of a full-size seal was conducted to measure leakage degradation over 8 months of compression. This test, which was at room ambient conditions, showed negligible leakage of the seal after the full mission time.

The axial hatch was opened and closed three times during the Skylab mission. No problems with the hatch were reported by the crew. The radial hatch was not used.

e. Pressure equalization system. The docking adapter hatch pressure equalization subsystem provides means of equalizing the atmospheric pressure between the command and service module and the docking adapter after docking and prior to workshop entry. Each docking port hatch is equipped with a visual differential pressure gage and a manually-operated equalization valve. Equalization of pressure across the hatch is achieved by opening the valve.

The differential pressure gage assembly consists of two gages mounted back-to-back in both pressure hatches. The assembly allows delta pressure monitoring from either side of the hatch. The basic gage was developed and used on the Apollo program and requalified for Skylab use. The gage had a range of ± 1.0 psi and a required accuracy of 0.1 psi.

The pressure equalization valve provides a flow path for equalizing the atmospheric pressure prior to astronaut entry into the docking adapter. The valve is operated by depressing a button in the valve handle and rotating the handle. The valve is capable of being operated from either side of the hatch. A cap provides redundant sealing capability. This valve is also used on the airlock.

Qualification tests of the differential pressure gage assembly consisted of CCOH, temperature, life cycle, vibration, shock, vacuum, and pressure. A performance test, consisting of proof pressure, accuracy and leak check, was performed after each environmental test.

Each gage assembly received an acceptance test to demonstrate suitable quality, correct assembly, and required performance. The gages were visually and dimensionally inspected, checked for pressure indicating accuracy, submitted to a proof pressure integrity test, and subjected to a leak test. All tests were completed satisfactorily.

Qualification tests on the pressure equalization valve were successfully performed according to requirements set forth for the airlock. These tests consisted of the following environmental tests: High and low temperature, oxygen atmosphere, vacuum, vibration, shock, cycle test, limit load, proof pressure, ultimate load, burst pressure, salt, fog, and humidity. A performance test was conducted after each environmental test.

Each valve received an acceptance test to demonstrate suitable quality, correct assembly, and required performance. Each valve was visually and dimensionally inspected, checked for flow rate capacity, subjected to proof pressure, tested for internal and external leakage, and the detent spool was examined for centering adjustment. All valves tested met the acceptability requirements.

Design operating life of the equalization valve was 30 cycles for orbital usage. Only five cycles were actually accumulated for the three manned phases.

No problems were encountered with the differential pressure gage or the pressure equalization valve during the Skylab mission.

5. Mounting Provisions and Installations. The docking adapter provides structural support for various experiments and crew equipment. These include the M512 materials processing facility, the S009 experiment, the radio noise burst monitor, speaker intercoms, fire sensor, fire extinguisher, fans, and various items of crew and experiment equipment. All items are permanently attached to longerons or to intercostals suspended between longerons.

These support fittings and structure were all designed with a factor of safety of 3.0, and were consequently not tested. All these items performed without failure throughout the mission.

Additional equipment and experiments installed in the docking adapter are described in the following paragraphs.

a. Telescope mount control and display panel. This panel is mounted in bay 4-A of the docking adapter. It is supported on shock mounts from two beams that are located at vehicle stations 3445 and 3510 The shock mounts protect the panel from launch vibration excitation and reduce the environment for components in the panel to acceptable levels.

Structural supports for the panel were designed using a factor of safety of 3.0 to eliminate the requirement for testing.

b. Earth resources/observations hardware. Various types of structure are used to mount earth resources/observation equipment in the docking adapter. Base plates for S191 and S192 are installed in cutouts in the shell and became part of the primary structure. The S190 experiment is mounted on four fittings that attach to hard points on the docking adapter structure. Various electronic equipment is mounted on three trusses which are attached at frame-longeron intersections. The S194 antenna and electronics package are mounted externally on the L-band truss.

All earth resources/observation support structure was designed using a factor of safety of 3.0, and no testing was performed to qualify these items. The S191 and S192 base plates were installed in the static test article during performance of all structural tests. These plates were also installed in the docking adapter flight article before proof pressure and leakage tests of the module were conducted.

The earth resources/observation support structure supported all equipment during boost and throughout the mission without failure.

c. L-band truss. This truss consists of aluminum tubular members welded and bolted together. Additional frames, brackets, etc., are attached to the truss to provide interfaces for the antennas, L-band electronics, proton spectrometer, and the inverter lighting control assembly.

All truss members are wrapped with aluminized mylar tape to help maintain thermal balance.

The truss is configured to interface with the docking adapter at three points: One point is an existing lifting fitting at the cone-barrel joint located to one side of the Z axis. A second point is on the opposite side of the Z axis at the cone/barrel joint. The third attach point is to an adjustable link that attached to the axial docking port-cone intersection.

A factor of safety of 3.0 was used in designing the truss, and no testing was required.

The L-band truss supported all its equipment without failure.

d. Foot restraints. Two foot restraint platforms are provided in the docking adapter for astronaut support when operating experiments. The platforms were fabricated from standard astrogrid to interface with the astronauts shoes. The telescope mount control and display platform was designed to be used by two crewmen and is approximately 17 by 49 in. It is adjustable, with three using positions and a stowage position. The using positions provide a vertical adjustment of 12 in. This platform is used to support the telescope mount console body restraint operator's chair when in use. The earth resources/observation/M512 platform was designed for use by one crewman working either at the earth observation panel or the M512 experiment. Two mounting locations are provided and two platform orientations are possible. A third location is used for platform stowage during launch.

These foot restraints were designed for a specified ultimate concentrated load of 140 lb. Launch conditions were also critical for some components. When requirements were identified to use the foot restraints for launch stowage of the extravehicular activity hatch window cover, the S183 kick plate, and the scientific airlock windows, it was necessary to reduce the design environment in order to retain an adequate factor of safety. Vibroacoustic data were used to define new vibration load factors, and the required factor of safety of 3.0 was demonstrated by analysis.

Both foot restraints successfully withstood the launch environment and were used without problems throughout the Skylab mission. The telescope mount foot restraint was used to support the console body restraint only sparingly during the second and third manned phases. The earth resources/observation/M512 foot restraint was moved from its stowage location and was used in both the EREP and M512 positions.

e. Control and display console body restraint device. The restraint assembly was designed for use by the crewmen during operation of the control and display console. The design consisted of a base plate that attached to the floor grid pattern (astrogrid) at the base of the console. This attachment consists of two cleats that fit at the grid corners and two clamps that pin the base plate to the grid. A seat with variable height adjustments is installed on the base plate with one quick release pin. The position of the back rest is made variable by rotating it on hinge pins and then securing the desired position with quick release pins. A lap seat belt is also provided (Figure VF-10).

The assembly was designed for a maximum working load of 150 lb in the fore and aft direction normal to the face of the control and display console and a maximum working load of 120 lb in the side-to-side direction parallel to the face of the console.

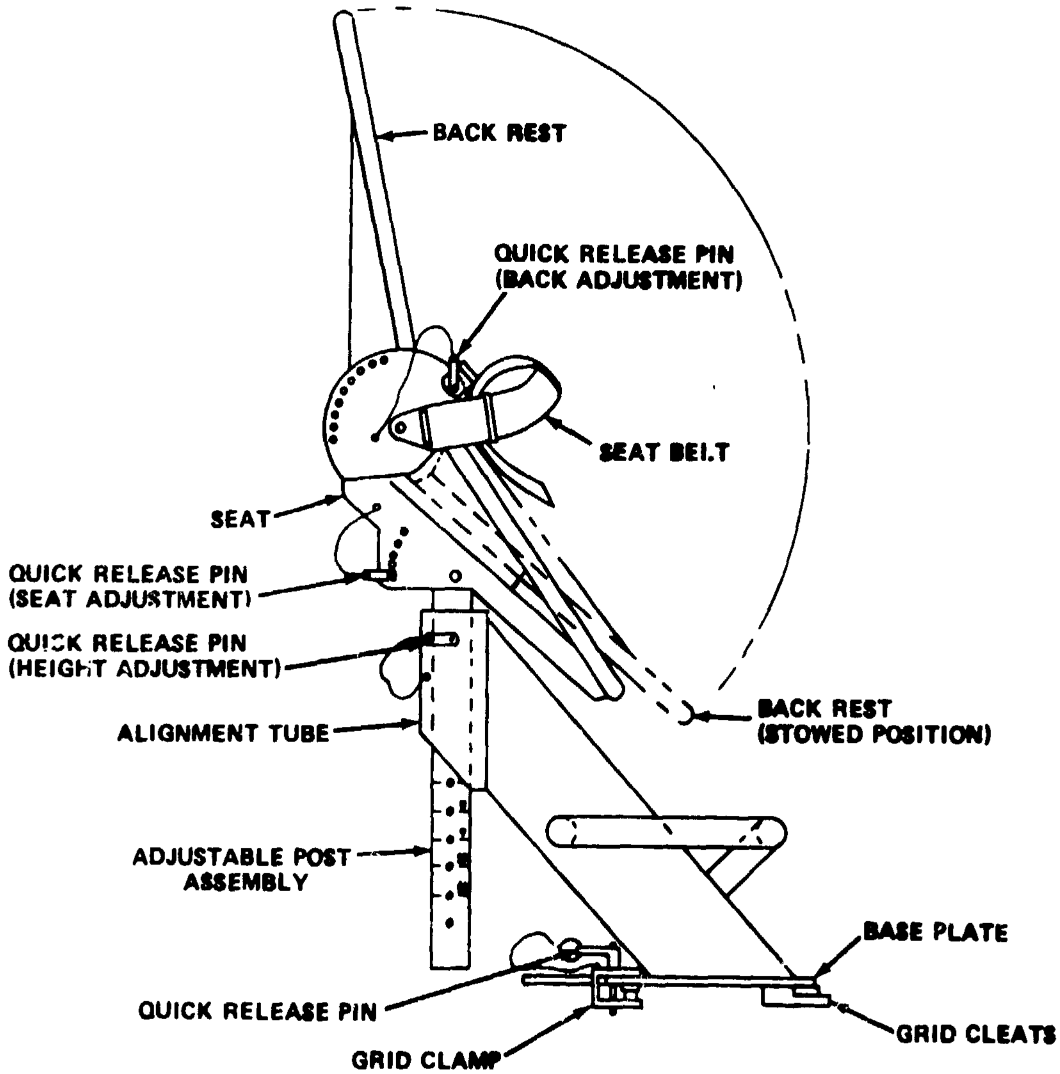


Figure VF-10. Control and Display Console Body Restraint

The assembly was stowed by installing it on the floor grid of the workshop with the seat in its lowest position and back rest folded over the seat. This provided the lowest center of gravity for the package and thus reduced the vibration and launch loads.

A qualification test item was vibration tested at MSFC to launch conditions, with the restraint assembly in the secured position, simulating the launch configuration. One of the welds between the back rest panel and the back frame cracked at a fillet weld during the first vibration test. The design was modified to add a reinforcement angle at this weld joint and the vibration test was repeated with no failures. Following the vibration test, the assembly was adjusted to all the possible positions with no problems and the assembly was considered qualified.

The body restraint was used by some crew members and not by others because they felt they would be more restricted at the control and display console if they used it. The use/non-use option was provided, through design, by making the restraint collapsible into a compact configuration when not in use. When used, the seat and back adjustment was optimized for the crew members so readjustment was unnecessary during each manned phase.

6. Stowage. The docking adapter provided stowage for various experiment and crew items used in the Skylab mission. Film vaults and stowage containers were used to stow many items. Some larger equipment, such as film cassette trees, fire extinguisher, and secondary oxygen pack, were stowed on brackets in the docking adapter. All such brackets were designed using a factor of safety of 3.0 to eliminate the requirement for testing.

a. Film vaults. Four film vaults are installed in the docking adapter to provide stowage for the telescope mount cameras and film, and miscellaneous items for the Skylab mission. The film vaults are of various sizes and wall thickness to meet physical and radiation requirements defined in their respective interface control documents.

The film vaults are located and supported at locations best suited for crew operation and to sustain launch loads. The vaults were fabricated from 6061-T6 aluminum. Doors attach to the basic box with a continuous piano hinge and lock in place for launch loads with expando pins. During activation by the first manned crew, the expando pins were replaced with pins. The doors are equipped with a friction device to control inertia forces on the door during crew operations in zero g.

The film vaults were designed using a factor of safety of 3.0 which elminated the requirement for design verification testing. Form, fit and functional testing was performed on all the film vaults to verify interface control document and crew interface requirements.

A typical film vault test was conducted on a simulated zero g flight to verify operability of the door friction device and stowed contents quick release supports.

Prototypes of film vaults #3 and #4 (the heaviest and lightest) were installed in the dynamic test article with prototype contents. After the vibroacoustic tests, the vaults and contents showed no degradation.

The film vaults performed satisfactorily during the Skylab mission with no anomalies reported. The second crew did state that a door (or doors) did not appear to have any restraining friction. The crew apparently did not find it necessary to use the adjustable friction devices that were provided for each door. A comment was made by the first crew during debriefing that the camera removal/installation and door operation performed better than during training on the ground.

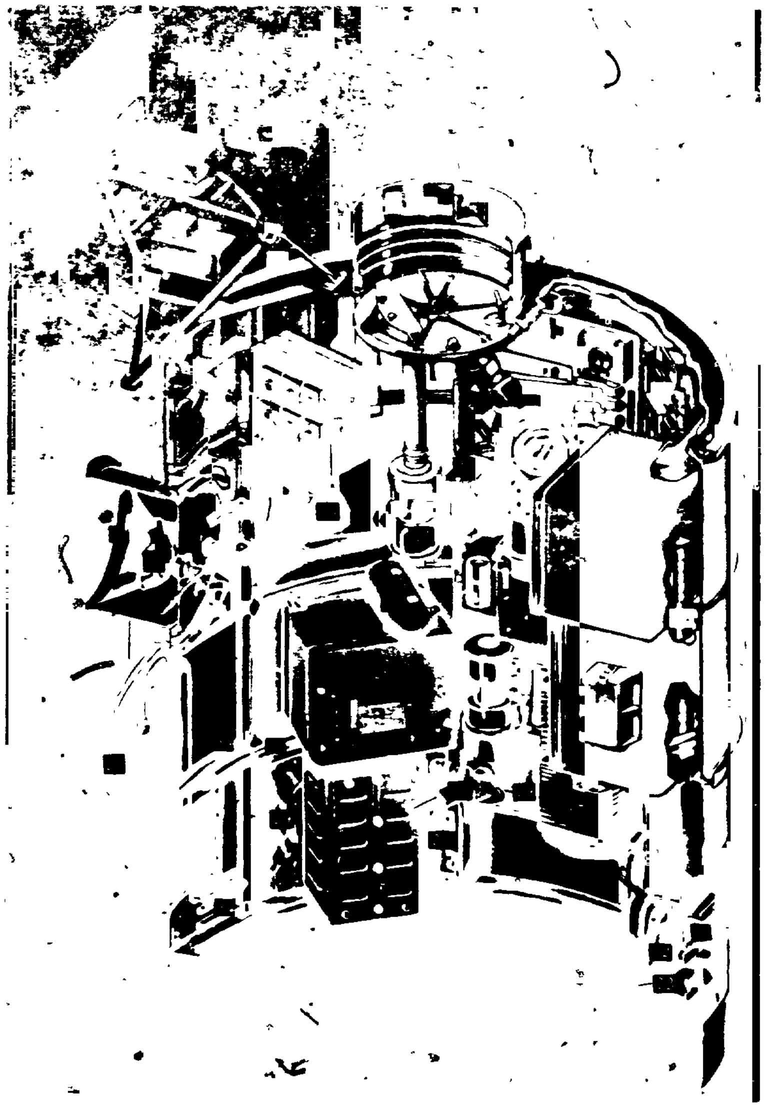
b. Stowage containers. Seven stowage containers are located in the docking adapter. These containers are used to store a variety of items such as CO_2 absorber canister, flight manuals, crew communication equipment, experiment support equipment, contingency tools, and in-flight maintenance tools and equipment.

The stowage containers are numbered in series according to their location, which aided the crew in finding a particular item. Each container has a decal listing the items and the quantities stowed inside. Locations of the containers and other docking adapter installations are shown in Figures VF-11 and VF-12.

The design and functional requirements for the stowage containers are:

- Containers and support structure must withstand launch loads; one-hand operation of container doors and removal of stowed items was desired.
- Design to a structural factor of safety of 3.0.
- Stowed items must be restrained in containers to prevent floating out in zero g.
- Restraint must be provided on doors to hold in any position.
- Good accessibility to containers and stowed items was desired.

All the containers were functionally tested by crew operations engineers to ensure operation of doors and fit of stowed items under



[illegible]

[illegible] trol [illegible] Container
[illegible] Stowed [illegible] Canister
[illegible] Cassette [illegible]
High and Low Power [illegible] Outlets
Miscellaneous Stowage [illegible]
Communication Headset
[illegible] Tube
Filter Bypass
Power & Control Umbilical
Film Return Bags
Shorting Plugs
Pointing Controller
Accutron Timer
Crewman [illegible] Mask

22 Fire Extinguisher
23 Radio Noise Burst Monitors
24 Environmental Control Air Interchange [illegible]
25 Experiment S054 Stowage Container [illegible]
26 Film Vault 2 Containing Experiment [illegible] Cameras and [illegible] Magazines [illegible] Astronomy)
27 TV Camera and Universal Mount
28 [illegible] Miscellaneous Stowage Container [illegible]
29 CM Condensation Blanket - Stowed
30 Wall Heater & Temperature Thermostat Control
31 Drogue Latch Set
32 CM/MDA Electrical Umbilical Connector
33 Electrical Cable Trays & Covers
34 Inner Pressure Shell & Primary Load Carrying Structure
35 External Shell Structure
Radiator Panels
Insulation Blankets
36 Micrometeoroid Panels
37 Docking Drogue and Radial Port Insulation Blanket

Figure VF-11. Docking Adapter Installations

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38. Environmental Control System Air Interchange Duct
39. Environmental Control System Area Fan
40. Window Heater Control Assembly
41. Speaker Intercom Assembly (EREP Work Station)
42. Earth Resources Experiment Viewfinder Tracking System and TV Optics (S191)
43. Earth Resources Experiment Infrared Spectrometer (S191)
44. Earth Resources Experiment Primary Tape Recorder
45. Earth Resources Experiment Control Console
46. Earth Resources Experiment Equipment Stowage Container (S190)
47. Earth Resources Experiment 10-Band Buffer Electronics
48. Earth Resources Experiment 10-Band Multispectral Scanner (S192)
49. Earth Resources Experiment Redundant Tape Recorder
50. Secondary Oxygen Pack and Mask Assembly - Stowed
51. Film Vault 3 Containing
    - Experiment S054 Camera and Magazine } Solar Astronomy
    - Experiment S052 Camera and Magazine } Solar Astronomy
    - Experiment S056 Camera and Magazine } Solar Astronomy
    - Experiment S052 and S056 Camera Handle Stowage
52. Inflight Maintenance Tools Container
53. Center Work Station Camera Cassette Tree Stowage and Cover
54. EREP/M512 Foot Restraint (Positioned for EREP C&D)
55. Apollo Telescope Mount Control Console Foot Restraint
56. Molecular Sieve Duct - Environmental Control System
57. Experiment S009 Cosmic Radiation Experiment (Nuclear Emulsion)
58. Vacuum Vent Panel (MDA & M512 Batteries Venting)
59. Docking Probe Stowage
60. Video Tape Recorder System
61. Carbon Dioxide Absorbers Container
62. Flight Data File
63. Backup Inverter-Lighting Control Assembly
64. Digital Address System
65. Apollo Telescope Mount Control Console - Solar Astronomy Experiment
66. Astronaut Hand Rails
67. Video Switch
68. TV Input Station
69. Utility Outlet
70. Speaker Intercom Assembly (ATM Console Work Station)
71. External Electrical Tunnel & Feed-Thru Cover
72. Docking Target
73. Running Lights
74. Debris Shields
75. EREP Coolant Bypass System & Control Valve

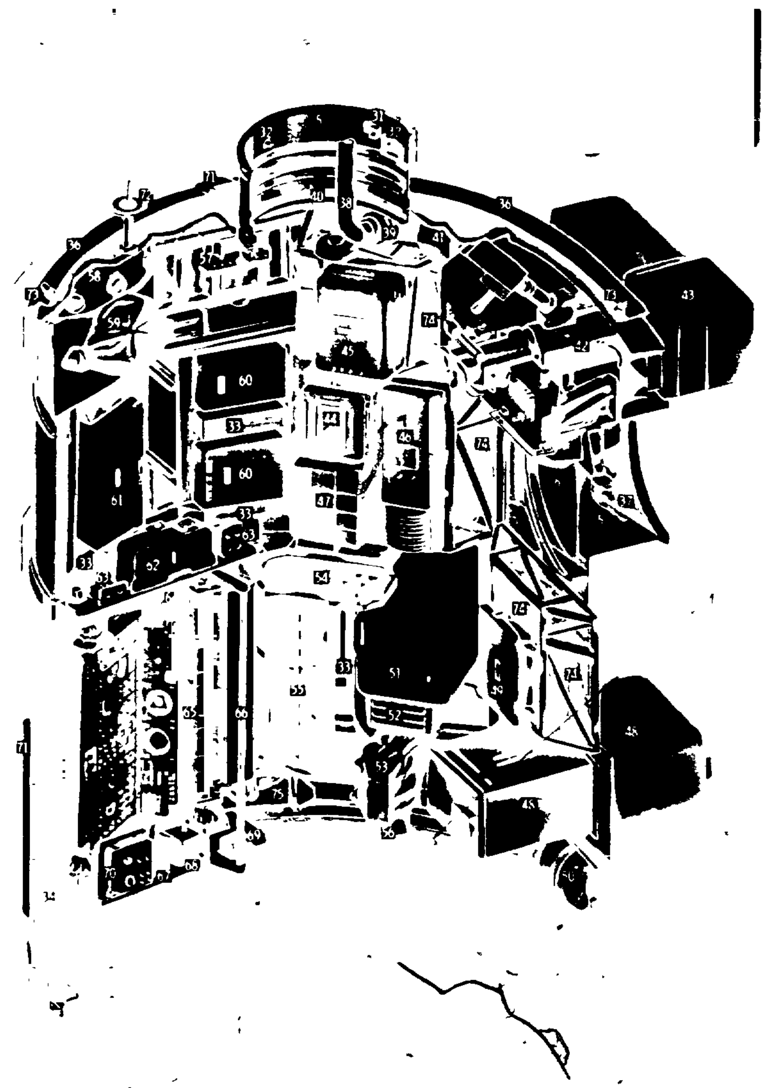

Figure VF-12. Docking Adapter Installations (Continued) simulated on-orbit operations. Removal and installation forces, restraint capabilities, and door operation were evaluated during several crew compartment fit and function reviews and altitude-chamber exercises. During altitude chamber tests, it was discovered that Mosite foam used for cushioning in the containers experienced growth when docking adapter internal pressure was reduced. This altered the dimension of stowage cavities. The problem was corrected by increasing the cavity size in the Mosite and revising the restraint system. No structural verification testing was required because of the large factor of safety used for the design.

The stowage container configurations, locations in the docking adapter, door arrangement, and methods of supporting and stowing the items in the containers were satisfactory for all applicable mission operations. The hardware performed during the mission without any problems.

7. Mechanical Systems. The docking adapter mechanical systems consist of the ventilation system, the telescope mount/earth resources/observation coolant system, the docking adapter ventilation system, and M512/M479 experiment vent system.

a. Ventilation system. The ventilation system design requirements consist of the following:

- Flow rate per cabin atmosphere hard duct (3 ducts)--55.7 cfm.
- Allowable pressure drop for cabin atmosphere ducts--0.0172 in. of water at 55.7 cfm.
- Flow rate for molecular sieve duct (two compressors)--62 $\pm$ 10 cfm.
- Allowable pressure drop for molecular sieve duct--0.035 in. of water at 62 cfm.
- Flow rate through the atmosphere interchange duct 100 to 170 scfm at 70 °F.
- Interface pressure--line pressure loss in command module portion of the atmosphere interchange duct not to exceed 0.07 inches of water at 150 scfm, 70 °F and 5 psia.
- Acoustic noise--no greater than 72.5 dB (sound pressure level) from all sources.

The docking adapter ventilation system consists of three fan/muffler assemblies, two adjustable diffusers to control air distribution in the docking adapter, and various ductwork to conduct air from the airlock to the docking adapter and from the docking adapter to the command and service module.

The docking adapter environmental control system ducts carry cooled atmosphere from the airlock structural transition section when the airlock environmental control system fans are on.

The docking adapter to command and service module fan/duct system introduce docking adapter ambient atmosphere to the docking adapter through the docking port tunnel.

The mol sieve duct introduces fresh (CO_2 and odors were scrubbed) atmosphere to the docking adapter. One or two airlock compressors can be used to deliver the conditioned air. The atmosphere can be diverted to the docking adapter or workshop depending on the damper position located in the structural transition section duct.

The atmosphere velocity at crew stations is controlled by placing one or both of the docking adapter cabin fans at high, low, or off settings and by adjustment of their attached diffusers. The diffusers establish the direction and shape of the existing atmospheric stream.

The docking adapter ventilation system is shown in Figure VF-13.

The development test for the flexible duct consisted of a flow and pressure drop test. The flow criteria being a pressure drop of less than 0.03 in. of water at a flow rate of 150 cfm. The data indicated a delta pressure of approximately 0.018 in. of water at 150 cfm, which was well under the 0.03 maximum requirement.

No problems occurred during development tests of the flexible duct assembly.

Qualification tests were run on two flexible ducts. The following tests were conducted: visual and dimensional inspection, temperature, altitude, storage and transportation, resilience, and vibration.

The two flexible ducts met all requirements of the qualification tests. There was no evidence of cracking, delamination, permanent deformation, deterioration, or physical damage as a result of the required tests.

Development testing of the muffler assemblies was conducted on a "slide tube" configuration that provided for a lever release of the fan. Development tests were conducted by MSFC which consisted of acoustic noise, contamination, flow, vibration and shock.

For qualification testing, the muffler assembly configuration was modified from the development configuration to a "hard" mounting which bolted the fan to the muffler bases. This design change was made to simplify the structure and to preclude tolerance and dynamic susceptability that was evident in the "slide tube" configuration.

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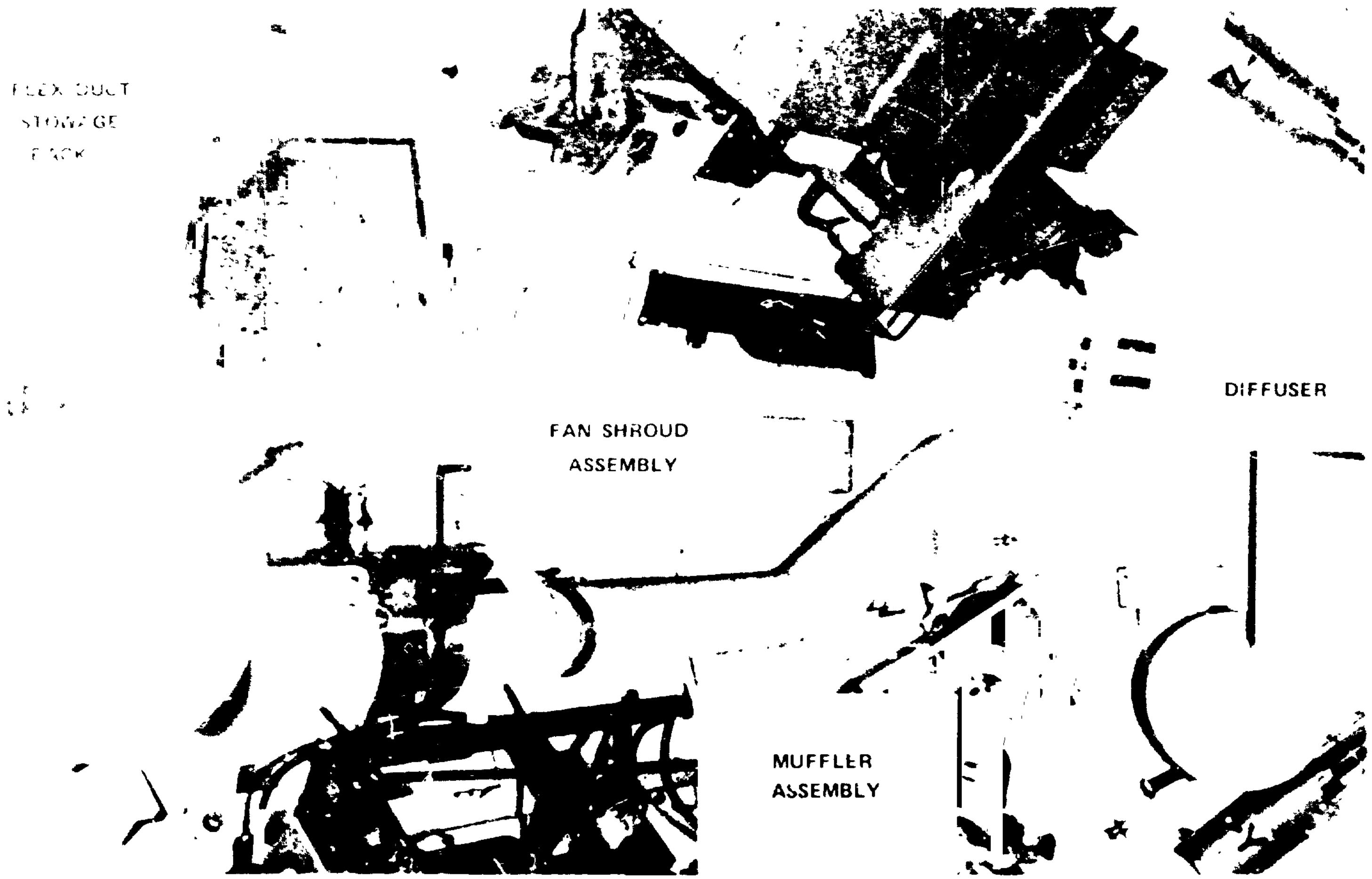


Figure VF-13. Docking Adapter Ventilation System

The qualification tests consisted of sine evaluation, vehicle dynamics and high and low level random vibration. A performance test was conducted prior to and after the vibration tests. The performance test consisted of an acoustic noise test. During the initial noise test, the muffler attenuation of fan noise did not meet the required level. The acoustic criteria were re-evaluated and new criteria were incorporated in the test specification. Testing was resumed. The test units successfully passed the tests.

Development tests of the diffuser were conducted to verify the capability of the diffuser to provide the required distribution of air flow in the docking adapter. This requirement specified that the air velocity range at the crew work stations be between 15 to 100 fpm. Initial development testing of the diffuser was conducted by MSFC. Flow tests were run at 5 and 14.7 psia. Based on the test results, it was concluded that a 5-in. neck diameter diffuser should be used for the docking adapter. A system air distribution test was conducted using an off-the-shelf diffuser of this size, modified to conform to structural and human engineering design requirements of the docking adapter. A mockup of the command and service module and airlock were mated with the engineering mockup of the docking adapter for the system air distribution test. The docking adapter and the airlock structural transition section were outfitted with the best available fidelity of internal hardware and experiment packages. The diffusers were installed and positioned in the flight configuration. A mapping of the docking adapter/structural transition section air velocity profile was made. The distribution characteristics were determined by observation of smoke cloud dispersion and by taking velocity measurements. The test results verified the system capability to maintain the air velocity requirements at the crew station.

The diffuser test consisted of a performance test, a vehicle dynamics low frequency sinusoidal test, a high level random vibration test, and a low level random vibration test. The performance test was conducted prior to and after the diffuser was tested in each axis of vibration. The performance test procedure was a manual adjustment of the control knob from one extreme position to the other.

A mechanical failure of the diffuser occurred during the high level random vibration tests. A failure analysis disclosed that the structural support hangers for the movable cone of the diffuser slipped out of their installed position. A modification was made to the diffuser design in which the hangers of the movable cone were welded to the interfacing detail part to form an integral assembly. A complete qualification retest of the diffuser was conducted without any subsequent failures.

The fan shroud qualification testing included vibration and shock tests of the fan assembly when mounted in the docking adapter muffler assembly. The fan assembly successfully passed these tests.

The crews evaluation of the performance of the ventilation system was; the air circulation was more than adequate, the screens on the inlet mufflers were an ideal size for collecting dust and debris, and the operation of the fan-muffler assemblies produced a low level acoustic noise output. It was very quiet. The fans operated as much as 1500-hr longer than the design life of 3360 hr because of unscheduled use as rate gyro-six-pack cooling fans and the extended third manned mission.

No problems existed with any of the system hardware.

b. Telescope mount control and display panel/earth resources observation coolant system. The telescope mount control and display panel earth observation/resources coolant system design requirements were specified in several documents because the system crossed several interfaces. The consolidated requirements are:

- Operating pressure--37.2 psia maximum.
- Inlet fluid temperature--49 to 78 °F.
- Fluid--high purity water plus additives.
- Flow rate--220 lb/hr, minimum
- Allowable pressure drop--6.75 psia at 220 lb/hr.
- Leakage--35 in^3 (from time of fill to end of 240-day mission).

The telescope mount control and display/earth resources/observation coolant system consists of the hard tubing, flexible lines, valves, and cold plates associated with conducting a flow of coolant to and from the telescope mount control and display panel, the earth resources/observation tape recorders, the earth resources/observation control and display panel, and the S192 electronics. Docking adapter mechanical components included in the system are a four-port manual selector valve and earth resources/observation flexible lines. The docking adapter system interfaces with the airlock system that contains the pumps, heat exchangers, and accumulator. Configuration

C - 3

of the components is shown in Figures VF-14 and VF-15.

Heat loads generated by components are transported by means of the coolant loop to the airlock coolant system from which heat is rejected to space by means of the airlock/docking adapter radiator system.

Fluid flow through the telescope mount control and display panel is controlled by positioning the four port manual selector valve. Flow control through the subsystem is accomplished by balancing flow orifices and the tape recorder selector valve.

The four-port selector valve is an adaptation of the Apollo glycol bypass valve. The Skylab design modification consisted of removing the electrical actuator and replacing it with a manually operated actuator. The fluid control portion of the two valves, including the pressure relief mechanism, is identical.

Development tests were performed primarily on the manual actuating mechanism. Life cycle tests were conducted to demonstrate capability to withstand the required number of operating cycles plus margin (475 wet cycles and 25 dry cycles). The actuating mechanism, including the position locks, performed all cycles without malfunction and only a slight marking was witnessed as indication of wear. Vibration testing was also conducted to verify that both the actuating mechanism and fluid control portion of the valve would not degrade under specified vibration conditions.

There was no structural degradation as a result of this test and the valve performed well during functional tests following vibration. There was no measurable internal leakage at operating pressures using volumetric displacement and nitrogen gas. Also, there was no degradation of the pressure relief mechanism as a result of the vibration testing. Relief pressures remained the same as pressures recorded prior to starting the test.

The four-port selector valve qualification tests consisted of physical inspection, proof pressure, over torque, ultimate torque, vibration, life cycle, and burst pressure. A performance test was conducted prior to and after each environmental test. There were no problems encountered during qualification testing.

At the completion of qualification testing, one of the two test units had no detectable internal leakage and the second unit had a leakage rate of 0.8 scc (GN_2) in 15 min. External leakages for the first and second units were 9.2×10^{-9} scc/sec (He) and 2.3×10^{-6} scc/sec (He), respectively. The maximum allowable external leak rate was 1×10^{-4} scc/sec (He).

Figure VF 14 Coolant Selector Valve

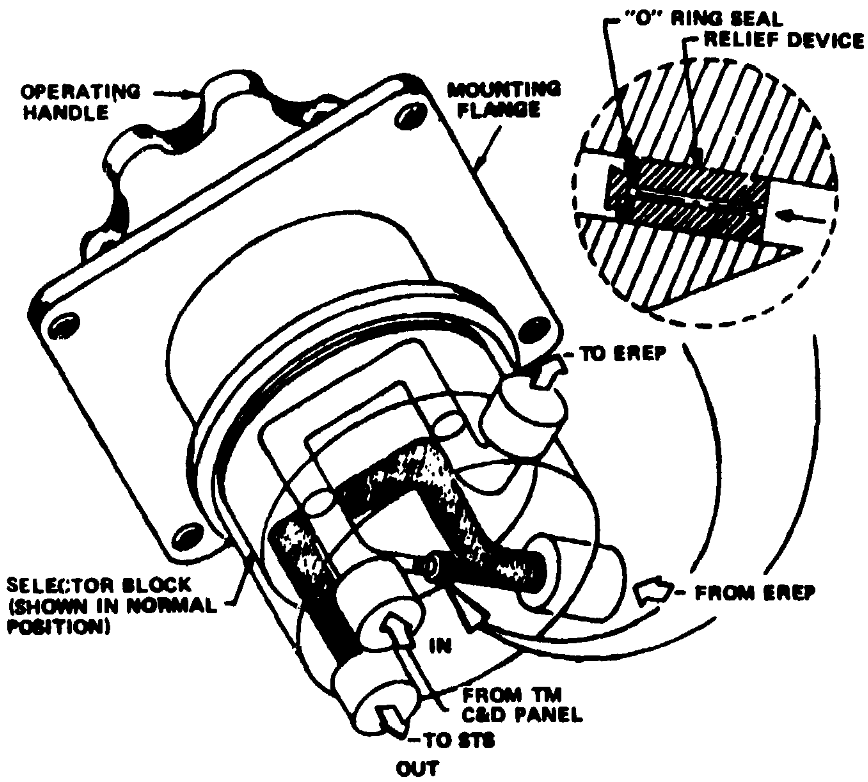


Figure VF-15. 4-Port Selector Valve

The qualification tests conducted on the telescope mount control and display panel flexible coolant lines were: pressure drop, first pressure fatigue, vibration, flexing, second pressure fatigue, and burst pressure. Performance tests consisting of proof pressure and a leakage test were conducted prior to and after each major test. There were no problems encountered during qualification testing.

At the completion of the qualification testing, the two test units were leak tested. A net leakage of 3.6×10^{-9} scc/sec (He) was measured on one unit and 1.0×10^{-6} scc/sec (He) was measured on the second. The maximum allowable leakage was 2.0×10^{-4} scc/sec (He).

Qualification of the earth observation coolant system flexible hoses was by similarity to several hoses qualified for use in the airlock. The sizes and configurations of the airlock hoses were sufficient to bracket the configurations selected for the docking adapter and the structural, environmental, and functional requirements either met or exceeded the docking requirements for the earth resources/observation coolant system hoses.

Development tests were not conducted on the earth resources/observation tape recorder selector valve because the design had been proven in previous development and qualification testing required by other NASA programs on similar valves.

Qualification tests consisted of a vibration test followed by visual inspection and leakage tests. There was no evidence of physical damage, water leakage, or change in handle position, as a result of the testing. Therefore, the tests were considered successful. The test unit was subjected to a high level random criteria of 10.4 g rms for 5 min in each of three axes.

Flow and pressure drop and leakage tests were the only complete coolant loop system tests performed on the flight article. No specific thermal loop flow tests were performed, since "off module" tests of experiment assemblies showed that thermal performance was satisfactory when the coolant flow and inlet temperature was within specified limits. Flow balance and distribution was demonstrated by power operation of the earth resources/observation modules with tubing surface temperature measurements during systems tests at MDAC-E.

Leakage tests "on module" were performed to a "volumetrics" measurement limit of 1.0×10^{-3} scc/sec of N_2. The volumetrics test method verified all joints (144) in the coolant system for the airlock interface. The "volumetrics" instrument showed a high sensitivity to temperature, such that the entire system had to be wrapped with a super insulation blanket, all lights turned off inside the docking adapter, and all personnel restricted from the test area during conduct of system leakage.

In order to achieve an acceptable "volumetrics" measurement, a helium mass spectrometer probe (sniffer) test to a maximum reading of 2.5 x 10 scc/sec was imposed as the system build was in process. During the system checkout, this probe test requirement was relaxed to 5 x 10^{-7} scc/sec helium, which still allowed sufficient volumetrics test margin.

Design cycle life of the four-port selector valve was 375 cycles. During the three manned missions, a total of 75 cycles were accumulated. The tape recorder selector valve has a design cycle life of 470 cycles. During the Skylab mission a total of 6 cycles were accumulated on this valve. The docking adapter coolant system operated in a normal manner with no anomalies. No leakage was evident during system operation.

c. Docking adapter vent System - The docking adapter vent system design requirements are: The docking adapter shall be equipped with series redundant, remotely-operated vent valves. The valves shall be sized to ensure that the maximum shell pressure will not exceed 6.2 psid during launch of the first unmanned phase. The vent valves shall be provided with a plug which shall be installed by the crew. Venting studies were performed to predict the internal docking adapter pressure during launch. The maximum pressure predicted by analysis was 5.3 psid. The actual results during launch/ascent phase were 5.25 psid.

The docking adapter vent system consists of two 4-in. motor operated vent valves mounted in series, one sealing device, and one stowage fitting. The vent valves provide a means of venting the docking adapter and airlock during prelaunch, launch, and ascent. The valves are opened prior to launch and closed during ascent, via instrument unit command, to maintain a positive pressure within the airlock/docking adapter.

The vent sealing device provides a positive sealing capability of the vent valves during orbital operation. The sealing device is installed by the astronauts upon their initial entry into the docking adapter during the first manned phase. The vent system is illustrated in Figures VF-16 and VF-17.

Examples of development tests conducted on the 4-in. vent valve (Figure VF-18) are: internal and external leakage, electrical checks, high and low temperature operation, life cycle and vibration. This unit successfully completed the tests with results that either met or exceeded the specification requirements.

The qualification testing consisted of the following environmental tests: CCOH, vibration, shock, vacuum storage, explosive atmosphere, burst pressure, and life cycle. A performance test was conducted after each environmental test. The problems encountered during qualification testing are summarized below.

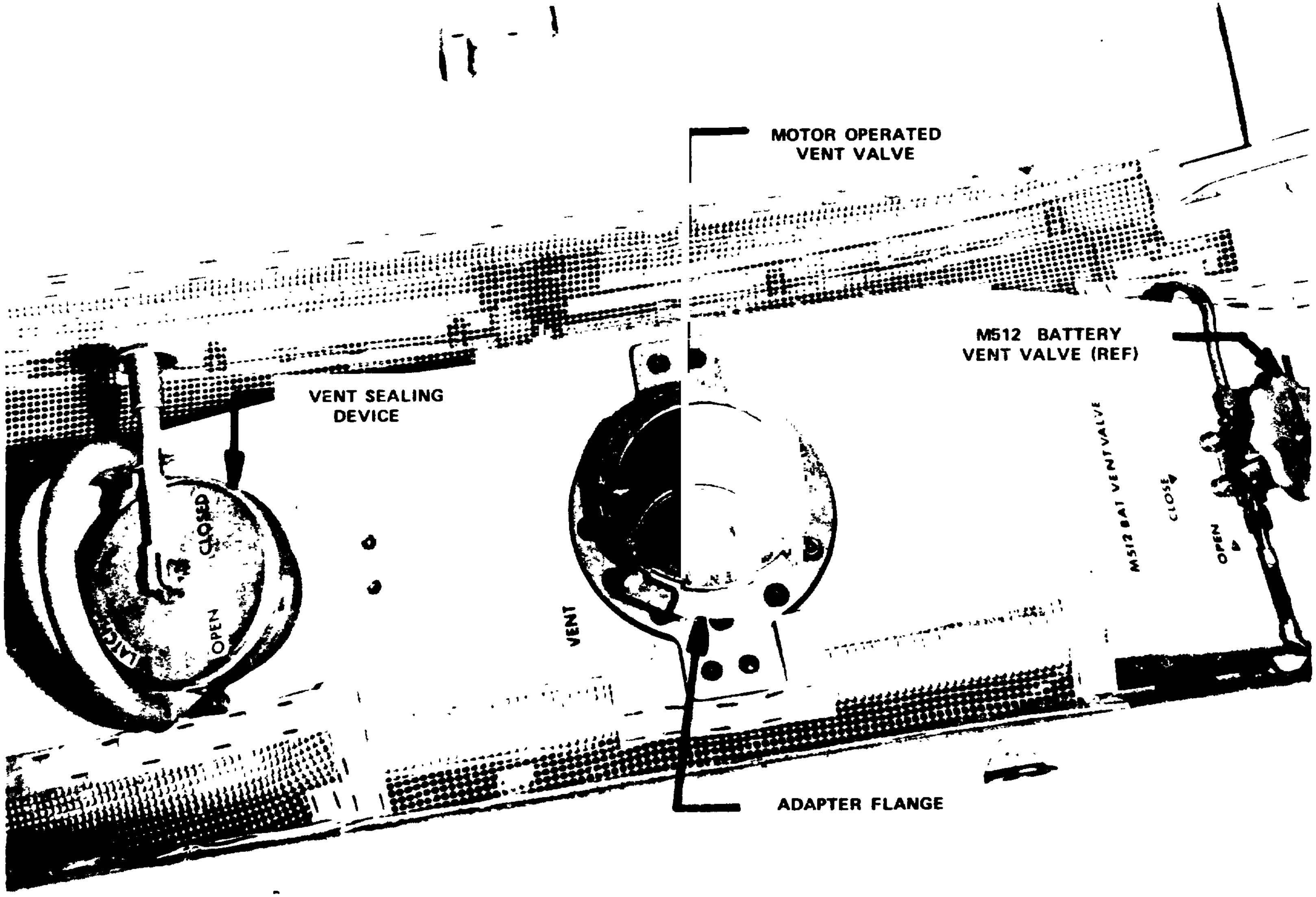


Figure VF-16. Vent System

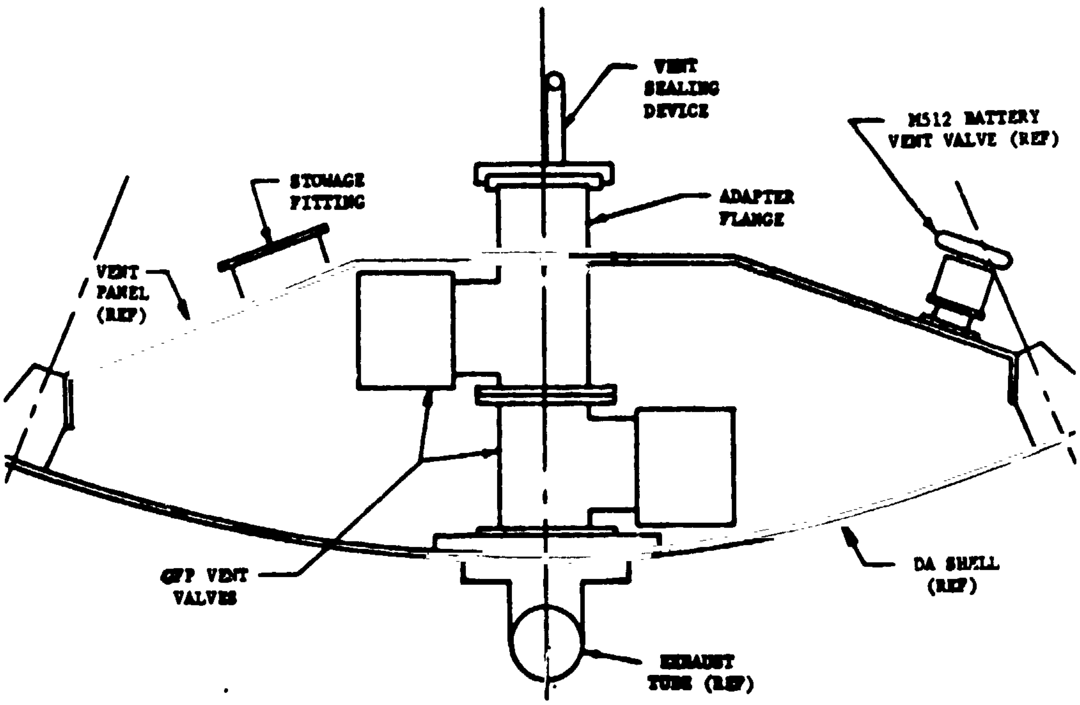


Figure VF-17. Vent Panel

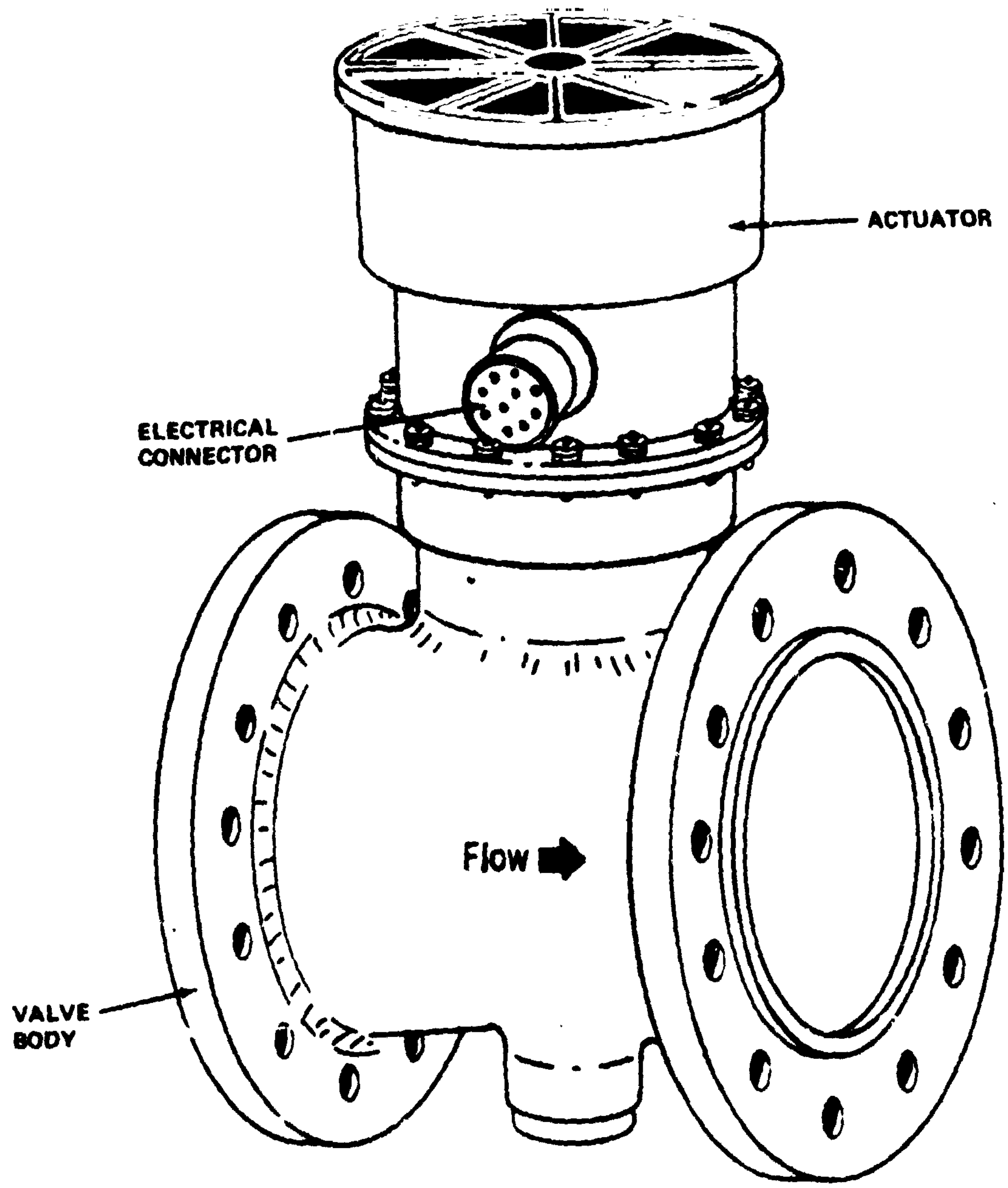


Figure VF-18. 4-inch Vent Valve Assembly

During functional test, prior to qualification test, with external pressure of 10^{-4} torr, the valve failed to operate when 22 Vdc power was applied. The failure was attributed to the deflection of the end plate resulting from the 14.7 psi differential pressure, therby causing the brake to hang up. The corrective action taken was to reduce the internal pressure of the hermetically sealed actuator to 7.0 $\pm$1.0 psia, thus reducing the deflection of the end plate.

During high level random vibration testing in the "Y" axis, one of the test units exceeded the response time requirements (14.4 sec instead of the required 8-sec maximum). During the same test, a second test unit had both "open" and "close" indicator lights on simultaneously. Fixture evaluation revealed an amplification of input leveled at the secondary mounting bracket of the vibration fixture of approximately 9 to 1 over the input level. The specimen failure was due to overtest. The fixture was redesigned and "Y" and "Z" axis vibration tests rerun. Both units passed the post-vibration functional test requirements.

Delta qualification tests were conducted on the two test units. These delta tests consisted of vibration fixture evaluation, functional test, and vibration tests in X, Y, Z axes. During fixture evaluation it was found that the control accelerometers for the previous qualification tests were not located near the valve mounting flange, therby creating an undertest condition. The accelerometers were relocated in order to simulate a realistic test during succeeding vibration tests. During subsequent functional tests, one of the two test units failed to operate. Valve failure was attributed to overtesting resulting from the excessive number of tests conducted during qualification and fixture evaluation test programs. This valve was used as a dummy mass for succeeding vibration tests. During Z axis high level random vibration the test unit failed to cycle properly. The unit had cycled properly subsequent to the vibration environment. The requirement to cycle the valve during vibration was re-evaluated. It was determined that the vibration levels were negligible when the valve had to cycle closed after launch, therefore the requirement was changed to cycle subsequent to vibration. Vibration testing in each of the three axis was then completed successfully.

Qualification testing was performed on the vent outlet sealing device, stowage fitting, and adapter flange. The qualification tests consisted of the following environmental and functional tests: handle locking force, proof pressure, leakage, previbration life cycle, vibration - sine evaluation and random, post-vibration life cycle and thermal vacuum. All the above tests were conducted with the sealing device mated to the adapter flange except for vibration and post-vibration life cycle during which the sealing device was mated to the stowage fitting.

The problems encountered during qualification testing are summarized below.

During the life cycle test, the handle locking pin fell out of the handle when the retaining ring, which retains the locking pin in place, broke during the latching portion of a cycle. Subsequent examination of the failed part revealed a design deficiency wherein the retaining ring could catch in the latching slot of the handle. Redesign of the unit changed the locking pin assembly to a press fit plus a locking nut design instead of a retaining snap ring. Retest of the redesigned unit was completed successfully.

Following the completion of previbration life cycle testing, aluminum particles were discovered on the sealing device O-ring. The particles caused no failure of the tested units but an investigation revealed that the particles were generated, during normal installation and removal, by contact between the aluminum sealing device and a detent spring tab used on the adapter flange. The adapter flange was revised to change the spring tab from steel to aluminum. Subsequent post-vibration life cycle testing resulted in only normal wear of the two parts and no generation of particles. All other testing was completed with no other problems encountered.

