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ABBREVIATIONS AND ACRONYMS

Å	Angstroms
AC	Alternating Current
ACE	Acceptance Checkout Equipment
ACQSS	Acquisition Sun Sensor
ACS	Attitude Control System
ADP	Acceptance Data Package
ALSA	Astronaut Life Support Assembly
AM	Airlock Module
APCS	Attitude & Pointing Control System
ARC	Ames Research Center
ASAP	Auxiliary Storage and Playback
ATM	Apollo Telescope Mount
ATMDC	Apollo Telescope Mount Digital Computer
BTU	British Thermal Units
CBRM	Charger Battery Regulator Module
CCB	Change Control Board
CCOH	Combined Contaminants, Oxygen, Humidity
CCS	Command Communication System
C&D	Control and Display
CEI	Contract End Item
CFE	Contractor Furnished Equipment
CG	Center of Gravity
C_L	Centerline
Cluster	SWS plus CSM (used synonymously with "Orbital Assembly")
CM	Command Module
CMG	Control Moment Gyro
CMGS/TACS	Control Moment Gyros Subsystem/Thruster Attitude Control Subsystem
C/O	Checkout
COAS	Crew Optical Alignment Sight
CO_2	Carbon Dioxide
COFW	Certificate of Flight Worthiness
COQ	Certificate of Qualification
cps	cycles per second
CRS	Cluster Requirements Specification
CSM	Command Service Module
C&W	Caution and Warning
DA	Deployment Assembly
db	Decibel
dc	Direct Current
DCS	Digital Command System
DCSU	Digital Computer Switching Unit
DDA	Drawing Departure Authorization
DDAS	Digital Data Address System
deg.	Degree
DTCS	Digital Test Command System
DTMS	Digital Test Measuring System

ECP	Engineering Change Proposal
ECS	Environmental Control System
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EPCS	Experiment Pointing Control Subsystem
EPS	Electrical Power System
ERD	Experiment Requirement Document
ESE	Electrical Support Equipment
ESS	Experiment Support System
ETR	Eastern Test Range
EVA	Extravehicular Activity
°F	Degrees Farenheit
FAS	Fixed Airlock Shroud
fc	foot candles
FM	Frequency Modulation
fps	feet per second
FSS	Fine Sun Sensor
ft.	Feet
g	Acceleration due to Earth's Gravity
GFE	Government Furnished Equipment
Grms	G Level, root mean square
GSE	Ground Support Equipment
H_2O	Water
He	Helium
HSS	Habitability Support System
Hz	Hertz
ICD	Interface Control Document
IOP	In Orbit Plane
IU	Instrumentation Unit
IU/TACS	Instrument Unit/Thruster Attitude Control Subsystem
IVA	Intra-Vehicular Activity
JSC	Johnson Spacecraft Center
KHz	Kilohertz
KSC	Kennedy Spaceflight Center
LCC	Launch Control Center
LCG	Liquid Cooled Garment
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LRC	Langley Research Center
LV	Launch Vehicle
LVDC	Launch Vehicle Digital Computer
MDA	Multiple Docking Adapter
MGSE	Maintenance Ground Support Equipment
MHz	Megahertz
MRD	Mission Requirements Document
MS	Margin of Safety
m/sec.	Millisecond
MSFC	Marshall Space Flight Center
MSFN	Manned Space Flight Network
MSOB	Manned Spacecraft Operations Building

N_2	Nitrogen
NASA	National Aeronautics and Space Administration
NHB	NASA Handbook
NiCd	Nickel Cadmium
NM	Nautical Miles
O_2	Oxygen
OA	Orbital Assembly (SWS and CSM - Used synonymously with "Cluster")
OWS	Orbital Workshop
ΔP	Differential Pressure
PCM	Pulse Code Modulation
PCS	Pointing Control System
PMC	Post Manufacturing Checkout
POD	Planning Operational Dose
psi	pounds per square inch
psia	pounds per square inch absolute
psid	pounds per square inch differential
Q	Heat
RCS	Reaction Control System
RF	Radio Frequency
S-IB	First Stage of Saturn I-B Launch Vehicle
S-II	Saturn II
SAL	Scientific Air Lock
SAS	Solar Array System
SCN	Specification Change Notice
SL	Skylab Program
SM	Service Module
SWS	Saturn Workshop (PS/MDA/ATM/AM/OWS/IU/ATM Deployment Assembly)
ΔT	Differential Temperature
TACS	Thruster Attitude Control System
TCRD	Test and Checkout Requirements Document
TCSCD	Test and Checkout Specification and Criteria Document
UV	Ultra Violet
VAB	Vehicle Assembly Building (HI-Bay)
Vdc	Volts direct current
VHF	Very High Frequency
WMS	Waste Management System
WSS	Water Subsystem
Z-LV(E)	Z Axis in Local Vertical (Earth Resources Attitude Mode)
Z-LV(R)	Z axis in Local Vertical (Rendezvous Attitude Mode)

SECTION 1 - INTRODUCTION

1.1 PURPOSE AND SCOPE

This document reviews the technical aspects of the Skylab-Orbital Workshop, including the original concepts, goals, design philosophy, hardware and testing. Discussed is the evolution from a "Wet Workshop " (one flown as the fuel tank of a rocket into orbit, drained, purged and then converted to a habitation area by the crew) to a "Dry Workshop" (one launched completely outfitted in orbital configuration).

The final flight configuration, overall test program, and mission performance are discussed in detail.

Each of the major systems will be identified and described. The design requirements and systems description will be reviewed. Areas such as contamination, flammability, toxicity, safety and reliability are evaluated and their tradeoffs discussed.

The testing program is reviewed. The major problems uncovered during test and their solutions are detailed.

Mission results and performance during launch and flight are provided. Special tests and analysis to support the mission are included and reviewed. Long term orbital effect on systems and hardware are evaluated. Of special interest are the conclusions and recommendations made for future programs.

Objectives and methodology of the reliability program are reviewed in detail. Items such as failure mode and effect analysis, critical items list, reliability model, trade studies and design reviews are discussed in detail. Supplier evaluation along with nonconformance reporting, analysis, corrective action control, and alert investigations are reviewed. In addition a section on mission reliability is included.

The crew safety program is reviewed step by step from the design phase through the testing phase and mission performance. The safety studies performed and their affect on requirements are reviewed. The controls used during testing are also explained.

The overall testing program is discussed in detail. Items such as the planning of test flow, compliance with test requirements and test procedures are reviewed. Component testing, both supplier and in-plant, and component qualification and development are described. Structural static and acoustical vibration testing along with the acceptance testing of flight systems and the performance testing of various support systems is reviewed. Integrated vehicle testing and KSC testing with the other vehicles, the performance of interfaces and overall compliance to requirements is discussed

Engineering program management is reviewed with descriptions of the planned and actual controls used and their effect on program performance.

The implementation and results of configuration control on the vehicle and GSE through the program including the management of OWS interfaces is described.

A section is included on new technology describing new methods and procedures as applicable to aerospace and non-aerospace industries.

A final section on conclusions and recommendations discusses the systems and system elements which performed notably above and below nominal, reviewing the contributing factors and making recommendations for future system designs. Program planning, testing, and mission support are also discussed and recommendations for future programs are made.

1.2 SUMMARY

1.2.1 Design Goals - The primary objective of the Skylab Program was to demonstrate that man could survive in space for a long period of time and do useful work. To this end, the 100 ton (90,718 kilograms) Skylab experimental space station was developed through many conceptual changes to provide a laboratory for extended manned spaceflight in which man could perform inflight experiments to:

- o Obtain biomedical data for evaluating the effects of zero-g missions of 28 to 56 days on crew members.
- o Determine the feasibility and advisability of manned zero-g spaceflights for durations greater than 56 days.
- o Obtain solar and stellar astronomy data to continue and extend studies beyond the limits of terrestrial observations.
- o Obtain data on the earth's surface to evaluate its resources.
- o Obtain data for the development of operational procedures for extended manned orbital operation.
- o Obtain engineering and technological data for development of advanced space vehicles and equipment.

The Skylab which came into being is shown in Figure 1.2.1-1. Figure 1.2.1-2 identifies the modules and their function as follows:

- o Command and Service Module (CSM) - Provides the vehicle that transports the crew to and from Skylab 1 (SL-1). The CSM is docked/undocked with SL-1 in essentially the same manner as it was accomplished with the lunar landing vehicle in the Apollo Program.
- o Multiple Docking Adapter (MDA) - Provides SL-1 docking capability for the CSM and control and display panels for Solar and Earth Resources Experiments (EREP).
- o Apollo Telescope Mount (ATM) - Provides a solar observatory for the study of sun activity free from distortion caused by the earth's atmosphere and approximately one-half of the Skylab electrical power.
- o Airlock Module (AM) - Provides the "nerve center" for the orbiting assembly or cluster; i.e., control and distribution for electrical power and oxygen-nitrogen crew atmosphere, equipment for voice, real time or taped, and digital command communications between Skylab and ground stations. Further, it provides external access for ATM film servicing and other extra vehicular activity.