The two motor operated vent valves were opened at T-5 hr, 15 min, according to the normal countdown procedure for the first unmanned launch phase and were commanded closed at T + 280 sec. The valves closed in 6.2 sec. The specified closing time was 8-sec maximum. The maximum docking adapter shell pressure achieved during ascent was 5.25 psid, well below the specified limit of 6.2 psi. The vent sealing device was installed during the activation.

Design cycle life of the vent valves was 10 cycles. Only one cycle was accumulated during the Skylab missions. No problems were encounterd with the docking adapter vent system during the mission.

d. M512/M479 experiment vent systems - The M512/M479 experiment chamber vent design requirements were:

- The system shall include manual redundant valves.
- One valve shall be located on or near the docking adapter bulkhead, the second valve shall be located at the experiment vacuum chamber.
- The valve system shall be capable of providing a variable orifice system.

The M512 battery vent design requirements were:

- The battery vent shall include a 1/4 in. shutoff valve and the interconnecting system tubing.
- Battery vent allowable leakage of 1×10^{-4} scc/sec N_2 at 15 psid internal pressure and 1×10^{-3} scc/sec N_2 at 5 psi external pressure.
- Chamber vent system allowable leakage of 1.85×10^{-4} scc/sec helium at 1×10^{-5} torr and 5×10^{-4} scc/sec N_2 at 20 psid.

The M512/M479 experiment chamber vent system provides a conduction path from the experiment chamber overboard to space. It consists of two, series mounted 4-in. manually operated valves separated by a metal bellows assembly with a short section of hard duct penetrating the docking adapter shell. The vent system has the capability of providing a conduction path for venting experiment contaminants overboard as required.

The M512/M479 battery vent system incorporates a redundant manually operated valve for vent or shutoff capability of the battery case. The 1/4-in. valve is mounted on the docking adapter vent panel. The M512/M479 experiment vent system is shown in Figures VF-19 and VF-20.

Development tests on the M512/M479 vent valve were limited to bearing lubrication evaluation and valve shaft bearing load tests under vacuum conditions. Other tests were not conducted because of previous test experience on similar valve configurations developed for the Titan propellant systems.

Three bearings were utilized in the test. Two were lubricated with Vac-Kote grease and one was lubricated with Microseal 200-1. The bearings were mounted in test fixtures designed to load the bearings radially from 0 to 500 lb when cycled, simulating vent valve operation.

Tare torque values were obtained prior to starting the test. The test items were then placed in a vacuum chamber and maintained at a pressure of 1×10^{-6} torr or less for 100 hr. After the 100-hr soak and while maintaining the vacuum chamber pressure at 1×10^{-6} torr or less, the bearings were load cycled 1000 times with torque measurement made every 20th cycle.

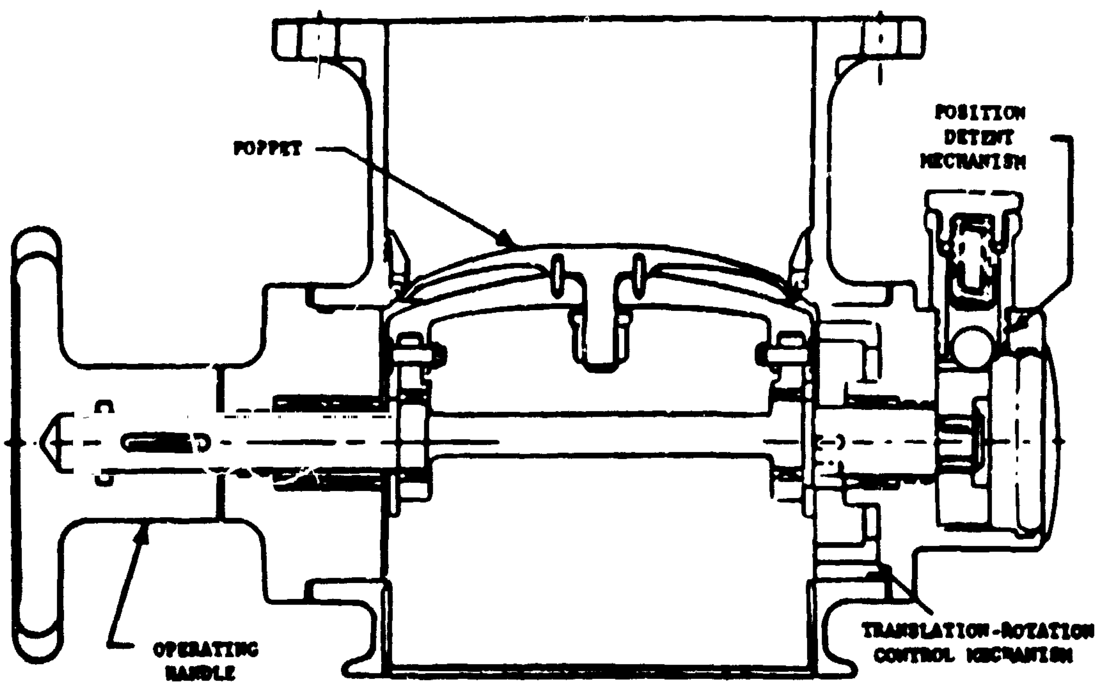


Figure VF-19. 4-Inch Chamber Vent Valve

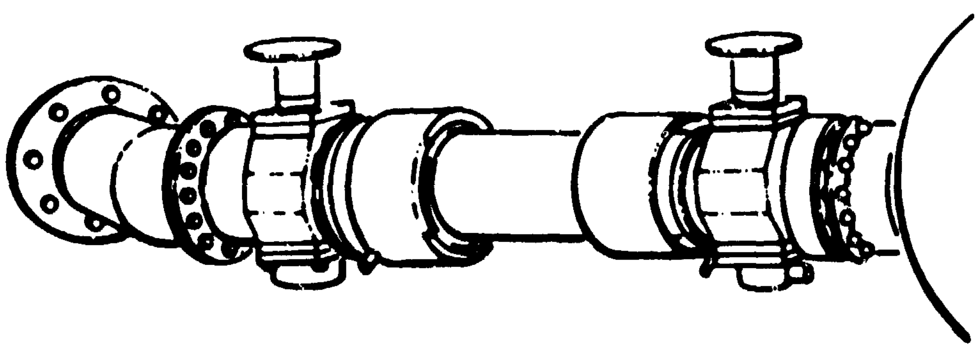


Figure VF-20. M512/M479 Vent System

The results indicated that both lubricants performed satisfactorily but Vac-Kote provided the least increase in torque and the smoothest operation during load cycling. Vac-Kote was selected for use in the production valve.

The valve qualification tests consisted of the following environmental tests: vibration, consisting of sine evaluation, vehicle dynamics, high and low level random; shock; thermal vacuum; temperature, altitude, storage and transportation; burst pressure; vacuum storage; life cycle; and CCOH. A functional test was conducted prior to and after each environmental test. Two problem areas were encountered during qualification testing. These problems are summarized below.

During vibration testing (the first environmental test following initial performance testing), the operating handle became loose and began rattling. The test was stopped and the handle setscrew examined. It was found to be peened and subsequent examination revealed the setscrew material to be too soft for the intended usage. The setscrew design was modified to alleviate the peening problem and also to provide positive handle positioning even if the setscrew became loose. The valve was then placed back into vibration test with the entire test being rerun.

During performance of the external leak portion of the functional test, following the CCOH test, the leakage exceeded the allowable level. Examination of the valve during failure analysis revealed no evidence which would indicate leakage was caused by exposure to the CCOH test. Failure was attributed to contamination of a static O-ring seal located between the operating handle adapter and valve body. The contamination of the seal occurred during the original build cycle of the valve. The contaminated seals were replaced and the valve placed back into qualification test at the point testing was terminated (functional test following CCOH) since the CCOH test was not considered a contributing factor in the failure. The entire functional test was completed without further difficulty.

During performance of the qualification test program, each of the two valves tested was manually cycled 1000 times. Three hundred of the cycles were conducted at pressures less than 1×10^{-8} torr. At the completion of all testing, the units still operated within the allowable operating torque limits of 40 in.-lb and the maximum internal leakage of the units was 5.4×10^{-7} scc/sec helium with 2.67×10^{-4} scc/sec allowable and maximum external leakage of the units was 1.02×10^{-8} scc/sec helium with 1×10^{-7} scc/sec allowable.

Qualification test of the bellows vent line consisted of the following environmental tests: proof pressure, spring rate, leakage, vibration, life cycle, and burst pressure. A proof pressure test, leakage test, and inspection of welds was performed before and after the vibration and life cycle tests.

The qualification tests were all successfully accomplished, without difficulty, on both bellows tested. The two bellows passed the internal leakage rate of 1 x 10^{-8} scc/sec (max.), the 1000 cycles test, the proof pressure test of 32 psig for 3 min, and the internal burst pressure of 52 psig for 1 min. One bellows was subjected to an external pressure test (the tube collapsed at 60 psig). The other bellows was subjected to an internal rupture test (at 300 psig, the bellows did not rupture), only the convolutes of the bellows "squirmed" to a permanent set.

Development tests of the M512 battery vent valve (1/4 in.) consisted of the following tests: weight, proof pressure, external leakage, operating torque, internal leakage, life cycling, pressure drop, vibration, shear torque, and burst pressure. The unit completed the development tests with results that met or exceeded the specification requirements.

Subsequent to development testing, the vibration levels were revised and it was determined that the Vespel seal material was incompatible with potassium hydroxide (KOH) vapors given off by the M512/M479 battery. Consequently, the design specification was revised to reflect the new vibration levels and the valve seat material was changed from Vespel to Kel-F.

The qualification consisted of the following tests: salt spray, pressure drop, temperature, life cycle, vibration (closed), vibration (open), thermal vacuum, flow versus handle position, shear torque, and burst pressure. Subsequent to each vibration test, a functional test consisting of proof pressure, internal leakage, and external leakage was performed.

The only anomaly occurring in qualification testing was excessive leakage attributed to frost buildup on the valve seat during a post-vibration functional test. The test personnel were cautioned to follow specific evacuation and purge times prior to performing a test.

Three units successfully completed qualification testing with results that met or exceeded the design specification criteria.

During a docking adapter flight article system leakage test, anomalies occurred on the M512/M479 work chamber vent valves and on the battery vent valve.

The purpose of the test was to verify that the internal leakage of each of the two work chamber vent valves had remained within the allowable range, that the external leakage of the total chamber vent system was within the allowable range, and to verify that the battery vent line and valve leakage rates were within the allowable range.

During performance of the system leak checks, valve #1 (at the work chamber) had no detectable leakage using the volumetrics leak detector and valve #2 (at vent elbow) had a leak rate too large to measure. The allowable rate is less than 1×10^{-4} scc/sec (GN_2).

Valve #2 was removed for failure analysis. Since there was obvious contamination on the rejected valve seat, the bellows assembly was also rejected for suspected contamination. Both rejected items were replaced in the vent system. Retest of Valve #2 (replacement item) resulted in no indicated leakage.

External leakage of the work chamber and vent line is measured by evacuating the chamber and vent line through a CEC helium leak detector, by externally bagging the chamber and vent line, and by filling the bag with helium. The measured leak rate for the total chamber vent system was 1.4×10^{-8} scc/sec (He) and the allowable rate is less than 3.7×10^{-6} scc/sec (He).

Leakage verification of the M512/M479 battery vent valve resulted in rejection of the vent valve for excessive internal leakage. The measured leak rate was 1.06 scc/sec (GN_2) and the allowable leak rate was 0.4 scc/sec (GN_2). The valve was rejected and failure analysis conducted. The results of the analysis revealed that the excessive leakage was caused by contamination imbedded in the valve seat. The rejected valve was replaced in the system with another valve and the leak test rerun. The re-test was successful with a measured leak rate of 0.47×10^{-7} scc/sec (GN_2).

Design cycle life of the 4-in. vent valve was 100 cycles; 69 cycles were accumulated during flight. Cycle life of the 1/4-in. battery vent valve was 10 cycles; one cycle was accumulated during flight.

The M512 vent system operated properly during all Skylab missions. No leakage or hardware problems were encountered.

The chamber pressure was maintained at less than 1×10^{-4} torr as required during all experiment operations.

8. Rate Gyro 6-Pack

During the first manned phases and the second unmanned period, there were indications that the Skylab rate gyros located in the telescope mount were not performing well and it was speculated that their environment could be the problem. A recommendation was made by the technical area responsible for the gyros, Astrionics Laboratory,(MSFC), to install a new set of rate gyros somewhere inside the Skylab that would replace the malfunctioning rate gyros. The telescope mount rate gyro system consisted of two gyros in each of the three axes or a total of six; thus, the name "Rate Gyro 6-pack".

The most feasible installation location for the gyro pack was determined to be in the docking adapter. The specific area chosen was the area reserved for the NRL film retrieval tree launch/mission stowage. The tree could be stowed elsewhere. This location in the docking adapter was selected because: (1) it was on one of the axes, (2) there were four hard point locations for attachment to the main docking adapter structure, (3) the installation required no special tools, (4) the gyro package would have minimal effect on the other system, and (5) a power connector was adjacent to the location, thus cable routing from the gyro pack to the C&D console could be made with minimum difficulty.

Responsibility for the design of the six-pack primary base mounting assembly was assigned to the S&M MSG. One axis system (2 gyros) would be installed on this plate. The other two axis systems (4 gyros) would be mounted on another plate assembled to the primary base plate, using alignment pins and captive fasteners. The design responsibility for these other two axis systems was assigned to the Astrionics Laboratory.

One of the early requirements was to make the base plate adjustable in three axes to provide precise alignment. A preliminary design was made for this configuration consisting of a firm base; two intermediate plates, which provided for two axes adjustment; and a gyro mounting plate, which provided the third axis adjustment. This design would require a considerable amount of operational time to install. An additional requirement was that the installed assembly have a natural frequency of greater than 10 Hz. This meant that the adjustable components must necessarily be heavy which led to making a tolerance study of the "worst case" misalignments to be encountered from the structural design of the docking adapter mounting provisions and a non-adjustable base plate. The results of this study indicated that the "worst case" misalignment could be 0°42' in the worst axis. The Astrionics Laboratory indicated that a 1° misalignment could be tolerated. Based on this study, the six-pack rate gyro base was made a non-adjustable plate.

The final configuration for the base plate assembly (Figure VF-21) consists of a 1-in. thick aluminum plate milled to 0.125 in. in the low stress area. Two six-pack alignment lugs/pins are provided for insertion into two guide pin holes on the existing docking adapter structure. Also, two clevis fittings are provided to secure the base to the docking adapter structure and the two associated "expando" pins, which are tethered to the base plate, are also provided. A compression screw with a hand grip knob is threaded through the center of the base plate. When the compression screw is tightened against the docking adapter structure, it takes up any clearance of the guide pins in their holes and makes the base plat fit securely. Two of the rate gyros ("Y" axis) are attached directly to the base plat and provisions were made to align and assemble the additional four-gyro ("Z" and "X") mounting plate assembly to the base plate.

To determine its natural frequency, the complete six-pack rate gyro was assembled using high fidelity models for the gyros, then installed on a test fixture and subjected to a sine test. The test fixture had been previously used for vibration tests on the NRL cassette tree. The lowest resonant frequency was found to be 22 Hz, which was well above the 10 Hz minimum requirement. It also agreed closely with the 24 Hz analytical prediction.

The six-pack assembly was subsequently fit checked in the 1 g docking adapter mockup at MFSC and in the docking adapter backup unit at St. Louis. No problems were encountered.

The base plate with two gyros attached and the four gyro adapter assembly were stowed separately under the couches in the command module for launch with the second crew.

On mission day 3 of the second manned phase, the crew assembled the gyro mount in the docking adapter with no difficulty. On mission day 28, during extravehicular activity, the crew assembled the electrical cabling, and the six-pack gyros were switched "on". They then became operational for the remainder of the Skylab mission.

9. Conclusions and Recommendations

All docking adapter mechanical hardware performed satisfactorily during all manned and unmanned phases of the Skylab mission. The design of the pressure equalization valve allowed it to be used for an unscheduled atmosphere sampling during the first manned phase. The M512 chamber vent valves also provided an excellent variable area flow and were unaffected by contamination generated by experiment operation. It is recommended that these valve designs, in particular, be used on any future applications of this kind. The docking adapter muffler screens, mesh size 0.14 by 0.14 in., was a good size to collect debris and provided a convenient area for cleaning during housekeeping activation.

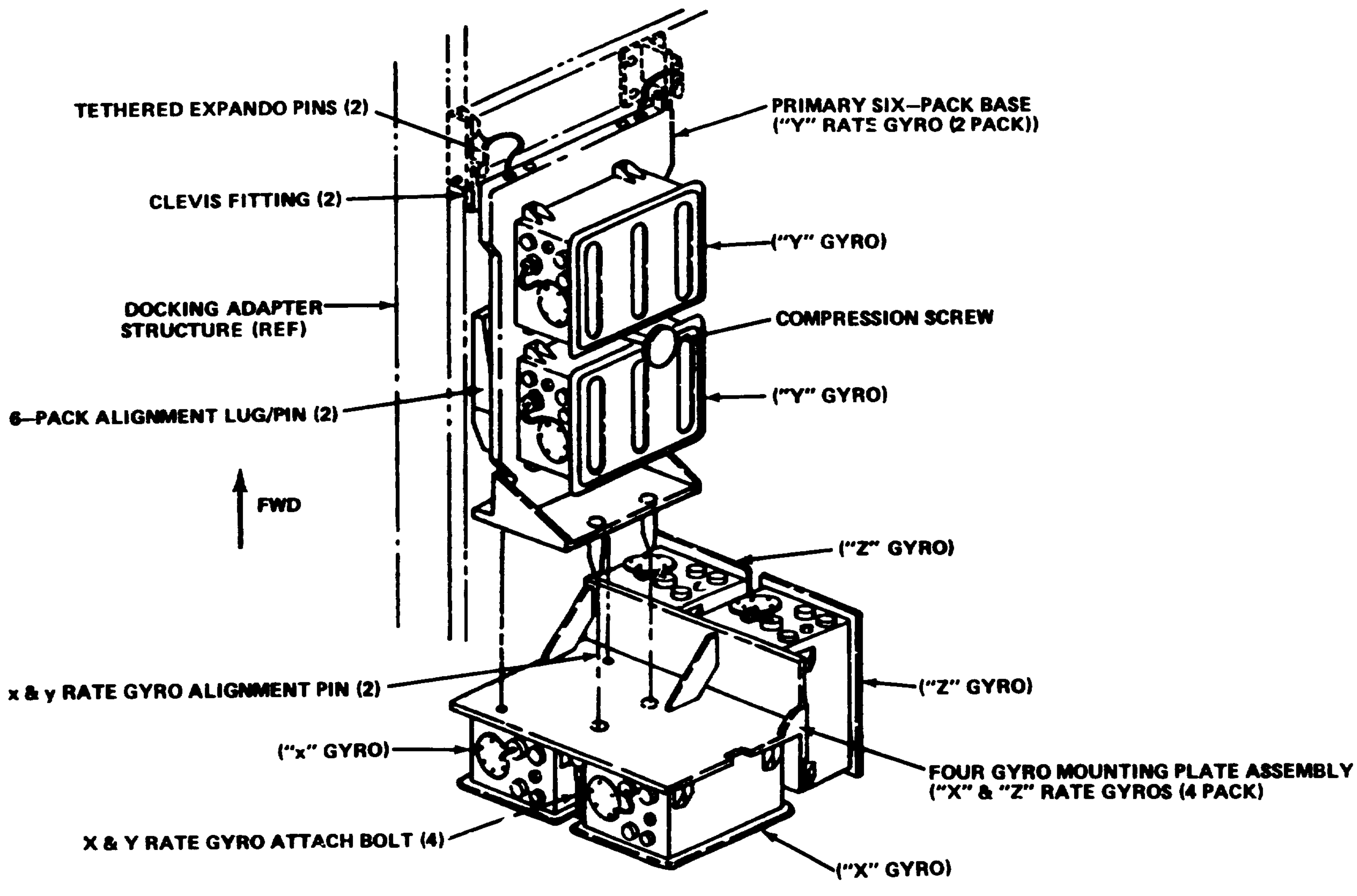


Figure VF-21. Six-pack Rate GYRO Assembly

Although all systems met all on-orbit requirements, there were problems encountered during ground testing of the telescope mount coolant system and hatch pressure gage; therefore, the following recommendations are made:

1) All metals used for construction of active coolant systems should belong to the same family group of dissimilar metals; this would eliminate unnecessary discussion and testing as to the effects of corrosion, and 2) if a visual mechanical pressure gage is required, one must pay close attention tc the shock environment and the accuracy requirement; if the maximum shock is on the order of 1500 g's, then the accuracy requirement should be at least 0.1 psid; this would allow a gage to be designed with a greater pressure range and, therefore, a stiffer bourdon tube which would be less susceptible to gravity effects and shock testing practices.

The mechanical actuator and latch for the S190 window cover functioned reliably throughout the mission. The second Skylab crew remarked that the mechanism operated easily. The use of such mechanical actuators is recommended for future vehicle designs as a simple and reliable means for controlling external components.

The importance of screening structural glass windows to detect invisible flaws was discovered in the process of designing the S190 window. Consequently, all windows in the docking adapter were flaw screened to eliminate any glass containing flaws that might grow to a critical size during the Skylab mission. The preferred method for flaw screening pressurized windows is pressure testing to simulate the operating stress distribution. Thermal shock testing had also been used for flaw screening, but it overstresses the edges of the window and may result in unnecessary test failures. All docking adapter windows functioned normally during the Skylab missions.

Another lesson learned in designing the S190 window was that structural windows needed prelaunch protection to prevent damage following acceptance testing. Such damage had occurred on earlier programs. In the docking adapter program, all windows were supplied with protective covers from time of installation until just before launch. Removal of the covers to perform tests or maintenance was controlled by procedures. Consequently, no docking adapter windows were damaged after installation.

Some of the blind nuts used to attach equipment to the longerons caused problems during manufacture of the docking adapter. The reason for this was removal and replacement of some items causing unexpected re-use of the nuts which exceeded their capabilities. This resulted in galling in the self-locking threads, and occasional bolt failure. Rework of the failed nuts involved careful controls to prevent contamination by chips and loose parts, which complicated the manufacturing process. It is recommended that all equipment that may be replaced prior to launch be attached with easily replaceable hardware.

G. Workshop

1. Introduction. The workshop is located aft of the instrument unit and interfaces with the booster vehicle. The workshop contains the crew living quarters, provisions, areas for food management and waste management, experiment storage, and work areas. The exterior is fitted with an aluminum meteoroid shield, solar array system, radiator for the refrigeration system, thruster attitude control system, and the pneumatic control system. The major structural assemblies are shown in Figure VG1-1.

The workshop had very little instrumentation on the main structure. The only dynamic instrumentation available was an accelerometer mounted on the film vault that was located on the forward floor. All other vibration data were obtained from the instrument unit. No deflection or strain gages were installed on the structure of the workshop with the exception of the strain gages used on the meteoroid shield position indicators.

The vehicle is a converted S-IVB modified in accordance with "Loads and Structural Design Criteria" DAC-56612B, September 1972 and "Orbital Workshop Acoustic, Shock and Vibration Test Criteria" DAC-56620C, May 1971.

There can be no direct evaluation of the workshop vibration and acoustic response since only one single axis low frequency accelerometer was present. This accelerometer was used to measure the response of the floor mounted film vault and was not capable of measuring frequencies of greater than 40 Hz. The workshop structural requirements were based on the vibration, acoustic, and acceleration levels experienced on previous flights. The instrument unit recorded a maximum axial sustained (steady state) acceleration at the predicted level of 4.4 g's for SIC outboard engine cutoff. The seals on the numerous penetrations held leak rates well below specification even after being subjected to the high temperatures prior to the installation of the JSC parasol. In all the fly-around photographs there is no visual evidence of structural deformation and the crew found no evidence of deformation within the workshop when stowing, relocating, and operating the equipment. From this it is concluded that it was not subjected to vibration, acoustic, and acceleration levels greater than anticipated.

Workshop subsystems such as the whole body shower, the wardroom window, the suit drying system, etc., will be evaluated from a mechanical point of view, on a component/system basis in subsequent paragraphs.

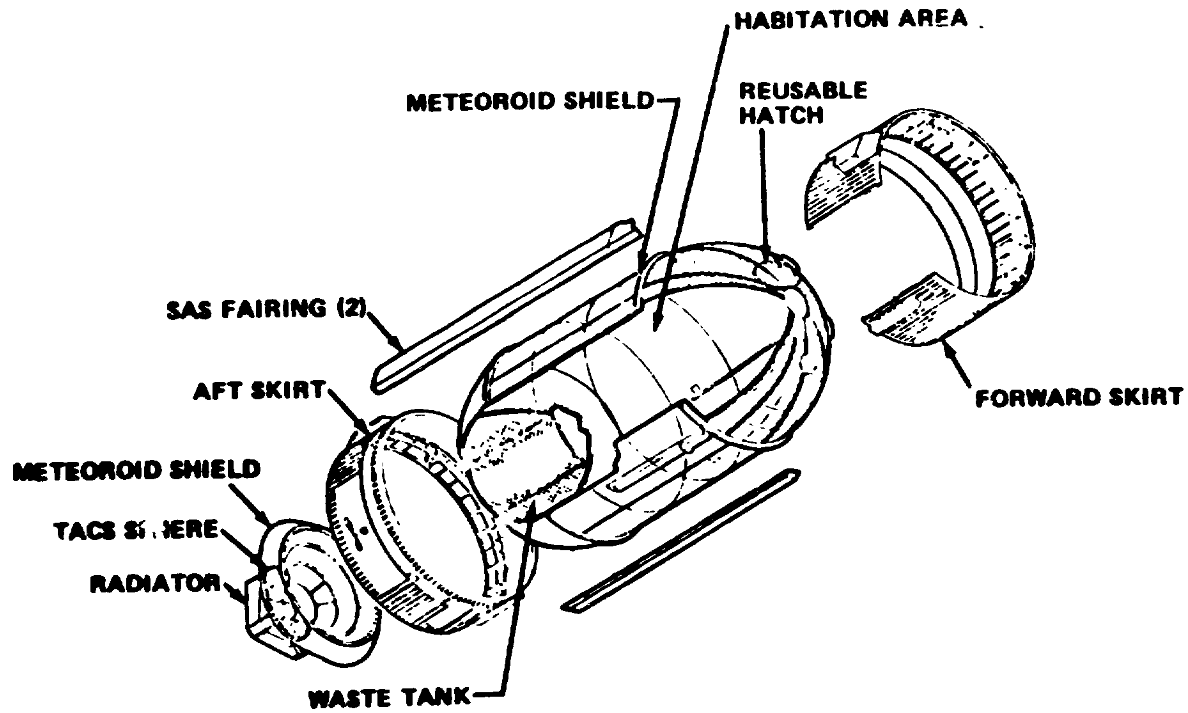


Figure VG 1-1. Workshop Major Structural Assemblies

2. Wardroom Window.

a. General requirements. The wardroom window, approximately 18 in. in diameter, was incorporated in the workshop crew quarters area to allow crew viewing as well as hand held and S063 photography.

The window is fused silica dual pane glass with the inner pane heated to prevent condensation on the inside surface. An internally mounted metal cover sustained launch and ascent pressure and provided an extra measure of protection during unmanned periods. Also included is a shade to shut out light when desired and a transparent impact shield for protection when not involved with photography.

Figure VG2-1 presents an exploded view of the window with associated hardware.

b. Mission performance. The wardroom window withstood the launch and boost environments and performed all required functions for the mission. The occurrence of moisture and ice within the cavity on the inboard side of the outboard pane did require periodic removal of condensate by the crew as described in detail in the following paragraph.

c. Anomalies.

(1) Workaround. During initial activation, the first manned crew observed the presence of ice and moisture in the cavity between the panes of the wardroom window. Several ground studies were conducted in an effort to solve the problem utilizing new procedures and on-board equipment. It was determined that the improvement of visibility would not justify the required impact on the crews time utilizing available equipment. An evacuation fitting was developed, flown up by the second crew, and installed in the purge fitting of the window (Figure VG2-2). The procedure called for venting the cavity to vacuum through the -Z scientific airlock and backfilling the cavity with desiccated air from the -Z scientific airlock desiccant canister. When the moisture was removed, an area of discoloration or residue remained. The moisture reappeared after about 7 days and the procedure was revised to eliminate the backfill, but the moisture continued reappearing every 6 to 10 days. The final procedure called for the evacuation fitting to remain installed in

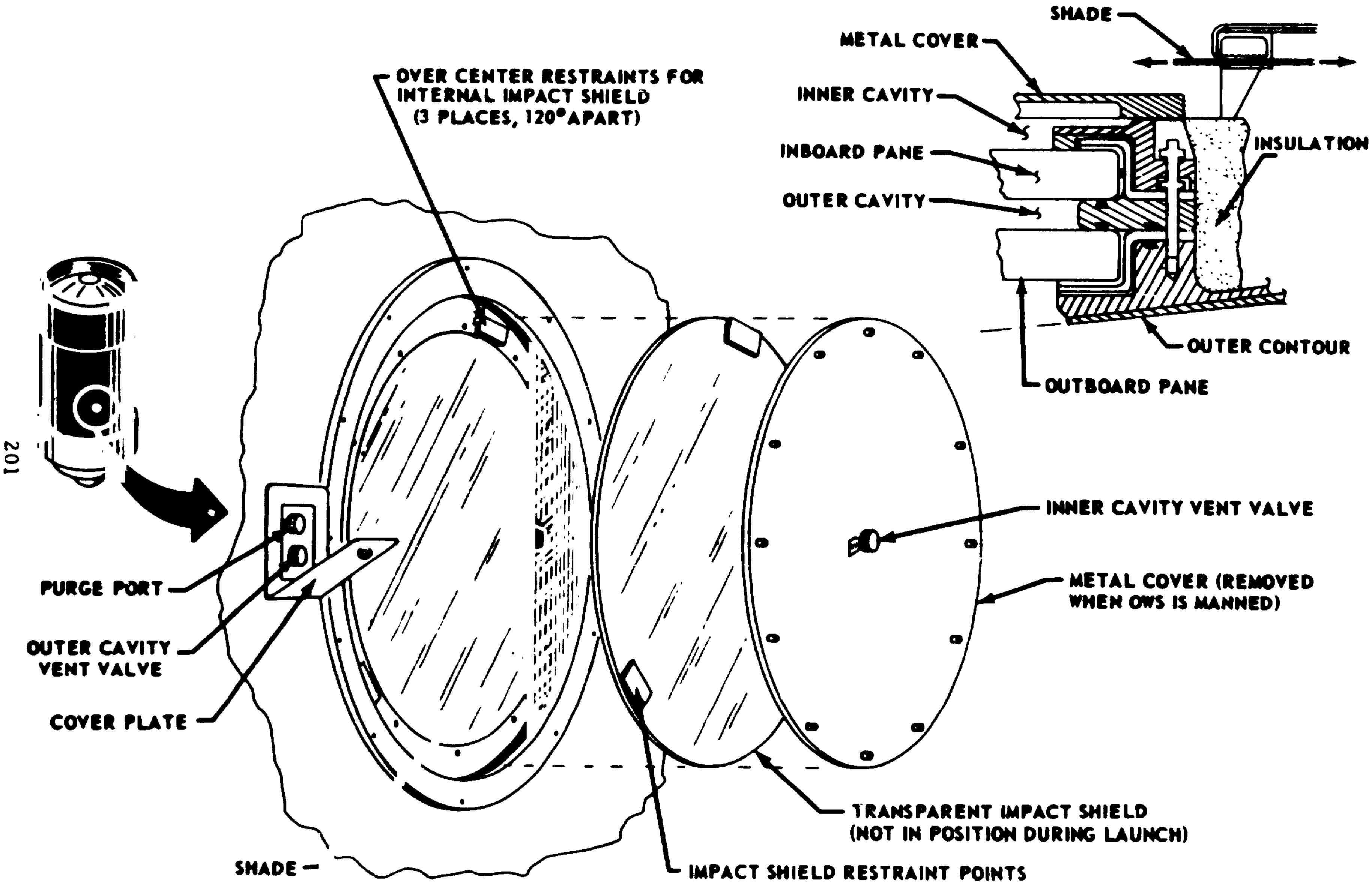


Figure VG2-1. Wardroom Window

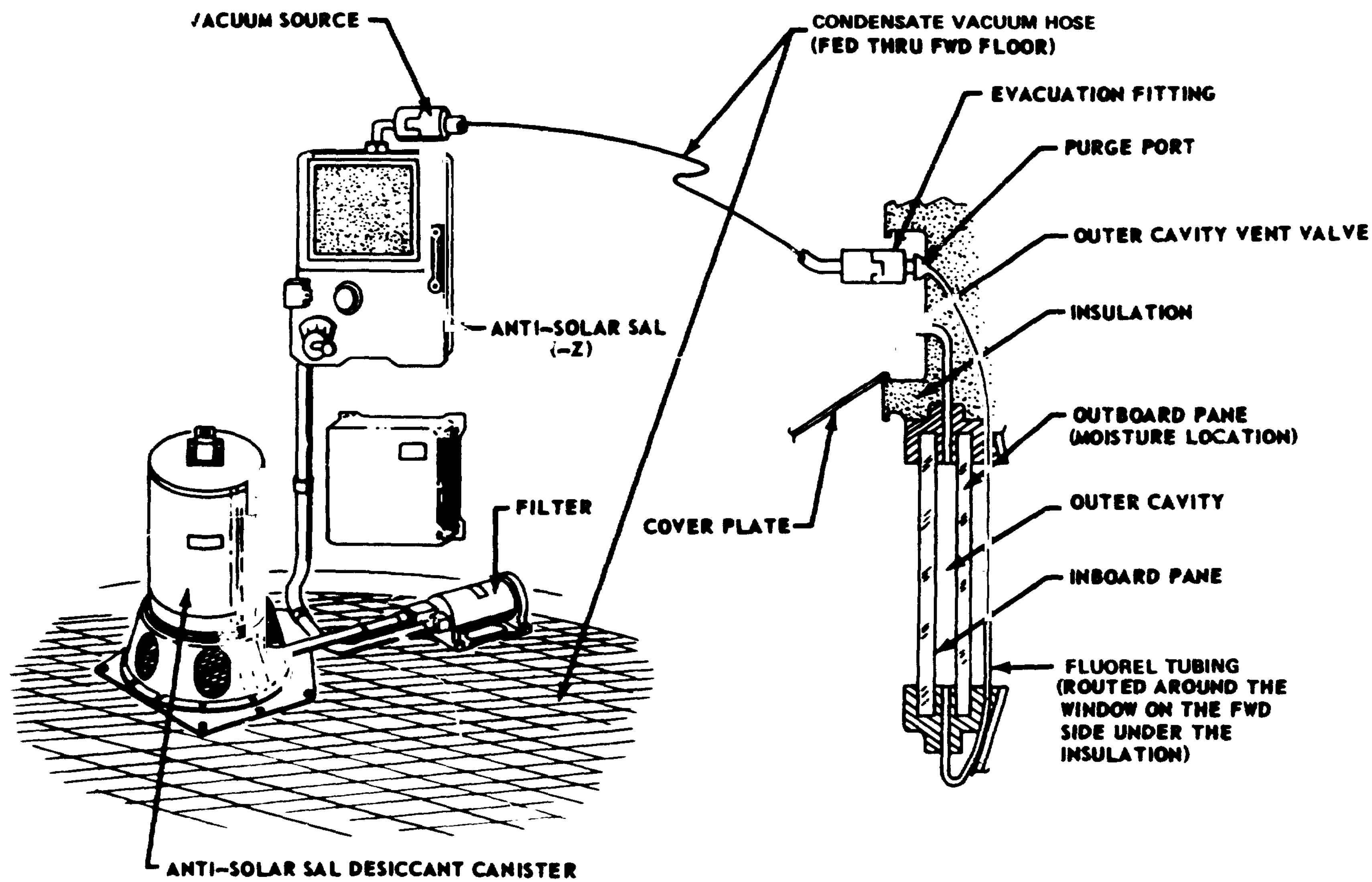


Figure VG2-2. Evacuation Configuration

the purge port and the line to be connected to the -Z scientific airlock by the crew when required. When the procedure was completed, the line from the evacuation fitting to the -Z scientific airlock was disconnected at both points. The removal of ice and moisture by sublimation and evaporation took 5 to 10 minutes, however the cavity remained vented to vacuum up to 3 hours to establish as complete an evacuation of the cavity as possible.

(2) Troubleshooting. The history of the window was researched to determine how and when the cavity contamination might have occurred. The procedures for handling, cleaning, testing, and installation indicate that contamination was not present up through installation of the window assembly in the workshop prior to closeout and shipment to KSC. Subsequent workshop procedures included no requirement for a contamination check between installation and final KSC closeout. Therefore, no checks were made to detect its presence.

It is pointed out that the valve O-ring could have possibly been damaged after the installation leak test was accomplished. Also, since the outer cavity vent valve (used to equalize outer cavity pressure at initial activation) had no positive mechanical lock to prevent tampering between window installation and final closeout at KSC, tampering is a possibility. The residue or discoloration mentioned in paragraph (1) was possibly the result of vapor deposition occurring gradually whenever moisture preset in the cavity condensed on the window. Subsequent evacuation removed the moisture, and conditions were set for repetition of the cycle.

The production acceptance tests for the wardroom window were conducted at MDAC-W, Huntington Beach, when the window was fully assembled but not installed. The specification leak rate of 1×10^{-6} standard cm^3/sec (sccs) was met. Long term leak testing that would uncover design or material deficiencies was not conducted during these tests. However, during checkout prior to shipment of the workshop to KSC, with the window assembly installed, the window was checked again and the specified leak rate was again met. (The test conditions in both cases were He at 75 °F and a delta pressure of 15.2 ± 0.5 psi.) After each test 1 atm of dry N_2 was locked in the cavity.

Analysis shows that no significant leakage of moisture into the cavity would occur on orbit with a specification leak rate of 1×10^{-6} sccs. However, the analysis did show that a leak rate of 7.5×10^{-2} sccs would allow a spot of moisture 0.001 to be formed in the cavity in 6 to 10 days.

d. Recommendations. On future projects, provisions to evacuate cavities between glass panes should be included. Also, our analysis shows that specifications, if not compatible with mission duration, can lead to moisture ingestion. Existing cleaning and handling procedures are acceptable but periodic checks should be made to detect the presence of contamination after final installation.

3. Solar Array System. The general requirements for the solar array system, parts of which are a carry-over from the wet workshop program, are:

a. General requirements.

(1) The solar array system shall deploy automatically, utilizing an ordnance system similar to that developed for the meteoroid shield, which shall not contaminate or affect the solar cells and other cluster systems.

(2) The time for deployment to vary from 9-min maximum (for deployment initiated 20 min after lift-off) to 14-min maximum (for deployment initiated 105 min after lift-off).

(3) The solar array system shall be capable of deployment through backup commands via the airlock module digital command system.

(4) The solar array system shall withstand the loads, vibrations and shock levels associated with launch, ascent, docking and maneuvering.

(5) During boost, the wing section cavities within the beam fairings shall vent through acoustically actuated vent valves.

b. Development and testing.

(1) SA-14 wing release. The purpose of this test was to develop the specific expandable tube and tension strap, or link, for releasing the stowed solar array panels. Although the flight configuration requires expandable tubes that are 31 ft in length, extending through the three 10-ft bays in the solar array system fairing, the test assembly was made 39-in. long. This was done to permit vibration of the tubes that are supported only at cinch-bar positions, that is, approximately 30 in. apart. The required X-ray inspection following vibration environment revealed that the fuse became separated within the expandable tube. On disassembly it was found that the fuse was free to slide within the assembly and not constrained as intended. The minor redesign that corrected this problem increased the diameter of the spacers on the fuse so that when the tube is flattened to the required dimension, the fuse is gripped at each spacer along its entire length.

Since early in the development phase, the only time the expandable tube failed to break a tension link in the entire test program occurred during this test. It was caused by a combination of two conditions that were subsequently corrected with added controls. First, it was found that the link had been incorrectly manufactured. The grain direction of the tension link, required by the engineering drawing to run lengthwise (i.e., in the same direction as the expandable tubes) was, in fact, transverse on several tension links, making the part more difficult to break. Tension links having a break groove material thickness of 0.021-in. were tested and compared to the meteoroid shield tension strap that had a 0.013 to 0.016-in. break groove.

In addition, it was noted that the fit of the expandable tubes within the failed tension link was extremely loose. This meant that some of the tube expansion was wasted since it did not work against the link.

(2) SA-15 wing release. The solar array system test was originally planned as a qualification test of four specimens, but was revised in scope and was actually performed with one specimen. The single test specimen consisted of two 10-ft long expandable tubes. They were installed in the one-third length solar array system beam fairing development model and exposed to dynamic environments. For this test, both the primary and backup systems were operated with a programmed delay between firings of 100 msec.

The system performed as required, severing the five attach links that secured the stowed solar array system wing section in a rigged condition. An anomaly was experienced at each tension link; strips of the aluminum link broke loose either as a primary system functioned or during functioning of the backup system. The strips resulted from the straps breaking at the secondary relief grooves paralleling the intended line of fracture. All loose elements were retained by a retainer, which is a part of the tension link assembly.

(3) SA-17 solar array system beam fairing release. This test was established to develop the configuration of expandable tube and tension strap for solar array beam deployment. This usage requires a very short expandable tube assembly (approximately 9 in. overall length) and a tension strap capable of carrying high loads.

The functional test specimen consisted of two expandable tubes and a tension strap assembly mounted and preloaded in a structural yoke assembly. The tension strap was adapted from the meteoroid shield design by increasing the break groove material thickness to 0.021 in. Other areas of the strap cross section were proportionally increased.

Results from firing the first nominal-thickness strap indicated that further modifications to the tension strap were required because portions along the fracture groove were broken loose during firing, a result similar to those of SA-15. All pieces were retained within the yoke and were not construed as fragments.

Following revision to the design to reduce the depth of the machined grooves (allowing the element along the fracture line to bend), the remaining five specimens with redesigned tension straps were operated with no further anomalies.

(4) ST-24 expandable tube and tension strap. Special test 24 was conceived as an expeditious way of determining (prior to performance of solar array system test SA-4) if parts would break off the tension straps during strap severance. There was concern because it had been judged during the manufacture of the expandable tubes for the SA-4 specimen that the annealed stainless steel tubing was softer than the prior product and would, therefore, yield greater expansion per given explosive charge. This would make the tension strap more susceptible to developing loose parts along the fracture line during firing. Special test 24 was run concurrently with the early position of SA-4.

The tension strap in each specimen was covered with Scotch 850 aluminized Mylar tape to simulate the passive thermal protection on flight hardware. In five specimens, aluminum tabs were broken off by the expandable tube firing, although so little energy was imparted to them that most were retained by the tape. One room-temperature specimen had no loose parts resulting from firing.

Final system rigging and qualification testing was successful from an ordnance standpoint. Qualification testing was performed at ambient temperature and there were no ruptures or loose pieces (tabs). As a result of successful firings at ambient temperature, it was decided that taping the separation joint to retain the tabs would not be necessary.

(5) Acoustic vent module. The vent module (Figures VG3-1 and VG3-2) was designed to open at one-half the expected acoustic pressure generated at launch, and to withstand, without opening, an acoustic pressure level of twice the maximum ambient expected in the area of the launch site. A relief valve (Figure VG3-3) was incorporated to allow venting in case the ground purge pressure exceeded 0.1 to 0.4 psid and as a backup to the main acoustic vent doors.

Site test at KSC, with the test unit, during the launch of Apollo 16 showed the unit functioned as predicted and designed.

(6) Actuator/damper. The actuator/dampers (Figure VG3-4) used on the beam fairings and wing sections were similar in construction, but differed in stored energy and deployment rate. Extensive testing under various environments was used to develop and qualify the actuator/dampers.

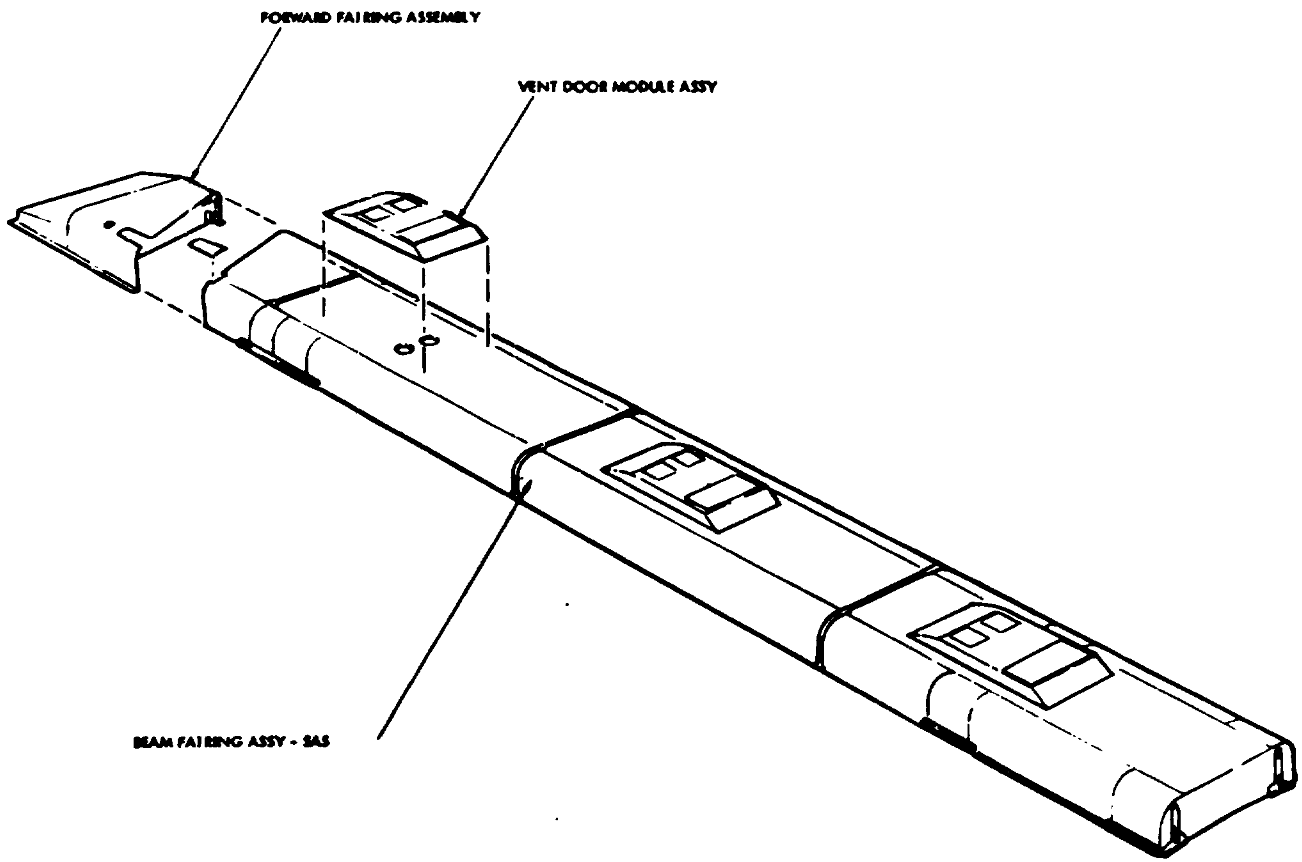


Figure VG3-1. Acoustic Vent Module

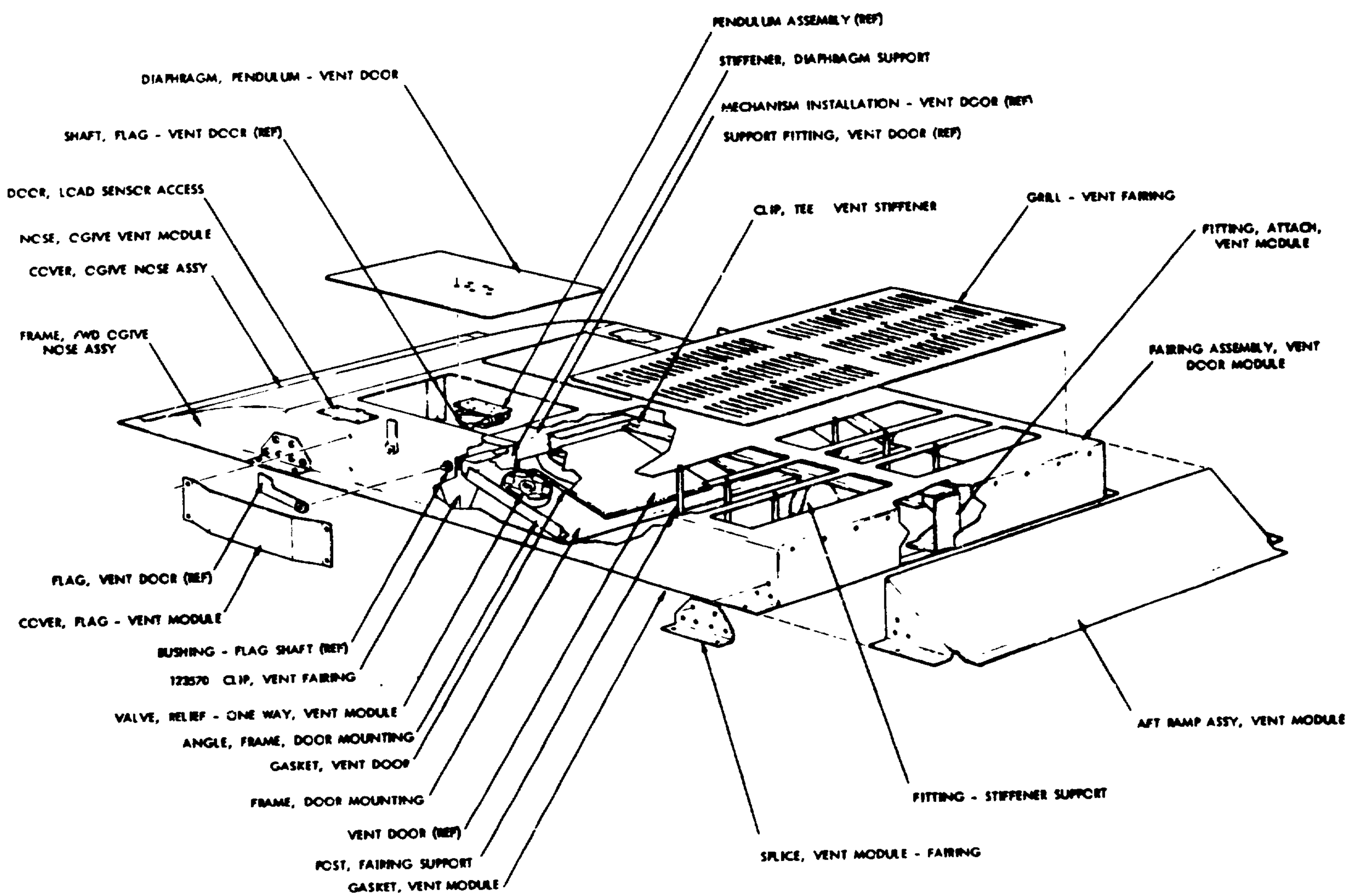


Figure VG3-2. Acoustic Vent Door Module Assembly

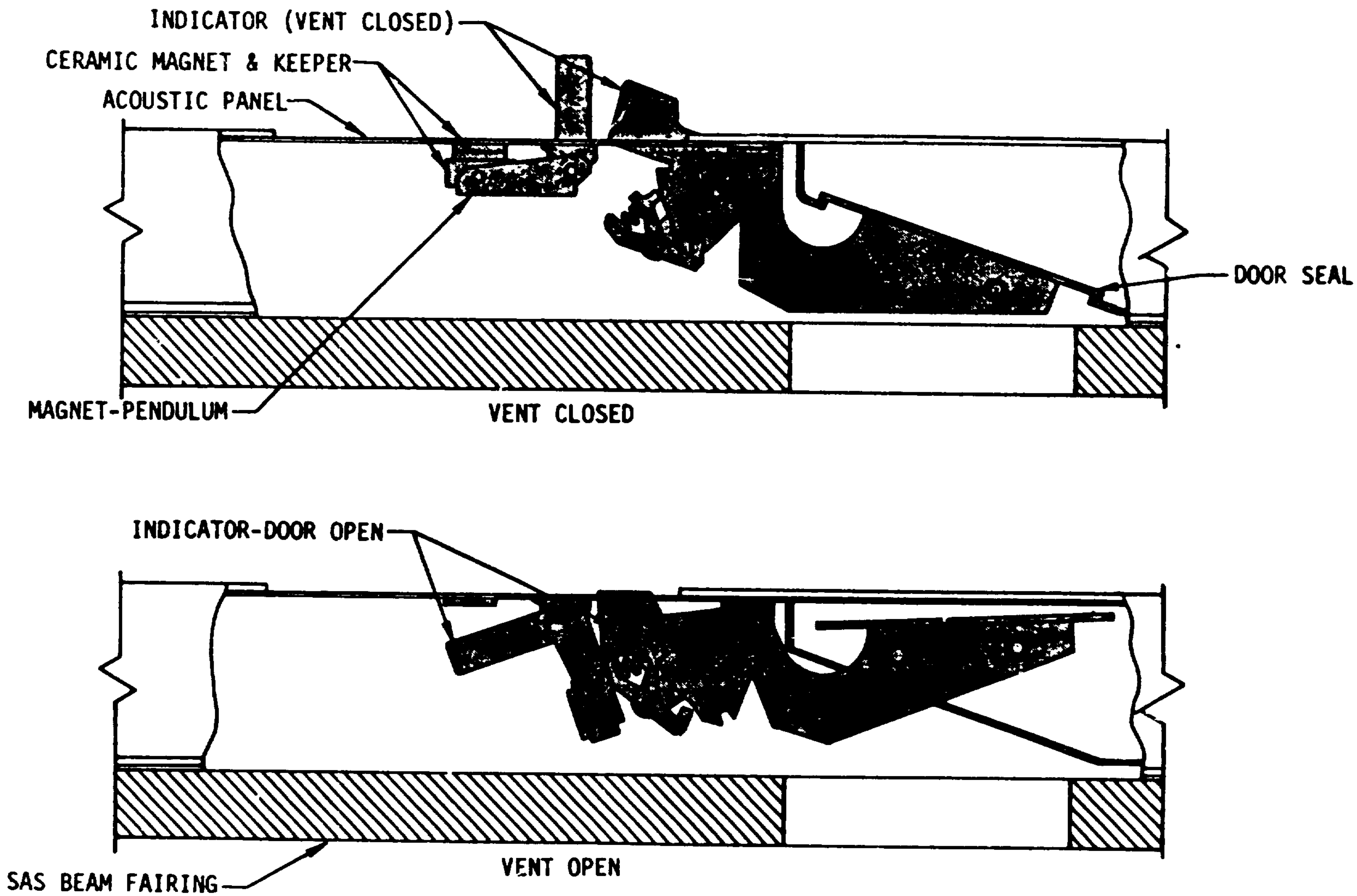


Figure VG3-3 Acoustic Vent Module Relief Valve

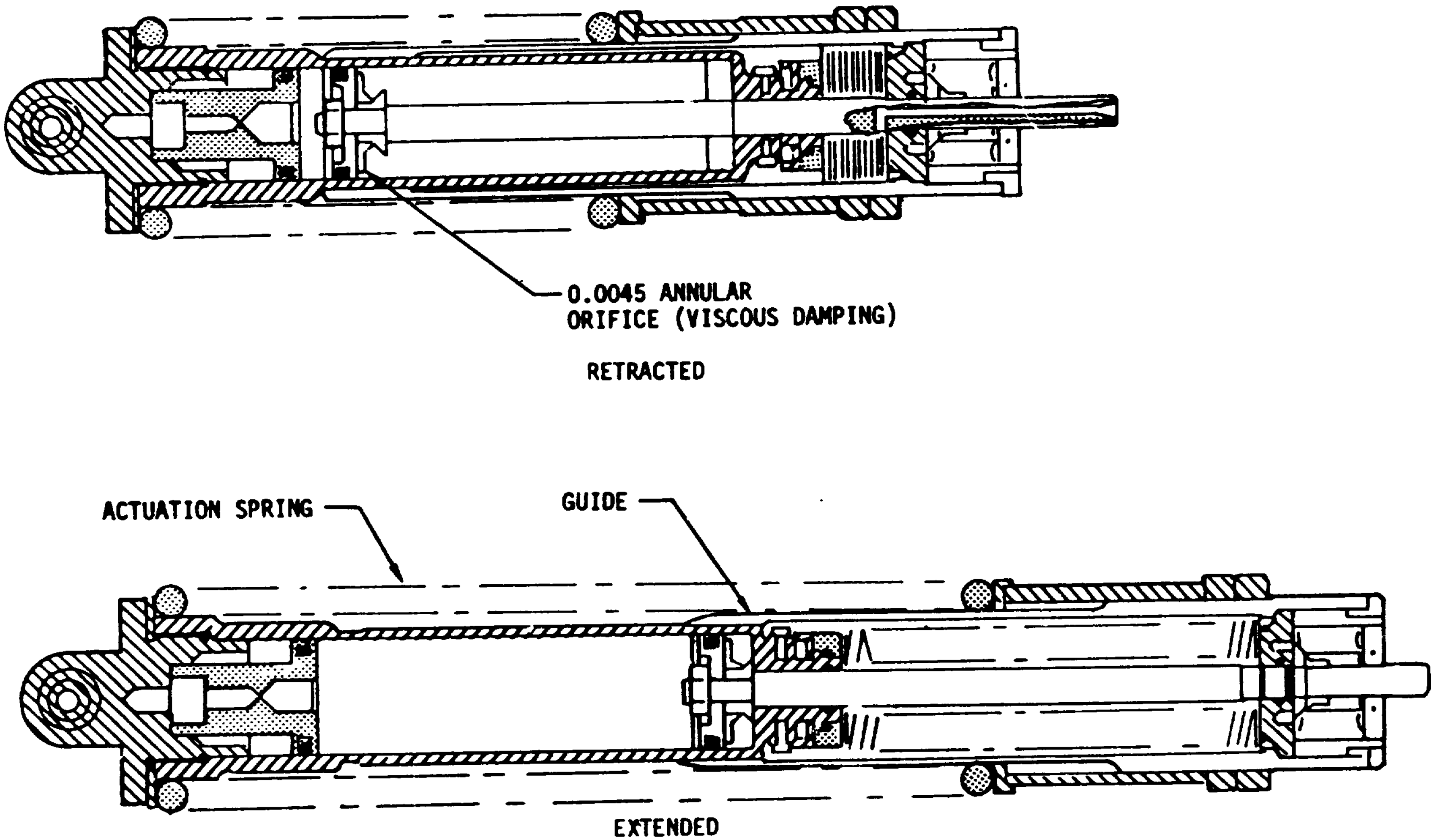


Figure VG3-4. Actuator/Damper

The beam fairing and wing section deployment time varied greatly as a function of temperature, as shown in the following plots (Figures VG3-5 and VG3-6).

c. Mission performance. The ordnance profile in Figure VG3-7 shows the nominal traces for charge and trigger command. The firing voltage specification and actual levels are given as well as the nominal and actual times of firing. Because of an absence of ground station coverage on solar array system beam fairing and wing section deployment, a comparison cannot be made of actual and specified firing times. The voltage level (on M7061-404) that was recorded onboard and later down-linked shows no activity at the nominal deployment time [Figure VG3-7, note (1)]

The solar array system beam fairing #1 released automatically and started to rotate to full deployment but was restrained by a section of the meteoroid shield panel joint. This is seen in Figure VG3-8. After the restraint was removed and beam fairing and the wing sections were deployed, they withstood the docking and cluster maneuver loads as required. No direct instrumentation existed to confirm this but review of cluster photos shows all solar panels on wing #1 were exposed and appeared undamaged.

All three vent modules on solar array system beam fairing #1 were seen open as the vehicle lifted off and passed the swing arm camera. Film coverage was not provided to record beam fairing #2 or actual vent module opening when the acoustic level built-up before launch release.

d. Anomalies. No known solar array system anomalies were the direct result of solar array system component malfunctions. The loss of solar array system beam fairing #2 was caused by the events that followed the structural failure of the meteoroid shield during launch.

Solar array system beam fairing #2 was separated from the vehicle approximately 593 sec (T + 593) after lift-off. The first indication occurred as an "unsecure" indication. The final separation of the beam fairing was confirmed when loss of telemetry (offscale high and offscale low on various transducers) occurred on all wing temperature and power measurements. As a result of this anomaly no evaluation can be made on the ordnance or deployment mechanism for this solar array system beam fairing.

Solar array system beam fairing #1 and wing section deployment would probably have been nominal had it not been restrained by a piece of the meteoroid shield (Figure VG3-8).

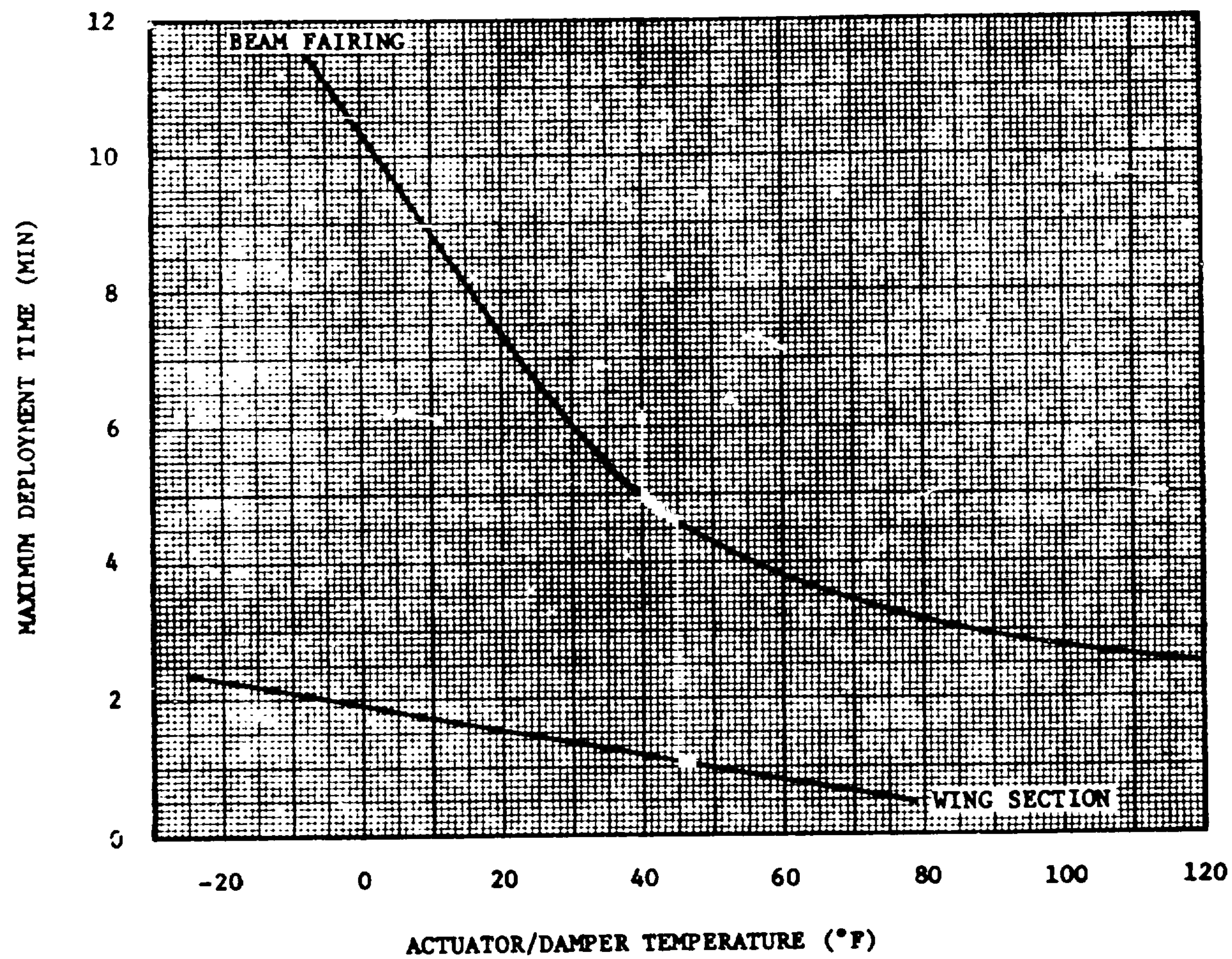


Figure VG3-5. Beam Fairing and Wing Section Deployment

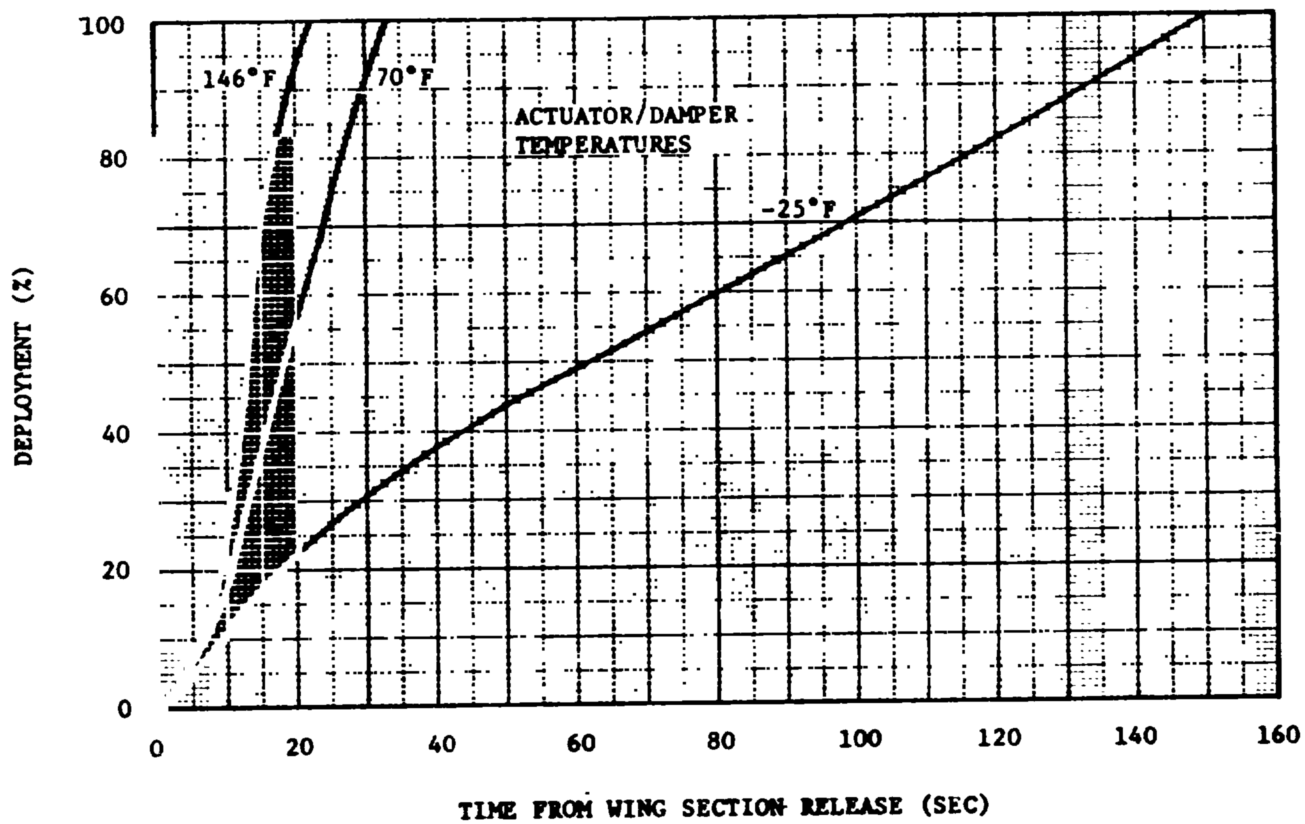
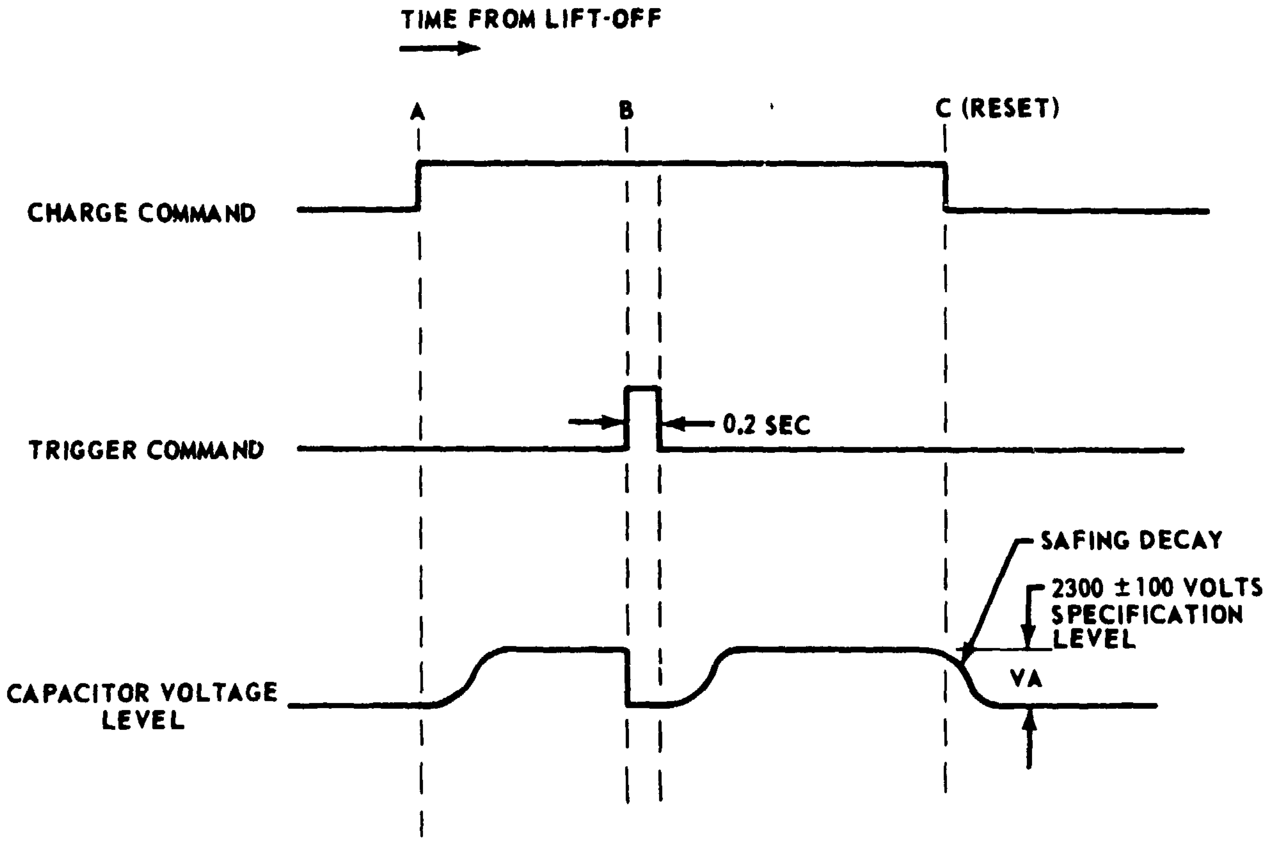


Figure VG3-6. Beam Fairing and Wing Section Deployment



	A		B		C		ACTUAL (VA) CAPACITOR VOLTAGE LEVEL	MEASUREMENT NUMBER
	SPECIFIED	ACTUAL	SPECIFIED	ACTUAL	SPECIFIED	ACTUAL		
BEAM FAIRING REL. PRIMARY	18:11:05	(1)	18:11:10	(1)	18:11:15	(1)	(1)	M7061–404
BEAM FAIRING REL. BACKUP	(3)	19:08:22	(3)	19:08:42	(3)	19:09:08	2391 VOLTS	M7060–404
WING SECTION REL. PRIMARY	18:22:00	(1)	18:22:05	(2)	18:22:10	(1)	(1)	M7067–411
WING SECTION REL BACKUP	(3)	19:20:56	(3)	19:23:26	(3)	19:24:01	2361 VOLTS	M7066–411
METEOROID SHIELD REL. PRIMARY	19:05:58	19:05:59	19:06:03	(2)	19:06:08	19:06:09	2361 VOLTS	M7000–411
METEOROID SHIELD REL. BACKUP	(3)	20:12:24	(3)	20:12:42	(3)	20:13:19	2361 VOLTS	M7001–411

(1) TELEMETERY NOT AVAILABLE, DATA OCCURED BETWEEN SAMPLE POINTS.
(2) COMMAND INHIBITER, INTERLOCKED WITH SAS B/F FULLY DEPLOYED.
(3) COMMAND TIME FOR BACKUP NOT SPECIFIED IN ADVANCE

Figure VG3-7. Ordnance Profile

Figure [illegible] S Wing #1 Fairing Restrained by a Piece of the Me[illegible] Shield

Prior to loss of signal, the solar array system wing #1 fairing secure measurement showed "secure" at the last ground station coverage before the automatic switch selector charged and fired the beam fairing ordnance. When no indication of solar array system wing #1 fairing deploy occurred at acquisition of signal, the backup airlock module digital command system signal was sent to charge and fire the backup ordnance. The backup signal for wing section ordnance was necessary due to an interlock that prevents the primary wing section ordnance from firing if the beam fairing has not fully deployed. The wing section position indicators showed some movement but full deployment readings did not occur until after the beam fairing was freed. The first manned crew deployed the beam fairing during an extravehicular activity, after breaking the actuator damper, by exerting the necessary moment ($\approx$ 750 ft-lb) with a beam erection tether attached between the aft end of the beam fairing forward vent module and the intersection of the deployment assembly and discone antenna support brace (Figure VG13-4). The vehicle was then maneuvered to an attitude to allow the three wing section actuator dampers to be warmed by solar heating. At the end of 5 1/2 hr the wing sections were fully deployed.

e. Recommendations.

(1) Instrumentation should be provided to indicate position or condition of deployment for critical systems.

(2) Consideration should be given to the manual deployment of solar power systems in the event a malfunction occurs in the primary mode.

(3) Acoustically actuated vents are good for environmental protection on the ground and activated by the normal launch environment.

4. Meteoroid Shield. It is not possible to include a mission evaluation of the meteoroid shield as it was torn off during ascent. Consequently, since some of the subsystems may be applicable to future programs, more detailed preflight data are included.

a. General requirements. The general requirements for the meteoroid shield are:

(1) Provide protection such that the probability of no pressure loss in the habitation area will meet or exceed a cluster requirement of 0.995 for 8 consecutive months. The meteoroid environment shall be as specified in NASA TMX-53957.

(2) Encompass the cylindrical section of the workshop and provide closures at each end to cover the annulus when deployed.

(3) During prelaunch, be no more than 6 in. radially from the outer surface of the habitation tank.

(4) Be free of flutter or other divergent instabilities during the launch phase.

(5) During orbit, initiate deployment by a signal from the instrument unit over the instrument unit switch selector with a backup deployment command from the airlock at the workshop/airlock interface.

(6) Single point failure shall not inhibit deployment.

b. Subsystem configuration. The original "wet" workshop concept envisioned a scroll-like shield, wrapped tightly around the cylindrical section of the S-IVB LH_2 tank and held together mechanically. The shield was to be released in orbit, by its own stored energy, so that it would provide an annulus between shield and LH_2 tank. Later, the idea of ordnance release was incorporated for the "wet" workshop; first the S-IVB type "mild detonating fuse", and then the "expandable tube" which was first used in military programs.

Four basic methods of meteoroid shielding were considered during the conceptual design of the "wet" workshop: (1) single armor plate (2) spaced sheet (3) spaced sheet with foam filling, and (4) spaced sheet with multilayered foam/Mylar filling. Trade studies on these methods included considerations such as: minimum modifications to the vehicle, weight, flutter, vibration, purge gas consumption, transportation, and matching of the shield to the stage (the flight shield had to be mounted on a propulsion test article for firing at SACTO, then removed and shipped to KSC for mounting on the workshop flight vehicle). It is pointed out that the temperature region considered on the "wet" workshop during boost was from -423 °F on the inside of the tank to 475 °F on the outside. The large amount of helium purge gas required for a spaced shield filled with insulation to combat these temperature extremes made further consideration of this meteoroid shielding concept, items (3) and (4) above, impractical. This left only the single armor plate or the spaced sheet, items (1) and (2) above, as acceptable design concepts to pursue.

The NASA meteoroid flux model required a shield, equal in protective capability to a single sheet of 2024-T6 aluminum 1.43 cm thick, in order to obtain the probability of no penetration of 0.995, but weight was a primary consideration on the "wet workshop." Thus, using the 1.43-cm single plate thickness as a base, and V.C. Frost's equations to obtain equivalent sheet thicknesses at discrete spacings, a minimum distance of 3.5-in. for a 0.025-in. thick sheet was calculated. It was felt that, since the function of the spaced sheet is to disintegrate meteoroids, and once a meteoroid is disintegrated its debris spreads into a conical shape, more spacing would be advantageous. More spacing would allow the debris to spread further, thus reducing the force per unit area on the tank wall. This was accomplished by going to a 5-in. nominal spacing. Furthermore, a deployable shield required no "Z" stiffener frames; only relatively light hardware was necessary at each end, forward and aft, along the main tunnel, and along the ordnance train. Trade studies showed that the 0.025-in. thick deployable shield with a 5-in. spacing would weigh slightly over 1,000 lb, while a fixed shield with a spacing of 1.65 in. had to be 0.045-in. thick and would weigh 1,700 lb. Any spacing further than 5 in. would increase the weight because of the larger size of supporting frames. Based on the foregoing, the 0.025-in. thick, 5-in. spacing, deployable meteoroid shield concept was designed for the "wet" workshop. When the workshop was coverted from "wet" to "dry", only a minimum amount of change was permitted; concept was to remain the same. Additionally, it is pointed out that even though not listed in the CEI specification, the deployable meteoroid shield, with its thermal protective paint pattern together with the low emissivity gold Kapton coating on the habitation tank exterior, provided a natural radiant heat barrier.

During ground handling and boost, the deployable shield (Figure VG4-1) was to be held in intimate contact with the habitation area cylinder wall by circumferential tension provided by the spring force of 28 titanium frames that were part of the auxiliary tunnel. Following orbital insertion, the workshop pressure was to be blown down and the solar array system deployed. Then firing of a confined detonating fuse inside an oval "expandable tube" would round the tube and rupture six tension carrying ordnance straps (Figure VG4-2) along the length of the shield. This event would allow eight deployment links each, at the forward and aft shield end, driven by the stored energy of preloaded torsion bars (Figure VG4-3) and mounted to the tank skirt flanges, to deploy the shield in a translation/rotational mode (Figure VG4-4). The final shape of the deployed shield was to be a cylinder concentric to, and spaced at, a nominal distance of 5 in. from the habitation cylinder wall (Figure VG4-5). The additional shield circumference required to assume the larger diameter was provided by a folding panel assembly located under the redundant ordnance assembly (Figure VG4-6). The shield was assembled from 16 curved, 135-in. radius, 0.025-in. thick, 2014-T6 aluminum preformed panels, some of them with smaller panel inserts to allow use of the wardroom window and scientific airlocks on orbit, one provided access to the ground access panel. The shield was, by way of butterfly hinges, connected to 12 straps running under the main tunnel and bonded to the tank (Figure VG4-7). In the vicinity of the ordnance assembly, two ends of the shield overlapped. They were joined with 14 trunnion bolts which were tightened for rigging (Figure VG4-8). The tension caused by the trunnion bolts together with the spreading of the auxiliary tunnel frames was to provide the hoop tension required for intimate contact during boost. A series of 0.005 CRES performed "fingers" were riveted under the forward and aft end of the shield forming a "boot" (Figure VG4-9) that would close off the annulus

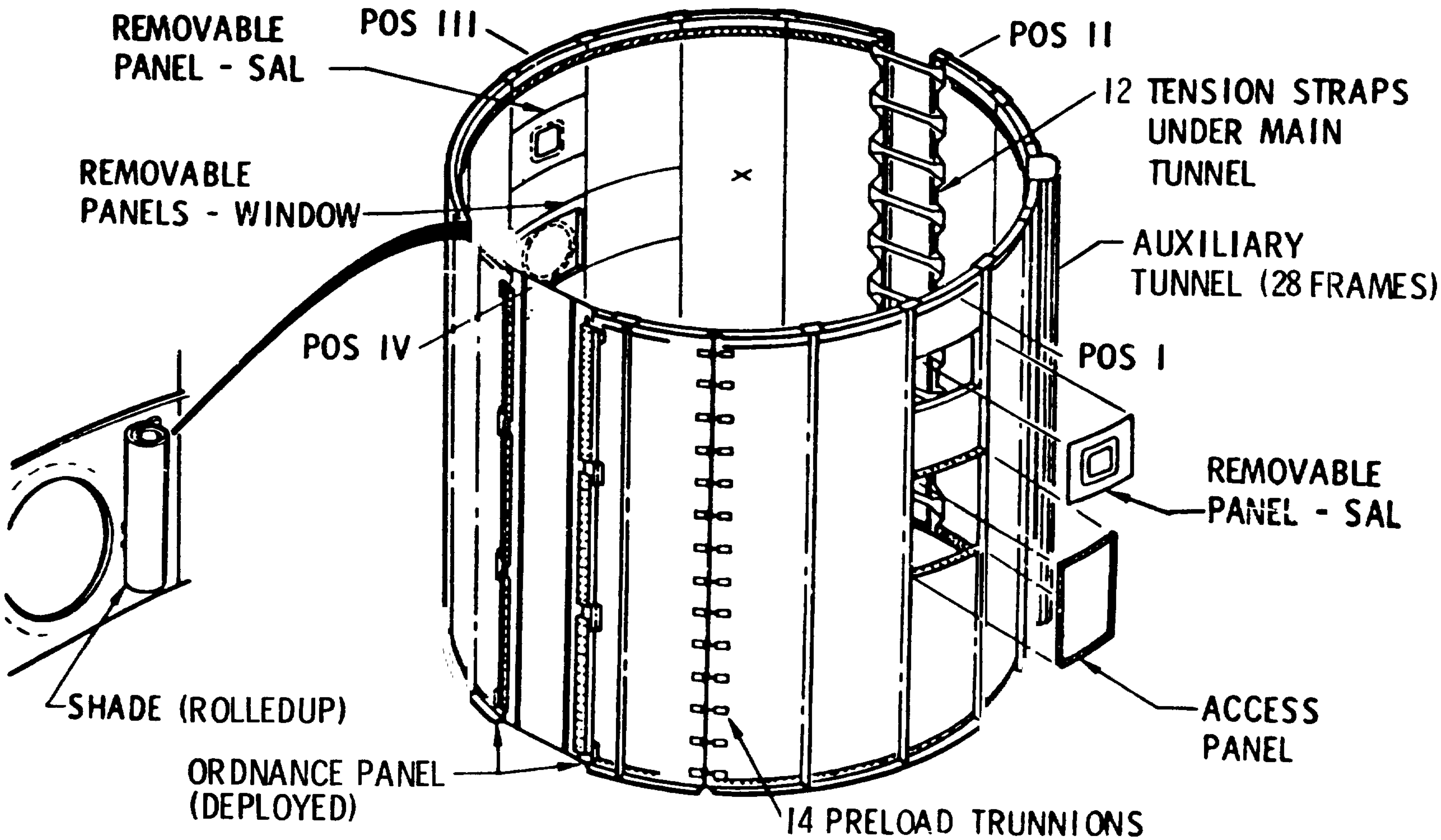


Figure VG4-I. Meteoroid Shield

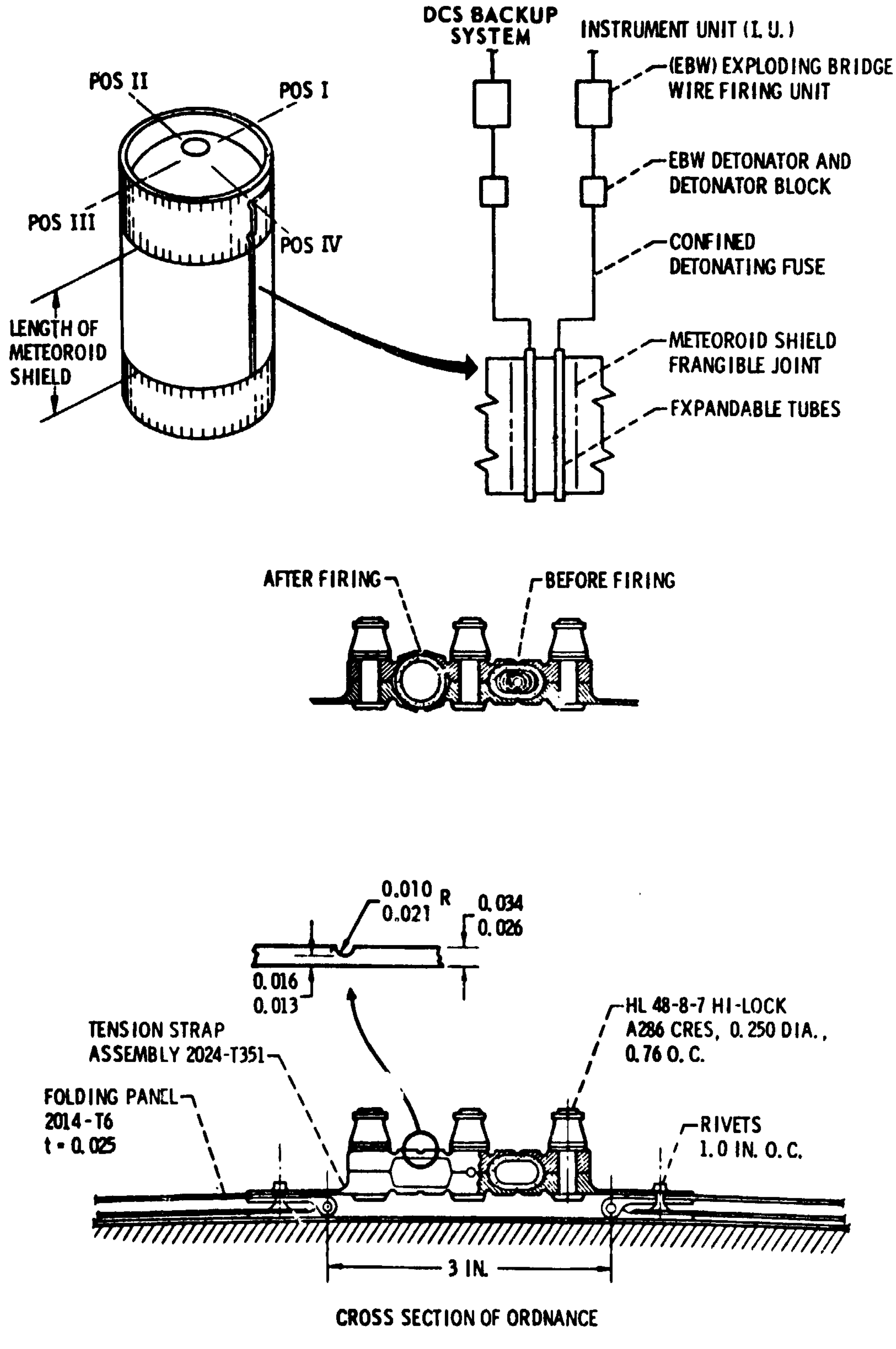


Figure VG4-2. Meteoroid Shield Ordnance Schematic and Cross Section View

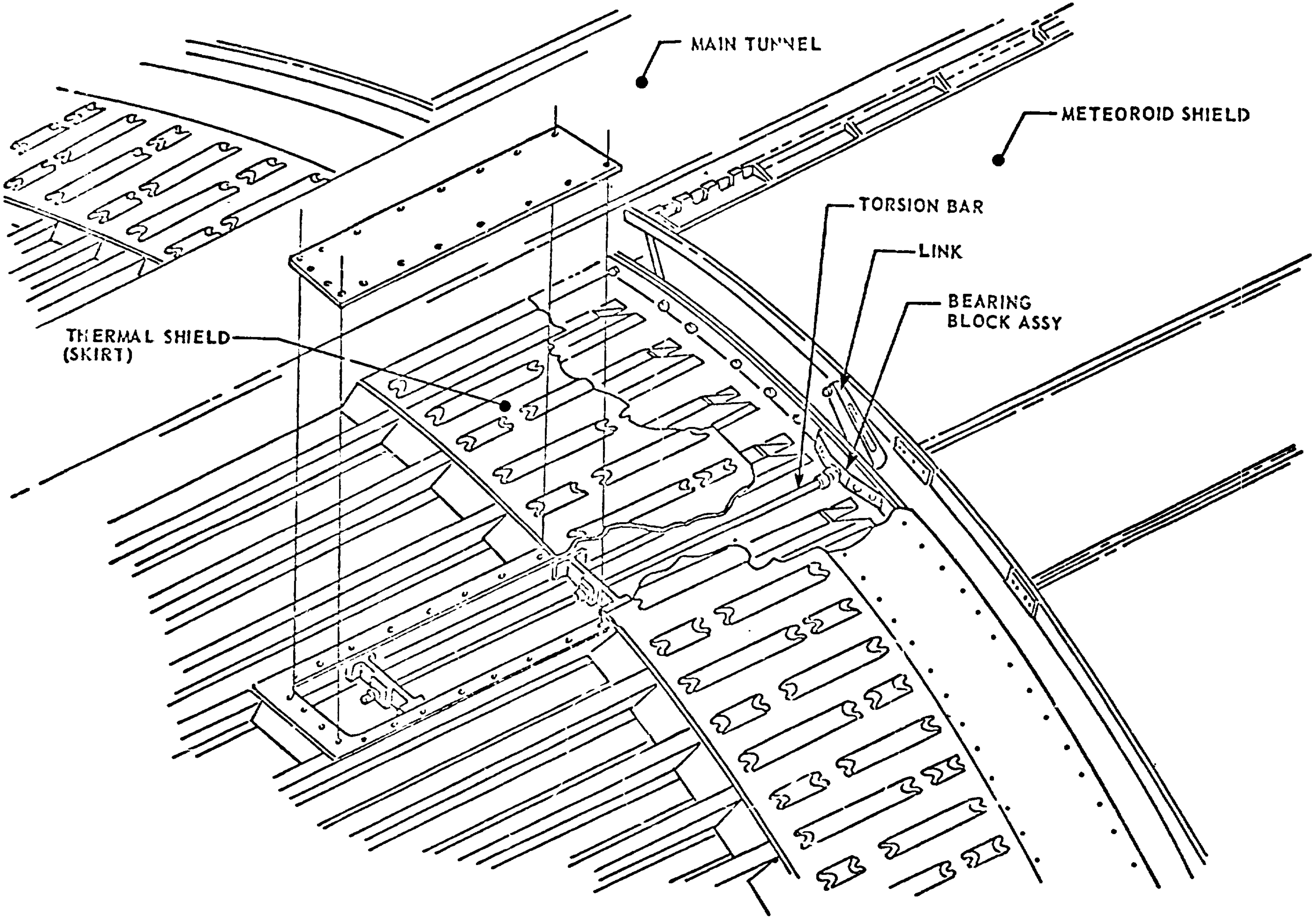


Figure VG4–3. Meteoroid Shield Deployment Mechanism

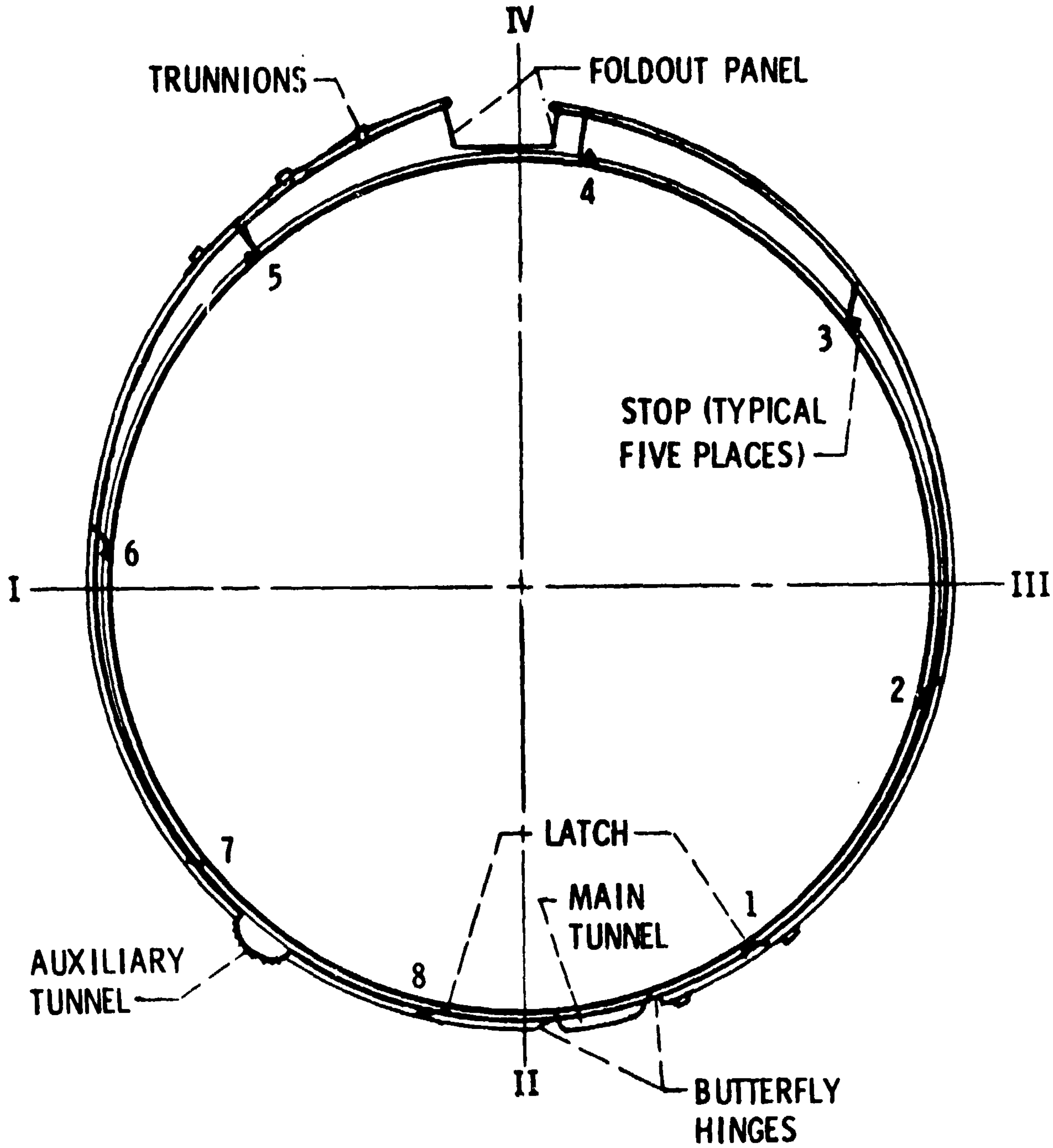


Figure VG4-4. Meteoroid Shield Partially Deployed

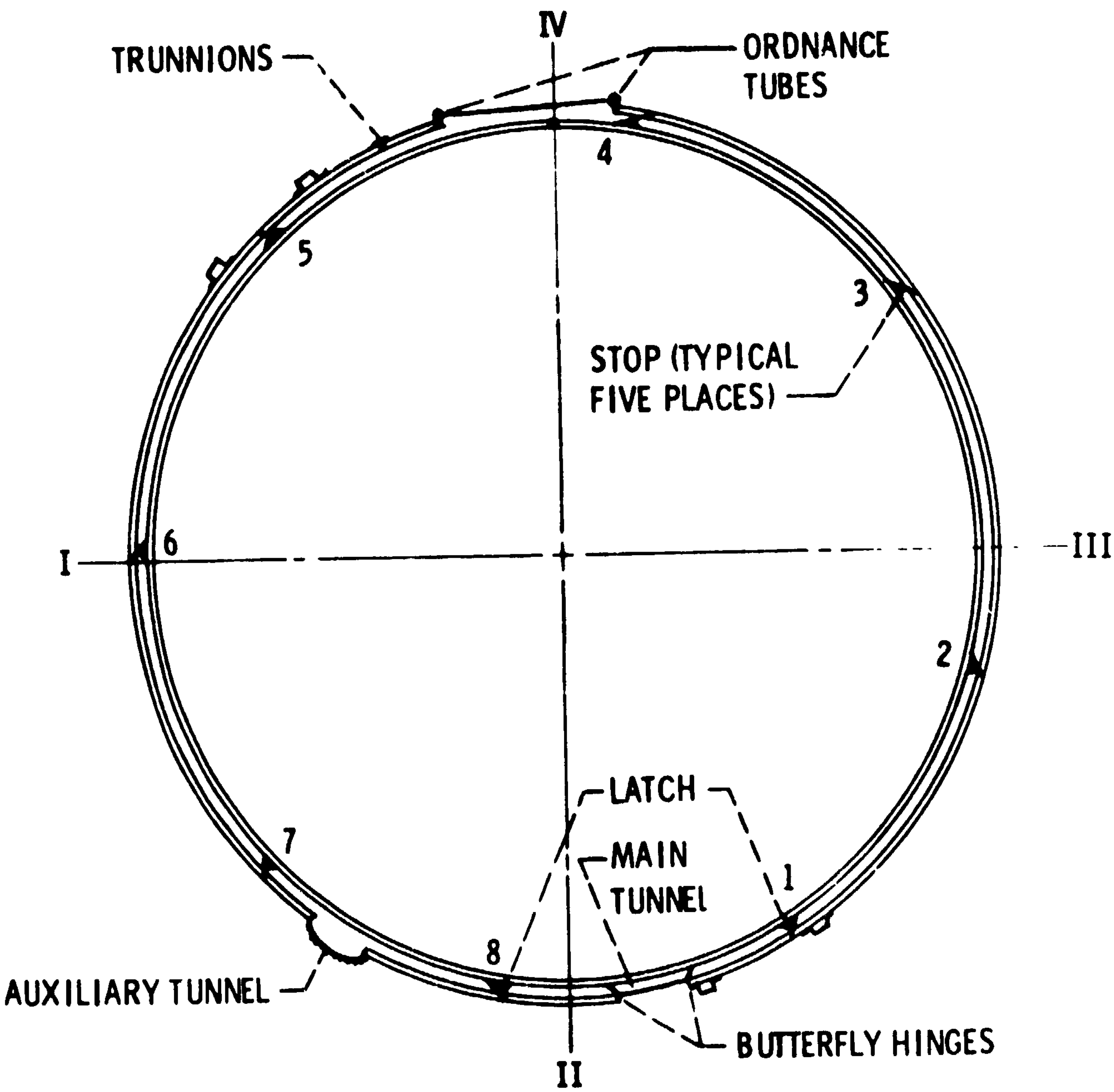


Figure VG4-5. Meteoroid Shield Deployed for Orbit

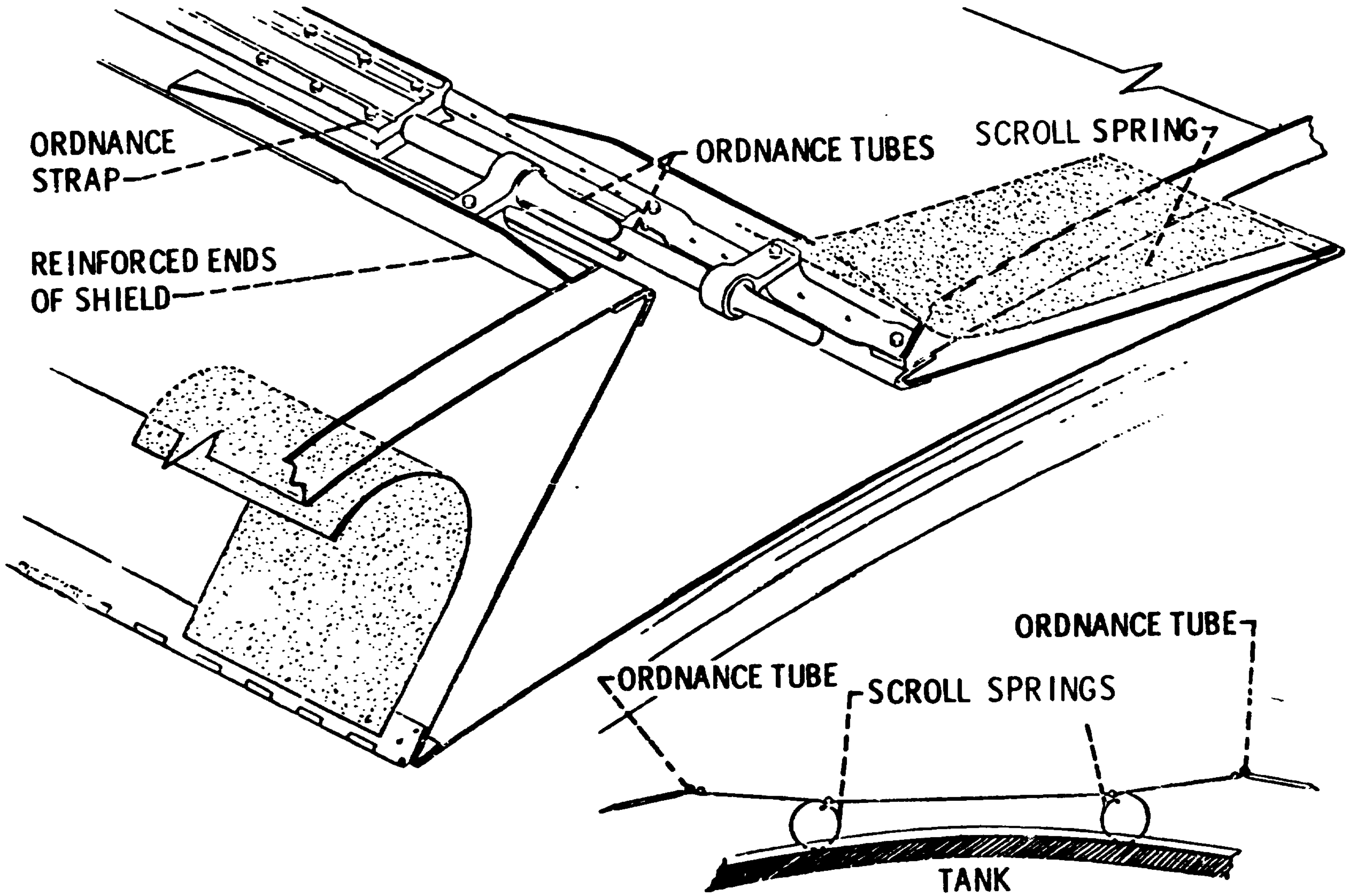


Figure VG4-6. Meteoroid Shield Deployment Ordnance and Foldout Panels

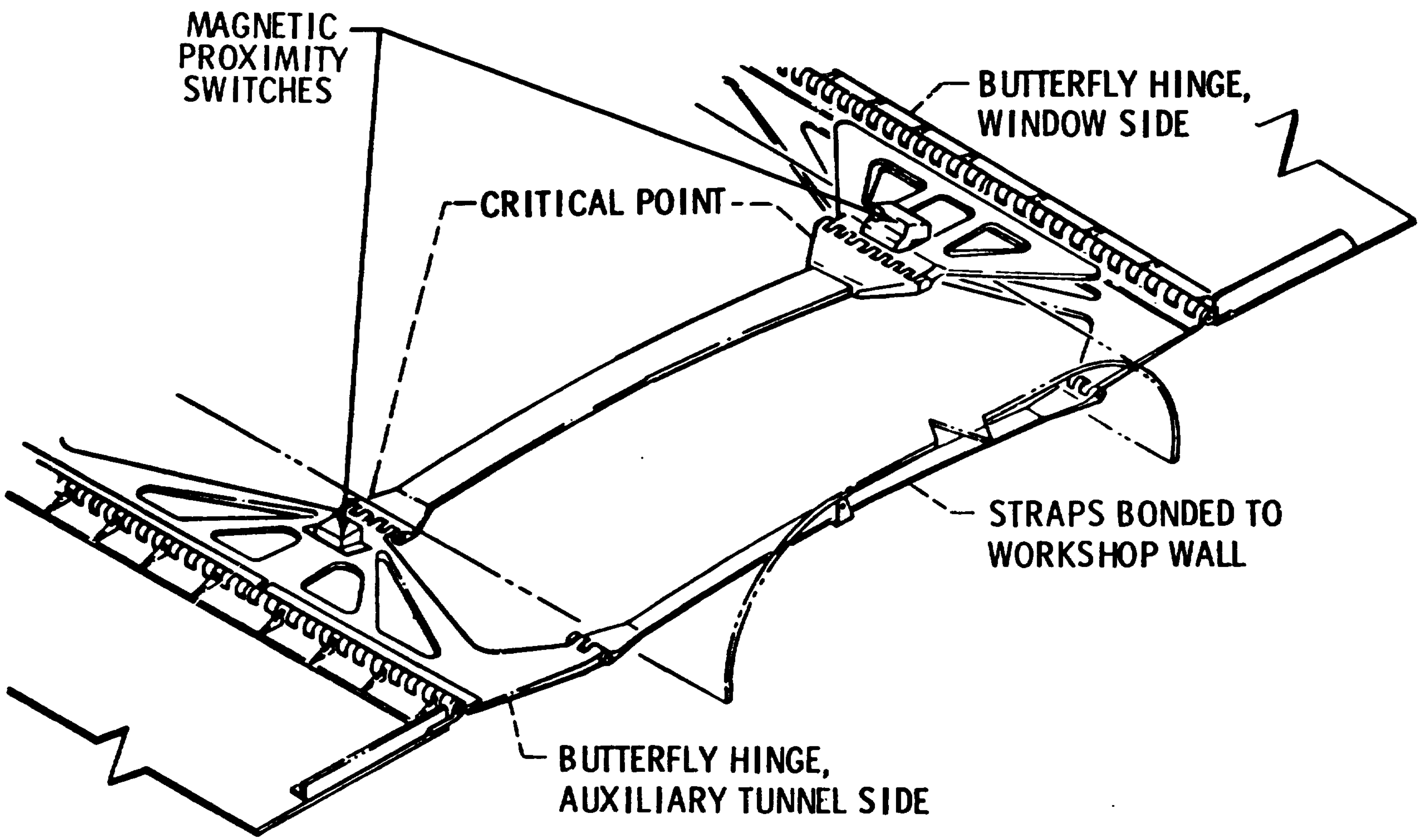


Figure VG4-7. Meteoroid Shield Tunnel Straps and Butterfly Hinge Assembly

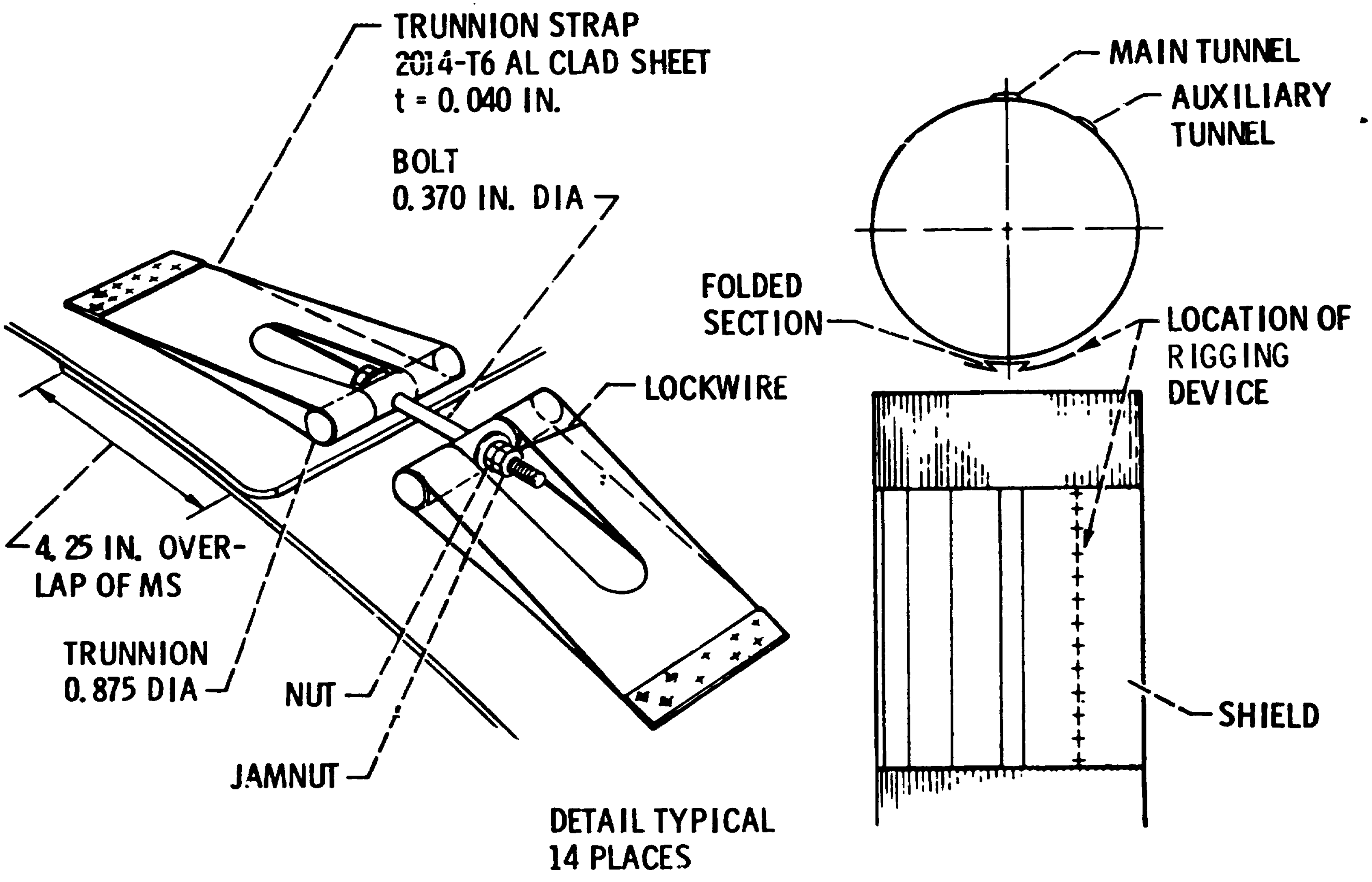


Figure VG4-8. Meteoroid Shield Trunnion Assembly

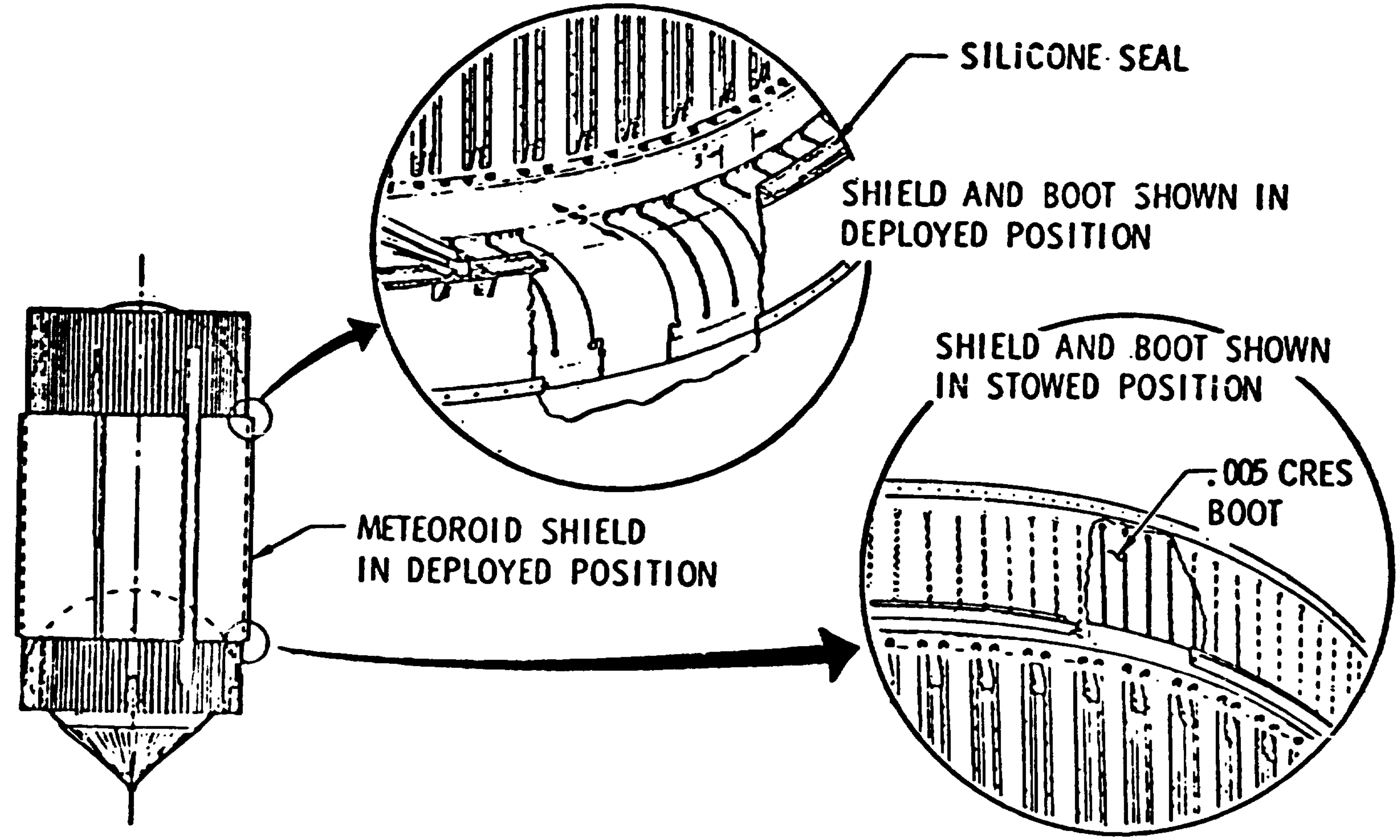


Figure VG4-9. Meteoroid Shield Boot

between shield and tank and provide meteoroid and thermal protection in these areas when deployed.

c. Development and testing. A meteoroid shield deployment verification test was performed at MDAC-W on May 9, 1971, to demonstrate proper separation of the three ordnance tension straps and deployment of the meteoroid shield. Two significant anomalies occurred in the test:

(1) The three ordnance tension straps separated properly and released the shield, but the expandable tube ruptured along a 5-in. length approximately one-third up from the aft end of the shield.

(2) The window side of the meteoroid shield came to rest circumferentially short of the fully deployed position because of friction and sagging caused by the 1 g effects.

Subsequently, an integrated redesign and retest program was initiated on the shield, the deployment mechanism and the ordnance, and was conducted at MDAC-W, MSFC, and KSC. The main items of redesign were:

(a) Corrections to the ordnance separation system (Figures VG4-10 and VG4-11):

1 Incorporated use of a drill-rod as an aligning device when drilling the bolt holes.

2 Replace 3/16-in. diameter bolts and locking inserts with tight fitting 1/4-in. Hi-lok fasteners; closed up bolt spacing to 3/4-in.

3 Reduced the length of the tension strap to reduce tolerance buildup and provide better alignment.

4 Added a smooth Teflon coating to the strap cavities.

5 Relocated the fracture groove from the inside to the outside of the strap to improve separation characteristics.

6 Increased the thickness of the material in the tension strap fracture groove from 0.013 to 0.016 to accommodate higher rig loads.

7 Lowered the nominal explosive core load in the expandable tube from 15.5 to 14.0 grains/ft to lower the expansion pressure.

8 Added the step of partial forming of the expandable tube during assembly, followed by a reannealing operation to improve material properties.

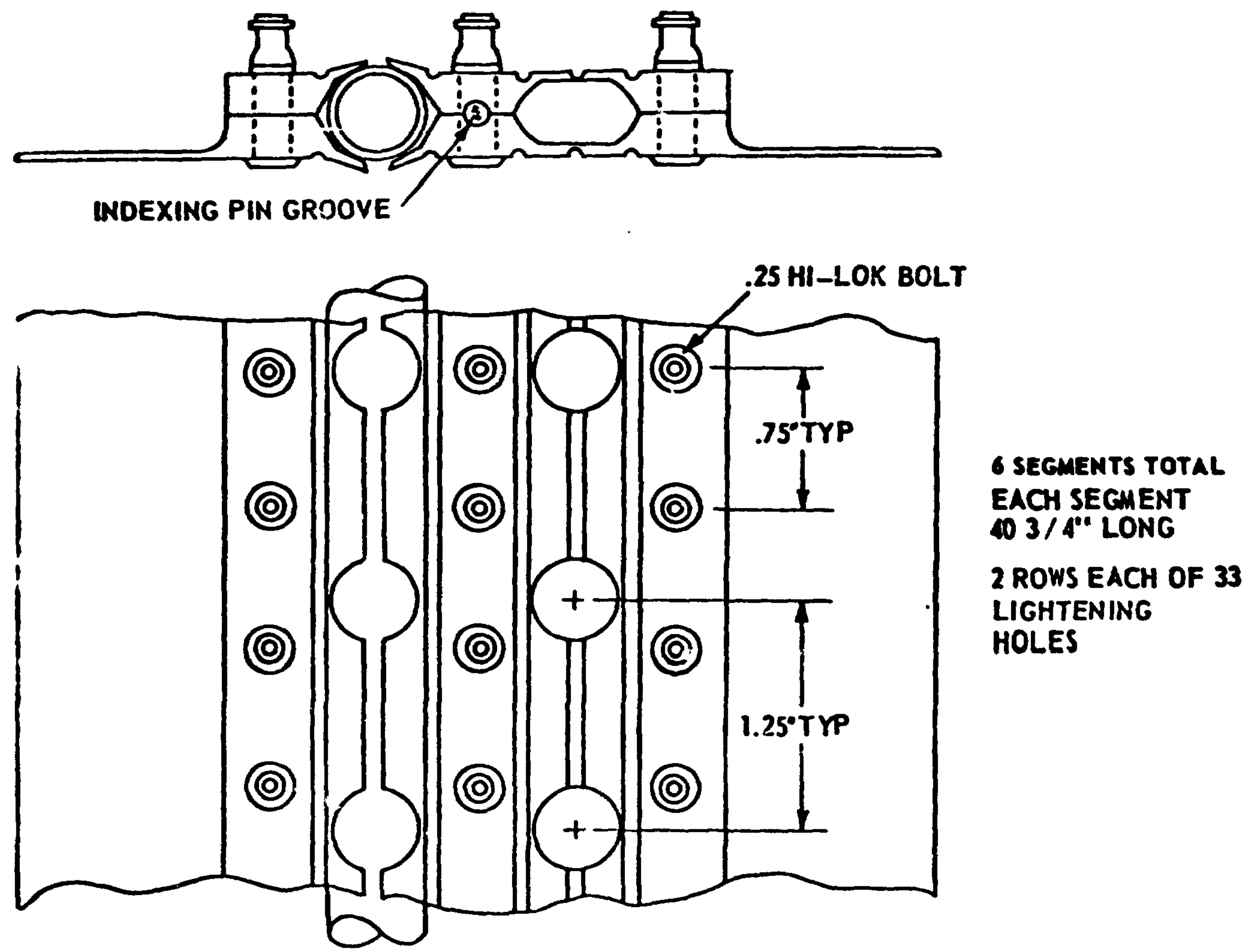


Figure VG4-10. Meteoroid Shield Ordnance Tension Strap Assembly

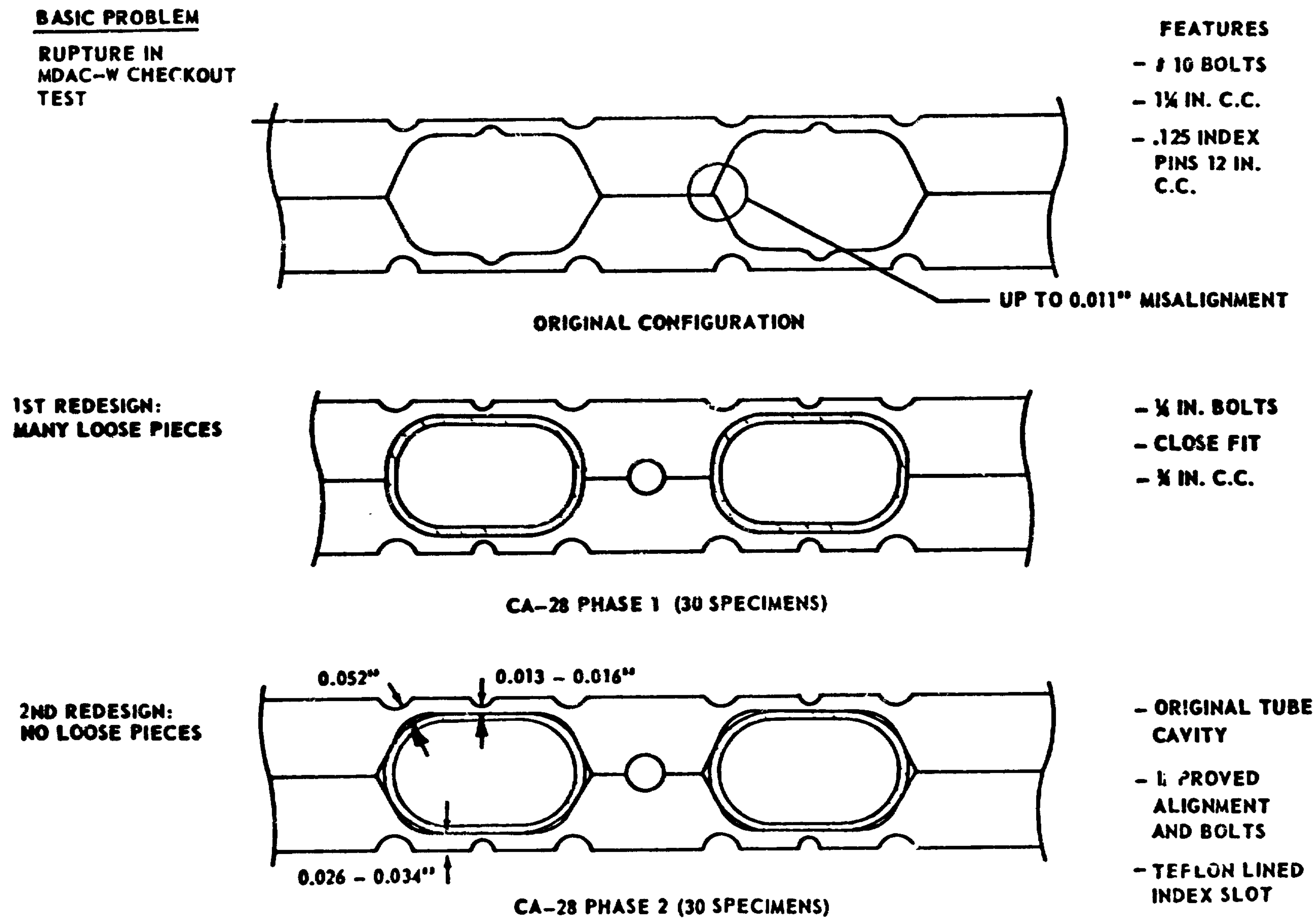


Figure VG4-11. Meteoroid Shield Ordnance Strap Design Evolution

9 Added a requirement to 100 percent metallographic inspect each deliverable length of expandable tubing, as well as checks of the tube hollows from which these lengths are drawn.

(b) Corrections to the deployment mechanism (Figure VG4-12.):

1 Incorporated latches at the four swing arms near the main tunnel to prevent shield rebound during deployment (Figure VG4-12).

2 Incorporated redundant strain gages on each torsion bar to measure the angular rotation/position of the shield.

3 Increased torque of the #1 and #8 forward and aft torsion bars and shortened some drive slots in the shield flanges to achieve more positive drive.

4 Added torsion springs in all hinge areas to assist in the deployment.

(c) Corrections to the meteoroid shield:

1 Incorporated several vertical stiffeners on the window and ordnance trunnion panel to prevent flutter during boost.

(d) Corrections to test hardware (Figure VG4-13):

1 Incorporated a counter balance "zero g kit" for ground deployment to minimize gravitational effects.

The main items of hardware development and test programs were:

(1) CA-28. Sixty subassemblies, consisting of primary and secondary 12-in. long expandable tubes with 4-in. long straps were tested to verify the tube ordnance charge change from 15.5 grains/ft to 14.0 $\pm$0.3 grains/ft and the redesign of the tension straps to prevent tube rupture. Some tabs (small metal pieces) along the fracture groove came off the tension straps during phase one of this test program. Testing was successfully completed in August 1971.

(2) CA-30. Three subassemblies consisting of primary and secondary 7-ft long tubes with tension straps were test fired, after acoustic and flight temperature (125, -140 °F) testing, in order to qualify the redesign. Testing was completed successfully in April 1972.

(3) CA-31. Two subassemblies consisting of primary and secondary full length tubes (22-ft long), tension straps, and fold-over panels were tested under rigged conditions. The tests were

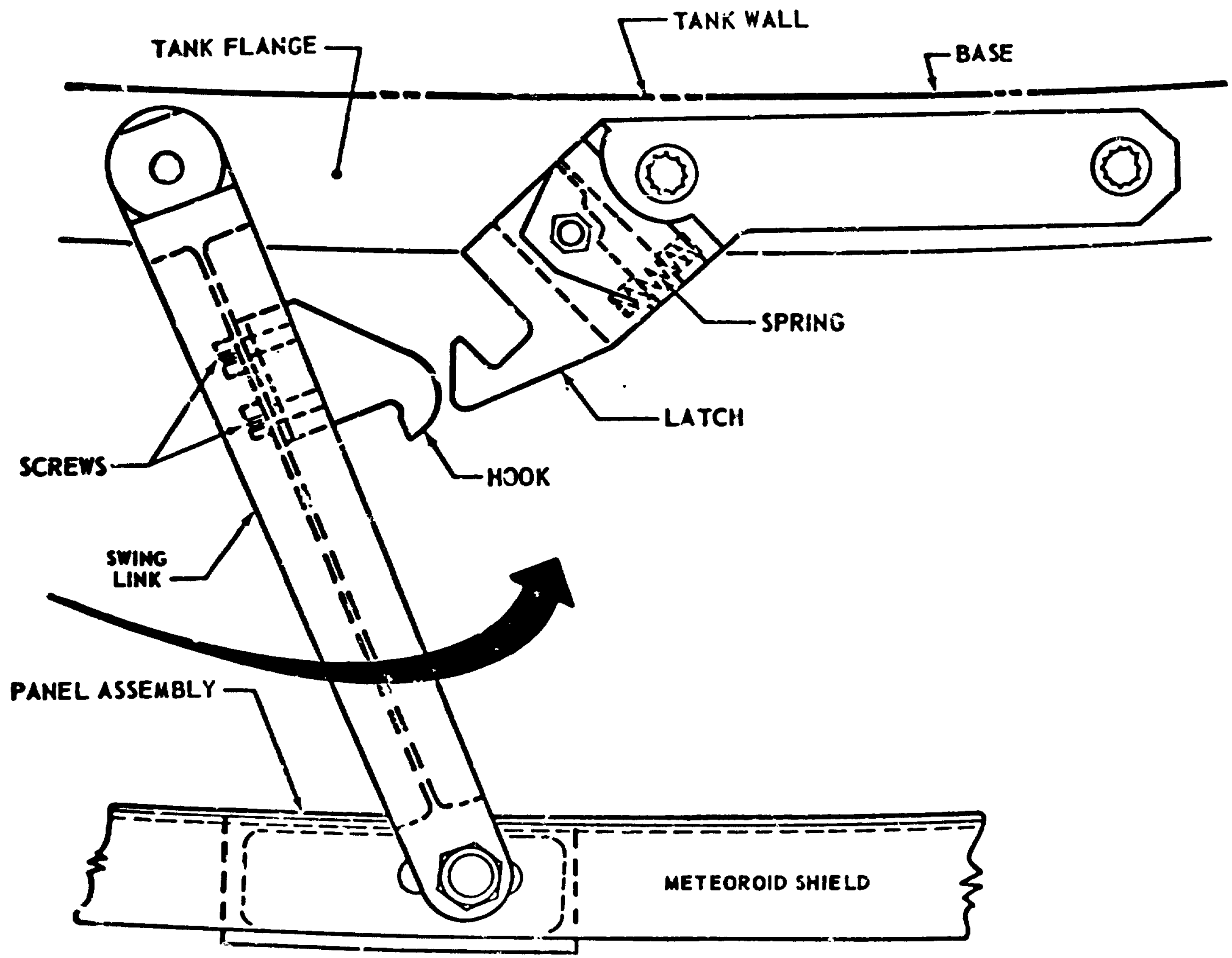


Figure VG4-12. Meteoroid Shield Deployment Latch Assembly

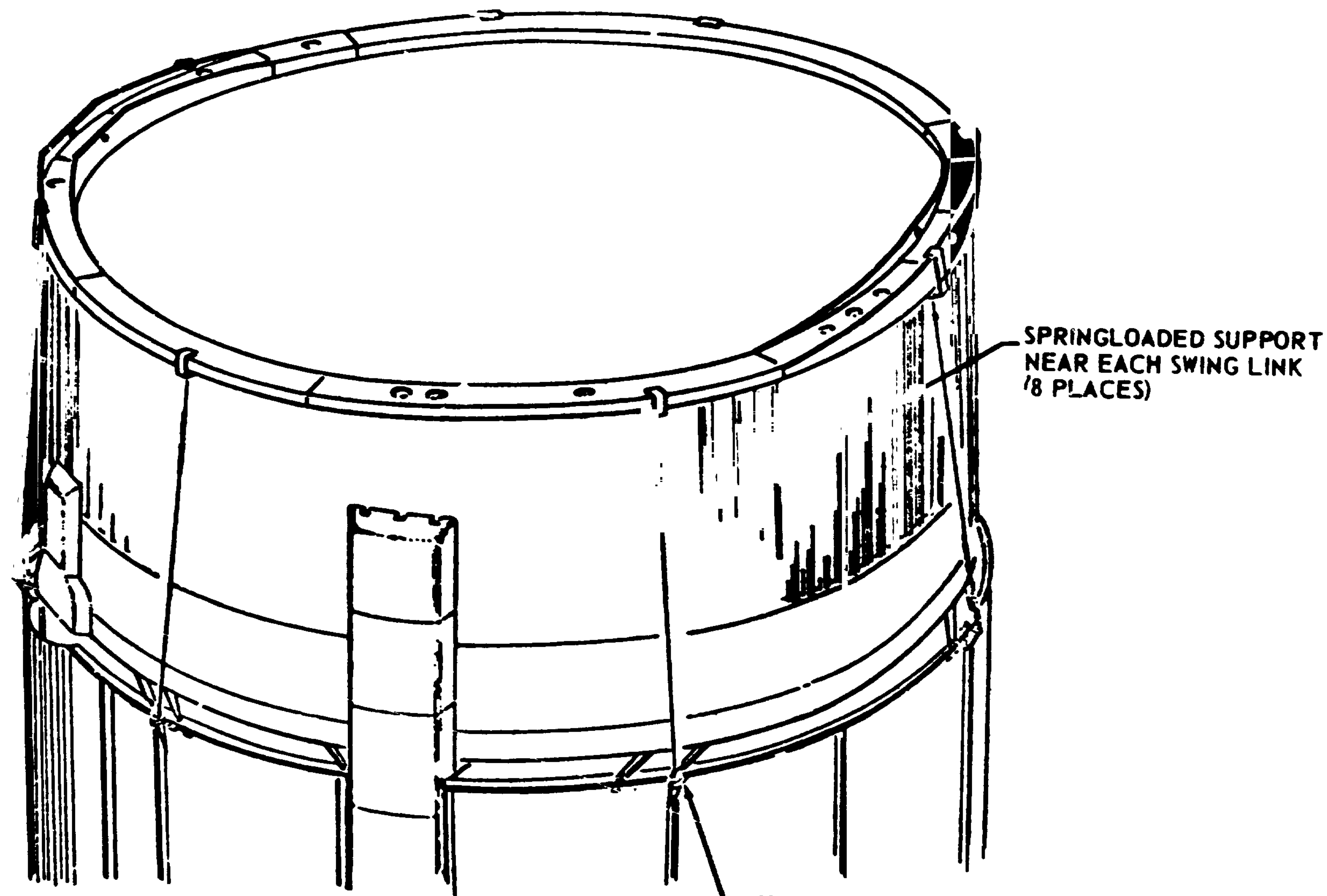


Figure VG4-13. Meteoroid Shield Zero-g Deployment Kit successful at ambient temperature except some tabs along the fracture grooves at the end of the forward and aft tension straps came loose when the backup tubes were fired. This was corrected by changing the hole pattern slightly at the ends of the tension straps. Testing was successfully completed in February 1972.

(4) CA-32. Eight strain gages mounted on a deployment torsion bar were tested under temperature, humidity, and torque environment to determine if they could maintain adequate stability and for an extended period of time with the bar in a torqued position. Results showed that the strain gages must be extremely well sealed to prevent drift at elevated temperature and humidity. Testing was successfully completed in January 1972.

(5) CA-34. Several swing links with latch hooks attached were mounted on an oscillating flywheel assembly and subjected to a series of latching cycles in order to verify proper latch operation over an expected range of meteoroid shield kinetic energy and momentum levels. Twenty-one tests were completed successfully in February 1972.

(6) ST-14. The workshop static test article at MSFC was fitted with a flight-type shield. Five mechanical deployment tests, using a pin puller mechanism (Figure VG4-14) rather than ordnance, and three ordnance deployment tests were performed utilizing the zero g deployment kit. During the first two mechanical and the first ordnance test, the forward end of the foldout panel leaned against the tank wall at the forward end. (When properly deployed, the foldout panel forms a chord away from the tank wall.) Four scroll springs were subsequently incorporated under the foldout panel to achieve proper standoff from the tank wall when deployed. Additional testing was performed to: (a) develop an adequate rigging procedure; (b) find the exact location and configuration of the capture hooks and latches; (c) find the right location for the deployment indication switches; (d) find ways to eliminate friction and interference; and (e) find a proper mounting arrangement for a silicon bulb seal at the forward and aft end of the shield. This seal acted as a weather seal. Also, adequate position indication of the strain gages was verified. Testing was performed during the period of February through April 1973.

(7) ST-28. As a result of an ultimate pressure test of 32.5 psig on the static test article at MSFC with the meteoroid shield in the rigged condition, three butterfly hinge lobe failures occurred. A laboratory test at MDAC-W repeated the failure under controlled conditions. Doublers were designed and riveted to the butterfly hinges in order to increase the number of total lobes per hinge. Matching doublers were bonded to the tunnel strap side for the same purpose. The result was an effective increase in hinge lobe strength. Laboratory testing showing a margin of 1.81 over the base line capability was successfully completed in June 1972.

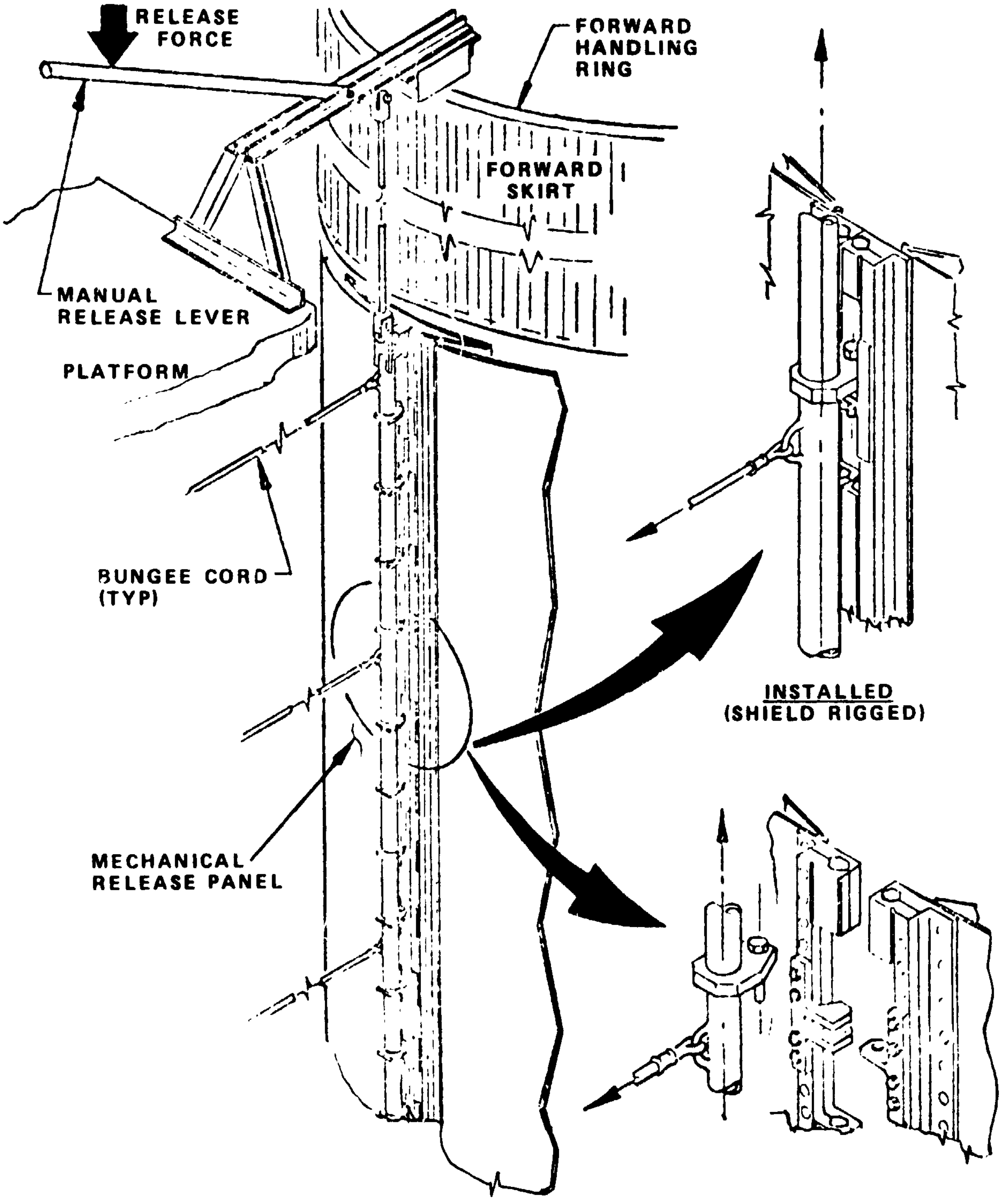


Figure VG4–14. Meteoroid Shield Mechanical Deployment Panel

(8) ST-38. During subsequent ultimate pressure testing to 31 psig on the structural test article at MSFC, strain gage measurements indicated that the #5 bonded tunnel strap doubler debonded between 28 and 30 psig. Inspection indicated that most other bonded doublers had partially debonded. An actual structural failure of the whole butterfly hinge/tunnel strap system had not occurred. Six control specimens, three specimens representing current bonding techniques and three specimens having controlled unbonded areas, were tested in the laboratory. An additional three specimens having a 50 percent bond defect, a 100 percent unbonded area, and a 100 percent unbonded area with a 4-layer resin-fiber glass "band-aid" fix were tested at MDAC-W. Test results demonstrated that the partially unbonded doubler and the "band-aid" fix provided a substantial margin above the required functional loads. An inspection was performed on the workshop at KSC. Some small partial debonds were detected but a fix was not deemed necessary.

As part of the KSC workshop rigging and checkout procedure KO-3018, performance of a mechanical deployment test using a pin puller mechanism rather than ordnance was required during October 1972 in order to verify the integrity of the redesigned shield and its components on the flight article. The shield was semirigged for transportation from Huntington Beach to KSC, i.e., the torque rods were slightly torqued in reverse direction to provide slight pressure of the shield against the tank. However, during the rigging for transporation, one torque rod was overtorqued because the bracket for the torque rod keeper was installed backwards. Also, one torque rod was sheared off because of torquing in the reversed direction. The flight item was reworked to correct these discrepancies. During the mechanical deployment checkout at KSC, the two capture latches on the window side of the shield did not engage. A subsequent failure investigation showed the following discrepancies:

(1) A thermocouple wire bundle running from the shield over the butterfly hinge line into the systems tunnel was too taut when the butterfly hinge folded.

(2) The capture hooks were not properly shimmed to their required position.

(3) The forward bulb seal caused too much friction against the swing links. This was mainly due to the whole shield being mounted 0.15 in. too far forward (above maximum tolerance). The tunnel straps were not properly adjusted during bonding at Huntington Beach. This required both of the 22-ft long butterfly hinges to be cut approximately in half.

This condition created a very flat V-shaped hinge line, which in turn introduced some friction.

(4) Several torque jam nuts on the torsion bars were not properly adjusted, causing additional friction in the swing link bearings.

(5) The zero g kit was slightly out of adjustment, causing additional friction just prior to the desired latch engagement.

(6) The magnetic deployment switches were readjusted since they failed to indicate deployment.

Upon correction of these problems, a "mini deployment test" was performed at KSC: with the torsion bars torqued and the shield in a deployed position, the butterfly hinge on the window side was manually moved toward the rigging position approximately 5 to 6 in. and then released. The capture latches engaged. Another full scale deployment test was scheduled as final verification and performed on October 22, 1972. During this test the forward capture latch on the window side again did not engage. A waiver was written to accept the engagement of any three capture hooks and latches for flight. All material review actions were worked and the shield was flight rigged from December 1972 through January 1973. During flight rigging, it became apparent that the meteoroid shield made only 62 percent contact with the tank wall, the thin aluminum sheets causing large bubbles. Several vertical joints of the shield were opened, the shield manually pressed against the tank wall, and the joints retightened. When the habitation tank was subsequently pressurized to 8 psig for leak check, the shield was remapped for contact and the contact area was then determined to be 95 percent.

d. Mission performance. The meteoroid shield did not meet the requirements to be free of "divergent instabilities" as required in the specification. The shield was forced out of the boundary layer and into a region where it was caught by the slip stream and torn from the vehicle about 63 sec into the flight. The most likely cause has been established as inadequate venting of the auxiliary tunnel that allowed pressure to buildup under the auxiliary tunnel and adjacent shield, forcing the shield into the slip stream. An extensive evaluation of the shield performance during the boost phase is provided in "NASA Investigation Board Report on the Initial Flight Anomalies of Skylab 1 on May 14, 1973."

The meteoroid shield primary ordnance flight data show a charge trace issued almost within 0.6 sec of the time specified. The safing command came 10 sec later and indicates that the ordnance command system was still functional. Then the backup command for meteoroid shield deployment, issued through the airlock digital command system, was given.

The data also show traces indicating charge fire and safing (Figure VG3-7). Since the ordnance train was no longer present this is only an indication that the primary and backup firing mechanisms and electrical bridge wires were still functional.

Although the meteoroid shield did not survive the launch environment, its loss did not drastically affect the probability of crew safety with regard to meteoroid penetration. The probability of crew safety for the longest planned manned segment of 56 days was 0.9999998 without the meteoroid shield. This is based on the crew having less than 4 min to evacuate before the pressure dropped to 3.6 psia. The probability of a meteoroid not causing leakage of internal pressure for the workshop was reduced for the various mission times as shown below:

	28 Days	56 Days	240 Days
With Meteoroid Shield	0.999+	0.999	0.995
Without Meteoroid Shield	0.995	0.990	0.958

These probability values applied at the start of the time segment and represented the projected success expected for the time segment. During the 8 months of the Skylab mission, no penetration was noticed or reported and no habitation area pressure loss was recorded.

e. Conclusions. Given the requirement for meteoroid protection on a new Saturn V type vehicle, no additional protection (justified on the basis of mathematical analysis and present experience together with habitation tank wall thermal coatings) or a rigidly mounted shield would be selected. Large and thin sheet metal structures with minimum support are extremely difficult to handle. Also, adequate tolerance provisions must be made as the material deforms easily. The venting of thin overlapping structures must be properly considered when exposed to rapidly changing pressure environments to prevent buckling or rupture.

The expandable tube separation system concept, when properly designed (in regard to explosive grain size, tube wall preparation, and tension straps), provides a clean and excellent mode of separation and is well suited for structural separations where contamination is an important factor.

Torque rods, properly designed and utilized, are a good way of storing deployment energy over an extended period of time without relaxation effects.

5. Workshop Entry Hatch.

a. General requirements for workshop hatch operation and function are:

(1) Withstand workshop launch and ascent pressure; leakage to airlock shall not exceed 1.90 lb/hr with airlock at a pressure of 0.5 to 14 psia.

(2) Check valves shall ensure that airlock pressure not be more than 1.0 psi greater than workshop pressure.

(3) Leakage during extravehicular activity shall be 26 sccs maximum.

(4) Handle operating force shall be no more than 25 lb.

(5) Be capable of 20 functional cycles minimum.

b. Mission performance. The performance of the hatch during the Skylab mission, with regard to the above requirements, is difficult to assess since no actual measuring devices were directly associated with the hatch. It is pointed out that several qualification tests were performed and the results are shown in the following paragraphs. During the launch and ascent phase, the hatch performed as designed with no excessive leakage. When the entry hatch was used as the aft airlock extravehicular hatch, no leakage was detected. During the debriefings, the crew stated that the handle loads for each of the nine extravehicular activities were not excessive.

The nine extravehicular activities during the three manned phases constitute the total cycling (open-close) imposed on the hatch. The hatch was left open for the unmanned storage periods between the three manned phases and was also left open at final deactivation.

c. Anomalies. During initial pressurization cycles of the workshop during the first unmanned period, the airlock/docking adapter was being prematurely pressurized by some unknown leakage source from the workshop. Normally, the workshop should be pressurized to 5 psi followed by equalization of the airlock/docking adapter pressure. However, the airlock/docking adapter was being simultaneously pressurized with the workshop, but was lagging the workshop pressure by approximately 0.2 to 0.3 psi, until the pressure reached 5 psi at which time the airlock/docking adapter/workshop pressure then equalized. Although this leakage was of no great concern during the initial pressurization, it was a consideration for extravehicular activity operations because at that time the hatch was used as the aft closure of the airlock. Therefore, if the leakage was determined to be through the hatch seals or check valves, it could be of considerable consequence. The check valves were suspect since they were opened during initial workshop blowdown. As a precaution that they might be leaking, the first manned crew devised makeshift "flapper valves" from onboard materials (mosite and tape) which they applied to the workshop side of the entry hatch check valves when the hatch was closed. It is doubtful that the flapper valves were ever effective. The nine subsequent Skylab extravehicular activities were performed with no leakage reported. Several possibilities exist for the source of the pressurization leak: (1) Dirt or some foreign material did not allow the check valves to seat, causing the leakage. The foreign material could have become dislodged when the hatch was initially opened; (2) The hatch seal could have leaked, however, inspection by the first manned crew indicated that it was smooth, uniform and in good shape. Test data concerning leakage of the hatch and check valves show that the requirements listed in paragraph a. were not exceeded. The test results are:

(1) Access Hatch Qualification Test Report MDC G3363 Production acceptance test on the qualification hatch at 10.5 psig with helium gas (no check valves on hatch): leakage was 0.51 sccs. Pressure cycle test and subsequent leakage at 10.5 psig with helium gas: leakage was 0.68 sccs.

Two previbration leak tests were made at 10.5 psig. Leakage was 1.87 and 2.45 sccs. A post-vibration leak test showed the leakage to be 1.96 sccs.

(2) Access Hatch Repeat Cycle Qualification Test Report MDC G3379. Leakage at 5 psig after 0,25,50,75, and 100 cycles of opening and closing the hatch was 1.20, 0.50, 1.30, 1.00, and 0 scims, respectively.

(3) Access Hatch Check Valve Qualification Test Report MDC G3375. Previbration leak test at 3.3 psig with GHe in the check direction: leakage was 6.10 scims. At 26 psig the leakage was 1.71 scims. The test requirement for the previbration leak test of the hatch check valve was for not more than 350 scims. Post-vibration leak test of the valve at 26 psig with GHe in the check direction: leakage was 17.70 scims. At 20, 15, and 8.3 psig the leakage rates were 14, 9.46, and 5.50 scims, respectively.

The test results show that the leakage of the test hardware was less than the leakage rate allowed by the requirements. Therefore, if the pressurization leakage problem was through the hatch, the aforementioned speculation of foreign material in the valve seat area would be the most logical explanation.

d. Recommendations. Use of check valves to control pressure should be exercised with caution. Check valves are potential leak sources and should be considered as such.

6. Scientific Airlocks.

a. General requirements. The general requirements specified for the scientific airlocks (+Z and -Z) are:

(1) Withstand all the loads and environments associated with launch, boost, and orbital operations.

(2) Be capable of withstanding astronauts applied loads in operating the outer door crank in the installation and removal of experiments or from physical contact of the astronaut against the structure; or applied to an installed experiment.

(3) Have an operation life of 100 functional cycles.

(4) Have a maximum allowable leakage of 3.6×10^{-3} sccs.

(5) Eliminate condensation and contamination of experiments mounted on the scientific airlock by providing a desiccant canister and particulate filter which shall be plumbed to both scientific airlocks.

2. Mission performance. The scientific airlocks (+Z and -Z) were designed to mounting/deployment several scientific experiments. However, due to the loss of the meteoroid shield, the +Z scientific airlock was not used for the experiments intended, but for mounting/deployments of the JSC parasol that was the temporary fix for the meteoroid shield anomaly. It performed successfully. The location and configuration of the scientific airlocks are shown in Figure VG6-1.

Operation of the -Z scientific airlock was found to be satisfactory for the Skylab mission, and there were no known experiment anomalies that could be associated directly to it.

3. Anomalies.

(1) During the first manned phase, a momentum buildup indicated a small thrust from the area of the +Z scientific airlock. The vacuum source quick disconnect was inspected and no evidence of a leak was detected. The crew decided to take a cautious approach and installed the vacuum hose with its cap.

(2) During a condensate dump (HK-60B) of the second manned phase, one of the crew left the -Z scientific airlock outer door "open" and the valve in the "press" position after a vacuum was established in the tank. With the desiccant system valve in the "open" position, the outer door open and the valve in the "press" position, cabin air will bleed overboard through the desiccant cannister and out of the scientific airlock. The leak was discovered during the crew sleep period. When the crew awoke, they reconfigured the scientific airlock properly.

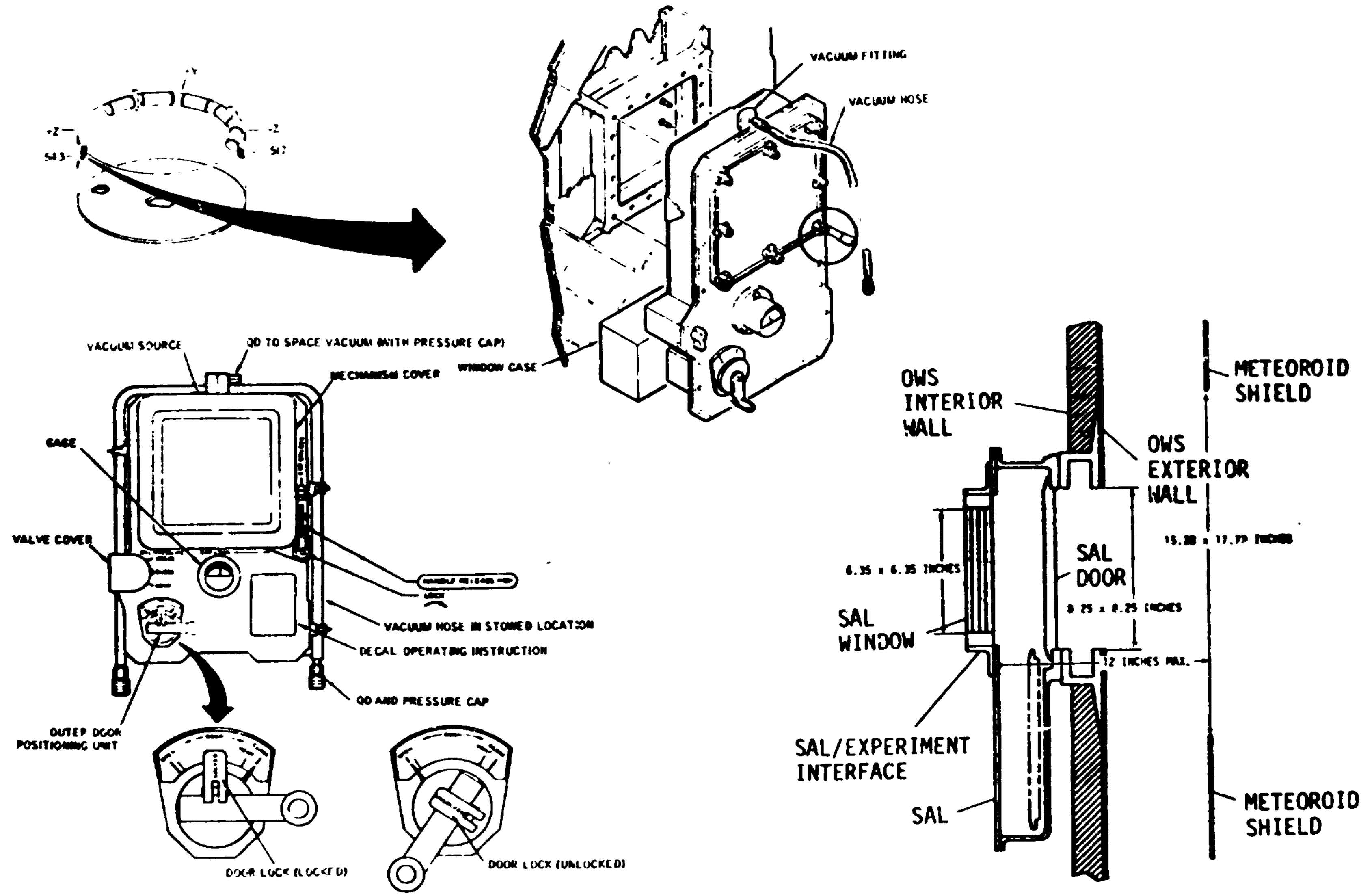


Figure VG6-1. Scientific Airlock

(3) During a wardroom window evacuation (HK-84K), during the first manned phase, the wardroom window cavity was back-filled with desiccated cabin air. The commander expressed concern for the experiment which used the scientific airlock because the desiccant system did not "appear" to remove the moisture and the "cold" experiments could be exposed to gas of a questionable moisture quantity. This concern was, to a large degree, based on the exposure time of the desiccant (about 10 hr).

However, an analysis indicated that the desiccant cannister was not saturated. This analysis used the manufacturers delivered condition, predelivery testing, and onboard use to compare unit capacity. The analysis is supported by the crew observation that the fogging in the wardroom window appeared to be emitting from the port opposite the desiccant supply port which would indicate a leak from the cabin. It was decided not to actually measure the quality since special equipment would have to be flown up for any accurate evaluation.

d. Recommendations.

(1) The crew suggested the pressure gage indicator and face be more visible because reading the pressure required a pen light.

(2) Although evacuation and repressurization times were in accordance with those specified, the crew felt larger lines should be used. This would reduce the unproductive crew time associated with the scientific airlock evacuation and repressurization.

7. Trash Airlock.

a. General Requirements for the trash airlock are:

(1) The Trash Airlock shall be designed to perform nominally in a 5 $\pm$0.2 psia external, and 0 to 5.2 psia internal at ambient temperatures.

(2) Its service life shall be 1500 complete functional cycles.

(3) The proof pressure shall be 10 psid and shall be capable of withstanding a malfunction pressure (either burst or crushing) of 26 psid.

b. Mission performance. The only instrumentation on the trash airlock is the absolute pressure gage. Based on crew comments and subjective evaluation, the trash airlock met or exceeded the structural, life, and function design requirements. It was cycled 656 times with no mechanical malfunctions. There were several operational difficulties that were successfully resolved. Two incidents of near jamming were attributed to overfilled trash bags. Further problems were avoided by better control of trash combinations during disposal. It was noted during the first manned phase that the valve handle was inadvertently kicked or left in an intermediate position, between PRESS and VENT, which caused a cabin atmosphere leakage of about 3.8 lb/hr. This problem was overcome by strapping the handle in the PRESS position between operations.

During the second manned phase, an operating characteristic of the trash airlock was highlighted. The force required to squeeze the lid during the initial portion of the latching operation was high. It was found that the high latching force could be overcome by technique or use of two crewmen.

c. Anomalies. On mission day 41 of the second manned phase, the crew reported the interlock push rod between the valve handle and the inner door latch (Figure VG7-1) was bent. The bent rod did not affect the operation of the airlock. Bending of the rod could have occurred if the door latch was operated when the valve handle was not in the pressurize position. Investigation of video tape on the trash airlock indicated that sufficient interlock operation remained to prevent placement of the valve/outer door handle in the vent position when the lid lock was not engaged.

d. Recommendations. Two improvements are recommended for future use:

(1) A means to positively maintain the valve handle in the PRESS or VENT position should be added to the handle.

(2) The squeeze force during the latching operation should be reduced.

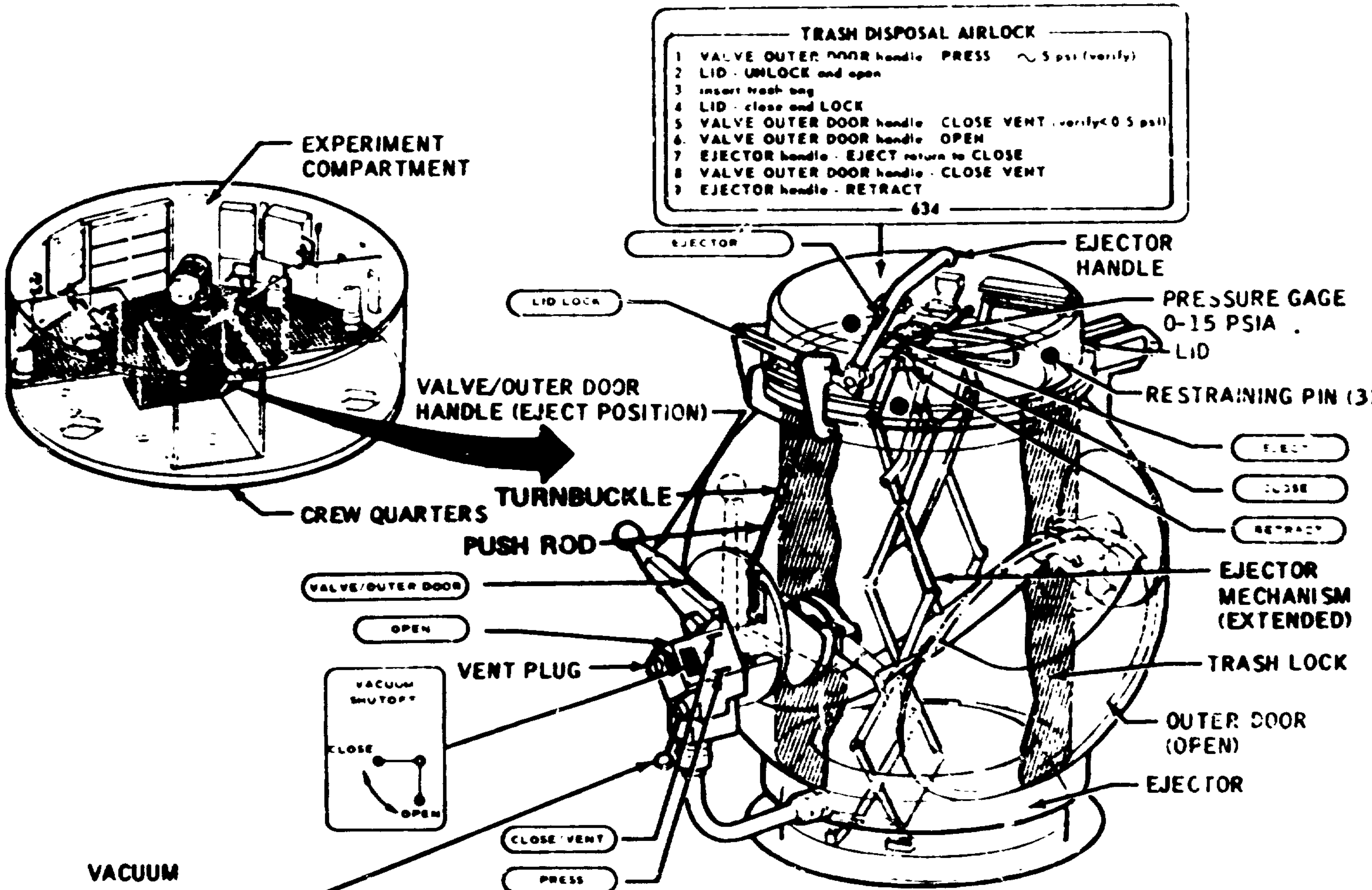


Figure VG7-1. Trash Airlock

8. Thermal Shield. The MSFC Skylab thermal shield was developed to be deployed during extravehicular activity over the sunside of the workshop. It would serve as a shield replacing the original thermal/meteoroid shield that was lost during the first launch phase.

a. Candidate designs. Several candidate designs as well as material types were evaluated. Among them were:

(1) Stretching a shield over the exposed outer skin of the workshop and by the use of hook-type bungees attaching the sheet to the corrugation on the workshop outer cover. This idea was discarded because of potential problems of extravehicular activity translation and the possibility of a crewman being entangled in the sheet.

(2) Constructing a shield made from an inflatable Tedlar tube and hanging it over the side of the workshop. Potential maintenance problems eliminated this.

(3) Various methods of attaching clotheslines to tubular booms and pulling a thermal shield down the side of the workshop from a position in the fixed airlock shroud.

(4) Attaching clotheslines to the bottom of the workshop with clips while conducting a standup extravehicular activity from the command module and deploying the shield from the fixed airlock shroud area.

From the two remaining possibilities above, the idea of using a tubular "A-frame" boom with attaching clotheslines came about. The "A-frame" boom would be secured from a position on the telescope mount outrigger. From here the thermal shield could be attached to the clothesline and subsequently deployed over the sun side of the workshop (Figure VG8-1).

b. Components. The MSFC thermal shield assembly consisted of the following components:

(1) Two deployment poles (Figure VG8-2): Each pole is assembled from 11 lengths of 1-in.-diam aluminum tubing, 4.7-ft long. A 3/4-in.-diam rod (guide pin) protrudes from one end of each tube, the other end of each tube is slotted to provide the means of fastening the poles together. A sleeve installs over the slot to increase the strength of the joint. The interchangeable poles are joined by inserting the guide pin of a rod end into the slot of a sleeve end and twisting the poles. A locknut, located on the rod, is screwed down against the sleeve and an O-ring is rolled into position behind the nut to prevent loosening.

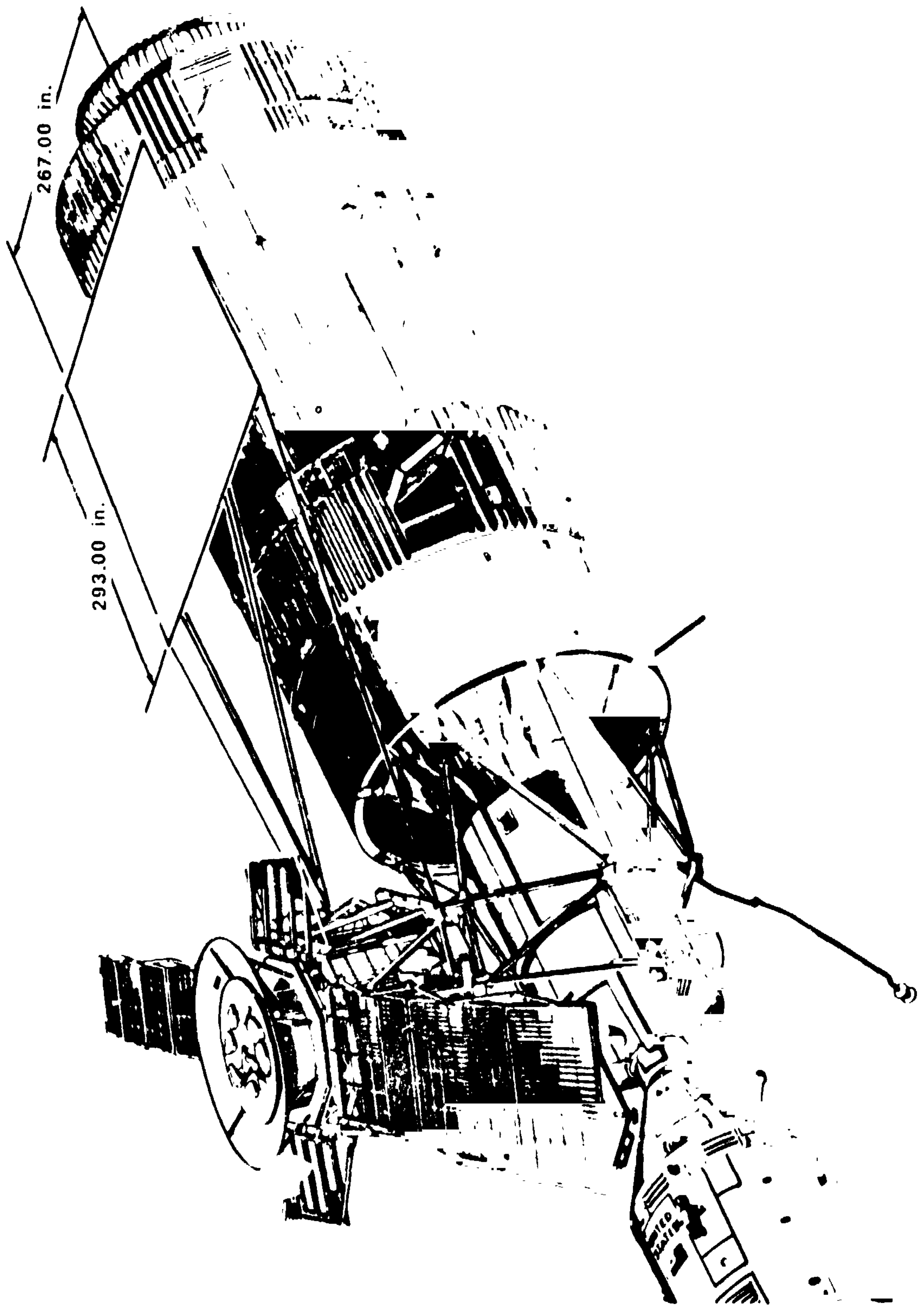


Figure VG8-1. MSFC Thermal Shield

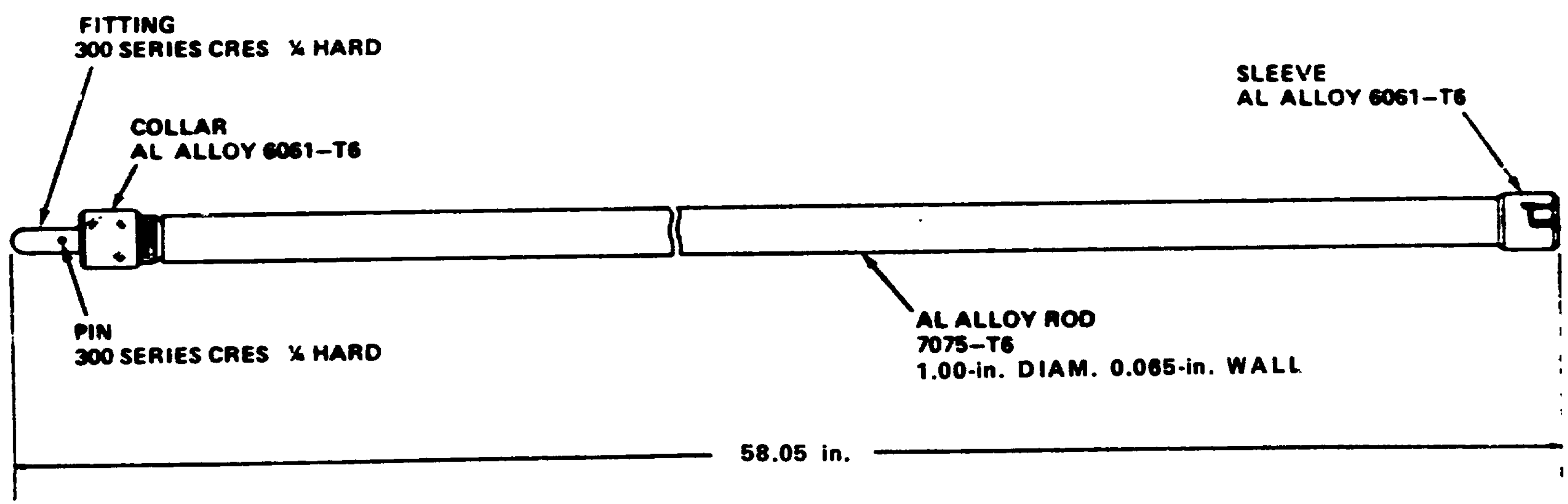


Figure VG8-2. Deployment Pole

Two "quivers" serve as storage pallets for the poles. The "quiver" is a 7.75- by 62.75- by 0.062-in. aluminum sheet and restrains 24 pole sections. Two extra poles were used with the special hardware cutting tools, developed by MSFC to free the undeployed workshop solar array.

(2) Two clothesline assemblies (Figure VG8-3): These serve as a means of deploying the thermal shield along the poles. The assembly consists of a 1/4-in. diam PBI (Polybenzimidazole) line approximately 105-ft long. The line is threaded through two "eye bolts" and the ends joined by sewing and wrapping with thread. The interior of each eye bolt is lined with teflon to reduce friction as the clothesline slides through them. One eye bolt has a male and a female fitting, for attachment to the shield poles and base plate, respectively. The other eye bolt has only a female fitting for attachment to the extreme end of the shield pole. The clothesline also has two steel "D" rings, spaced 25 ft apart, that are attached to the line by wrapping with thread. The thermal shield is attached to these rings during deployment.

Two clothesline assemblies are used. For identification of right and left poles, the metal parts of one assembly are colored green and the metal parts of the other assembly are colored red. Each assembly weighs about 5.8 lb and is packed in a similarly colored pallet assembly that weighs 4.6 lb.

The pallet assembly consists of a 1/8-in. thick aluminum base plate with elastic bands routed through holes in the plate. The elastic retains the folded clothesline assembly. A beta cloth cover, with one end bonded to the plate bottom, is placed over the packed clothesline assembly and again attached to the plate bottom by a Velcro fastener, thereby serving as an additional clothesline assembly retainer and providing protection as well.

(3) One base plate assembly (Figure VG8-4): The base plate assembly is mounted to the telescope mount +Y outrigger structure. The two deployment pole assemblies attach to the aluminum base plate assembly by a bayonet-type positive lock joint as described in paragraph (1). The two joined poles, with the apex at the base plate, form an A-frame boom with an angle of 24° between the poles. The base plate has the capability of accommodating a rotation of the A-frame pole assembly in order to position the poles and shield against the skin of the workshop. A locking mechanism was provided in order to ensure a positive lock for the pole/shield assembly in its final location.

(4) One thermal shield (Figure VG8-5): The thermal shield was made from a material of nylon ripstop over aluminized mylar and coated with an RTV based thermal control paint (S-13G). The finished shield measured 22 ft 3 in. by 24 ft 5 in. It was contructed by sewing (using double needle commercial sewing machines) 3-ft-wide strips of material with PBI thread. The edges of the shield were channeled to hold a 0.25-in.

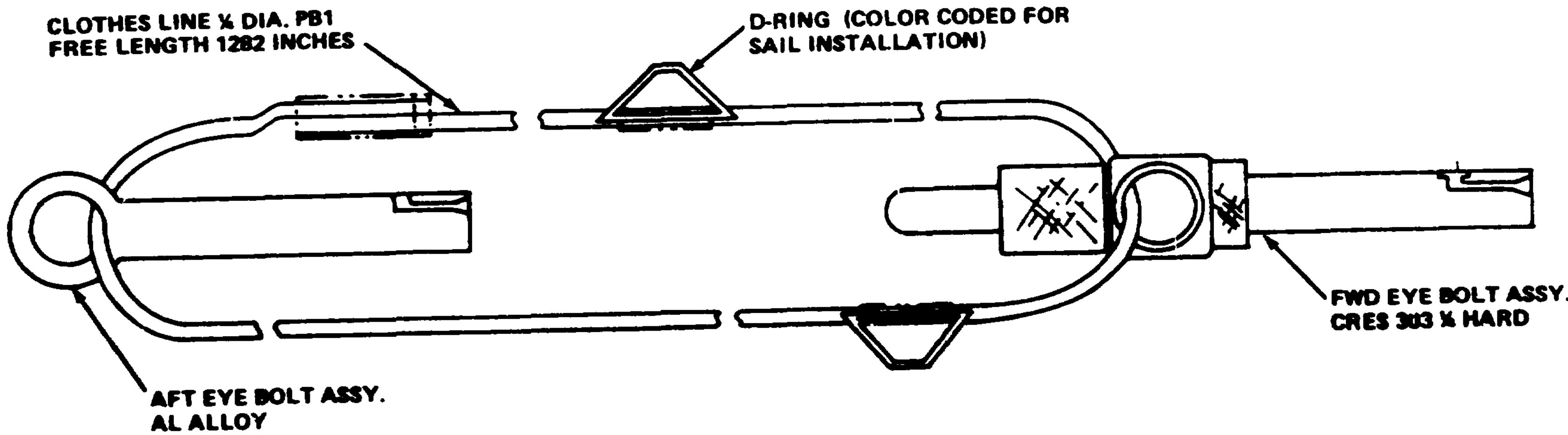


Figure VG8-3. Clothesline Assembly

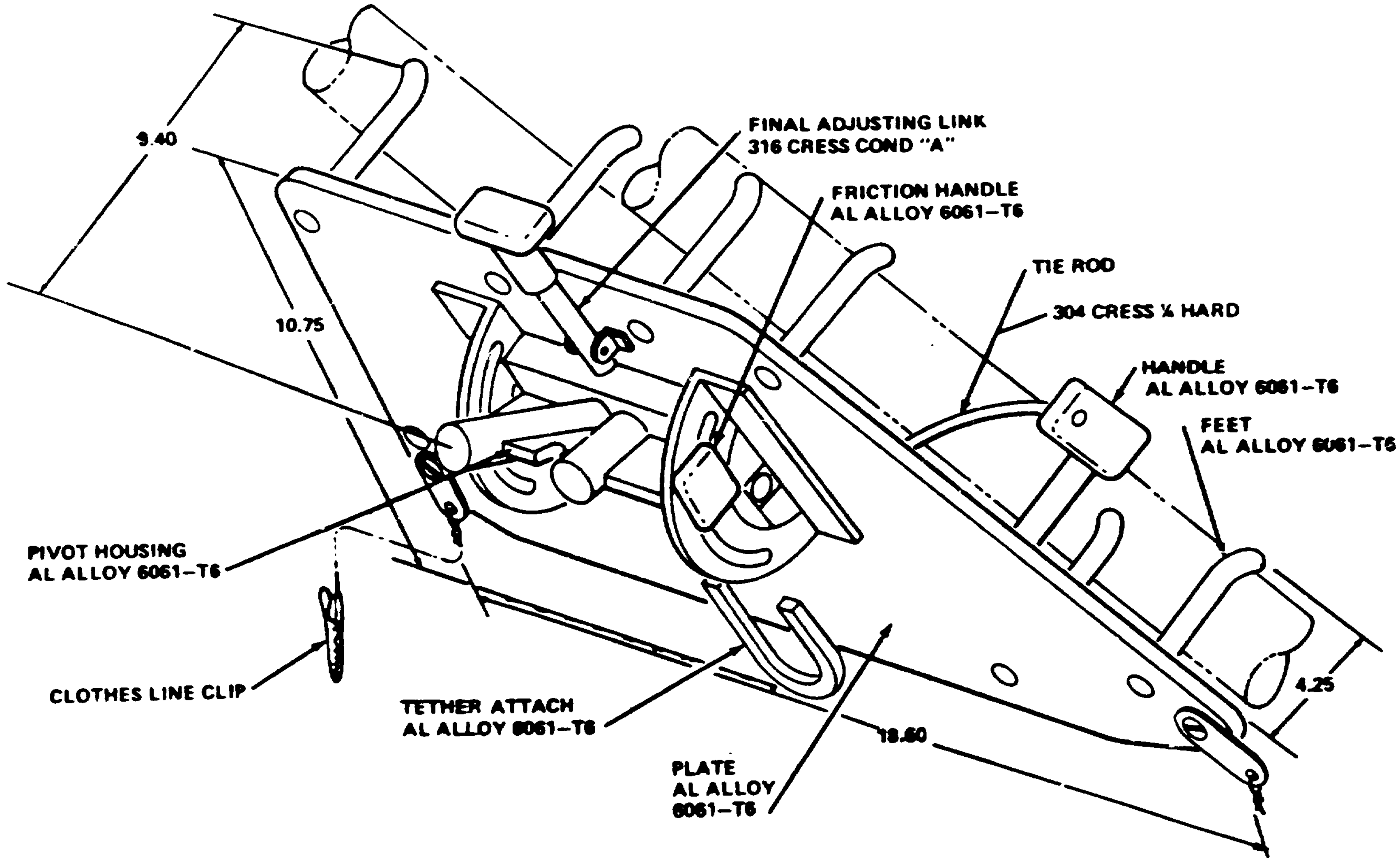


Figure VG8-4. Base Plate Assembly

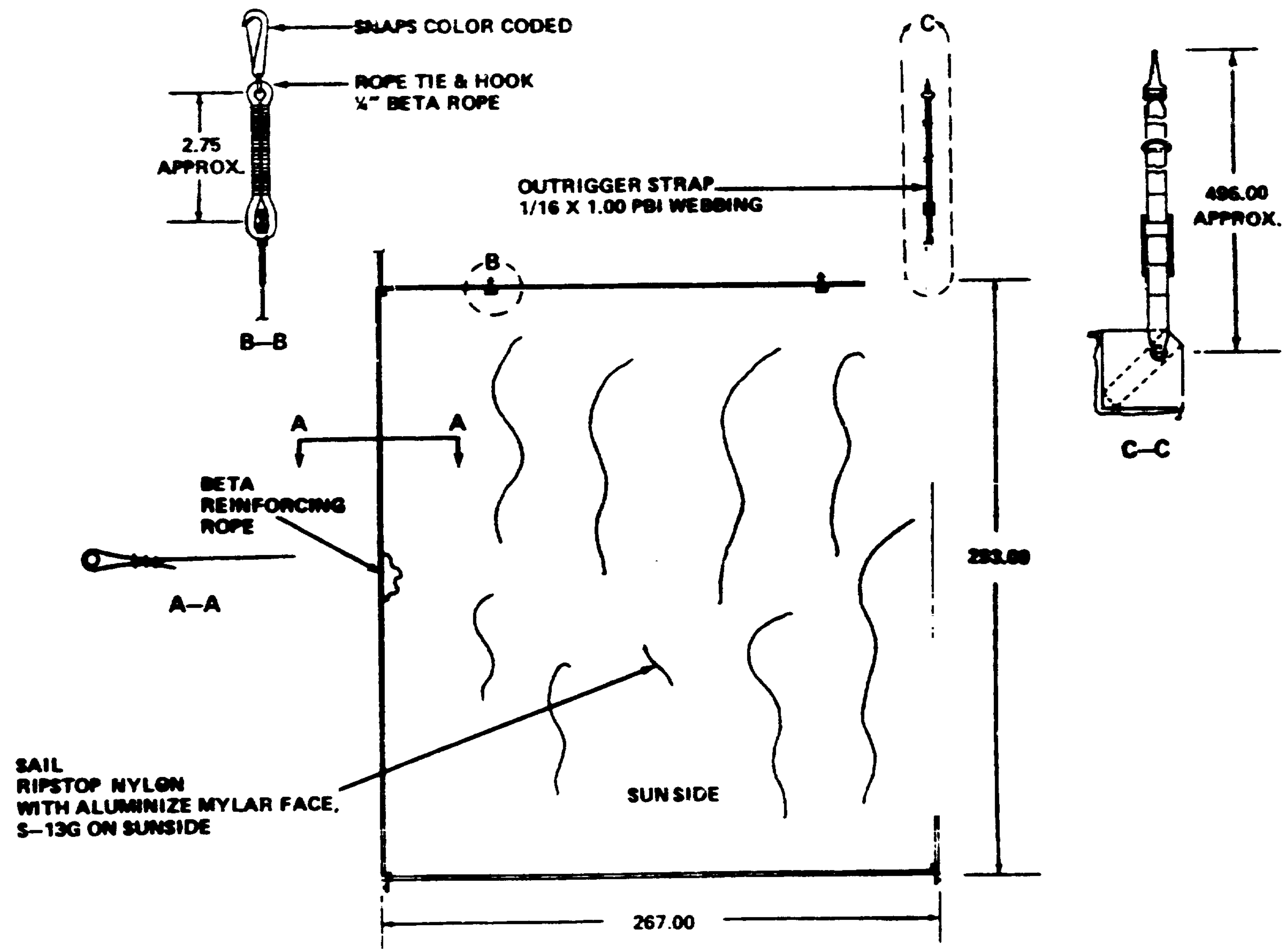


Figure VG8 5. Thermal Shield Assembly

diam teflon coated beta glass rope. Six grommets were sewn into the corners and edges for attaching PBI strapping. In the top two corners of the shield, a 1-in. wide by 0.16-in. thick webbing approximately 41 ft long was attached. The other ends of each of these straps were attached, at shield deployment, to the +Z and -Z outrigger structure, respectively. This allowed the shield corners to be stretched out. At the four other grommet locations, hooks were attached. These hooks were attached to the clothesline "D" rings at shield deployment. To ensure that the correct side of thermal shield was facing the sun, all hooks, straps, and corners were color coded, red and green, to match the color code of the clothesline assemblies.

The shield was folded for flight using an accordian fold to prevent air pockets in the package and also for ease of deployment. The folded shield with straps and buckles was packaged in a 14- by 14- by 11-in. beta cloth and Velcro zippered bag. Located on the outside base of the bag is a tether hook that is used to tether the bag to the base plate assembly.

(5) One foot restraint plate assembly (Figure VG8-6): The foot restraint plate assembly was developed to restrain the astronaut while at the thermal shield deployment work station located at the telescope mount. This assembly was composed of an astronaut extravehicular activity foot restraint and a foot restraint adapter assembly. The astronaut extravehicular activity foot restraint was an existing "onboard" piece of hardware, located in the workshop at launch, and thus was not additionally required as part of the overall thermal shield assembly. The foot restraint was used as is with no modifications required to fit the new adapter assembly. The adapter assembly was an aluminum and stainless steel structure designed to interface with the existing workshop foot restraint. The adapter was designed for attachment, by the astronaut, to the +Y telescope mount outrigger structure.

c. Testing. Extensive qualification testing was accomplished considering the tight time frame of design. Test requirements were established and static and dynamic structural testing of the deployed twin pole shield configuration was accomplished. The tests confirmed that the twin poles would successfully withstand the load conditions that would be experienced during shield deployment.

Tests were also conducted on the thermal shield material and the S-13G coating, regarding optical properties and breaking strength. Thermal cycling tests were also conducted and the results were adequate. The complete shield, folded for flight and bagged, was subjected to a vacuum chamber test at 5×10^{-2} torr to guard against the possibility of the shield trapping air and expanding rapidly when exposed to command module or space atmosphere.

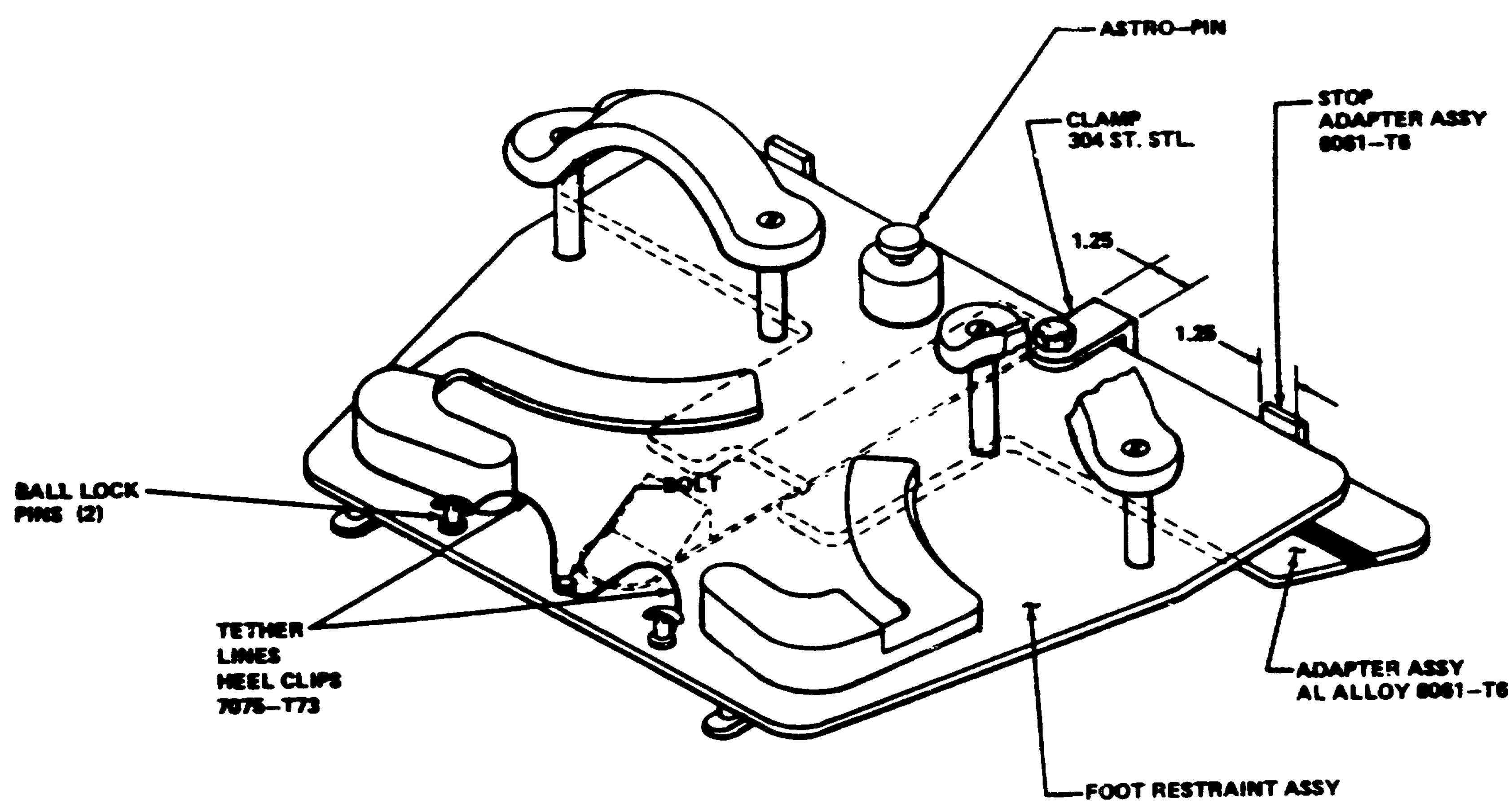


Figure VG8-6. Foot Restraint Assembly

The entire assembly was checked out in a full end-to-end simulation in the neutral buoyancy simulator facility.

d. Functional performance. Actual thermal shield deployment on the orbiting workshop went smoothly. The poles presented a small problem when they were unfastened from their storage pallets. The large "O" rings snagged when the poles were pulled through the elastic restraining straps. Also, during assembly of the poles, the clothesline was found to be twisted around the deployment pole. This was corrected by breaking a pole joint and rotating the outboard end of the poles. Concern was expressed that the handling and insertion of the assembled pole into the base plate in zero g might prove to be a problem as well as the possibility that an oscillation might occur in the poles that would be difficult to dampen. However, this never materialized and the assembly operation went smoothly. The shield itself was noted to have retained some of the accordian folds from packaging, but proved to be no problem. The twin-pole thermal shield was a major repair element that ensured successful completion of the Skylab mission.

9. Waste Management System. The waste management system provides the supplies and equipment necessary for hygenic collection, processing, storage, and return or disposal of waste products (feces, urine, and vomitus) for the three crewmen of each mission. A vacuum cleaner is supplied to collect free-floating debris within the Skylab.

a. Fecal collection and processing.

(1) General requirements.

(a) Fecal collection. The fecal collector shall provide the capability to collect and contain all consistencies of fecal matter. Requirements for the collector are:

1 The collector shall provide a positive means to ensure separation, collection, and containment of the feces and wiping material.

2 The collector shall not alter the constituents of of the fecal material (including water) until a mass measurement has been performed and its results recorded. After the mass measurement, the fecal collection shall be vacuum dried.

3 The maximum duration of each complete defecation cycle (excluding defecation) shall not exceed 15 min. The cycle shall include system preparation, initiation of processing, and preparation of the waste management system for the next cycle. The system shall be designed to limit initial preparation time to no more than 30 sec.

(b) Vomitus collection. Contigency fecal bags shall be provided to collect and contain vomitus material from the crewmen and shall interface with the waste processor and the specimen mass measurement device.

(c) Waste processor. The waste processor shall provide for vacuum drying the fecal and vomitus collections and debris collections (if necessary) so that the waste products therein are deactivated and bacterial contamination is prevented. Requirements for waste processing are:

1 All fecal and vomitus collections shall be vacuum dried.

2 Each processor shall have the capability of being individually controlled and shall include a display to indicate when each specimen has been deactivated and is ready for storage.

3 Six processors shall be provided.

4 The qualification and performance requirements for the waste processor module are:

a Operating temperature-- 58 to 90 °F.

b Service life--9,000 hr (including 3,360 hr active orbital environment); cycles--7,800 operational (280 cycles/chamber 1 to 1 1/2 hr on, 15 min off with manifold pressure 5 psi below ambient).

(2) Mission performance.

(a) Fecal collection. The fecal collection equipment worked successfully and the crews expressed general satisfaction (Figure VG9-5).

The airflow system of collecting feces was reported to be a good concept and worked exceptionally well. However, it was felt that higher airflow would provide even more satisfactory results. In order to obtain the proper seal required for good airflow collection the hand grips were used. It was reported that excessive pulling force on the grips was required to attain a seal.

Minor difficulties were encountered installing the bag in the fecal receptacle as the second cuff was occasionally difficult to install on the receptacle. It was also reported that several cuffs debonded; these bags were discarded.

The time required to accomplish bag sealing, mass measuring, and processor loading was not considered excessive. Odor control was satisfactory and noise level was acceptable except during sleep periods. There were no fecal bags damaged during use and no filter or seal leaks.

The bag/processor interface was satisfactory. The four top chambers were used to process feces; the two lower chambers were used to dry desiccants (PGA and film vault). At various times during the Skylab mission all chambers were in use.

(b) Because of a procedure used by the crews, the SMMD mass/time processing curves were never used. The SMMD was exchanged for the wardroom SMMD sometime during the first manned phase. There was no noted difference between the specimens dried with or without heat.

(c) Waste processor. The processor was used to dry feces and suit drying desiccants only. The processor module (Figure VG9-1) control panels, valves, doors, dampers, and pressure plate mechanism operated satisfactorily during the Skylab mission. No quantitiative evaluation can be made of the drying times. However, all returned samples were acceptable for medical analysis. It was determined that the number of collections during the mission was approximately half the design requirement. Also, it was determined during flight that feces can be dryed without a heater element by extending the processing times by approximately 50 percent.

(3) Anomalies.

(a) Collection module. No anomalies reported.

(b) Fecal bags. Several of the black rubber outer cuffs came loose from the fecal bags. These bags were discarded and replaced with new bags.

(c) Processor module. The processor chamber failed to evacuate when attempting to dry pressure suit desiccant during the first manned phase. It was determined by the crew that the desiccant thickness held the saver valve closed over the vacuum port. The desiccants were subsequently red-tagged and replaced with spare desiccants. No further anomalies were reported throughout subsequent manned phases.

(4) Recommendations.

(a) Fecal processing should be done without heat. The increase in drying time does not warrant the complexity of an electrical heater from the mechanical design point of view.

(b) Fecal bags should be designed with a single rubber cuff since smearing of the seal did not occur. This would eliminate the difficulty of installing the bag in its receptacle.

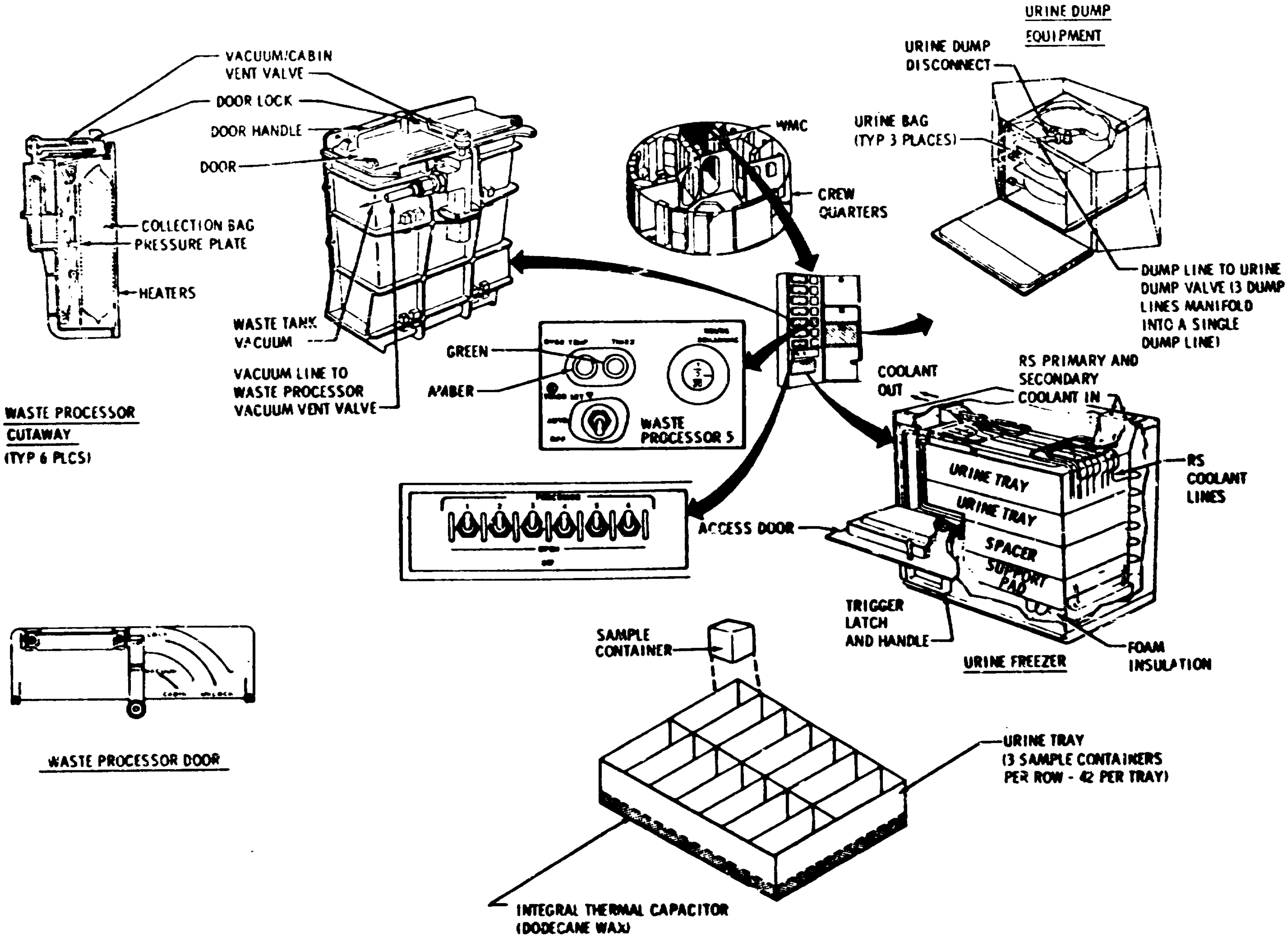


Figure VG9-1. Waste Processor Module

(c) Airflow through the fecal seat should be increased by approximately 50 percent.

b. Urine collection and sampling.

(1) General requirements.

(a) The urine collector shall provide the capability to collect, contain determine volume, sample, and dispose of excess urine. Requirements for the urine collector are:

1 The urine receiver shall completely enclose the urine stream during the collection process. The tailoff or dripping portion of the normal urination process shall be accommodated by the urine collector.

2 The urine receiver and the urine collection unit shall be operable while the astronaut is restrained in the seated or standing position.

3 The urine collection unit shall be designed to collect and contain the crew urine output for a 24-hr period.

4 The urine collector shall provide the capability to extract representative samples of 122 ml (minimum), from a homogenous pool, for freezing. The samples shall be frozen below -2.5 °F within 8 hr.

5 Urine remaining after sample extraction shall be disposed of into the waste tank. The system shall provide the capability to dispose of the urine at scheduled intervals.

6 The urine collection unit shall determine the volume of each 24 hr void to an accuracy of $\pm$ 15 percent.

7 The urine collector shall be designed to prevent cross-contamination between the users. A flushing capability shall be provided as a means of controlling cross-contamination between the 24-hr pooled urine collections for each user.

8 The 24-hr urine pool shall be maintained at or below 59 °F.

9 The maximum time for each urination cycle (excluding urination) shall not exceed 1- min.

10 The urine collection system shall interface with the command module for transferring, collecting, measuring and sampling, and dumping of the urine collected during command module operations prior to workshop activation.

11 A lithium chloride tracer shall be incorporated as the prime method to determine the volume of urine collected in each 24-hr urine pooling period. Lithium shall be added in the amount of 30 ±0.3 mg into each pooling bag prior to flight.

12 An alternate urine collection system shall utilize Apollo-type roll-on cuffs and adapters to accommodate urine collection directly into urine collection bags without the use of air entrainment.

13 The qualification and performance requirements for the urine separator are:

a Operating temperature--58 to -90 °F.

b Operation pressure--3 to 7 psia (external: 14.7 psia higher than internal pressure).

c Service life--9,000 hr (1,750 operational cycles/8-min cycle)

(2) Mission performance.

(a) Airflow and urine collection time was considered satisfactory by all crewmen.

(b) Noise level of the urine separators (Figure VG9-2) was not disturbing. However, when the system was used during sleep periods, the crewmen were occasionally awakened by the separator noise. This was not considered a problem. It is recommended, however, that future space vehicle design dampen or isolate such noises during crew sleep and/or nonactive periods.

The one occurrence during the first manned phase of separator filter changeout caused by the pilot's filter clogging, required about 30 min of maintenance time.

All nine separators used during the Skylab mission performed without incident. During the last week of the third manned phase, it was reported by the crew that urine salts were deposited on the case of all three separators. However, the separators continued to function and were used through the end of the mission.

(c) Urine drawer chillers (Figure VG9-5) operated normally throughout the manned phases. Examination of ADDT data and real time data for the duration of the Skylab mission also indicates an average temperature for the three chillers of 45 °F. The crews reported no excessive buildup of moisture on the heat exchanger plate. They did, however, wipe the plates daily.

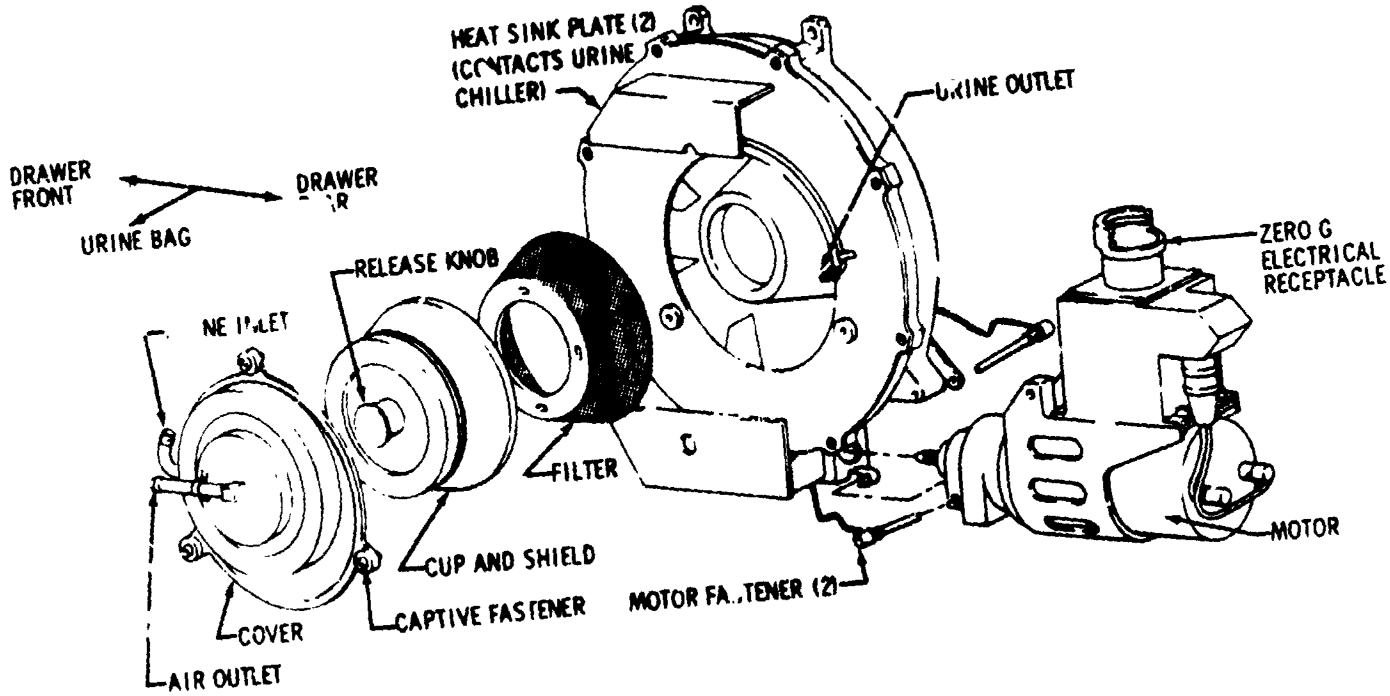


Figure VG9-2. Separator

No difficulty was experienced installing new urine bags in the urine bag box and on the separator. During the first manned phase, the pilot's urine drawer was reported to have a tendency to stick while closing; it was difficult to close the last inch of travel. Similar problems were encountered during ground checkout becase of close tolerance. The pilot of the first crew reported that he was reluctant to slam the drawer and, thereafter, he applied a force slowly that closed the door adequately.

The condensation in the urine drawers was minimal and confined to the chiller plate (primarily on the bumpers). This condensation was wiped daily.

Occasionally, the urine hoses were caught behind the separator motor but this was corrected by ensuring that the urine hose was not in a position to become pinched prior to closing the drawer.

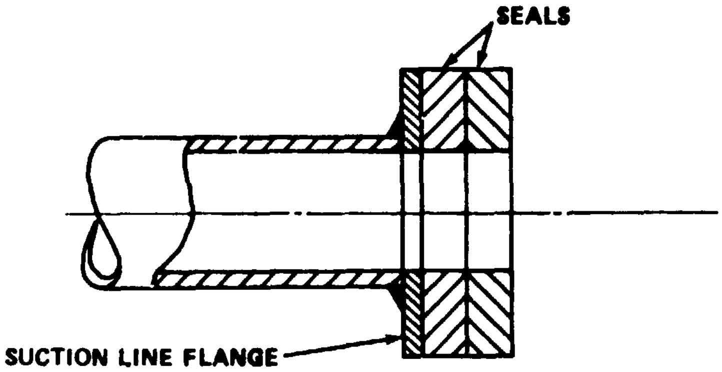
(d) The accuracy of in-flight volume measuring as compared to post-flight lithium chloride analysis varied randomly throughout the Skylab mission. However, it was clear that the in-flight measuring system was most accurate at the higher collected volumes. There appeared to be a gradual learning curve which improved the in-flight measuring slightly during the latter part of the missions. The sample bags arrived frozen as planned and none of them leaked after thawing.

(3) Anomalies.

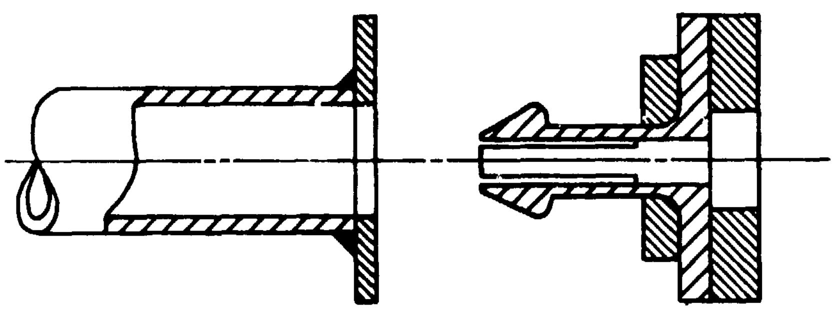
(a) During deactivation of the urine drawers by the second crew, the science pilot found that the suction line seal had debonded from the flange of the suction line. The seal was taped to the collector face. Examination of a photograph (taken by the crew) and of the qualification test collection module indicated that the failure may have been due to an improper bond. A replacement plug seal was fabricated for the third crew to install that solved the problem (Figure VG9-3).

(b) During the first manned phase, low airflow was reported in urine drawer #3. The crew changed the separator filter and the airflow became normal. Inspection of the filter by the crew did not reveal any visual blockage or wetting of the seal. Since the filter was not returned, the cause of failure is not known. A single large collection may have flooded the filter surface, blocking the flow and the continued operation of the blower subsequently dried the filter.

(c) Excessive air was reported in the urine samples taken by the first crew, resulting in low sample volume (90 to 100 ml average rather than 122 to 130 ml). Cycling the sample did not remove the air.



URINE COLLECTION DRAWER #2 SUCTION LINE FLANGE LAUNCH CONDITION: TWO INDIVIDUAL SEALS WERE BONDED TOGETHER AND TO THE SUCTION LINE FLANGE. THE SEALS BECAME DEBONDED AT BOTH INTERFACES DURING THE SECOND MANNED PHASE.



REPAIR TECHNIQUE FOR DEBONDED SEALS: A STAINLESS STEEL COLLET INSERTS INTO THE SUCTION LINE WITH PROVISIONS FOR SEALING AT THE SUCTION LINE FLANGE AND VALVE BODY INTERFACE.

Figure VG9-3. Urine Collection Drawer Suction Line Flange Launch Condition, Failure, and Repair

However, keeping the pressure on the urine bag as the sample was taken worked the best and was adopted by the second and third crews. Samples of 122 ml were easily obtained in 1 g testing which could be attributed to better separation. It is more likely, however, that the air in the collection bag was more evenly distribured in zero g and resulted in excessive air in the urine samples.

(d) During a trial run of stowing the urine trays in the return container, the third crew reported difficulty inserting the trays. Of the four positions, the outer two trays could not be fully inserted without encountering resistance. The science pilot elected not to force the trays further. He felt that the trays would be impossible to remove and return to the urine freezer. The cause of the problem was determined to be sample bags frozen above the top of the tray. Removal of all the cardboard spacers from the return container did not provide sufficient space for the trays. In further conversations with the crew, the science pilot stated that he thought he could force the trays into the container adequately. This proved to be true, for during deactivation no additional problems were reported.

(e) Late in the third manned phase, the crew reported a urine (ammonia) odor coming from the collection module. The odor was apparent when the blower was operating and indicated a failure of the odor control filter.

The odor control filter was designed for 28 days of operation, and housekeeping procedures required filter replacement halfway through the originally planned 56-day manned phase. When the third manned phase was extended to 85 days, no provisions were made for additional filter replacement. Although the filter was designed for 28 days, it was tested for 56 days. The qualification test filter failed on the 54th day. During the third manned phase, the second filter had been installed on about the 28th day. Therefore, on the day the crew reported the odor (day 79) this filter had been operating approximately 51 days and failed approximately when expected.

(4) Recommendations.

(a) Some means of monitoring critical systems airflow (delta pressure gages) should be provided to the crew to indicate system performance.

(b) Planned use of waste management systems during sleep periods would neccessitate special low noise design.

(c) A better method of more effectively counteracting the zero gravity effect, i.e., centrifuging the sample under pressure, could be developed to remove a larger portion of the gas.

c. Vacuum cleaner.

(1) General requirements--The general requirements for the vacuum cleaner are:

(a) It shall be capable of collecting debris (including free water) and particulate matter from the atmosphere of all accessible areas of the workshop.

(b) It shall be electrically powered, utilizing a universal electrical cable and preinstalled electrical junction boxes.

(c) It shall incorporate provisions to permit one-handed carrying of the unit and attachments and shall be capable of being attached to the grid floor.

(d) It shall have nonpropulsive exhaust vents and shall have a three-position switch for "On", "Off", and "Momentary".

(e) The qualification and performance requirements for the portable vacuum cleaner

1 Operating temperature--58 to 90 °F.

2 Touch temperature--55 to 150 °F.

3 Operating pressure--3 to 7 psia.

4 Voltage--24 to 30 Vdc.

5 Service life--9,000 hr (1,960 cycles).

The power module requirements are the same as those for the vacuum cleaner.

(2) Mission performance

(a) All crews reported that the vacuum cleaner (Figure VG9-4) worked satisfactorily. It was never used to collect wet debris but was used primarily to clean the debris screens on the mixing chamber and waste management compartment exhaust fan.

(b) No quantitative evaluation of the power module or vacuum cleaner can be made. However, all four power modules performed without a malfunction. The suit drying module probably was functioned more than the twenty 10-hr cycles that it was tested for, because of the extended mission.

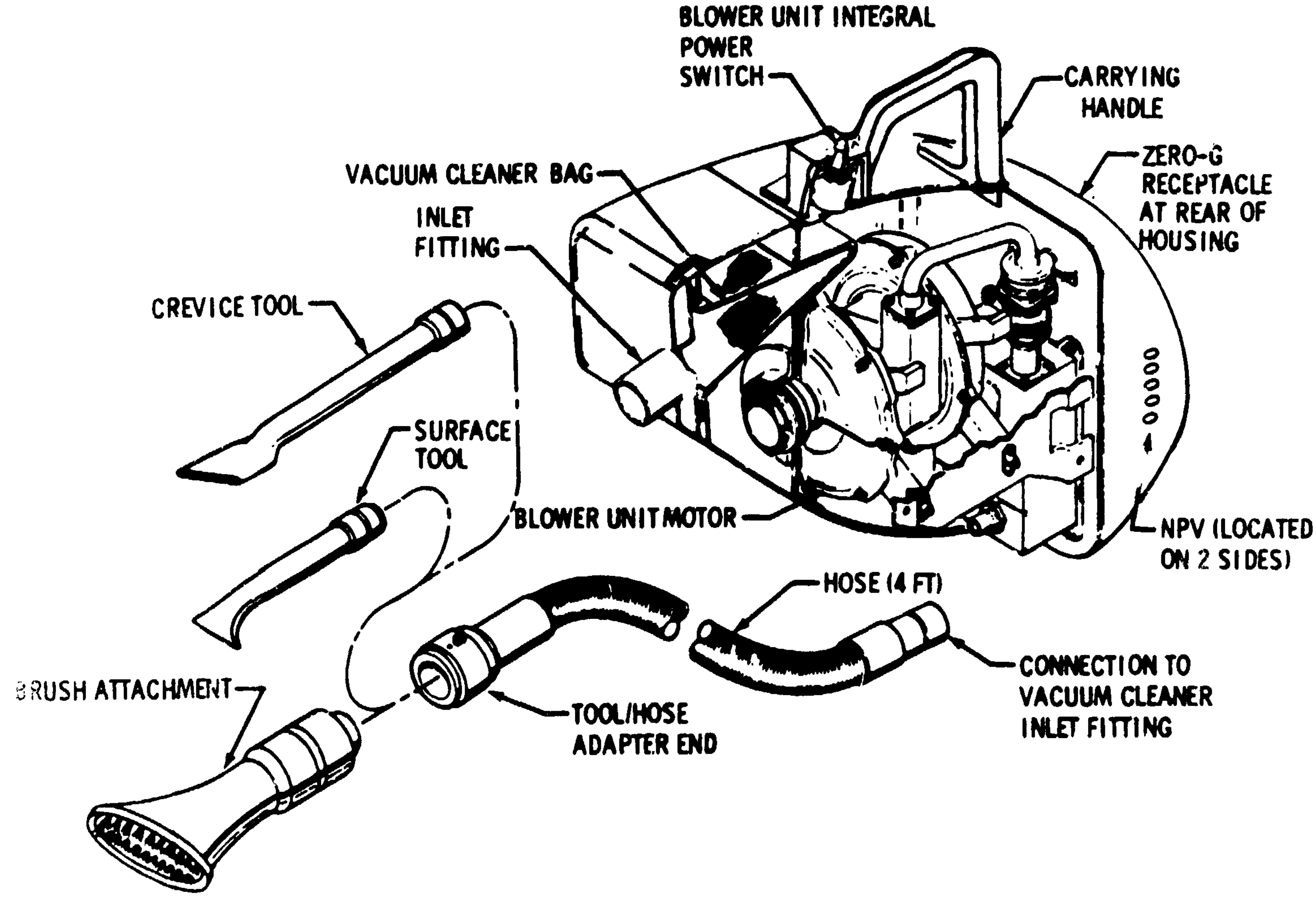


Figure VG9-4. Vacuum Cleaner

(3) Anomalies. The vacuum cleaner operated as designed and no anomalies or malfunctions were reported.

(4) Recommendations. Within the elctrical power limitations of the Skylab, the vacuum cleaner (110 W) performed extremely well, however, the crews strongly recommended that future vacuum cleaners be more powerful (higher nozzle delta P and higher airflow).

d. Component evaluation.

(1) Collection module (Figure VC9-5). The qualification and performance requirements for the collection module are:

(a) Operating temperature-- 85 to 90 °F.

(b) Operating pressure-- 3 to 7 psia.

(c) Thermal/vacuum

Launch: 30 to 140 °F, 26 to 0.5 psia (0.1 psi/sec blowdown rate).
Orbit: 58 to 90 °F., 0.5 to 1.3 psia.

(d) Service life--9,000 hr

(e) Cycles--700 cycles at 8-min/cycle per urine separator system, 140 cycles at 20 min/cycle/fecal/urine system.

The collection equipment worked successfully and the crews expressed general satisfaction with the collection module. Minor difficulties were encountered installing the bag in the fecal receptacle; the second cuff was occassionally difficult to install on the receptacle. Bag sealing was accomplished by the method which makes a 1-in. fold instead of 1/2-in. folds. Bag sealing was always done with the blower on, and although there were no seals that leaked, the crews commented several times on the "unforgiving" sticky adhesive on the bag. There were no problems at the bag/processor interface.

(2) Odor control filter (Figure VC9-6). The qualification and performance requirements for the odor control filter are:

(a) Temperature-- 58 to 90 °F.

(b) Pressure-- internal, 3 to 7 psia

(c) Service life--28 days effectiveness (each crew to install new unit at activation).

The odor control filter was effective for 51 to 54 days, as compared to a design requirement of 28 days. During the third manned phase, one unit remained in use for 51 days until an ammonia odor was noticed by the crew. The unit was then replaced with an onboard spare.

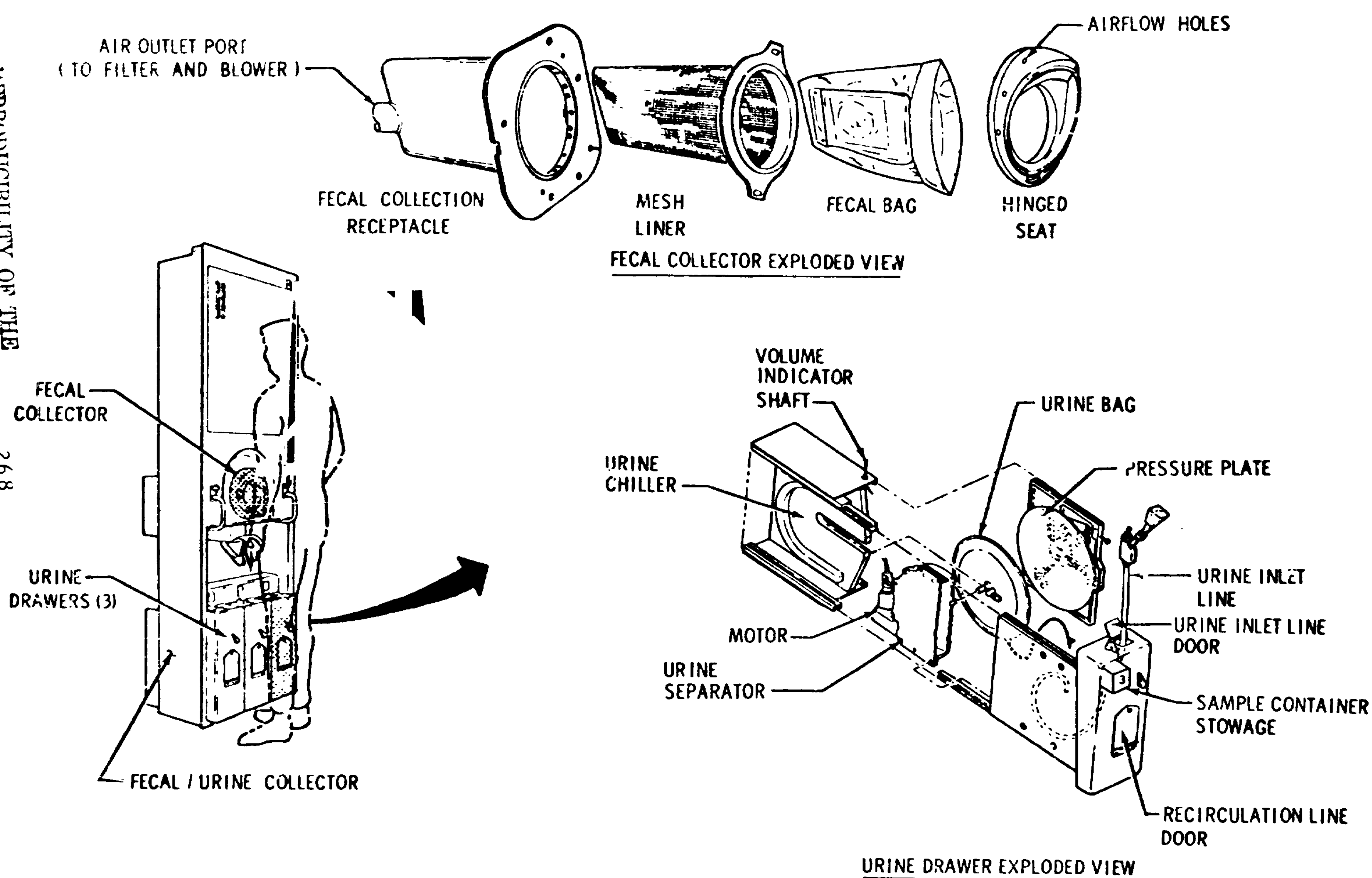


Figure VG9-5. Collection Module

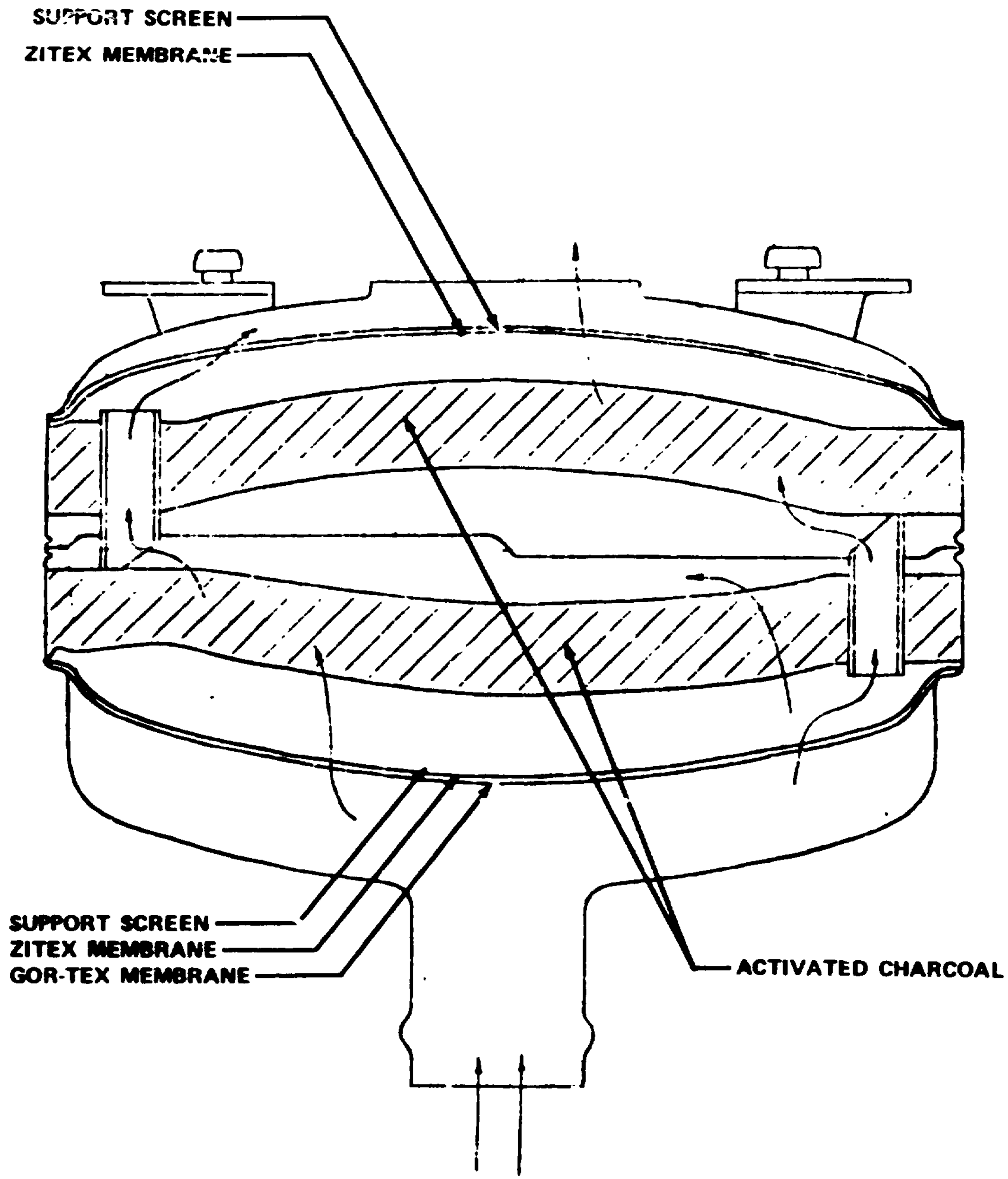


Figure VG9-6. Odor Control Filter

10. Water System.

a. General requirements for the water system are:

(1) 6,000 lb capacity.

(2) Compatibility with iodine with capability to monitor iodine concentration levels and ensure complete dispersal after iodine injection.

(3) Positive protection from freezing.

(4) Hot and cold water to prepare meals for three crewmen simultaneously.

(5) Visual indication of the amount of water dispensed.

(6) Warm water for personal hygiene activities and body cleansing.

(7) A urine flush system to dispense water for daily flushing of the individual urine collection modules.

b. Mission performance. The water system (Figure VG10-1) performed successfully during all phases of the Skylab mission. This is based primarily on crew comments since the only specific water system data available were the nitrogen supply pressure and the water volume for each of the 10 tanks. Bus current and voltage provided a limited evaluation of water heater operation.

(1) Water storage and pressurization equipment. The 10 water tanks loaded with the required 6,000 lb of water survived the launch environment with no apparent problems (Figure VG10-2). Integrity of water tanks was a consideration during the first unmanned period because of high temperatures in the vehicle. Elevated temperatures and related water expansion could have damaged the gas/water dome or bellows, but damage did not occur since the tanks were qualified at 55 to 275 °F. operating temperature. (The elevated temperature of the tank was only 130 °F.) The tanks were qualified at an operating pressure of 40 psia (max.). The bellows were also qualified at the above temperatures with an operating pressure of 37 psig. The service life requirement of the bellows (25 operational cycles; 100 min., 200 max. test cycles) was adequate. No problems were reported during activation or deactivation of the water tanks throughout the three manned phases.

The water tank blanket heaters were not required since the temperature level in the water containers did not go low enough to actuate the system.

Nitrogen regulated to 150 psig is supplied from the airlock and regulated in the workshop to 35 psig as part of the N_2 distribution network. The required operating range during activation was 38 to 44 psia. During the second unmanned period, telemetry showed the 35 psig

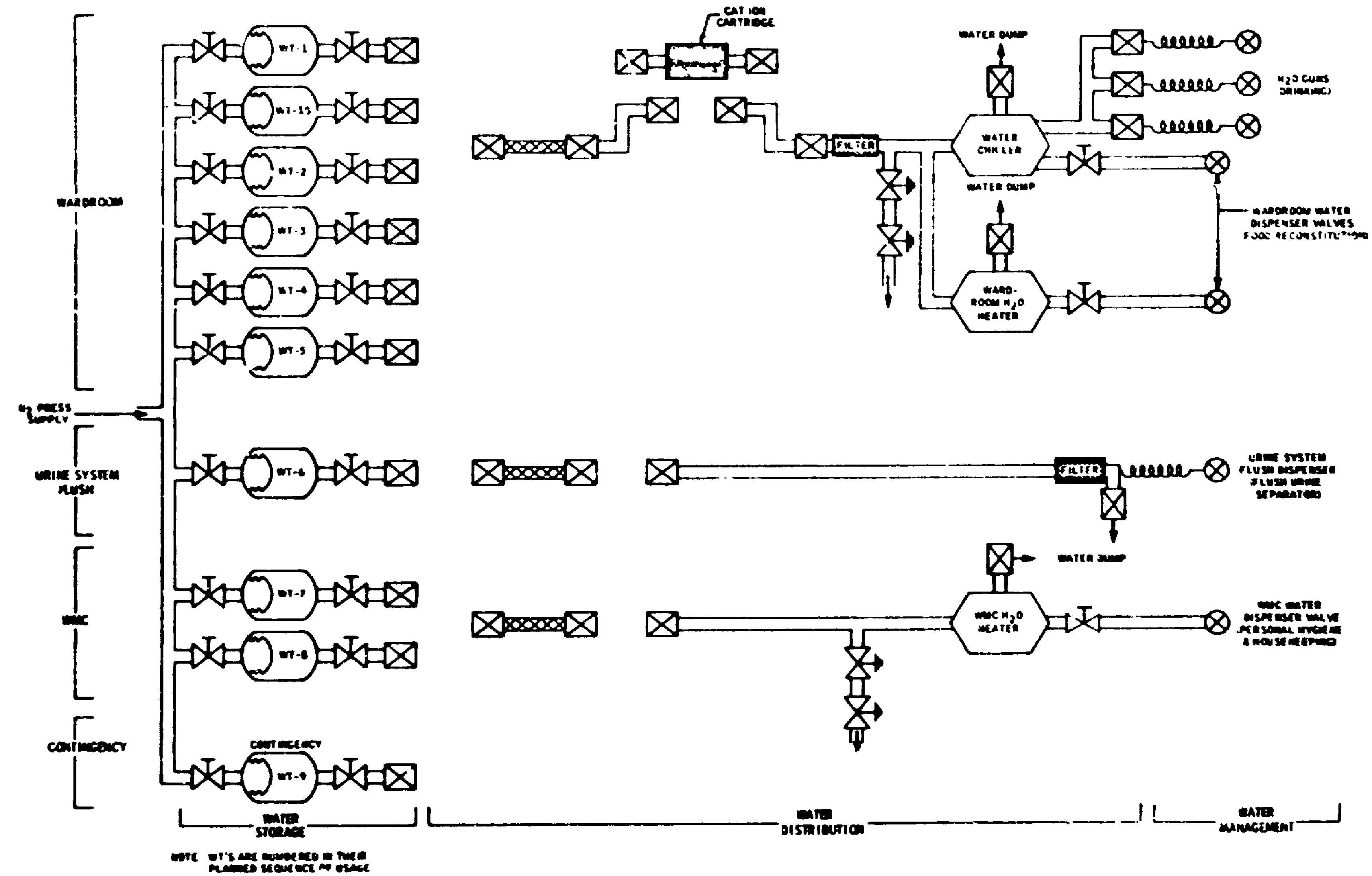


Figure VG10-1. Water System Schematic

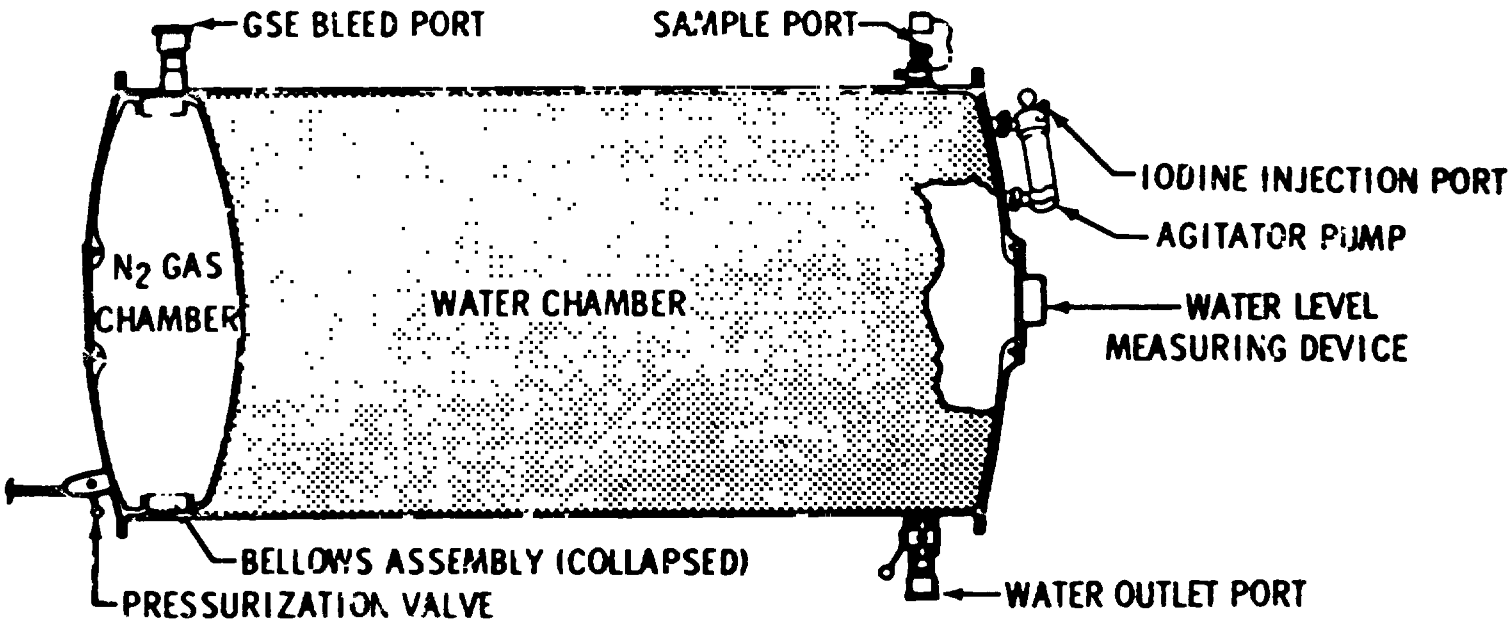


Figure VG10-2. Water Container Assembly

water tank gas pressure had decreased to 34 psig, but this was considered in the range of normal system leakage. During manned phases, the water pressurization system held steady at approximately 40 psia.

(2) Wardroom water equipment. Temperature measurements were not provided, but crew comments indicate temperatures were acceptable. Also, measurements were not available to evaluate the volume accuracy of each unit of water rationed by the drinking water dispenser. The three drink dispensers alloted 9,000+, 15,000+, and 12,000+ units during the Skylab mission. Each unit was required to be 0.5-oz, which was issued by one trigger actuation. The service life qualification and performance requirements were: (a) 13,000 cycles (production, test, checkout, and mission requirements) and 52,000 cycles (tests - three tests): 1 26,000 cycles 2 2,016 cycles (with biocide) followed by 3 23,984 cycles (rapid succession); (b) operating temperature -40 ±5 °F; and (c) operating pressure--40 psia (max.). The crewmen did not comment on operation during debriefings. It is assumed that both operation and dispenser tip changeout was normal. The spare drink dispenser was not used since there were no problems with the prime units.

The overall operation of the wardroom water dispenser was excellent. No operational problems were noted by the crew during any of the three manned phases. The qualification and performance requirements for the wardroom water dispenser were: (a) Operating temperature--35 to 155 °F; (b) operating pressure--40 psia (max.) and proof test--82 psig (max.); and (c) service life--5000 cycles (mission requirements) tested 11,000 cycles.

Based on the analysis of the water samples returned by the third crew, the deionization filter (Figure VG10-3) maintained the ion levels of the metals below specified requirements. Contrary to qualification test data, iron and chromium levels in tank #1 and tank #5 were also still within specification. Nickel was out of specified requirements but well within specification after passing through the filter. The filter qualification/performance requirements of: (a) operating temperature--55 to 90 °F; (b) operating pressure--40 psig (max.); and (c) service life--connect/disconnect, 40 cycles (design) 500 cycles (test) were adequate.

(3) Waste management compartment water equipment. Equipment performance was adequate. (See paragraph c. "Anomalies"). The water temperature was considered acceptable by crewmen, but measurements are not available.

(4) Urine flush equipment. The system was not activated or used. Microbial testing of the separators during system testing on the ground negated the need for daily flushing.

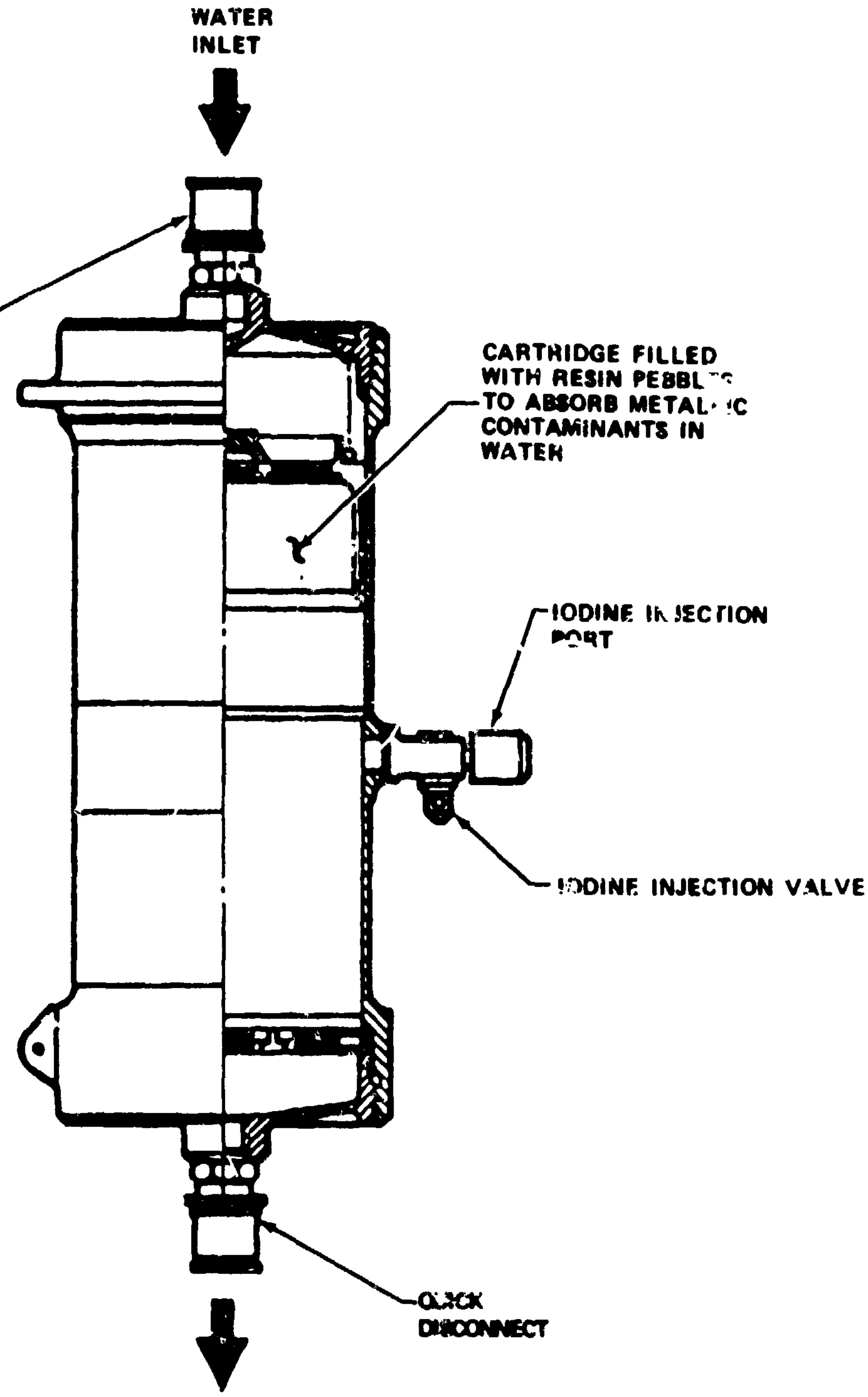


Figure VG10-3. Deionization Filter

(5) Water purification equipment. This equipment (Figure VG10-4) was provided to sample the iodine level in the water tanks (it was used 28 times) and to add iodine as required to maintain the level between 1 and 6 ppm (iodine was added 16 times). It was also used to inject iodine into the cation filter in preparation for each storage period. During debriefings, the crewmen commented that the system was straightforward and easy to use. The equipment was qualified to perform at 55 to 105 °F at 40 psia (max.).

(6) Portable water tank. The portable tank (Figure VG10-5) was used for sterilizing the water distribution system with 100 ppm iodine solution prior to the start of each manned phase, and to check the N_2 system pressure. Also, it was intended as a contingency water supply but was not required. The utility and flexibility was exploited for other functions: (a) ground spare was reworked and charged with coolanol for resupply of airlock coolant if rescue mission was required; (b) onboard unit N_2 side was used to purge S201 experiment 29 times. The following qualification/performance requirements were adequate:

1 Operating temperature--55 to 150 °F.

2 Operating pressure--40 psia (max.), 500 press cycles to 200 psig.

3 Service life--(production and mission requirements) 167 cycles (tested) 217 cycles.

c. Anomalies. During the first manned phase activation period on mission day 3, the pilot reported difficulty in connecting the wardroom supply hose on water tank #1. This was attributed to the elevated temperature of the tank (130 °F) and to thermal expansion of a small amount of water between the quick disconnect and the tank shutoff valve.

The following day, the commander reported the water system had gas in it. He stated that in filling a 7 1/2-oz coffee container, the container would not handle the water and the air, and to operate properly would require air removal. (It was believed that the system was initially free of air.) The problem was simulated in a 1 g environment using flight-type hardware. It was established that air could enter the containers from the cabin. Subsequent operatings had lesser air entrapment problems, therefore, no further action was taken.

During the first few days of the third manned phase, while using tank #2, the crew reported gas in the water, but after changing to tank #3, the gas problem disappeared in 2 to 3 days and no other difficulty was experienced. There was a possibility that the bellows developed a small leak, but this was unlikely, and no troubleshooting was done.

C-4

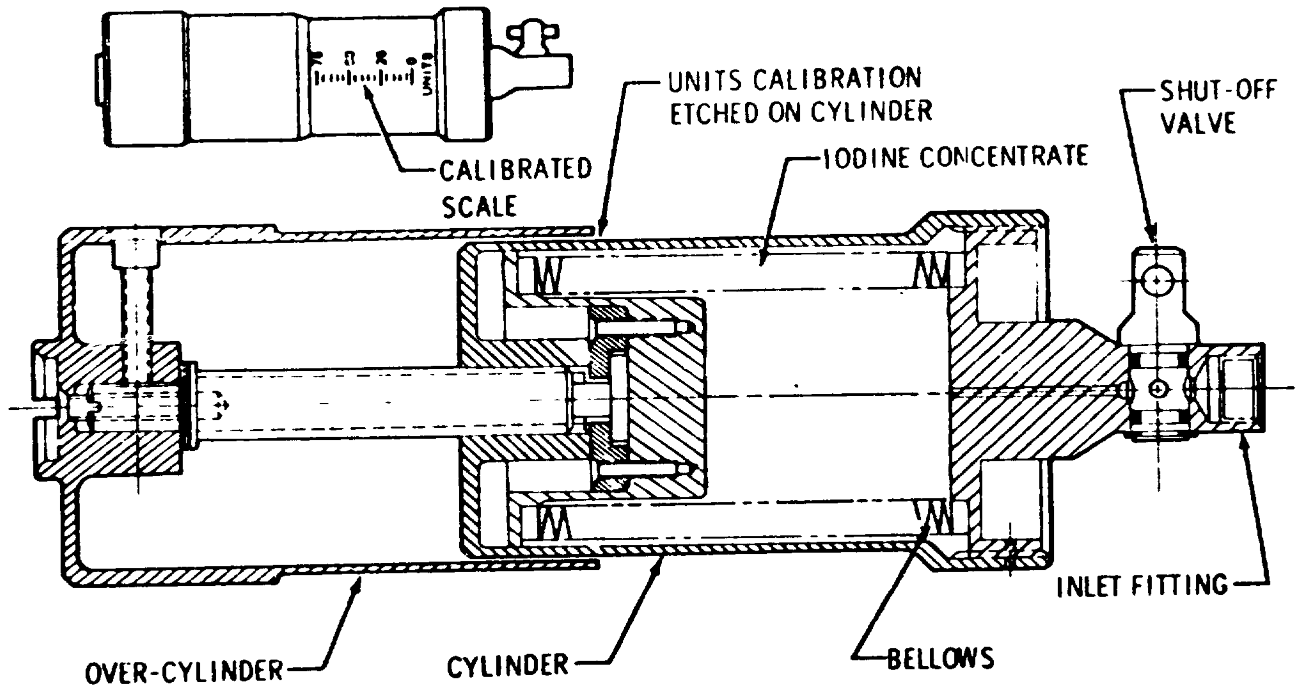


Figure VG10-4. Purification Equipment (Iodine Injector)

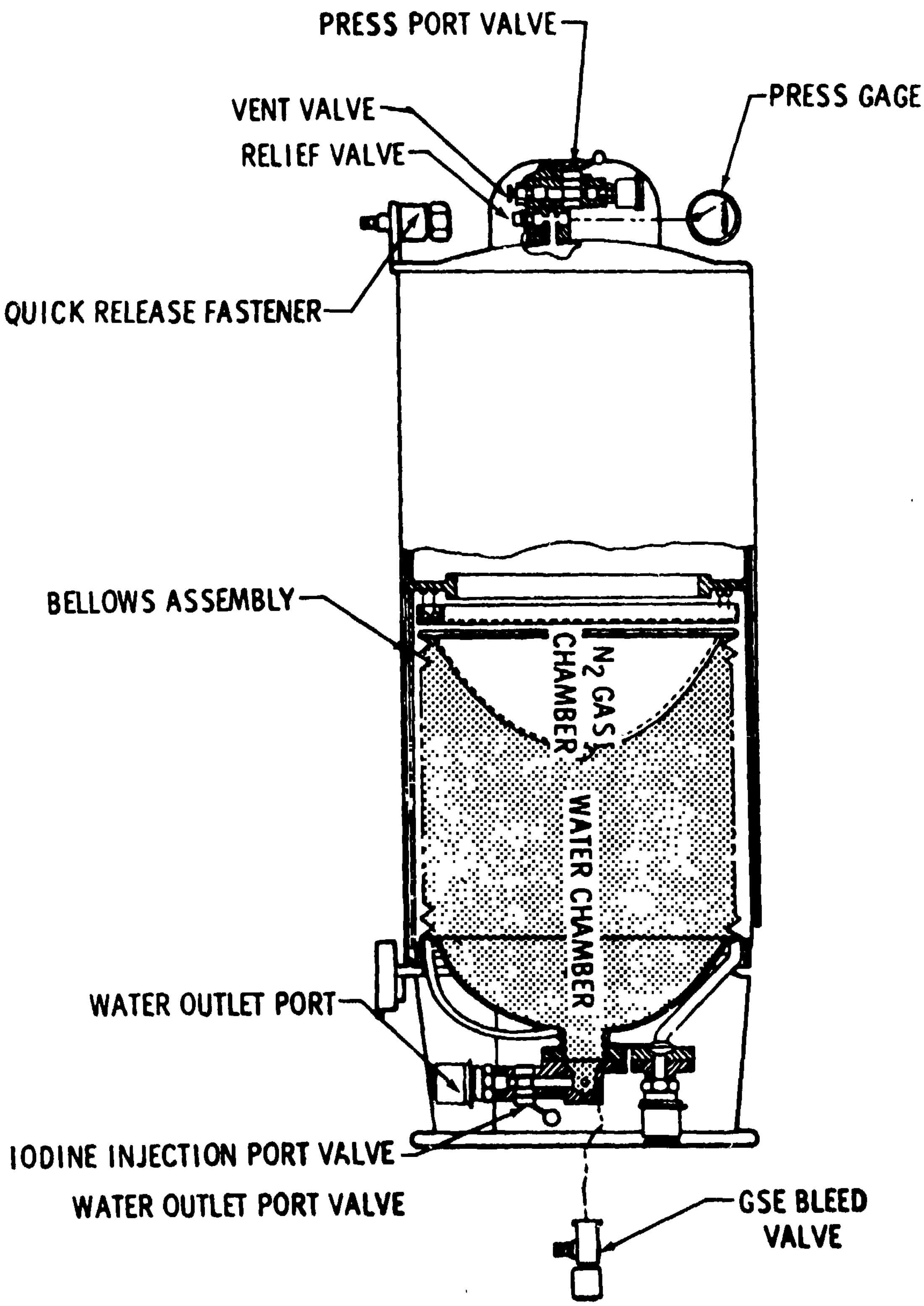


Figure VG10-5. Portable Water Tank (PWT)

Bus current data on day 56 of the third manned phase showed an abrupt increase in "ON" time of the wardroom water heater. The heater resistance was estimated and showed an abrupt increase. Behavior of the heater during qualification tests indicated that the silicon covered Inconel heater element could deteriorate in the iodized water and not meet the qualification requirements of: operating temperature for the wardroom heater - 150 +5°F for 3,360 hours and for the WMC heater - 127 ± 5°F for 3,360 hr . Service life: design requirement - 9,000 hr and mission requirement - 3,360 hr . Since the tests indicated that the silicon covered heater element could deteriorate, it was concluded that one of the two heater elements had failed and as a result the water temperature remained the same but recovery time was longer. A similar increase in "ON" time of the waste management compartment heater was noted 18 days later. Figure VG10-6 compares the heater resistance trend during the Skylab mission to qualification test data. Figure VG10-7 graphically depicts the heaters' configuration. Spare water heaters were onboard the workshop to allow replacement if complete failure occurred; however, replacement was not required because of the longer-than-expected heater life, which can be attributed to lower iodine concentrations and zero g effects. These spares were flown since delivery of redesigned elements with ceramic wire would not meet the workshop schedule.

During activation (mission day 3 of the first manned phase), a check of the waste management system line pressure was requested by the commander. Evacuation of the line to a specified 0.2 psia minimizes gas entrapment when the system is filled with water, but the minimum pressure obtainable during a previous condensate tank dump was 0.77 psia. This higher pressure allowed some gas entrapment, but was determined to be acceptable since it would be purged out during normal use.

Twenty days later, a decrease in flow of the waste management compartment water dispenser was reported. The crew replaced the assembly with the spare unit and reported flow to be normal. The unit was returned and failure analysis disclosed that the seal was smaller than required. Further investigation disclosed that neoprene rather than Viton was used. When the dispenser was disassembled, a white powdery residue was caked on the inlet snap ring and a white, flaky residue in all outlets. It appeared to be a soap residue, but analysis showed it to be a product caused by the attack of iodine on the beryllium copper retaining ring. New seals of the proper material were supplied and a reworked spare dispenser was launched with the second crew.

During the second manned phase activation, the dump line pressure transducer was found to be inoperative (off-scale high). Nothing conclusive as a failure is known since no troubleshooting was done. A work around procedure (timing of dump) was developed and activation continued.

On mission day 56 of the second manned phase, the washcloth squeezer piston seal (Balseal) (Figure VG10-8) was replaced because of the leakage that randomly began on mission day 15. Examination by the commander

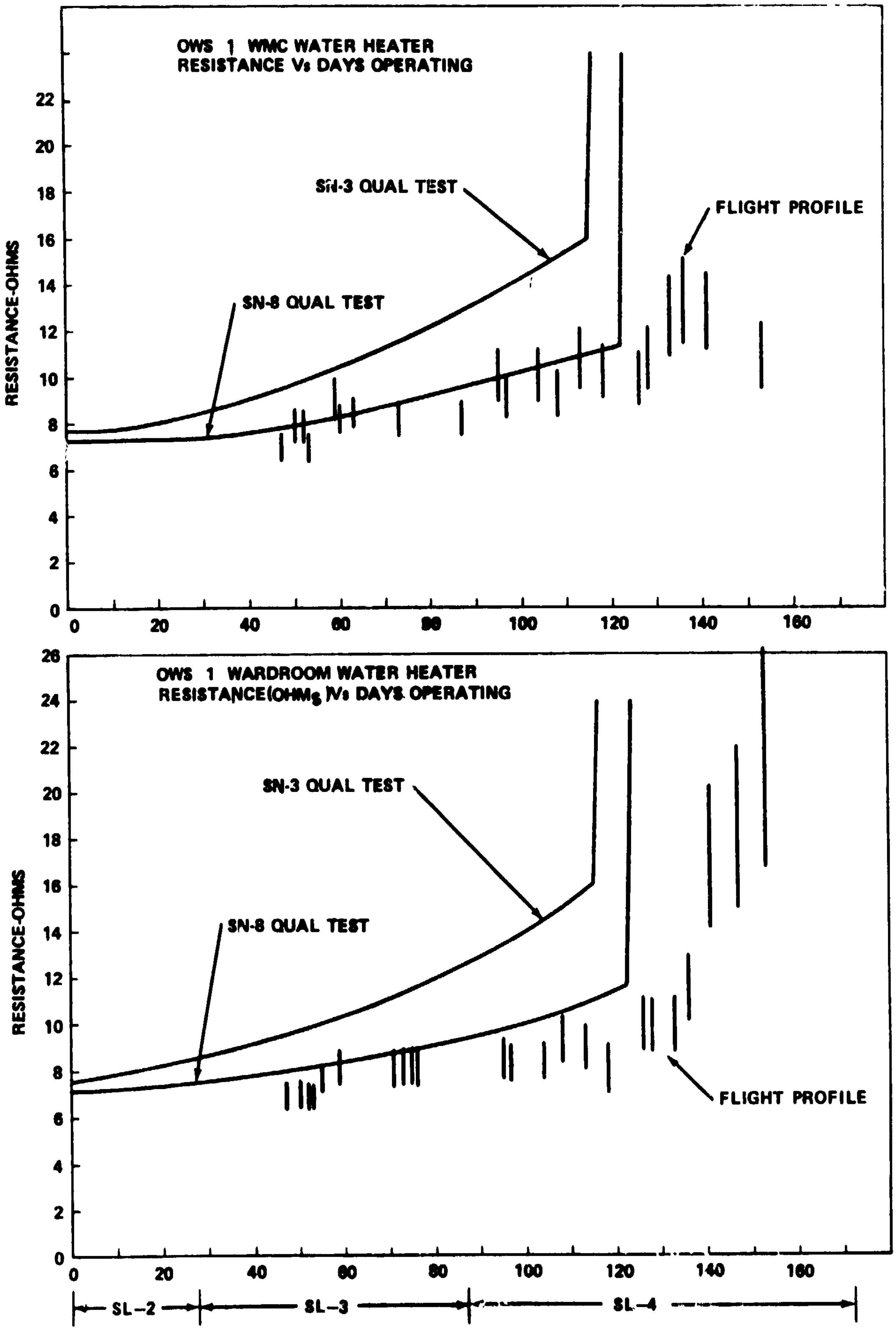


FIGURE VG10-6. Heater Resistance Trend (Qual Test verses Flight)

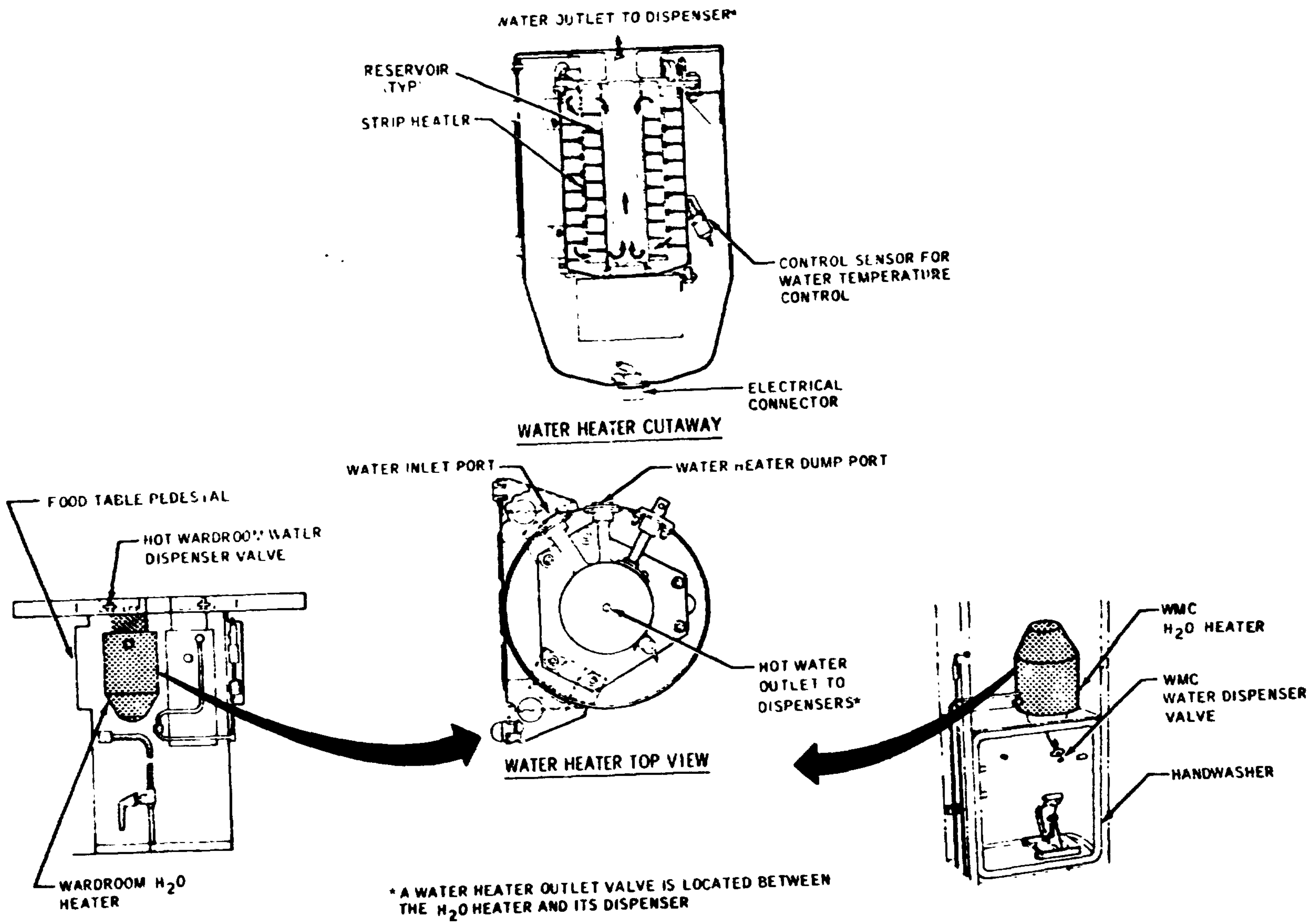
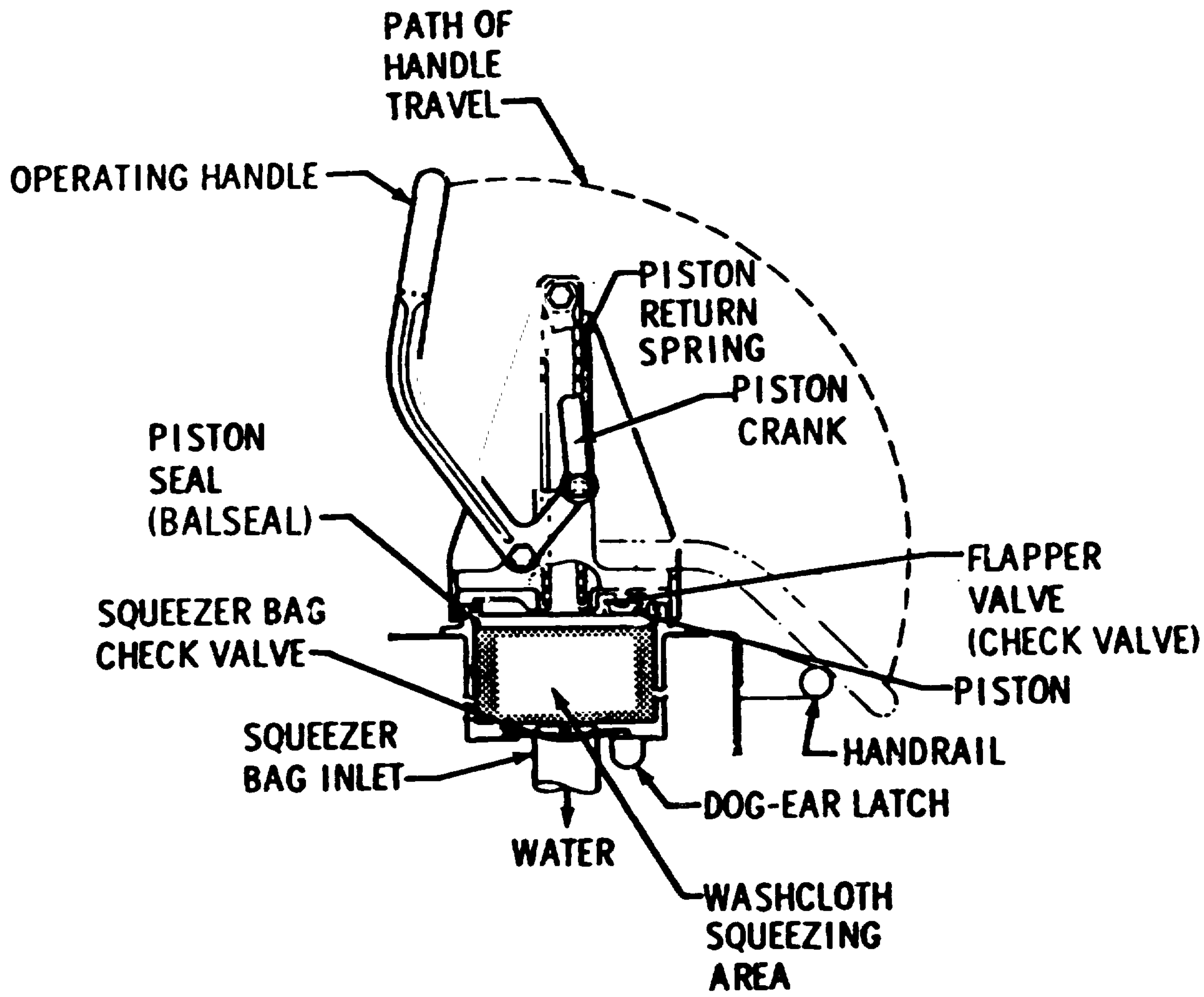


Figure VG10-7. Wardroom and WMC Water Heater



SQUEEZER SIDE VIEW

Figure VG10-8. Washcloth Squeezer

showed the seal to be folded back in at least one area, allowing water to leak past the piston. As a result, three additional spare Balseals were launched with the third crew who reserviced the washcloth squeezer and no further problems or anomalies were reported. This early maintenance was brought about by either or both: (1) inferior quality of initial lubrication; or (2) accumulation of grime during extended use. The crew did report increasingly higher operating forces. Consequently, a maintenance procedure was developed by MSFC to clean, lubricate, and adjust the bearing screws; once completed, operation returned to normal. The normal operating requirements were: lever hand load--2.6 to 30 lb for 1,176 squeeze cycles, mechanism--5,980 cycles (design) and 13,000 cycles (test), and a service life of 140 days.

d. Recommendations. The washcloth squeezer was intended to be a low cost improvement to the washcloth bathing scheme. The hardware performance and overall scheme were reasonably successful, but it is felt that this is an area for improvement for future designs. Future wet cloth squeezers should be designed with protected main seals and with mechanisms less susceptible to grime and contamination.

The problems related to the failure of the onboard water heater element were expected since they occurred during long duration qualification tests. Spare water heaters were onboard to allow replacement if required. A heater element encased in metal (Cal-rod type) was designed and tested for the backup workshop. The new configuration should be considered for any future project.

Material compatibility is a very important aspect in systems such as the waste management compartment water system and should be given special consideration.

11. Refrigeration System (Figure VG11-1).

a. General requirements.

(1) The temperature of the urine pool shall not exceed 59 °F for more than an accumulated time of 3 hr during a 24-hr period.

(2) A freezer shall be provided to freeze urine and blood samples.

(3) A return container shall be provided for transferring the frozen urine and blood samples from the workshop to earth by way of the command module.

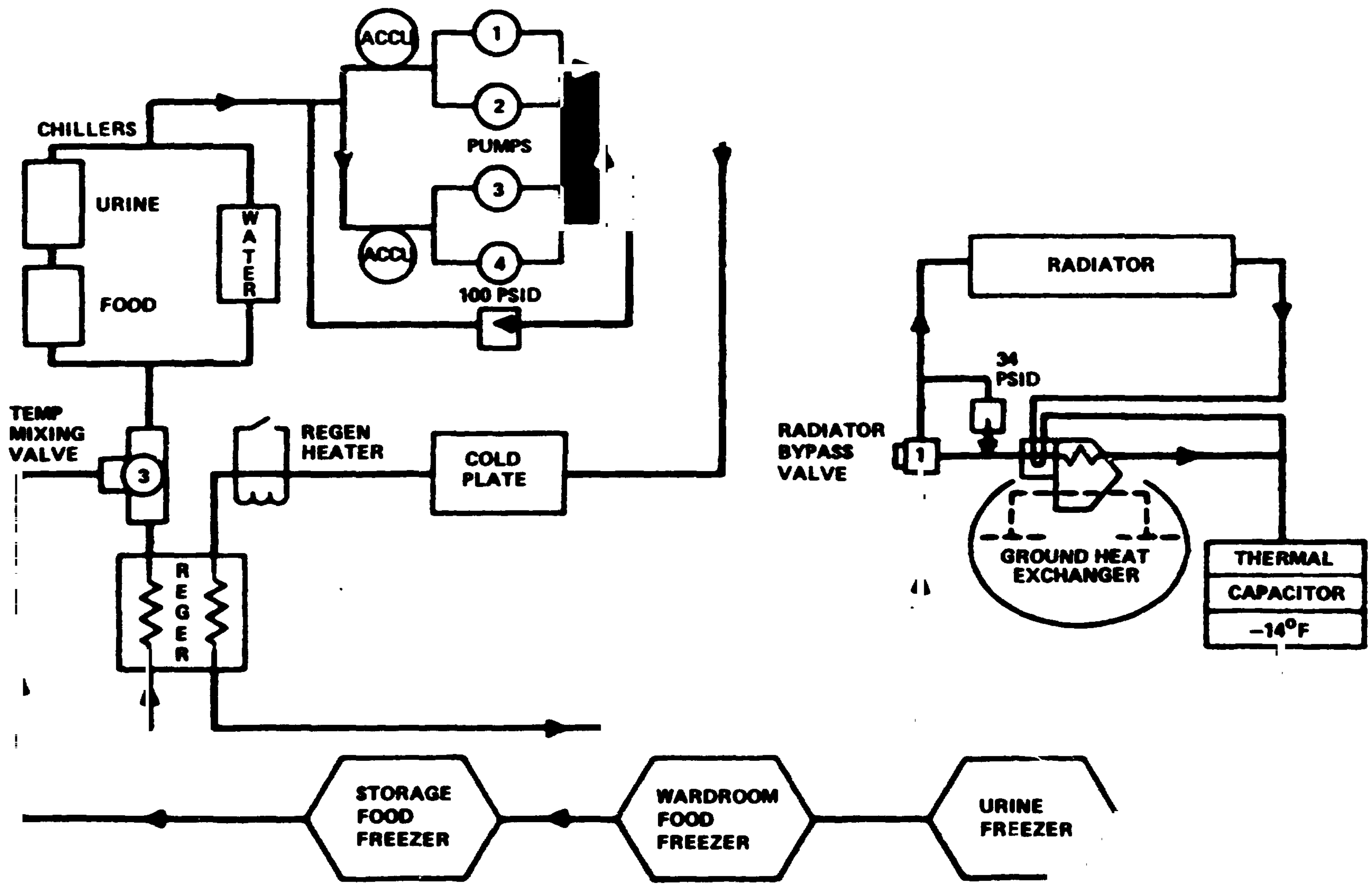


Figure VG11-1 Refrigeration System Schematic

(4) Positive protection against freezing during all mission phases shall be provided for the water chiller.

(5) Frost buildup in the food freezers shall not impair removal of the food packages during normal use.

(6) Dispensed water temperature shall be 50^{+0}_{-12} °F.

(7) The urine/blood return container shall be designed to maintain its content at 17 °F or lower for a maximum time span of 22 hr. The urine/blood container shall be designed to be interchangeable with the A-7 locker in the command module.

(8) Major system design parameters.

(a) Allowable leakage: 12 in^3/yr, per loop of Coolanol 15 (maximum).

(b) Radiator plume shield jettison velocity: 5 ft/sec

(c) Pump life: 2250 hr/pump.

(d) Radiator heat rejection capacity: 1680 Btu/hr (orbital average).

(e) Chiller valve control temperature: 39 ± 3 °F.

b. Mission Performance.

(1) Urine collector--Qualification test results indicated that as long as the urine chiller temperature, as measured by flight sensors, was less than 46 °F, the urine pool would remain less than 59 °F. The CEI requirements were less than 46 °F.

(2) Urine and blood freezer - The urine and blood samples freezing rates and temperature limits were satisfied by qualification tests. The measured freezer wall/sink temperature never exceeded -6.2 °F during Skylab missions.

Wardroom freezer #2 was used during the latter part of the second and third manned phase to store two trays of frozen urine and blood. At no time did the freezer wall temperature exceed -2.5 °F.

(3) Urine and blood return containers--Samples were returned in the container from all Skylab missions in a frozen state and with no known degradation.

(4) Water subsystem--Potable water was chilled during all manned missions and maintained above freezing by the chiller valve.

(5) Food stowage and use plan--Moderate frost buildup on the food freezers was observed but it was easily removed by the crew. No difficulty in food transfer or removal was reported.

(6) Water chiller--No direct evaluation of chilled water temperature can be made. However, crew comments were complimentary; and the chiller control valve, which supplied coolant to the water chiller, controlled the water temperature to its specified 39 $\pm$3 °F.

(7) Urine/blood return container--Mechanical interchangeability requirements with the command module were satisfactorily met. Transfer of the return container from the workshop to the command module was performed as scheduled at the end of each of the three manned phases, and contents were returned undamaged.

(8) The following relates system performance to design parameters in paragraph 11,a. (8) above.

(a) Leakage--There was no detectable Coolanol 15 leakage from the primary or secondary refrigeration loops.

(b) Radiator plume shield (Figure VG11-2)--The radiator plume shield jettison time could not be determined precisely from the available flight data, but there was verification that the instrument unit command for shield jettison was sent at the nominally prescribed time (00:09:57:04 GET). Refrigeration system temperature data indicated that the shield was jettisoned at, or near this nominal time. This conclusion was based on the following:

1 The refrigeration system bypass valve event data indicated a switching event from bypass to radiator surface temperature and within the predicted elapsed time for a nominal shield jettison.

2 The slope of the radiator surface temperature showed a continuous decreasing trend to its minimum, during the first revolution, following the HS-19 Test data.

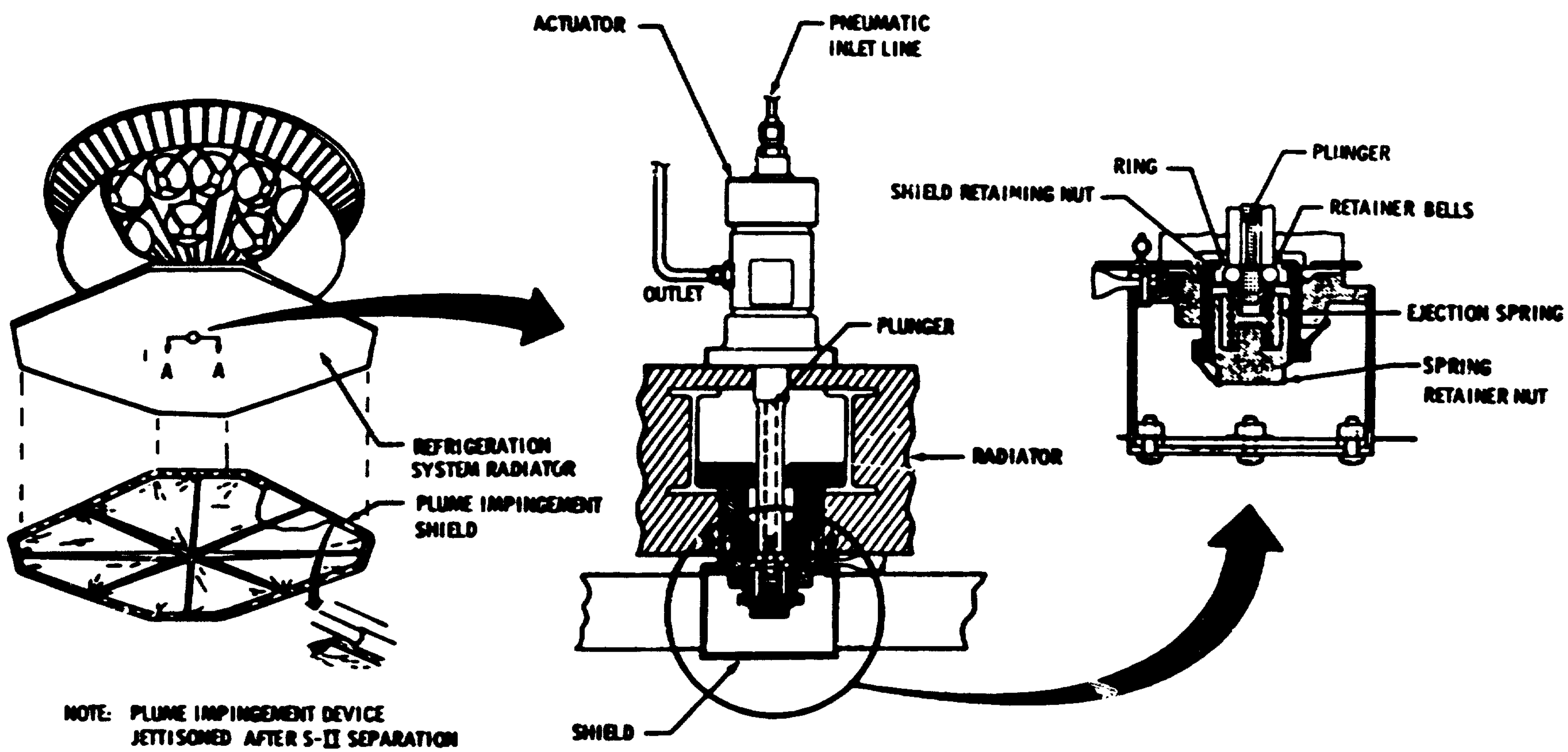


Figure VG11-2 Plume Shield and Release Mechanism

(c) Pump package (Figure VG11-3)--Primary loop pump #1 was used throughout the three manned phases except for a 300-hr period during the first manned phase. A total of 7,270 hr was accumulated on primary pump #1 before it was disabled during the third manned phase at 040:06:23 of mission day 74. Primary loop pump #1 far exceeded its qualified service life of 2250 hr in an environment of 1×10^{-6} torr at 80 °F.

(d) Chiller valve control (Figure VG11-4)--The primary loop chiller valve controlled within the specified 39 $\pm$3 °F range through the Skylab mission. The operating environment to which it was qualified was 1×10^{-6} torr with a -20 °F fluid medium. During the end-of-mission refrigeration system test, the secondary loop was activated and the secondary chiller valve also controlled within specification.

(e) Thermal capacitor (Figure VG11-5)--At launch the thermal capacitor was fully frozen at a temperature of approximately -26 °F. Following launch the refrigeration system heat load was absorbed by the capacitor until 0310 GET, at which time the radiator became fully effective and the capacitor once again was completely frozen.

The capacitor continued to function normally and held the outlet temperature to -14 °F, or less, until the occurrence of the first manned phase mission day 29 anomaly, during which it melted completely. By 174:22:30 GMT it began to recover, and by the seventh day after the end of the first manned phase, it was sufficiently frozen so that the outlet temperature again dropped to -14 °F, or less.

Throughout the remainder of the Skylab mission, the capacitor continued its normal function of absorbing all inlet temperature excursions while maintaining an outlet temperature less than -14 °F.

c. Anomalies.

(1) Mission day 29 anomaly (first manned phase). Refrigeration system operation was nominal and within the prescribed temperature range until mission day 29.

Data obtained by way of Mission Operation Planning System indicated that at 173:02:03 on mission day 29 (at a time when a radiator bypass valve (Figure VG11-6) switch from bypass to radiator position was expected) an abrupt 5 psid decrease in pump delta pressure was noted. This decrease indicated that the flow path of the coolant had suddenly changed, but not in the expected manner (a change to the radiator position would exhibit a rapid increase in delta pressure). Subsequent to this event, the thermal capacitor inlet temperature began a rapid rise.

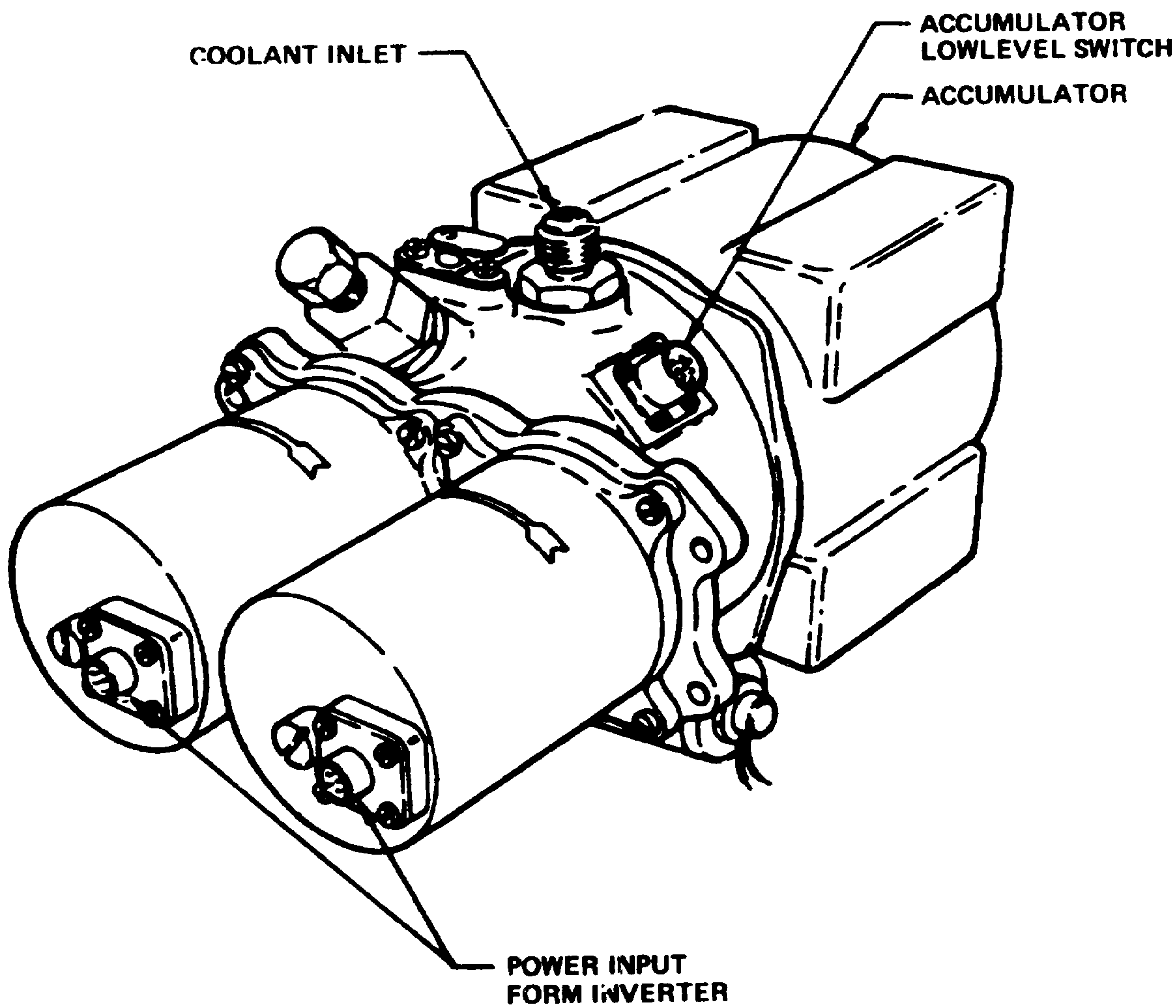


Figure VG11-3. Refrigeration Pump Assembly

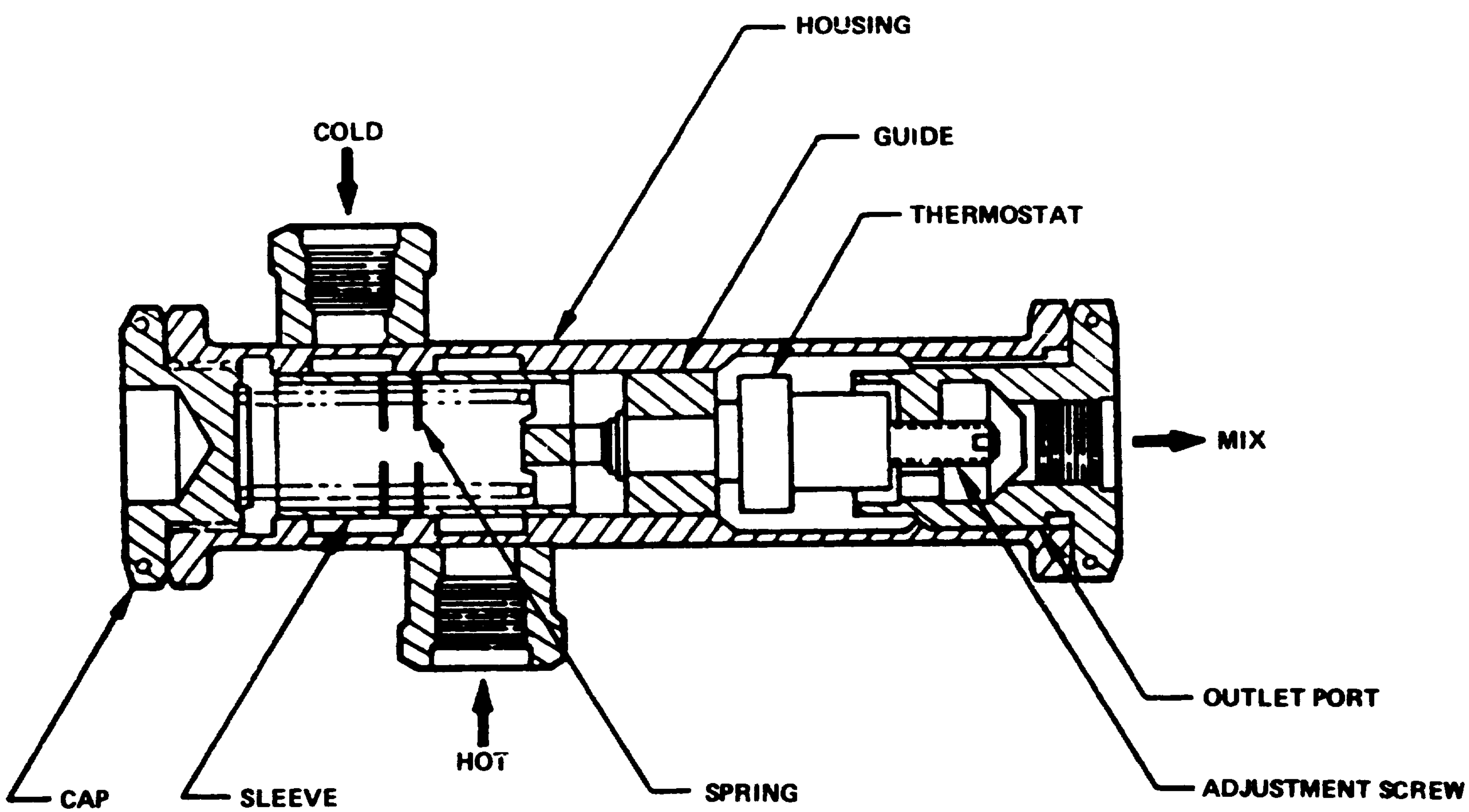


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Figure VG11-4. Chiller Control Valve

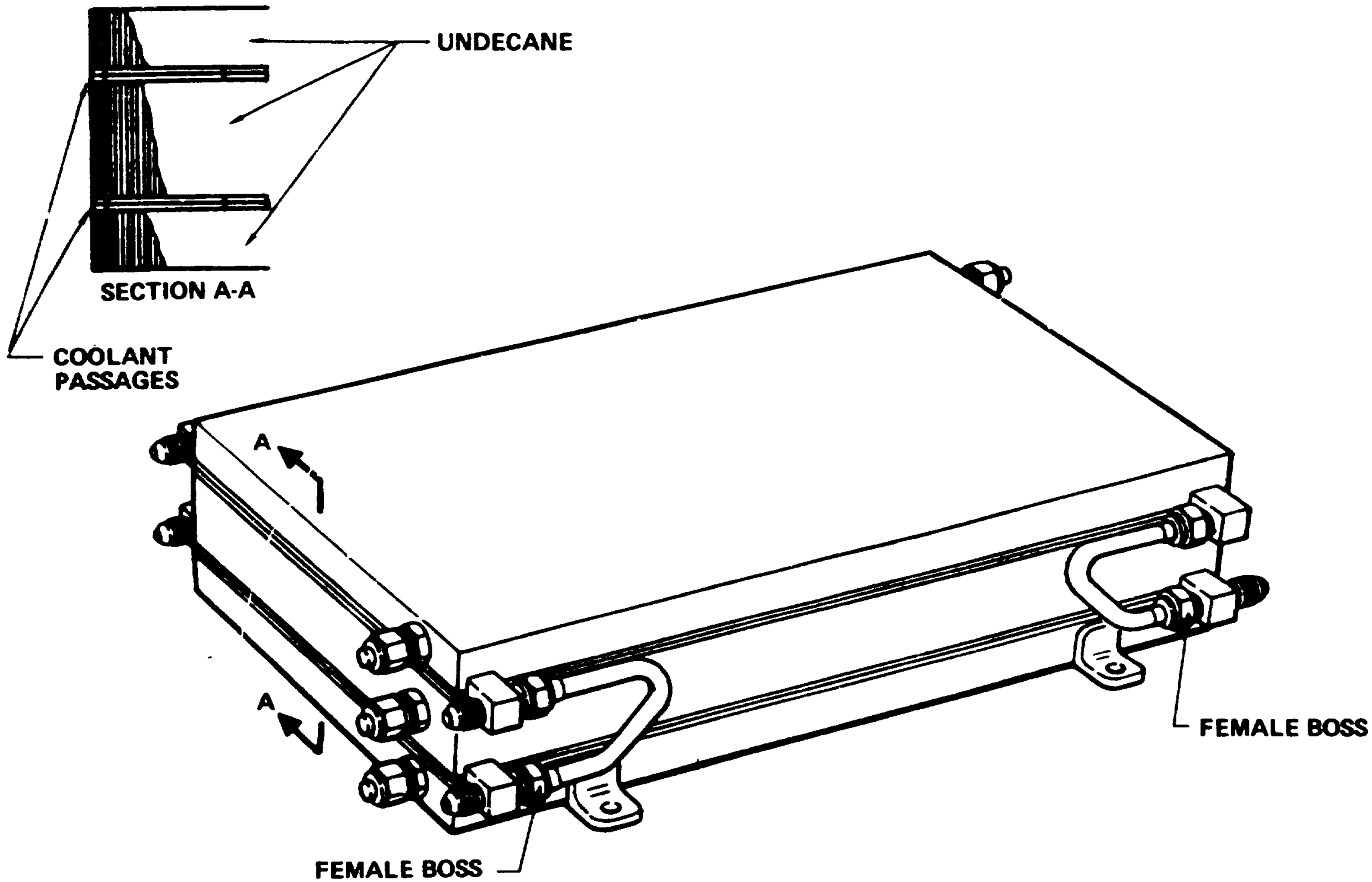


Figure VG11-5. Thermal Capacitor

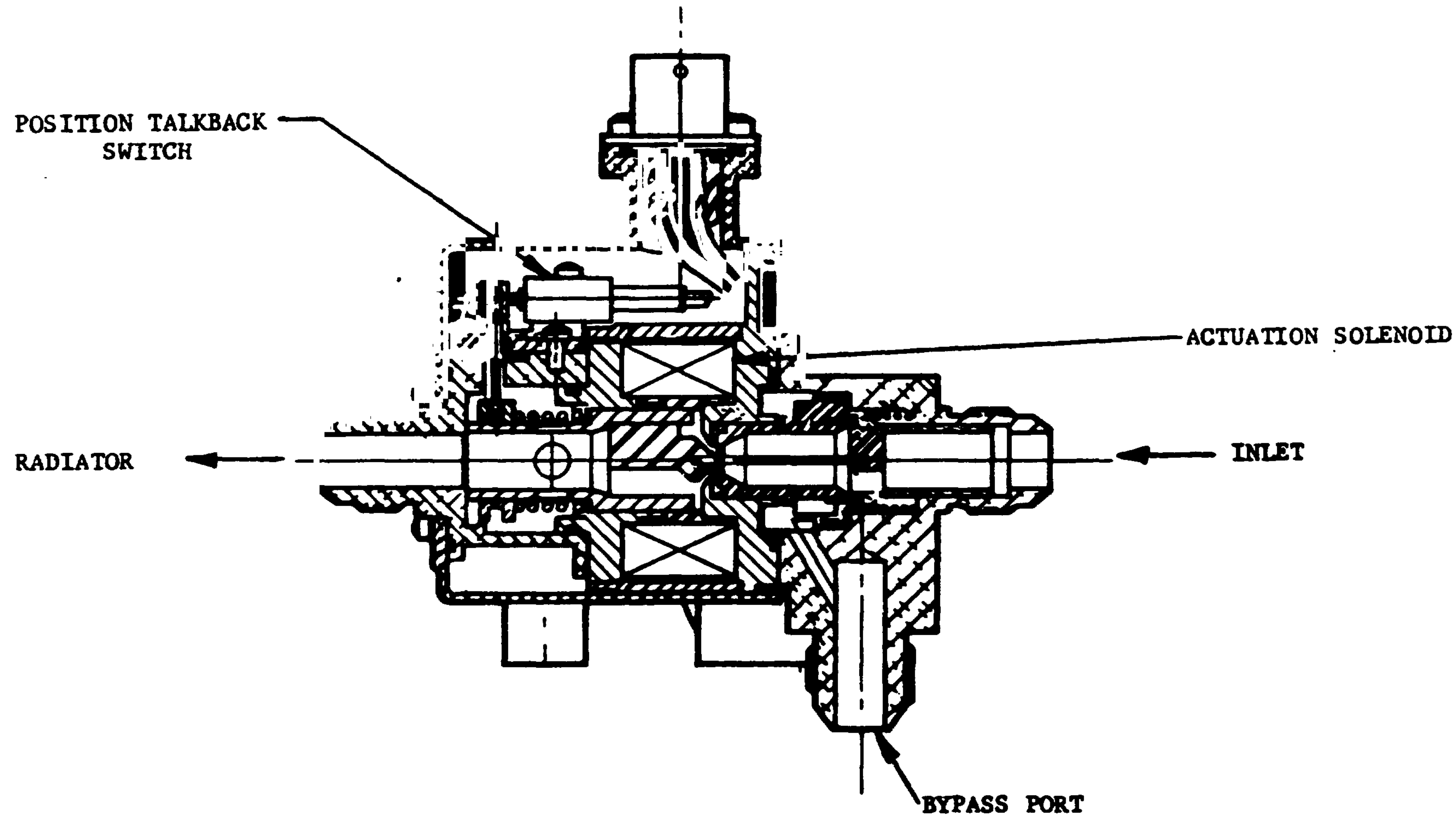


Figure VG11-6. Radiator Bypass Valve

This event led to thawing of the thermal capacitor and some of the refrigeration system freezers eventually exceeded specification (0 °F maximum).

The onboard logic eventually sensed this failure and switched loops when the coolant inlet temperature to the wardroom freezer reached 1 °F. However, the secondary loop, while operating for approximately 45 min, exhibited a more rapid temperature rise than the primary.

On the premise that the refrigerant contained contaminate causing the valve to "stick", the radiator bypass valves were cycled, by enabling and disabling the loops by ground command. This was done during the remainder of mission day 29 and the first 11 hr of the following day. When a loop is disabled, the bypass valve switches to the bypass position and when a loop is enabled, in this case, it switches to the radiator position. Therefore, this loop switching resulted in cycling the bypass valve. The primary loop was cycled 113 times and the secondary loop 41 times before the primary loop pump #1 was allowed to run continuously beginning at 174:10:50. Average system temperatures began to exhibit a slow but consistent decrease. By day 21 of the second unmanned period, the thermal capacitor inlet temperature was cycling around 14 °F and the urine freezer was cycling around -11 °F. The radiator bypass valve never again automatically cycled to the bypass position.

It appeared from data that coolant flow was being bypassed around the radiator and was mixing with the reduced quantity of radiator flow prior to entering the thermal capacitor. The net result was a higher temperature of fluid entering the thermal capacitor. A probable cause was leakage past the radiator bypass valve bypass poppet seat, with the valve positioned in the radiator flow position. Bypass valve seat openings in excess of 25 microns could cause significant valve leakage and would account for the observed on-orbit system performance.

End of mission flushing tests were conducted on the last day of the third manned phase and the following day, in an effort to confirm that the first manned phase mission day 29 anomaly was caused by contamination in the radiator bypass valve. No further information was gained.

(2) Low temperature excursion--During the third unmanned period following the anomaly on mission day 29 of the first manned phase, the radiator bypass valve circuit breaker was opened. This prevented the valve from switching from the radiator position. Consequently, the frozen food temperature varied as a function of workshop internal environment.

d. Recommendations.

(1) Prefiltration for valves. Large capacity filters, 15 micron absolute (or better), should be installed between sensitive valves with close tolerance fits and any potential contaminant generators (pumps, heat exchangers, etc.). They should be installed as close to the upstream valve port as is feasible.

(2) Filter location. To prevent filter blockage by ice crystal formation, as a result of minute amounts of water usually found in refrigerants, filters should not be installed at cold locations.

(3) Use of orifices. Systems using flow diverting valve should, if additional pressure drop is acceptable, use a flow-limiting orifice in the branch least sensitive to flow variations in the event of a flow split caused by a valve malfunction.

(4) Leakage. No discernable leakage was experienced during the Skylab mission. Maximum use of brazed joints and careful use of specialized mass spectrometer methods of leak testing will be necessary if similar success with future systems is to be achieved.

(5) Component life. Four pumps were needed in each workshop refrigeration system loop to meet Skylab mission life requirements. Since essentially only one pump in each loop was used during the mission, future system design need not be so conservative if pump life can be predicted or verified by test (Figure VG11-1).

(6) Redundancy. The philosophy of redundancy should be re-examined since each of the two identical workshop systems experienced similar anomalies with the possibility that the same component in each loop malfunctioned.

12. Thruster Attitude Control System (TACS).

a. The general requirements for the TACS are:

(1) The TACS shall provide attitude control after the S-II separation until CMG spinup and shall assist the CMG in control of the workshop during orbital flight.

(2) This system shall use cold gas for control impulse; high pressure spheres to provide the gas shall be mounted to the thrust structure.

(3) Control valves and interconnect tubing shall supply gas to two thruster modules, with three thrust nozzles each, located at positions 1 and 3 on the aft skirt.

(4) The minimum individual nozzle thrust at the start of the mission shall be 50 lb.

b. Mission performance. Because the specific impulse was higher than expected, system leakage was negligible. No instrumentation was available to determine the specific leakage.

During each unmanned phase there were several periods without TACS usage. Calculations of remaining GN_2 propellant using sphere pressure and temperature data for these periods indicated no discernible leakage occurred.

During preflight testing, it was noted that certain vibration frequencies caused the valve transducer to erroneously indicate valve chatter. These frequencies were predicted and did occur during portions of the S-IC and S-II burn and examination of flight data shows chatter. Except for this chatter indication, which was of no consequency, all data indicated that the TACS valves performed normally throughout the mission.

c. Anomalies. The TACS system performed as designed and no problems were incurred.

d. Recommendations. The negligible leak rate of the TACS system has verified the adequacy of bimetal joints and "in place" induction brazing for consideration in long-term storage of high pressure gases. The TACS valves remained leak tight after extensive orbital usage, which verified the valve design, materials, and testing program for future programs.

The unanticipated high propellant consumption during the early part of the mission caused concern that TACS might be depleted prematurely. Future designs should consider interconnecting systems using similar working fluids, such as the TACS and the airlock O_2 and N_2 supplies

13. Special Projects Hardware. The performance of special hardware developed as a result of the orbital workshop meteoroid shield/solar array system anomalies that occurred during launch is covered in this section. It was theorized, since no photographs, TV coverage, etc., was available during the first unmanned period, that the undeployed beam fairing on the solar array system wing #1 was restricted by a portion of the meteoroid shield debris. The special hardware subsequently developed and built by the MSFC to remove the restriction was based on this theory and the photographs taken later by the first crew (Figure VG3-8) bears the theory out as valid. It was also learned from these photographs that the restricting debris did in fact resemble a strap which was imbedded in the honeycomb fairing of the beam allowing only a 5° - 10° deployment.

A concerted design and build effort took place at the MSFC, as well as facilities of several contractors who were asked to join the effort to develop the special debris removal/cutting tools required to free the restricted solar array system beam fairing. An around-the-clock effort by civil service and contractor personnel was initiated during which several design concepts were evaluated. Among those considered were: the use of a bone saw; a chisel and hammer; an oxyacetylene torch; special tools for sawing (manual and powered); a solar parabolic mirror system for burning; cutting with mechanical shearing devices; and severing or prying with special designed instruments. The sawing, chiseling, and hammering-type tools were rejected bacause they were considered unsafe for space use because of their potential for causing space suit punctures. To use them the astronauts would be required to move too close to the torn/jagged edges and corners of the anomalous area for the proper and accurate operation of the tools, and the resultant residue cuttings would, themselves, be potential contaminate debris. The oxyacetylene torch and solar parabolic mirror system were also discarded because they were not readily adaptable to space use because of the nonexistance of: (1) a suitable gas regulator for the torch; and (2) a rigidly platformed sun alignment/following device for the mirror system. A matrix of the tools and the task accomplishment considerations is shown in Table 13-1.

As a result of the considerations listed above, only the cutters, shears, and the prying/severing (universal tool) concepts remained as candidates for further investigation and development. Commercial and Government suppliers of these type tools were consulted regarding availability of off-the-shelf hardware that could be expeditously modified/adapted for space use within the very limited time frame before the first manned phase lift-off (approximately 10 days). One prying (universal) and two cutting (shears and cable cutters) tools were designed and manufactured by the MSFC within the required time, were stowed aboard the first manned command service module, and subsequently flown to the orbiting unmanned laboratory. An evaluation of these tools follows:

a. Evaluation. The three special tools developed by MSFC were designed to fit the end of a tubular aluminum interlocking utility

TASKS / TOOLS	CUTTERS	SHEARS	UNIVERSAL TOOL	SOLAR MIRROR	BONE SAW	CHISEL & HAMMER	OXY TORCH	SAW
1. SAFE IN SPACE	+	+	+	+	(–)	(–)	+	(–)
2. ADAPTABLE TO SPACE USE	+	+	+	(–)	+	+	(–)	–
3. COMPLETE TASK IN	+	+	+	–	+	+	(–)	–
TIME ALLOWED								
4. PRY/SEVER	–	–	+	–	–	–	–	–
5. CAN BE OPERATED								
FROM AT LEAST 25 FT.	+	+	+	–	(–)	(–)	(–)	(–)
6. CUT;								
SHEET (2024-t4/.025)	+	+	–	+	+	+	+	+
ANGLE (7075-T6/.062)	+	–	–	+	+	+	+	+
PLATE (6061-T6/.095)	+	–	–	+	+	+	+	+
BOLT/NUT (NAS1003-3A, MS21043-3)	+	–	–	+	+	+	+	+
RIVET (MS20426)	+	–	–	+	+	+	+	+
7. SERVE AS CLAMP	+	+	–	–	–	–	–	–
8. POUND/HAMMER	–	–	+	–	–	+	–	–

NOTE: + = ACCEPTABLE ( ) = PRIME REASON(S) FOR REJECTION

– = UNACCEPTABLE

Table VG13-1. Tool/Task accomplishment comparison

pole (same as utilized for thermal shield), approximately 1-in. in diameter and 5-ft long, and to function on a long handle made by assembling two, three, or more sections of these poles. At this distance, approximately 25 ft, an astronaut standing in the command module hatch could safely attempt to remove or cut the obstructing debris from the solar array system beam fairing. The three MSFC developed tools are shown in Figures VG13-1, VG13-2, and VG13-3. The universal tool and the cable cutters, were developed from off-the-shelf electric power line hardware as used by electrical power linemen with modifications for lightness and use with space suits.

The universal tool is a general purpose tool designed for pushing, pulling, and prying. The shears are a thin sheet metal cutting tool. The cutting action is provided by retracting an attached line. The cable cutters are similar to the shears in appearance and operation but are intended for cutting thicker sheet metal, such as the strap that was later found to be preventing the solar array system beam fairing from deploying.

After a damage assessing flyaround by the first crew, a standup extravehicular activity was performed on the first day of this manned phase, using the open hatch of the command service module as a work platform. The crew was unsuccessful in their attempt to dislodge the beam fairing with the universal tool. It is theorized that the attempt was unsuccessful because the tool could not be maneuvered into a proper position on the strap to afford the astronaut the leverage moment needed. No attempt was made to sever the strap during this standup extravehicular activity using either the shears or cable cutters because of astronaut fatigue and time restriction.

Through simulation and test activities at MSFC, which were based on crew descriptions and television coverage, it was decided that onboard tools could be successfully used to free the beam fairing. The strap was determined to be in the general shape of a 90° angle with legs approximately 1.62-in. long, 0.75-in. high, and 0.182-in. thick. The short leg was comprised of three layers of aluminum sheets (types 7075-T4) riveted together in a staggered pattern on 0.75-in. centers. The cutting force required to sever the strap was analyzed to be between 40 and 50 lb.

Using the cable cutters, an extravehicular activity was successfully carried out by the crew on the 14th day of the first manned phase. The cable cutters were secured to a 25-ft pole made from five utility poles, and maneuvered to the restraining strap from a re-rigged work station at the edge of the fixed airlock shroud. The cable cutters were then secured to the strap as a clamp, without cutting it. The 25-ft pole was then used as a fireman's pole, allowing the astronaut to translate to the beam fairing. He secured a rope, called the beam erection tether (BET) to the beam fairing forward vent module (about 4 ft from the hinge); the other end was fastened to the telescope mount deployment assembly. The astronaut then translated back to the re-rigged work station, and the crew successfully cut the aluminum angle strap by activating the cable cutters. The crewmen were able to apply sufficient force through the

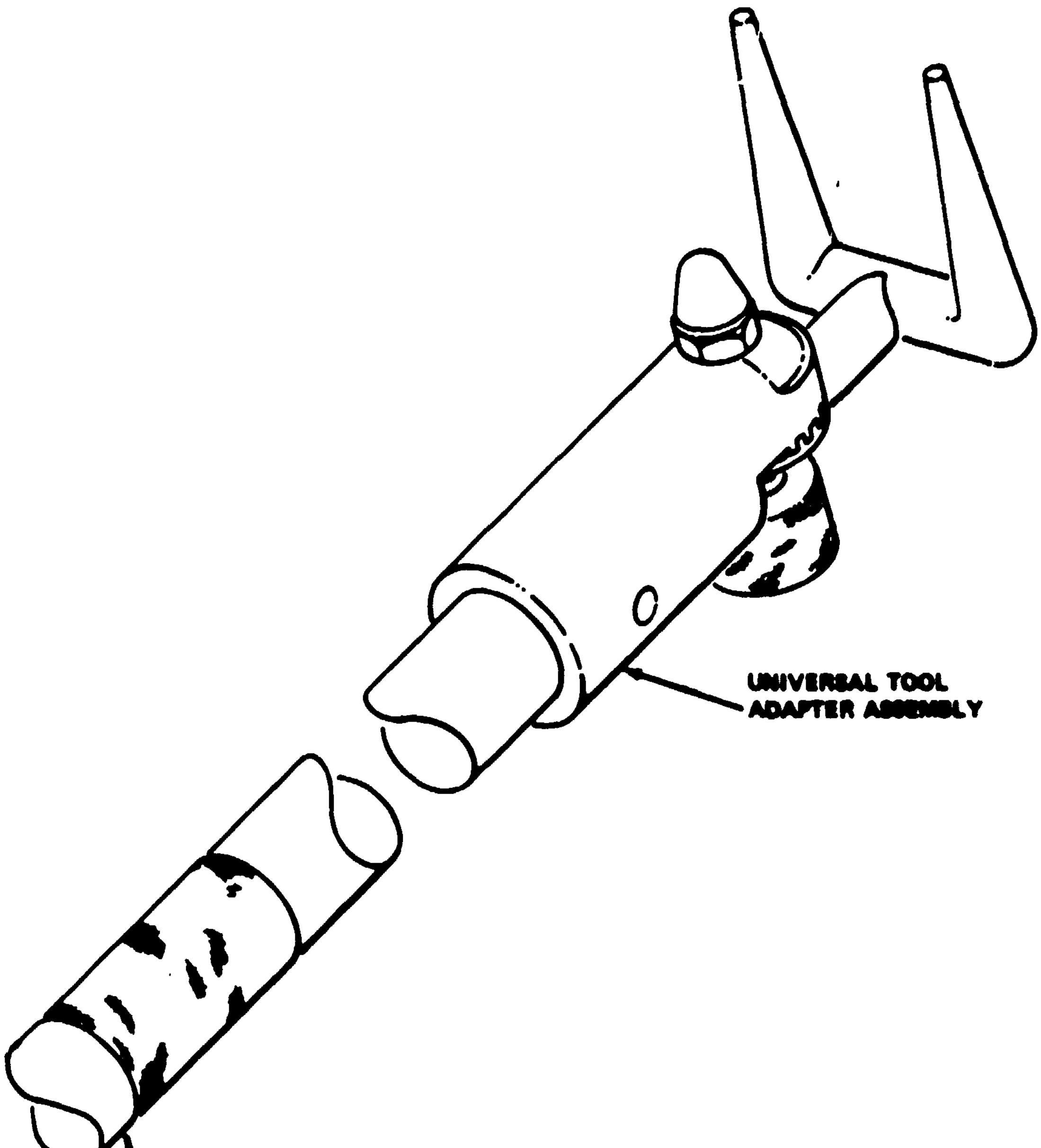


Figure VG13-1. Universal Tool

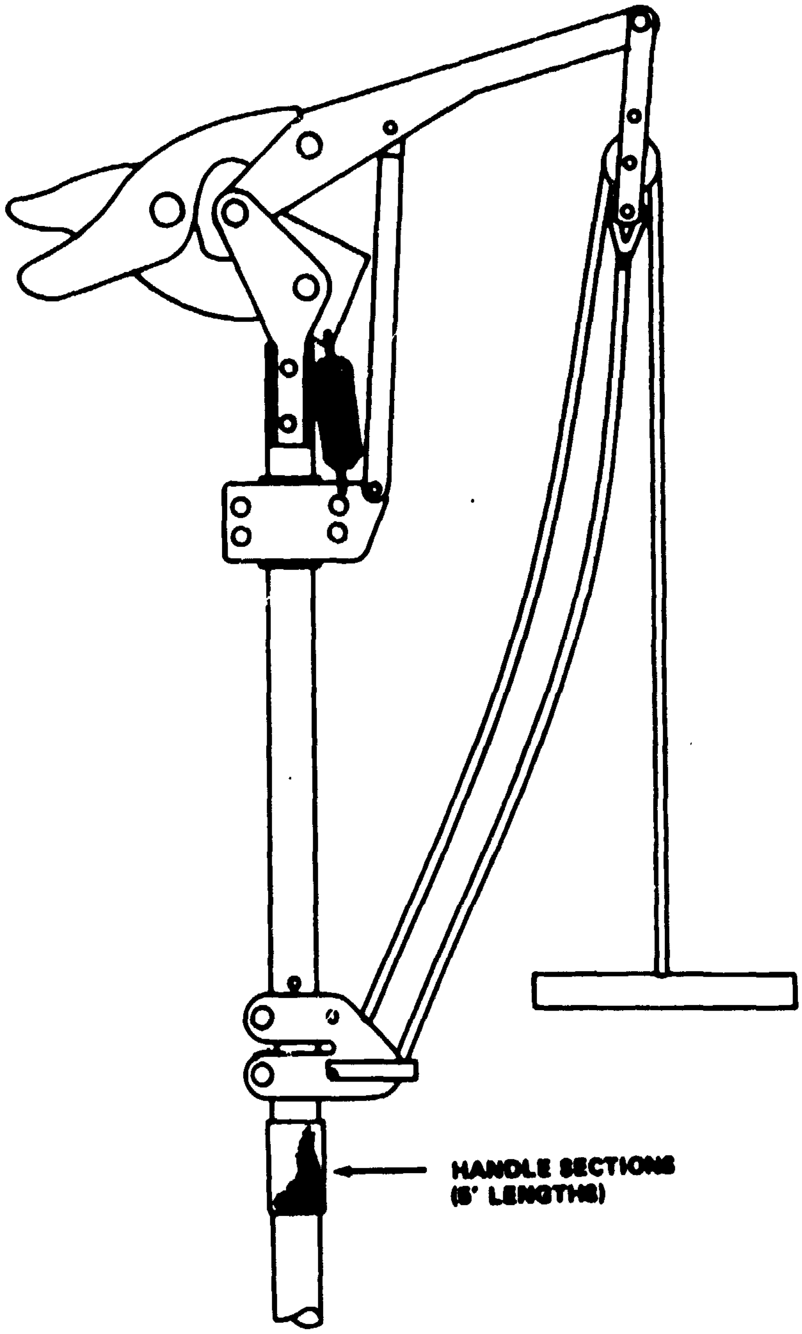


Figure VF13-2. Shear

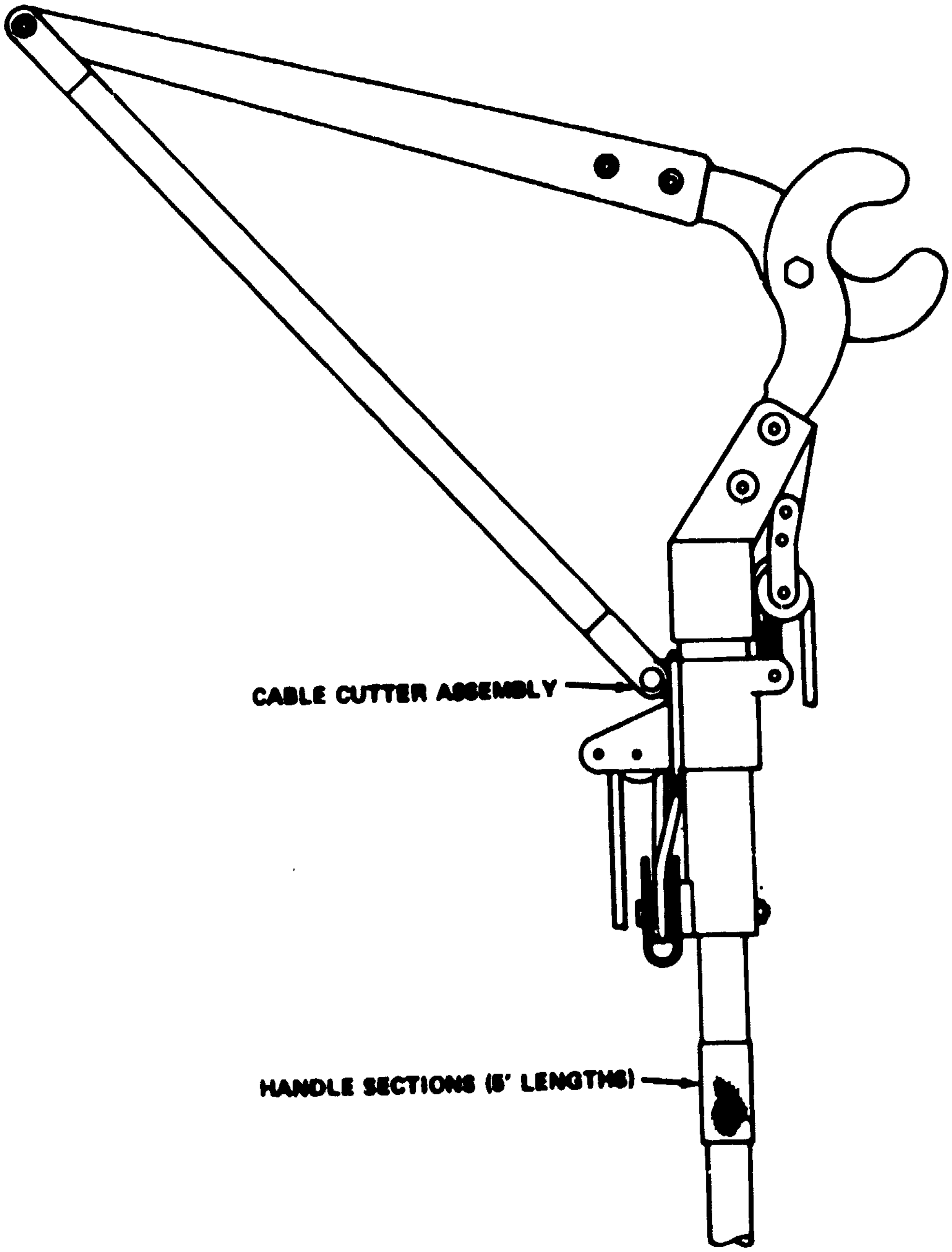


Figure VG13-3. Cable Cutter

approximate 100-lb mechanical advantage designed into the cutters, to cleanly sever the strap. The wing then deployed an additional 8° to 10°, but stopped, as expected, because of a frozen actuator damper. With the strap restraint removed, the astronaut then used the previously assembled beam erection tether to break the clevis on the beam fairing actuator damper by exerting the necessary moment (approximately 750 ft-lb), allowing the beam fairing to deploy fully and latch. Figure VG13-4 illustrates this breaking maneuver.

14. Suit Drying Station.

a. The general requirements for the suit drying system are:

(1) Circulate the workshop cabin air through the Pressure Garment Assembly (PGA).

(2) Maximum air temperature of 120 °F delivered at suit interface.

(3) Dry three suits within 48 hr (dynamically).

(4) Minimum air flow of 10 acfm (Actual Cubic Feet Per Minute) through the PGA.

(5) Maximum delta P of 3.5 in. of water through PGA.

(6) Minimum moisture removed during dynamic drying to 400 out of 500 g total.

(7) Provide static desiccant bags (two per PGA) to remove a minimum of 100 g of moisture at 10 percent RH at 75 °F.

(8) Static desiccant bags to maintain air in suit below 55 percent after 50 hr.

(9) Minimum number of drying cycles--23.

b. Mission performance. There was no instrumentation in the system to monitor performance parameters. However, the suit drying equipment performed as planned. All hardware operated satisfactorily with the exception of the high touch temperature on the power module. The crew reported that the suits were dried very well and that there was no odor to the suits after the drying process. The noise level during the long operational time of the blower was acceptable.

c. Anomalies. It was reported by the first crew that the suit dryer power module was too hot to touch. The second crew was instructed to leave the compartment door open for additional cooling. No further problems were reported.

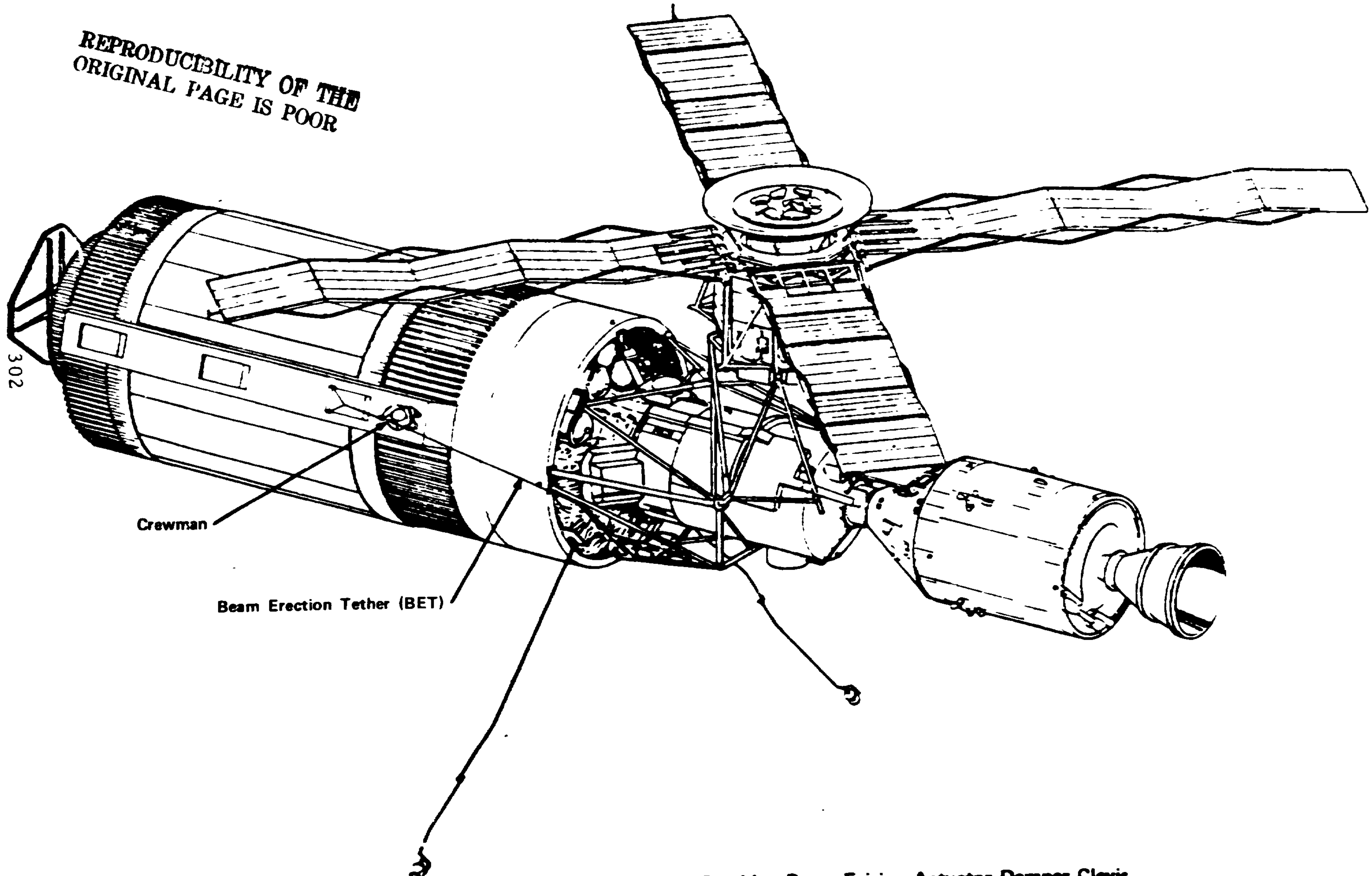


Figure VG13-4. Breaking Beam Fairing Actuator Damper Clevis.

During qualification testing in a 5 psia environment, the power module handle reached 96 °F.

15. Whole Body Shower.

a. Whole body shower description. The whole body shower (WBS) provides the crewmen with a means of cleaning the entire body similar to the 1 g environment. Operationally, water is obtained from the workshop waste management water heater and transferred to the crew quarters shower location by a water bottle module. A shower is taken by connecting the water hose, of the filled water bottle module, to the upper ring of the shower stall. This is subsequently connected to the spray nozzle in the upper ring. A flexible cylindrical enclosure is used to contain the water during the showering operation. A workshop power module is used to provide air circulation and water transfer from the shower stall enclosure. Air flow moves water from the stall by a pickup head attached to a flexible hose which routes water into a mechanical liquid-gas separator. Water is transferred from the separator into a flexible, removable rubber bag which is restrained by the collection box assembly. To assure proper pumping efficiency, the collection box provides a controlled back pressure to the collection bag by a pressure equalization line connected to the separator. A redundant liquid-gas separation capability is required to avoid the possibility of free water entering the habitable area. The hydrophobic filter assembly provides this backup capability in addition to furnishing an interface between the WBS and the workshop power module. Figure VG15-1 is a system diagram of the workshop WBS. After all showering is completed and the system is shutdown, the water that is collected in the rubber collection bag is removed from the collection box (by the bag). Collection bag and water are placed in an armalon overbag which provides structural support. This support is required to prevent the rubber bag from expanding to the sides of the workshop trash airlock as it is expelled into the waste tank. Miranol jem, contained in a syringe, is used to dispense the cleansing agent.

Qualification and performance requirements with a concise flight evaluation for various components of the workshop whole body shower are given in the following pages.

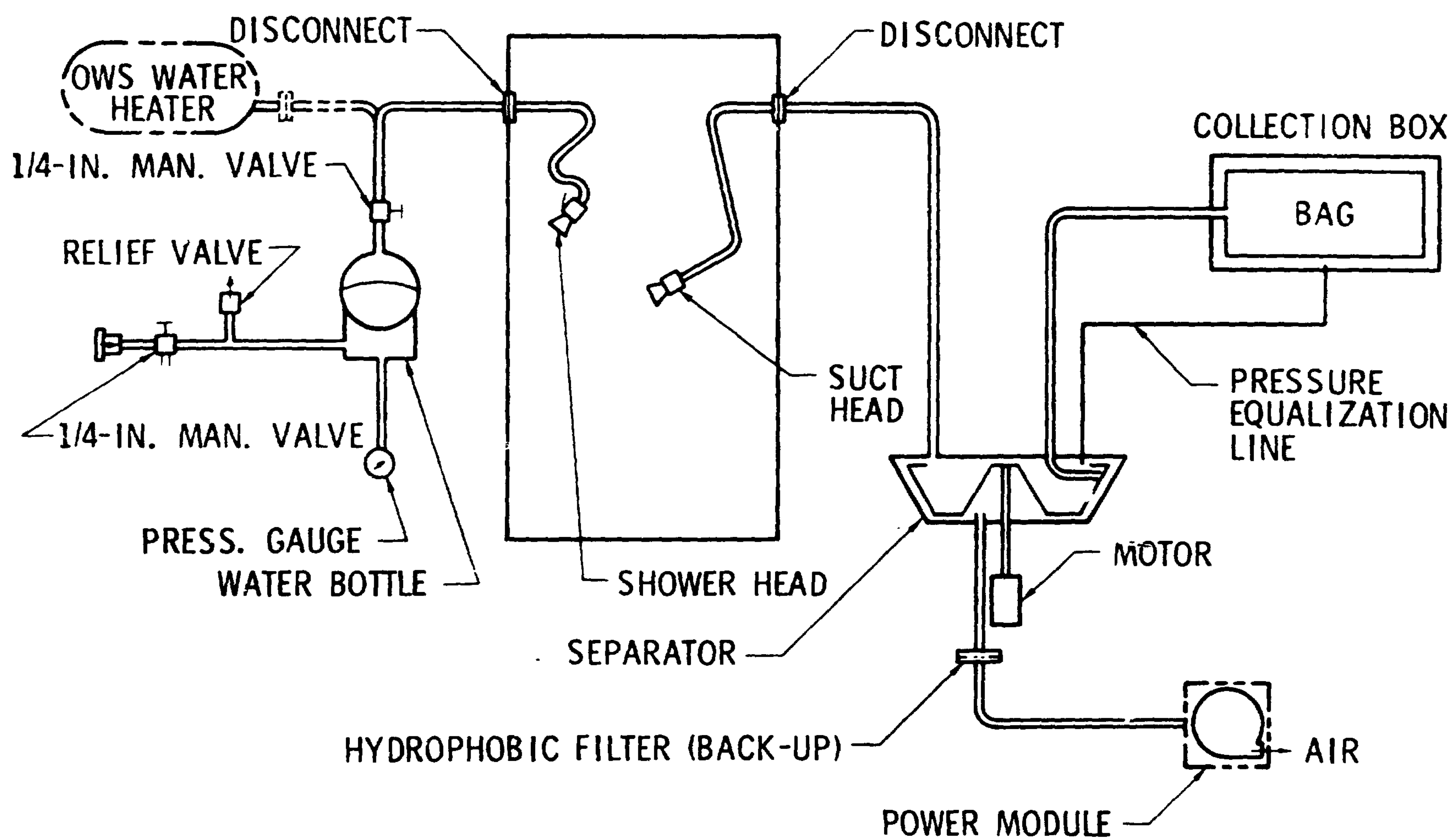


Figure VG15-1. Workshop Shower

WATER BOTTLE MODULE
(See figure VG15-2)

QUAL & PERFORMANCE REQUIREMENTS	FLIGHT EVALUATION
• CAPACITY: 6 lb of H_2O • PRESSURE: Nominal Operating--10 to 25 psig Maximum Design--49 psig • TEMPERATURE: Nominal Operating--60 to 130 °F Maximum Design--160 °F Minimum Design--0 °F	Functioned as designed. Crew comments indicate that water cool-down rate was faster than desired. Water should have been insulated from the metal container. Certain crew members expressed the desirability of a direct connection between the shower and the water system with a full range of temperature control. Operating pressure range reflects the pressure change in the precharged plenum, which maintained a consistant full spray pattern. Nominal operating temperature varied within requirements: the first 4 lb of water were removed from the waste management compartment water heater at approximately 127 ± 3 °F; the last 2 lb were removed at workshop ambient temperature.

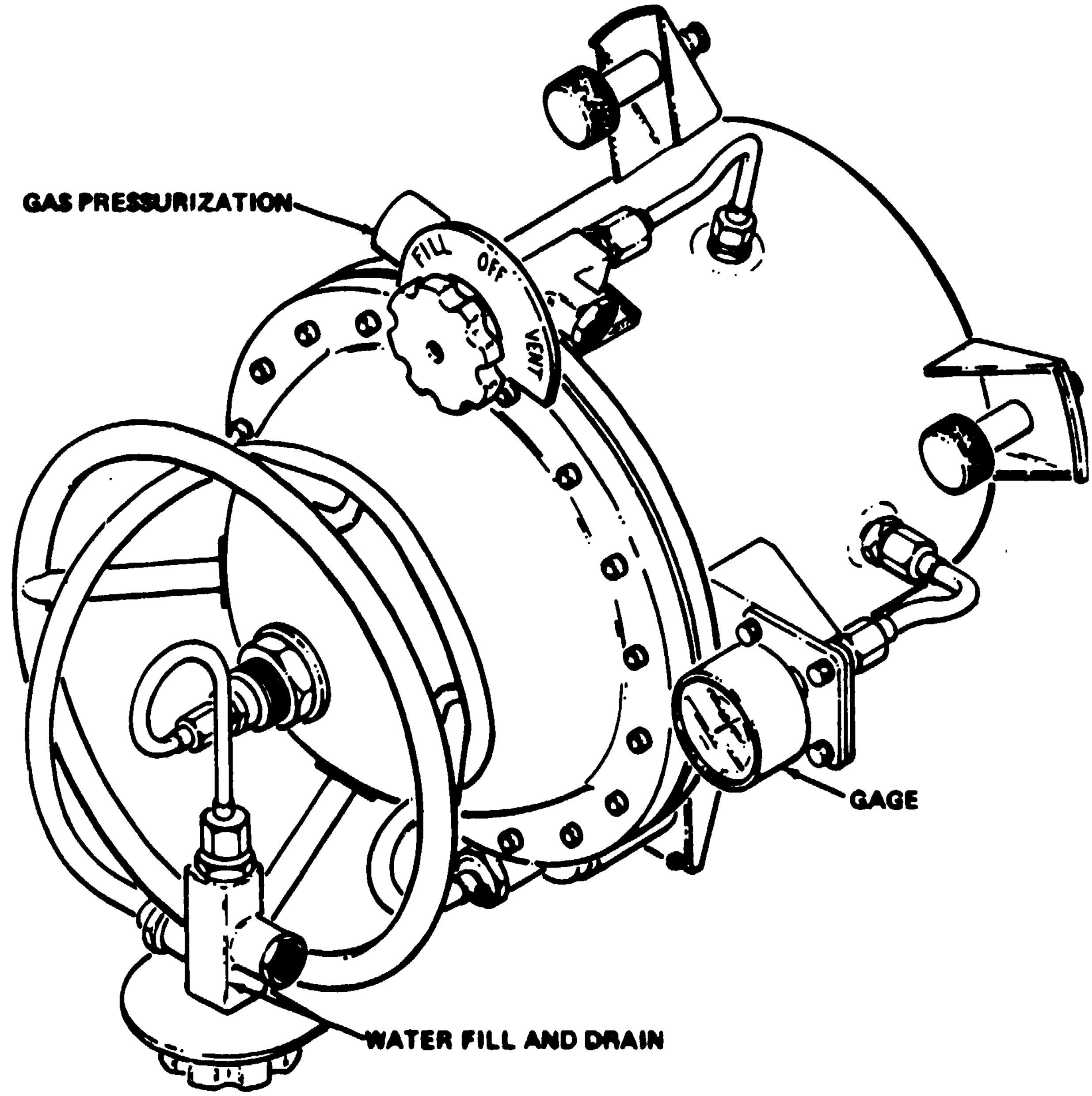


Figure VG15-2. Water Bottle Module

SHOWER STALL ASSEMBLY
(See figure VG15-3)

QUAL & PERFORMANCE REQUIREMENTS	FLIGHT EVALUATION
• COLLAPSIBLE CYLINDER: Dimensions: 36-in. diam by 74 in. Flow Rate and Delta Pressure: 1.0 in. H_2O at 10 scfm • LIFE CYCLE: Installation and removal--25 Shower Curtain—300	Collapsible cylinder was used to reduce launch dynamics problems and allow a larger operational volume in the habitable area when the shower was not in use. Dimensions were defined by human engineering testing and available hardware and proved acceptable. Flow rate and delta Pressure were adequate to contain the free water in the shower stall and maintain the CO_2 level at nominal. Life cycle testing was more than adequate with no curtain or latch failures. The third manned crew complained of wet walls being cold to the touch as they moved about to vacuum up the loose water.

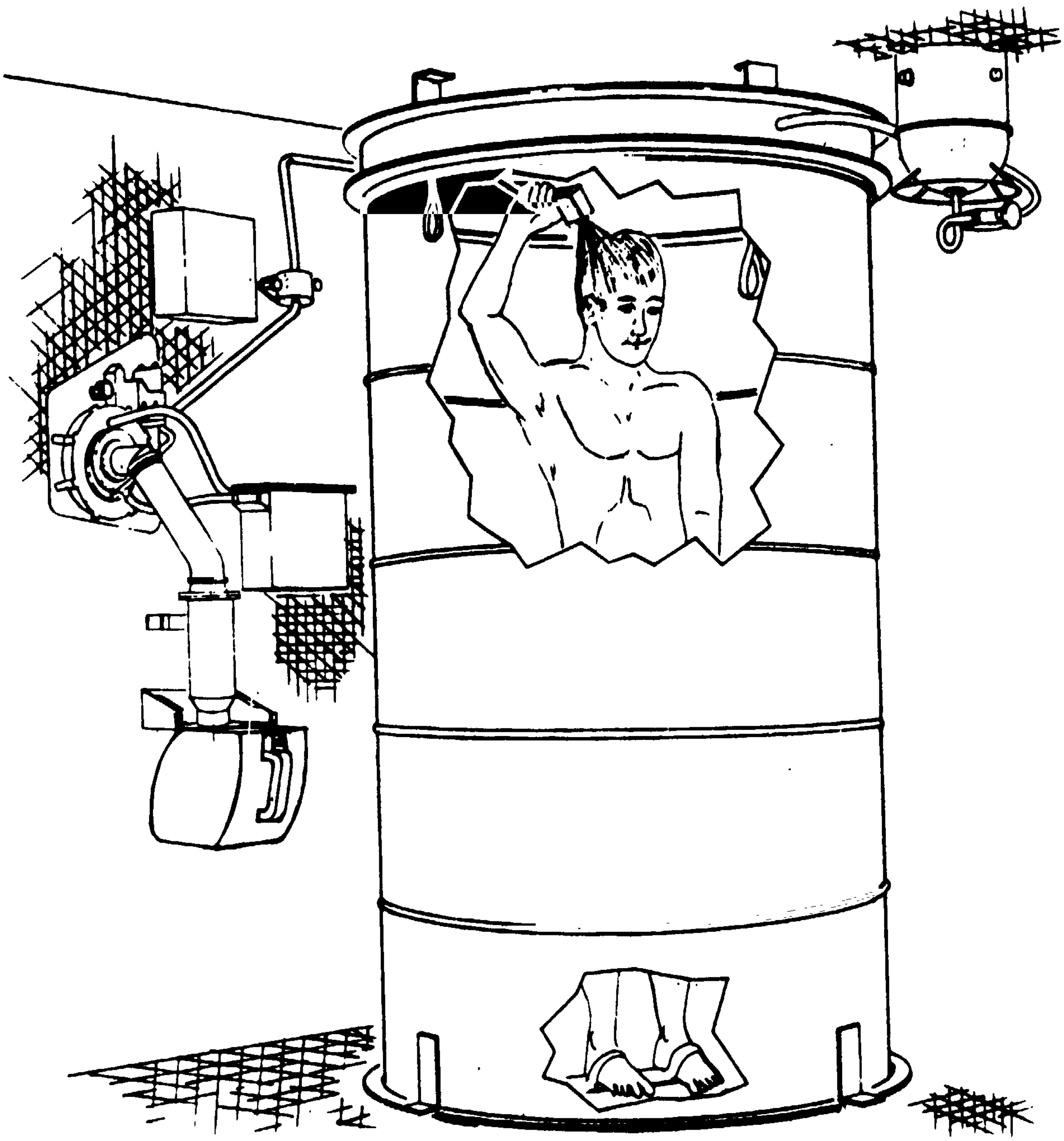


Figure VG15-3. Whole Body Shower Stall (Operational)

SHOWER HEAD ASSEMBLY
(See figure VG15-4)

QUAL & PERFORMANCE REQUIREMENTS	FLIGHT EVALUATION
• SPRAY PATTERN: Full Cone • PRESSURE: Nominal Operating--10 to 25 psig • FLOW RATES: 1400 ml/min at 26.2 psig 850 ml/min at 7.4 psig • LIFE CYCLES: 10,000	A full cone spray pattern was maintained throughout the nominal operating range. Flow rates and the spray pattern were found to be very comfortable and desirable. Design cycle life proved to be more than adequate. There were no crew complaints relative to the shower head.

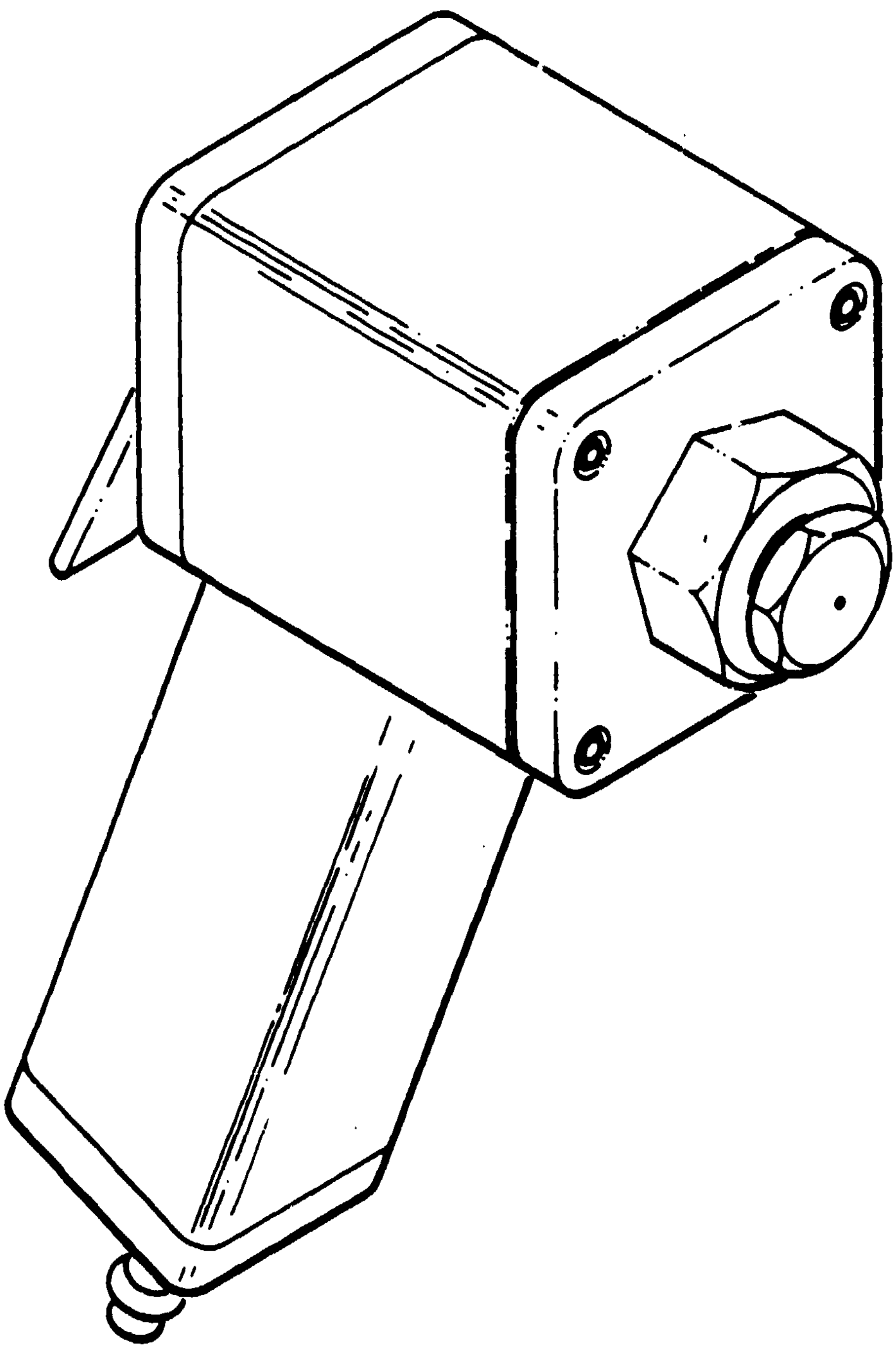


Figure VG15-4. Shower Head Assembly

SUCTION HEAD ASSEMBLY
(See figure VG15-5)

QUAL & PERFORMANCE REQUIREMENTS	FLIGHT EVALUATION
• DELTA P: 1.5 in. H_2O at 10 scfm • LIFE CYCLES: 10,000	Pickup efficiency was considered poor. Shower stall clanup was very time consuming. Higher air flows will be required to improve the pickup efficiency significantly (Higher air flows of 15 to 18 in. H_2O with the required blower head were not available on Skylab.) Cycle life of wiper was sufficient as no crew comments were recorded concerning wiper failure.

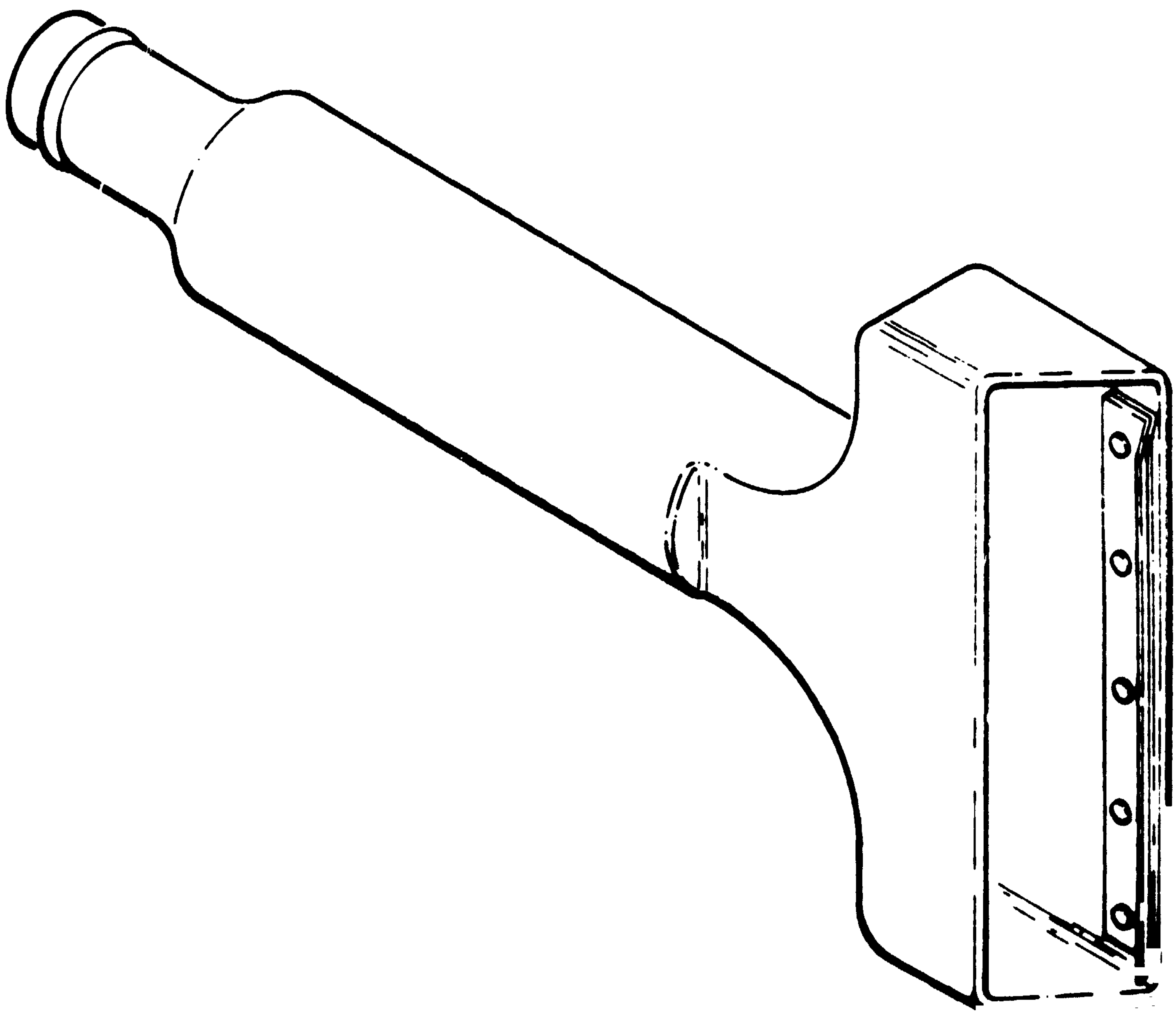


Figure VG15-5. Suction Head Assembly

COLLECTION BOX AND BAG ASSEMBLY
(See figure VG15-6)

QUAL & PERFORMANCE REQUIREMENTS	FLIGHT EVALUATION
• COLLECTION BOX: Contain the collection bag; provide 1.0 in. H_2O pressure on the collection bag during liquid gas separation. • COLLECTION BAG: Capacity: 6.9 lb of H_2O. No visual liquid leakage • Design Burst: 4 psig • OVER BAG: Provide structural support for the collection bag.	The collection box successfully contained the collection bag and it maintained approximately +1 in. delta P exterior to interior of the bag during shower operation. This delta P was maintained with a balanced set of orifices. Throughout the Skylab mission the collection bag contained all the waste water produced during one shower. After three showers were taken, the three filled, bag-over bag, units were disposed of through the workshop trash airlock. The over bag successfully contained the collection bag sufficiently to permit operation of the trash airlock.

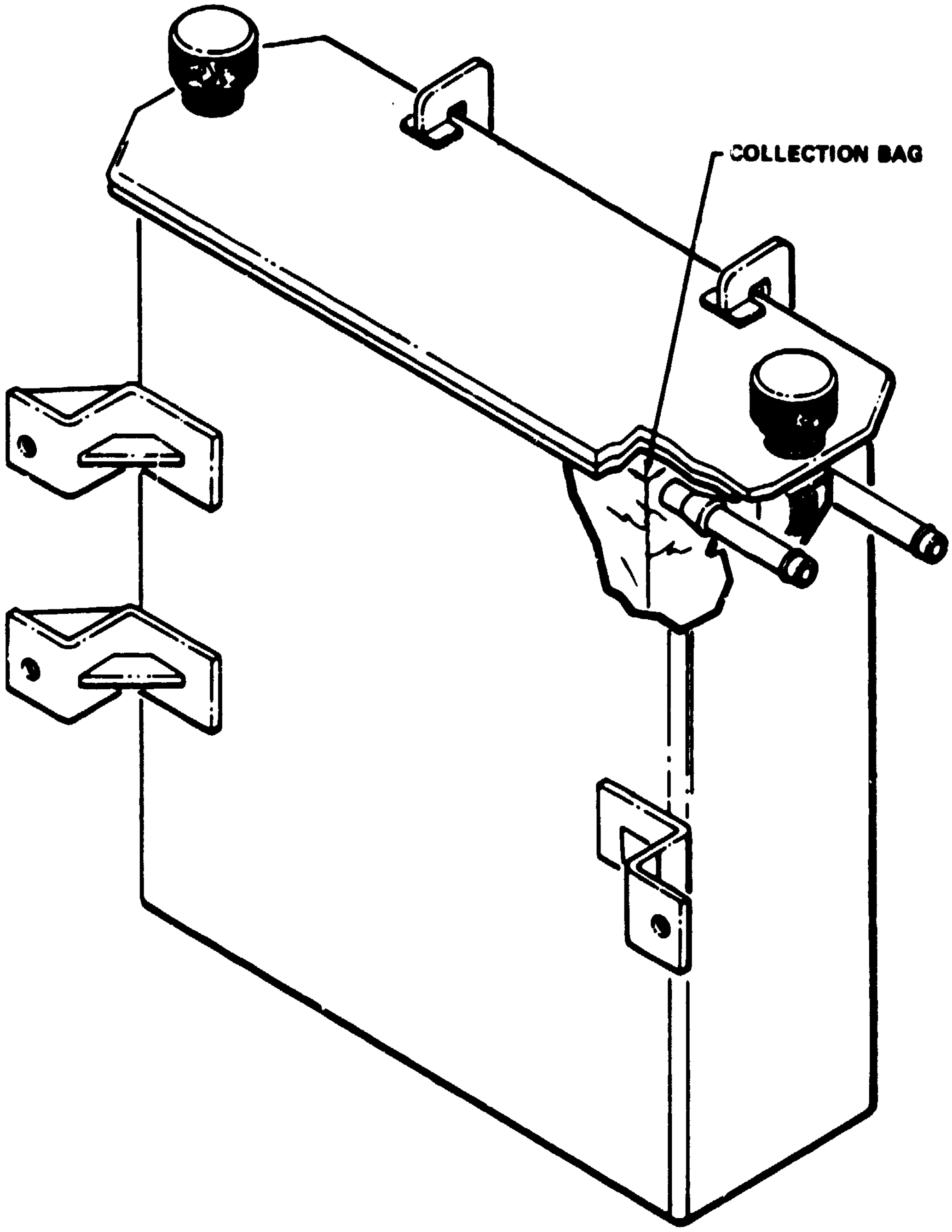


Figure VG15-6. Collection Box Assembly

LIQUID-GAS SEPARATOR
(See figure VC15-7)

QUAL & PERFORMANCE REQUIREMENTS	FLIGHT EVALUATION
• PUMPING RATE: 1100 ml/min of H_2O. Delta P: 5.0 in. H_2O at 10 scfm. • RPM: 250 at 28 Vdc	The unit performed as designed until day 77 of the third manned phase at 1856 CDT. At that time the crew reported the shower power module was inoperative. This condition appeared to be the result of the water in the module. Based on this assumption, failure of the liquid-gas separator was probable. Crew comments indicated that the shower soap supply was depleted and that they replaced it with Nutragena soap. The reduced pumping capability caused by high sudsing and plugging probably resulted in waste water being carried out the air outlet of the separator.

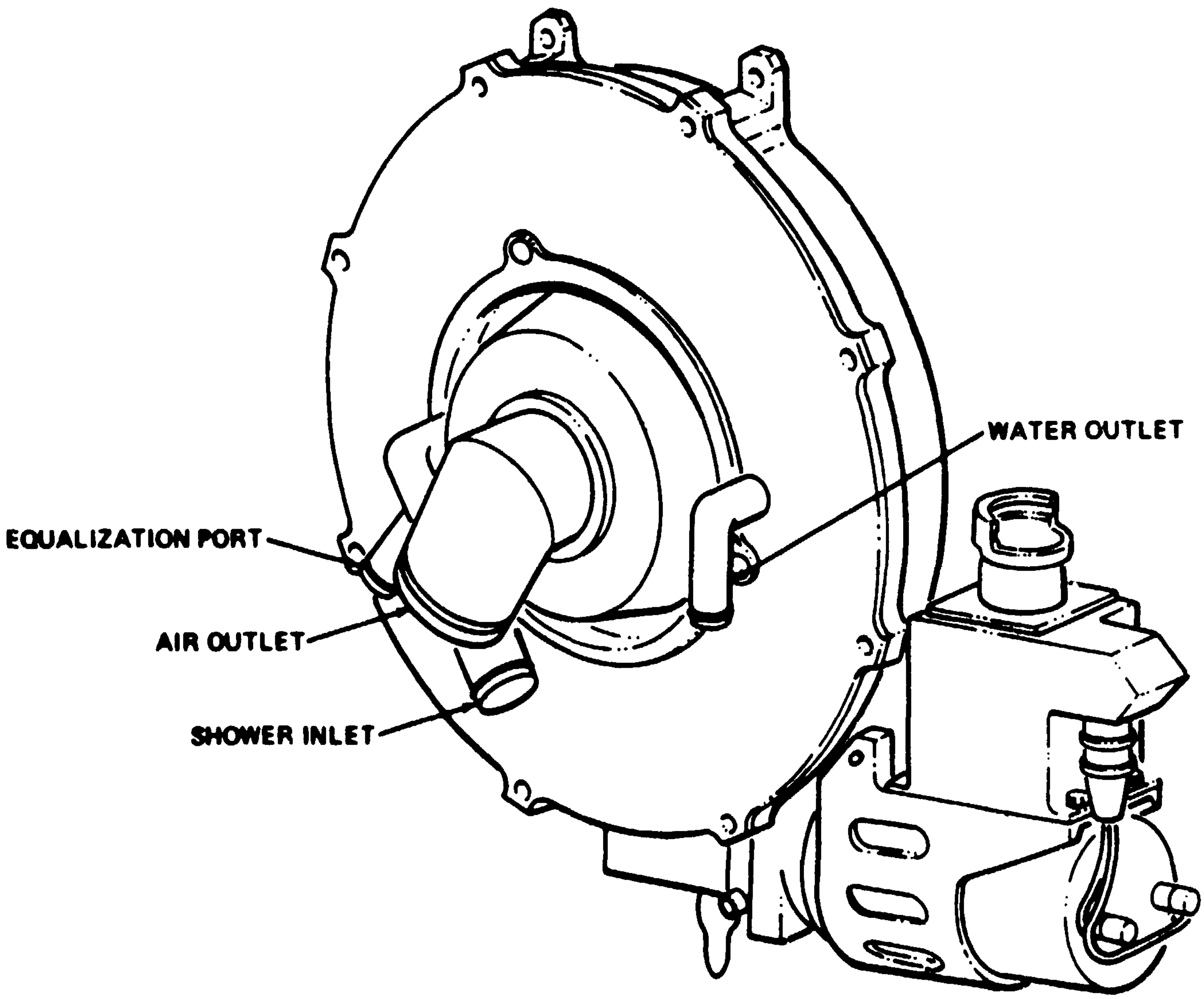


Figure VG15-7. Centrifugal Separator

HYDROPHOBIC FILTER ASSEMBLY
(See figure VG15-8)

QUAL & PERFORMANCE REQUIREMENTS	FLIGHT EVALUATION
• Contain A Hydrophobic Filter Element. • ASSEMBLY DELTA P: 3 in. H_2O at 5 scfm. • FILTER ELEMENT: H_2O. • HOLDING CAPACITY: 82 ml.	The unit performed as designed until day 77 of the third manned phase at 1856 CDT. At that time, the crew reported the shower power module was inoperative. This condition appeared to be the result of water in the module. It is assumed that water passed through the filter element. Crew comments indicated that a considerable amount of water was in the filter element when it was checked. Nutragena soap was used toward the end of the mission because of a lack of soap and this would quickly break down the hydrophobic filter element.

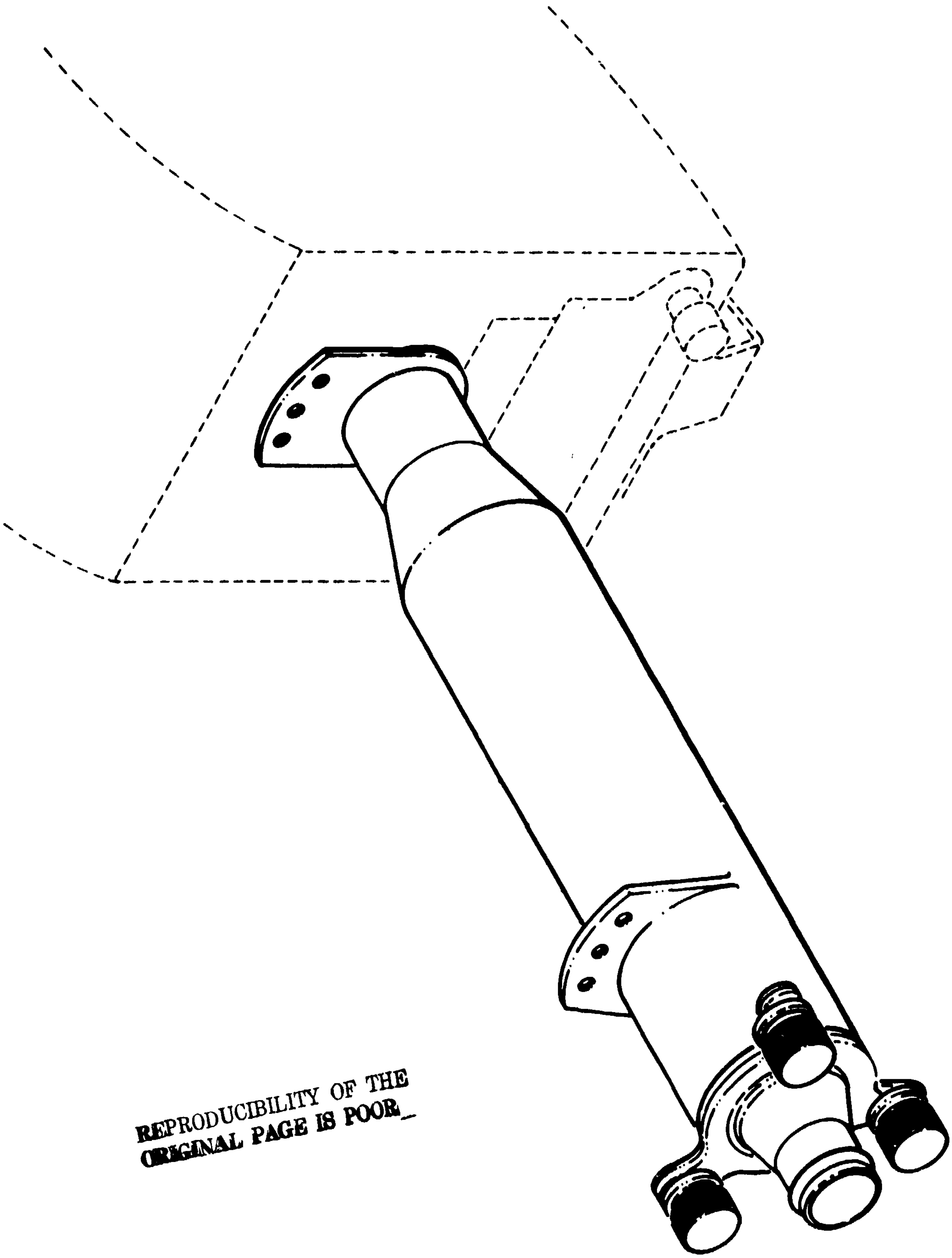


Figure VG15-8 Hydrophobic Filter Assembly

SOAP DISPENSER & SOAP

QUAL & PERFORMANCE REQUIREMENTS	FLIGHT EVALUATION
• DISPENSER: Capacity: 64 ml Pressure: Normal Operating - 0-12 psig Proof - 18 psig • LEAKAGE: 1.0 x 10^{-2} sccs at 12 psig • SOAP: Nonbacteriacidal Nonwetting of Hydrophobic Membranes Approval of The Food And Drug Administration Low Sudsing	Crewmen reported soap shortages. Leakage was never reported throughout the Skylab mission. The cleansing agent (Miranol Jem) performed as required. The odor was not desirable and crew comments reflected this opinion. In preflight development, adverse, cleansing agent comments were received only when greater quantities than planned were used. Crew comments reflect that they ran out of soap, which is indicative that larger portions than planned were used.

16. Liquid Dump Systems.

a. General requirements. The CEI specification required that the liquid oxygen tank be utilized as a waste tank for disposing of all wet and dry refuse; that provision be made for dumping airlock environmental control system condensate water into the waste tank; and that valves be provided to control the vacuum ports for the urine dump and water management dump systems. The workshop design, as described in the following, met these requirements.

The liquid dump system consists of three separate sets of plumbing for dumping liquid waste into the waste tank; The liquid urine dump system, the wardroom water dump system, and the waste management compartment water dump system (Figure VG16-1).

The liquid urine dump system was designed as a backup method for disposing of the daily accumulation of urine into the waste tank after an appropriate sample had been withdrawn.

The two water dump systems were used to evacuate the fresh water supply lines in the wardroom and the waste management compartment during flushing (with iodine biocide) and filling operations, and to drain the lines prior to the orbital storage periods.

The condensate dump system is composed of two parts. The backup workshop condensate dump system running from the forward hatch to the waste management water dump compartment and the holding tank that was connected to the waste management compartment water dump system by a flex hose, when a dump was required.

Each liquid dump system has a replaceable heated dump probe assembly with a separate control for each heater element. The two heating elements in each dump probe assembly are redundant. Only one element at a time can be operated. The heater elements are required to keep the probe free of ice formation and resultant blockage. The converging nozzle keeps the pressure inside the dump line above the triple point of water during a dump (Figures VG16-2 and VG16-3). The probe heaters are required only prior to and during liquid dumping operations.

b. Mission performance. The liquid urine dump system was used only to pull a vacuum on the urine bags prior to their use during the first and second manned phases. However, because of the lack of urine collection bags, and the problem of excessive force required to dispose of full urine bags through the trash airlock reported by crewmen, the liquid urine dump system was used to also dispose of liquid urine approximatly 17 times during the third manned phase. This was in addition to its design usage of evacuating the urine bags prior to their use.

The wardroom and waste management compartment water dump systems were each used once during each activation and once during each deactivation.

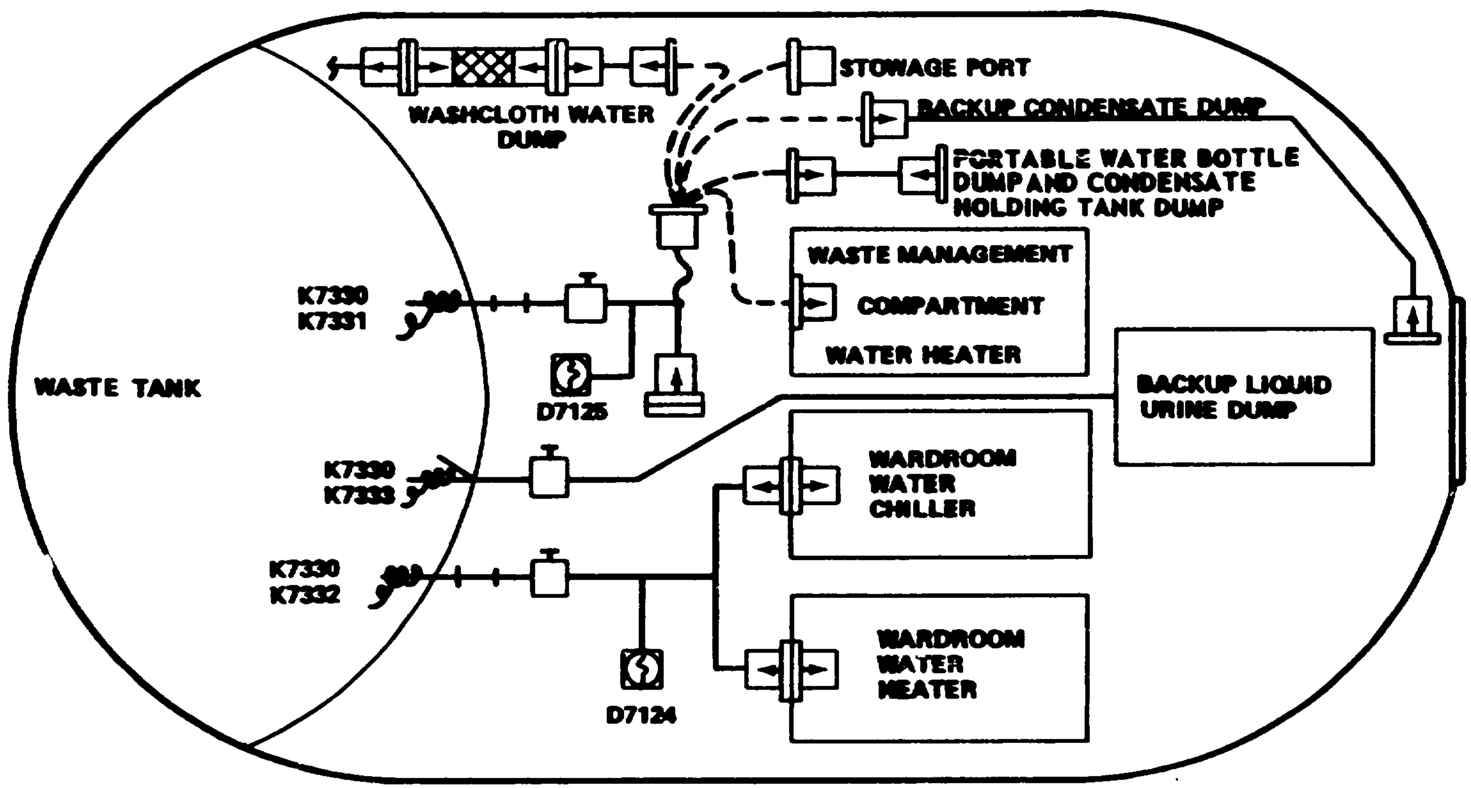


Figure VG16-1. Liquid Dump Systems

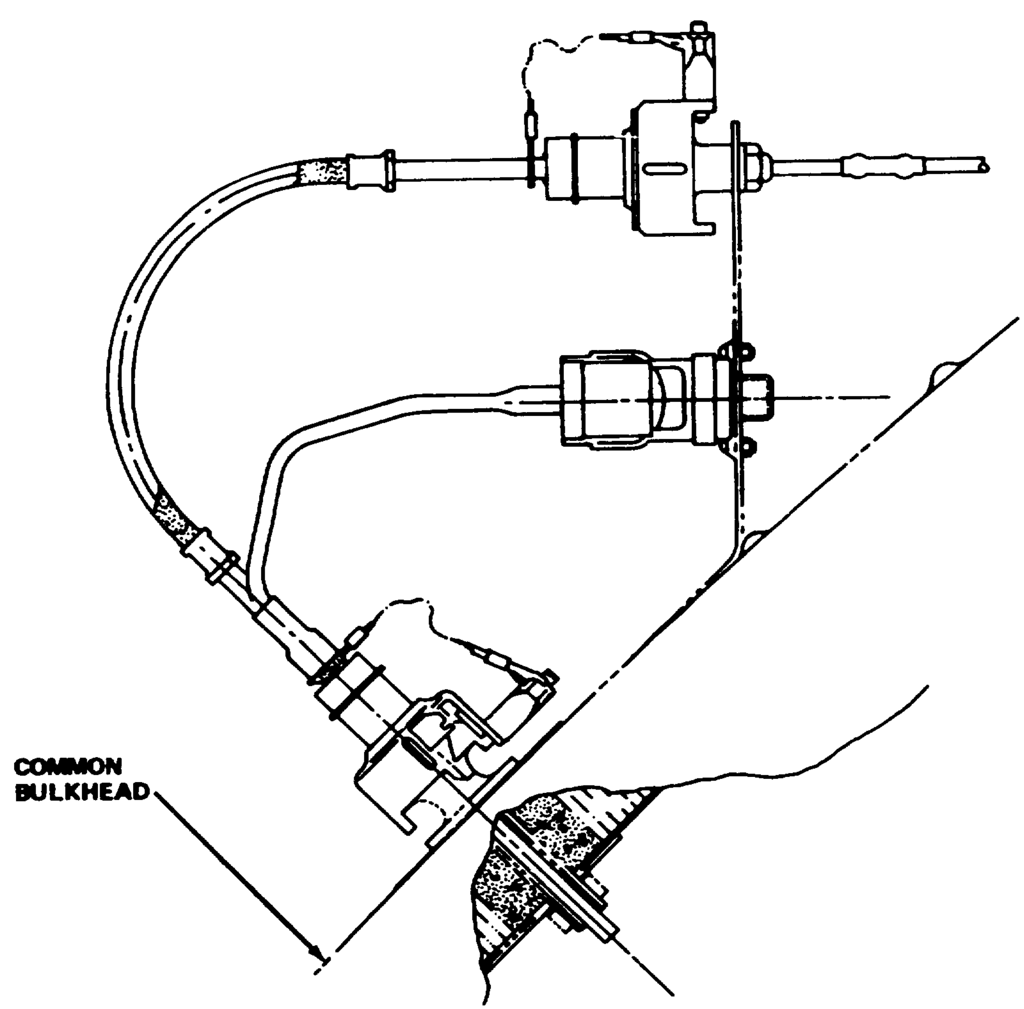


Figure VG 16-2. Liquid Dump Heater Probe Installation

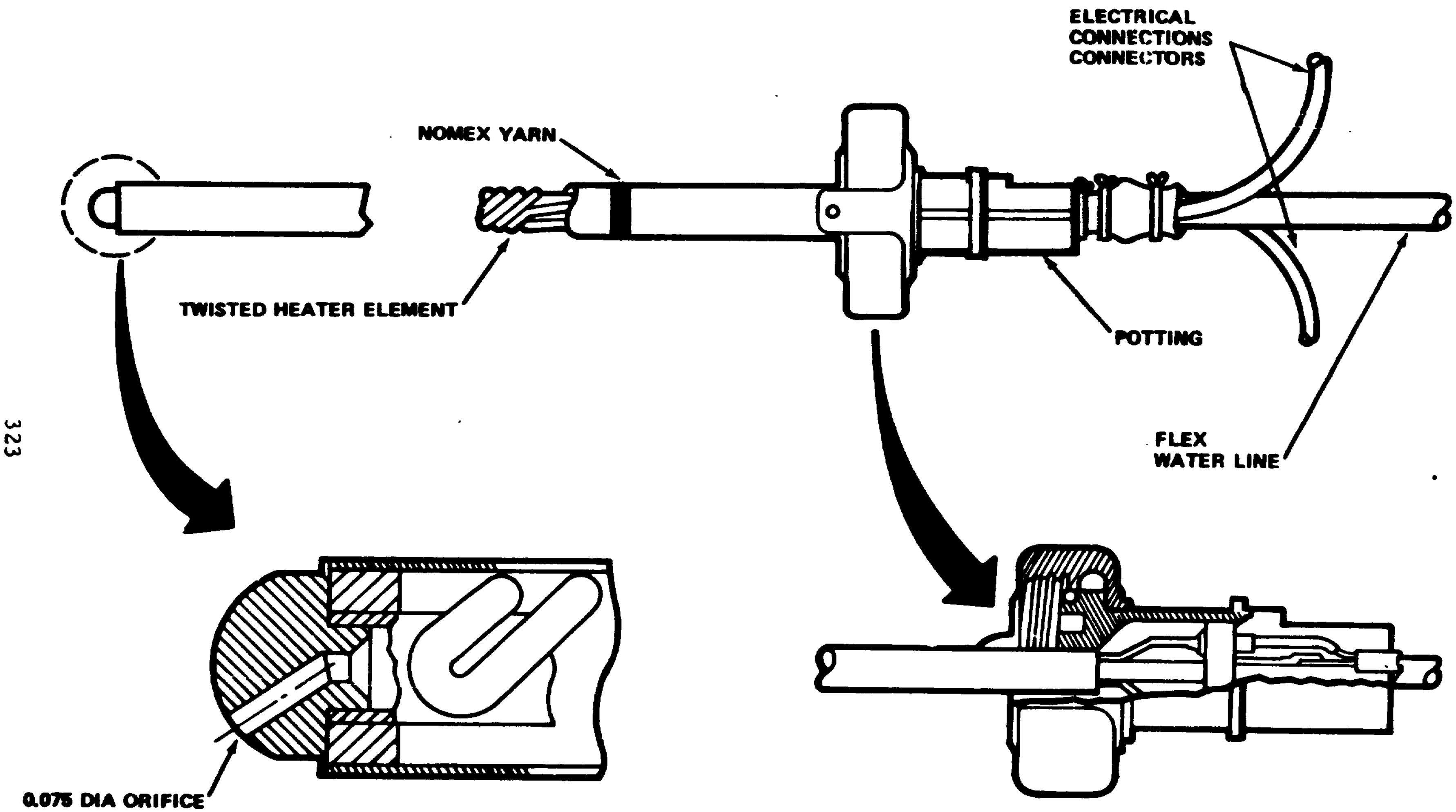


Figure VG 16-3. Liquid Dump Probe

Also, the waste management compartment water dump system was utilized approximately once every 3 days to dump waste wash water that had been collected in the wash cloth squeezer bag.

During the second manned phase, the waste management compartment water dump system was used approximately 35 times for workshop condensate holding tank dumps because of a leakage problem in the condensate system.

In addition to the above usage, water dump systems were used during the second and third manned phases for cabin atmosphere pressure management in support of experiments M509 and T020 (astronaut maneuvering units). By installing a purge fitting in the water dump line, cabin atmosphere was vented slowly to a desired pressure.

c. Anomalies. On day 23 of the second manned phase, the waste management compartment water dump system failed to completely dump the workshop condensate holding tank per crew procedure HK60T; maximum condensate system delta pressure was 3.5 psid. This is indicative of a line or probe blockage since the normal level following a complete dump is 4.0 to 4.5 psid. However, the washcloth squeezer bag water dump (which uses the same dump system) was successful. At this time, the dump probe was being operated on the Bus 2 heater. The following morning the commander turned on the dump probe Bus 1 heater for about 30 min, but the dump line remained clogged. The water dump valve was also cycled 10 times to no avail. A brief 35-psi hot water dump utilizing the waste management compartment water heater cleared the dump line; the dump probe Bus 2 heater was in use at this time. A condensate holding tank dump operation was then initiated, but the maximum delta pressure obtained was only 3.7 psid; a crew check of the dump line using the condensate pressure fitting revealed that the line was again clogged. Another 35-psi hot water dump was attempted, during which the dump probe Bus 2 heater was turned off and the Bus 1 heater was turned on. About 30 min after this operation, the condensate pressure fitting showed the dump line was clogged. Because of the elapsed time between this hot water dump and installation of the condensate pressure fitting, a third hot water dump was initiated still utilizing the dump probe Bus 1 heater. Immediately after this dump, the condensate pressure fitting was installed so as to purge the dump line. This cleared the dump line, after which a slow but successful holding tank dump was performed. On day 35 of the second manned phase, another successful holding tank dump was performed. This dump was very slow, indicating that the probe was partially blocked. The following day, the holding tank dump per HK60B (which pressurizes the gas side of the holding tank bellows) was unsuccessful. At this time the decision was made to replace the waste management compartment water dump probe assembly. Following replacement, the commander reported that the old probe had ice in the tip. The holding tank was then dumped per HK60B without incident, indicating proper operation of the new probe assembly. On mission day 42 of the second manned phase, the crew performed an electrical continuity test of the removed dump probe; all readings were normal. The crew also checked the probe for contamination by inserting and withdrawing a length of safety wire, and by blowing through the probe; no contamination was found and the probe was returned to stowage for use as a future spare.

Five potential mechanisms of ice blockage formation were identified based on previous ground test experience: internal contamination; ice buildup on probe tip; ice building back along normal flow path from waste tank wall; and ice buildup aggravated by high pressure dump (Figure VG16-4). Exact cause of probe freezup is not known; however, all subsequent water dumps through the new probe were successful. On day 51 of the third manned phase, the crew reported a problem in the liquid urine dump system. While attempting to evacuate a urine bag following a liquid urine dump, no flow was observed through the system. The dump heater probe was operating as evidenced by the indicator lights. The crew left the heaters on for several hours and later that day flow through the system resumed. No further problems were encountered during the remainder of the mission. Because of the possibility of blockage by a small piece of undissolved boric acid tablet, tests were conducted. The testing showed that although temporary flow stoppages lasting from 5 to 9 sec were observed during the two 6 test dumps, these stoppages had been seen during development and acceptance testing and were not considered unusual or detrimental to system performance. The most likely cause of the blockage was a buildup of ice on the probe that required a longer than normal heater-on time. This assumption is supported by the fact that no further problems were reported. The crew also stated that they left the liquid urine dump heater probe on for the rest of the mission to ensure that no additional problems would occur.

d. Recommendations. The probe concept is recommended for future systems of this type. Although some problems were experienced indicating need for further improvements, general success of this concept was the result of emphasis placed on system development and testing under conditions that closely simulated the flight environment.

17. Vent System

a. Habitability area

(1) General requirements.

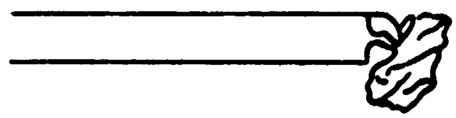
(a) The tank shall have the capability of being vented nonpropulsively to space environment.

(b) Venting shall be terminated by control from the workshop switch selector.

(c) Solenoid valves shall be installed in the habitation area vent system to provide the capability to vent the habitation area by a signal from the airlock on command from the ground.

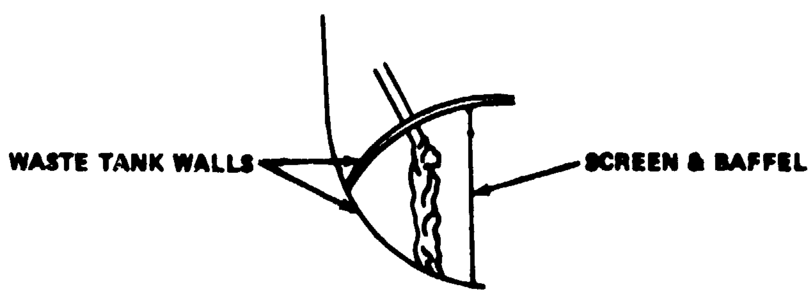
1. INTERNAL CONTAMINATION

2. ICE BUILDUP ON PROBE TIP



3. ICE BUILDING BACK FROM CLOSE OBJECT TO PROBE TIP

4. ICE BUILDING BACK ALONG NORMAL FLOW PATH FROM WASTE TANK WALL



5. ICE BUILDUP AGGRAVATED BY HIGH PRESS DUMP

SAME AS 4.

Figure VG 16-4. POSSIBLE MECHANISMS OF FLOW REDUCTION OR STOPPAGE

(d) A sealing device shall be provided for the LH_2 vent tank outlets to minimize leakage.

(2) Mission performance. The workshop pneumatic vent valves operated normally, as evidenced by the nominal blowdown of the habitation area following launch. The pneumatic vent valves were inoperative during the rest of the Skylab mission.

Because of the loss of the meteoroid shield during launch phase, and the subsequent high temperatures experienced in the workshop, the cluster was purged of any potential toxic outgassing by alternate pressurizing and venting, five times, prior to crew arrival. Solenoid vent valve operation was normal throughout these operations, except at the conclusion of the final vent, when the talkback for valves #1 and #3 did not indicate closed (146:02:02). Since valves #2 and #4 did indicate closed the only problem was apparent loss of redundancy. The probable cause was the talkback switch on the valve. Subsequent operation of the valves on all three manned phases was normal with proper valve talkback.

Each post-occupation vent was accomplished without expenditure of thruster attitude control subsystem propellant, thus verifying that the vent system was nonpropulsive.

The pneumatic vent port sealing device was installed during the first crew visit immediately after entering the workshop with no reported problems. The solenoid vent port sealing device was not installed because of the low cluster leakage rate and the desire to simplify emergency egress procedures.

(3) Anomalies. Two anomalies were associated with the solenoid vent system prior to the first manned phase activation. During the final vent cycles of the potentially toxic cluster gases prior to the first manned phase, the vent rate was slower than predicted. In addition, at the end of the cycle, solenoid vent valves #1 and #3 failed to indicate closed after the close command was sent. Valves #2 and #4 did close properly and allowed the cluster to be pressurized normally. The probable cause of the slow vent rate was the presence of debris in the system. The first crew verified the existence of debris in the solenoid vent port inlet filter and subsequently cleaned the filter per an unscheduled maintenance procedure.

Troubleshooting the failure of valves #1 and #3 to indicate closed was delayed until mission day 18 of the first manned phase. At that time all four solenoid valves were commanded open, and the crew reported flow. Then valves #1 and #3 were commanded closed; and the valves responded normally; valves #2 and #4 were then closed. Exact cause of the anomaly is unknown, but the problem failure mode was particulate contamination. Subsequent operation of the solenoid vent valves was normal.

(4) Recommendations. The value of a quad redundant system was evident in the anomaly with the solenoid vent system. Future design should carefully examine the factors of reliability versus complexity involved to determine the extent of redundancy required.

The use of filters at the inlet to vents is required to protect systems from debris that cannot be detected and removed prior to the vent operation. Each Skylab crew reported some debris in the solenoid vent port inlet filter following venting.

b. Waste tank.

(1) General requirements.

(a) A nonpropulsive vent shall be provided for venting the LOX tank.

(b) The waste tank (LOX tank) shall be pressurized at launch with nitrogen gas to stabilize the structural shell during launch and flight. The internal pressure shall be maintained within a 22 to 26 psia operating range during launch and ascent. The tank shall have the capability of being vented to space environment.

(c) The waste tank shall be continuously vented in orbit. The waste tank vent shall remain open after initial venting

(d) The LOX tank shall be used as a waste tank for disposing of all wet and dry materials and refuse collected in habitable areas internal to orbital assembly. Provisions shall be made for providing waste tank screen filters for limiting of ice particles and contaminants, in the order of 10 microns, from being vented through the waste tank vents.

(2) Mission performance. Waste tank vent system operation was normal throughout all three manned phase missions. Except for pressure spikes associated with water dumps during activation and deactivation of SL-2 and during the 35 psi water dumps performed in troubleshooting the waste management compartment water dump probe, waste tank pressure was maintained well below the triple point of water (0.29 psia). The high waste tank pressure was apparently due to a higher than expected rate of sublimation of the ice formed during the liquid dumps, but no adverse effects were observed because of this high pressure.

All flight data indicated that the waste tank system, consisting of filter screens and a nonpropulsive vent, was completely effective in providing for disposal of liquid and solid waste materials outside the habitation area without interfering with optical experiments or imposing a load on the attitude control system. Discussions with MSFC contamination MSG personnel have indicated that no traces of waste tank effluents were uncovered in any of their experiments. Evaluation of the attitude and pointing control system data shows no measurable unbalanced venting, even during the largest liquid dumps into the waste tank.

No anomalies or problems were experienced with the waste tank vents or screens.

18. Vacuum Support System.

a. General requirements. General requirements for the vacuum support system are:

(1) Vacuum outlet system shall provide a valve to control the vacuum port for the waste processor and refrigeration pump enclosure.

(2) Provide a pressure of 2.82 psia at a flow rate of 0.111 lb/min at the M092 lower body negative pressure device (LBNPD) interface.

(3) Provide a pressure of 10^{-5} torr at a volume flow rate of 0.4 liter/sec at the M171 metabolic analyzer vacuum line interface.

b. Mission performance. The waste processor vacuum line shutoff valve was cycled once during each manned phase (open on activation and closed on deactivation). The refrigeration pump enclosure hand valve was left open throughout the three manned phases. The experiment vacuum support system provided an adequate vacuum for operation of the LBNPD and the metabolic analyzer during the three manned phases. The vacuum lines saw almost daily usage from experiments M092 and M171. The LBNPD vacuum vent hand valve was cycled once during each manned phase (open on activation and closed on deactivation).

c. Anomalies. There were no reported hardware problems associated with the vacuum support system.

Because of the loss of the meteoroid shield during the first unmanned launch phase, overboard venting through the LBNPD vacuum line tended to be propulsive, causing momentum buildup that required careful scheduling of LBNPD operations. In order to simplify the scheduling operations, two adapters were made and flown on the third manned phase. These adapters allowed the crew to connect the LBNPD vacuum line to the waste tank by using the unused, onboard hoses (Figure VG18-1). This approach allowed LBNPD gas to be vented, nonpropulsively, through the waste tank vents. No problems were reported by the crew in performing these hardware modifications or in the use of the equipment.

19. Pneumatic System.

a. General requirements. The pneumatic system provides actuating pressure, on command, to the systems listed below within an appropriate period after lift-off of the first unmanned phase. The pneumatic system loading pressure band is 390 to 5 psia. The minimum actuation pressures for the various functions are:

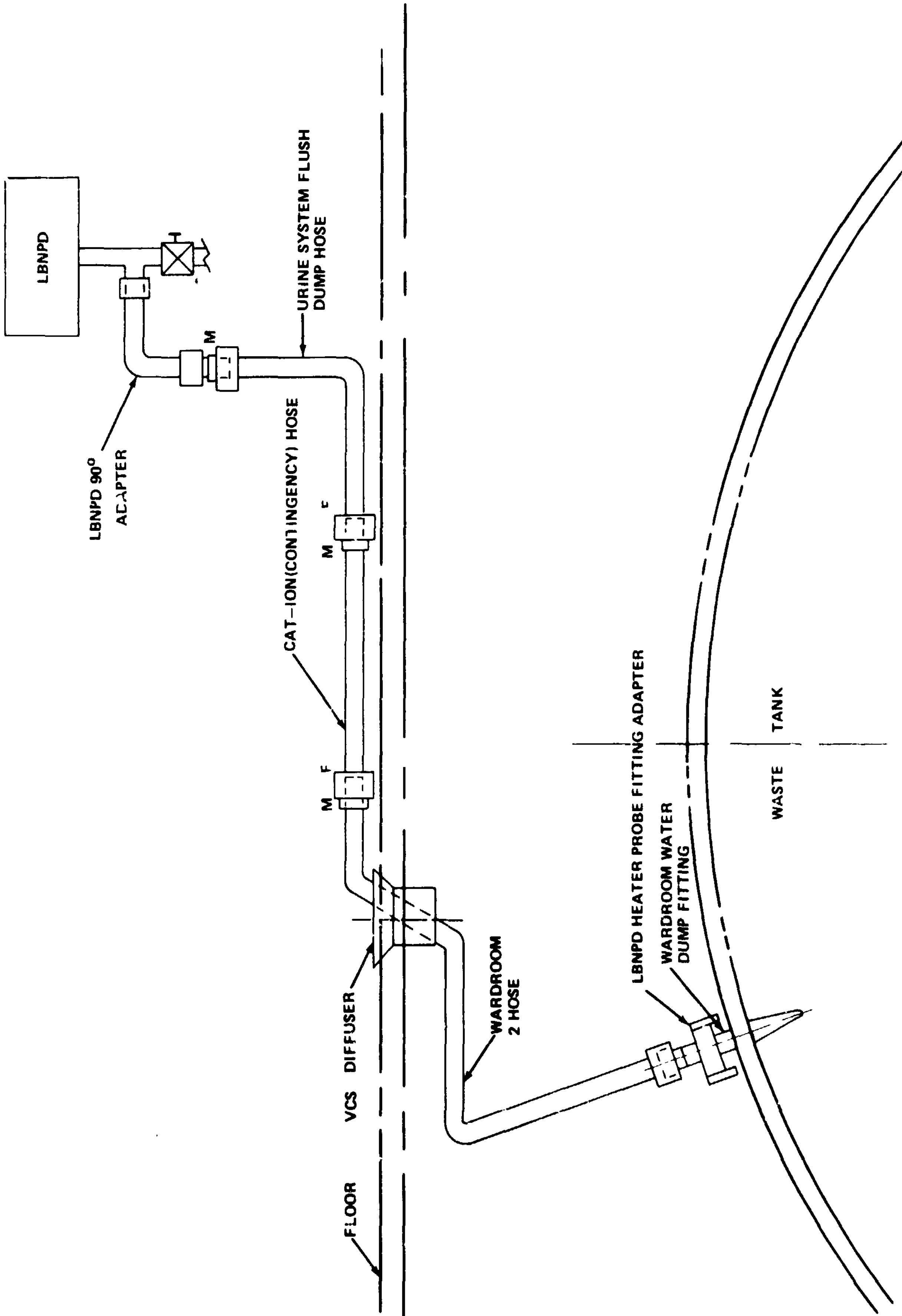


Figure VG 18-1. LBNPD Vacuum Source Modification

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Waste tank vent caps	111 psia
Refrigeration shield jettison	100 psia
Habitability area vent valves at 26 psia	230-psia

b. Mission performance. Post-Skylab mission flight data indicate that the pneumatic control system met all design requirements and that all events requiring pneumatic pressure occurred on time and within operational limits. The pneumatic sphere was pressurized to 441 psia just prior to the first launch. After completing all pneumatic functions but prior to the end of instrument unit lifetime, the pneumatic sphere was vented to 38 psia to safe the system. Prior to venting the pneumatic sphere (control system), pressure was 437 psia, indicating that system usage and leakage were negligible.

APPROVAL TMX-64824

MSFC SKYLAB STRUCTURES AND MECHANICAL SYSTEMS MISSION EVALUATION

By

Structures and Mechanical Mission Support Group

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report has been determined to be unclassified in its entirety.

This document has also been reviewed and approved for technical accuracy.

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for H. R. Coldwater
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