ORBITAL WORKSHOP
SKYLAB

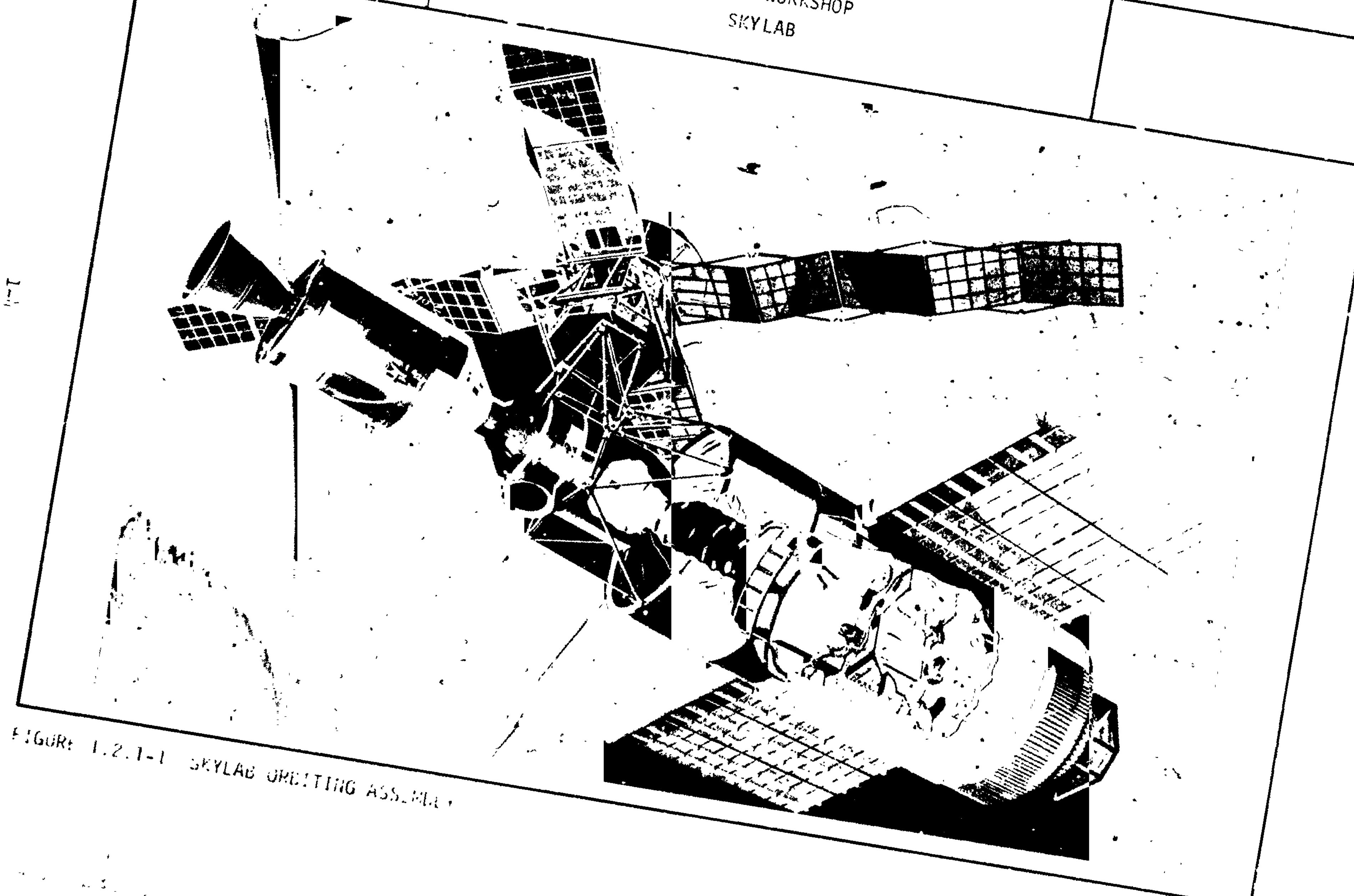
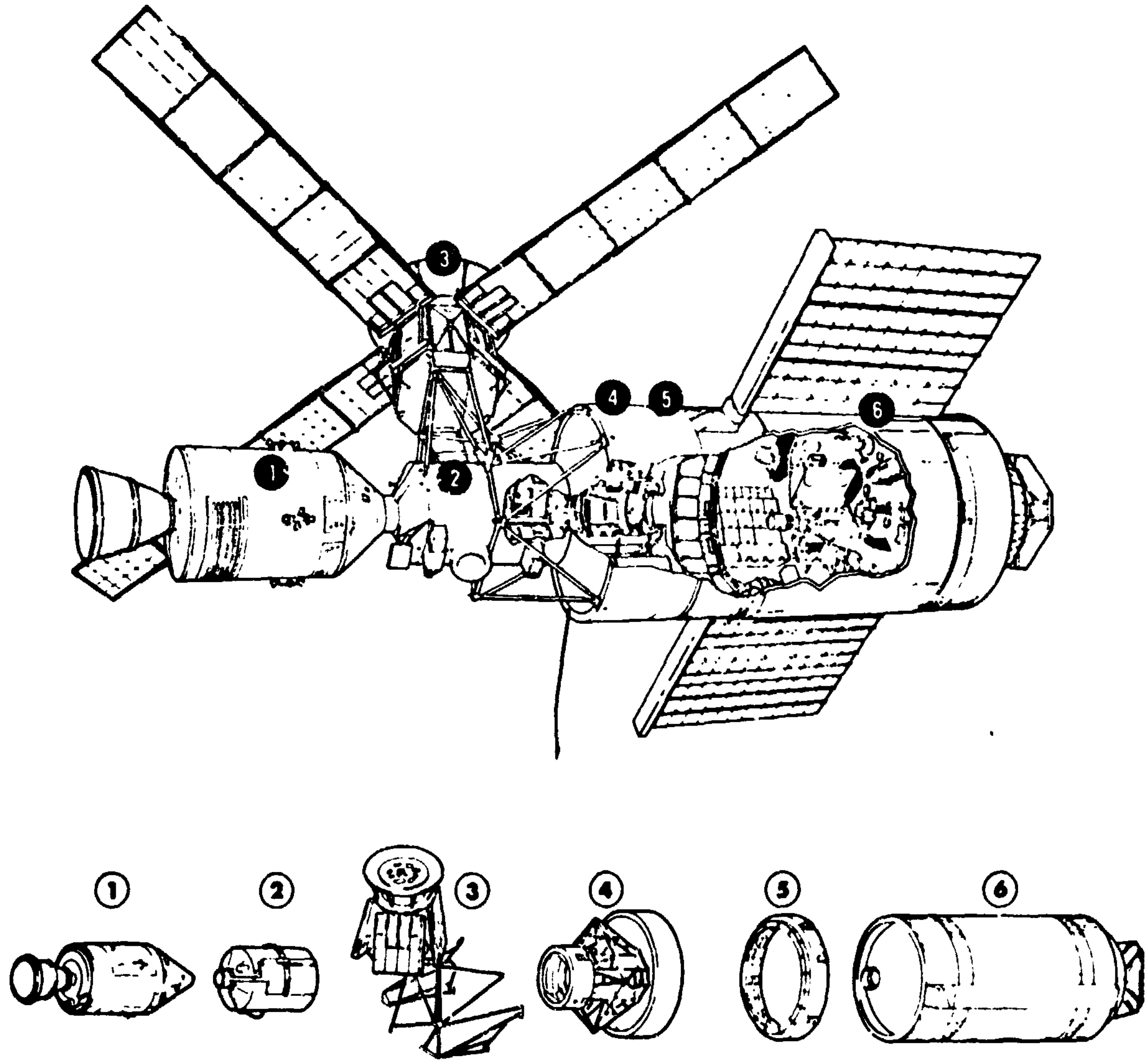


FIGURE 1.2.1-1 SKYLAB ORBITING ASSEMBLY



Element	Command and Service Module	Multiple Docking Adapter	Apollo Telescope Mount	Airlock Module/Fixed Airlock Shroud	Instrument Unit	Orbital Workshop
Function	Crew Ascent & Descent	Docking Interface ATM/EREP Controls & Displays	Solar Observation	Power Control & Distribution Environmental Control Utility Center Data System Extravehicular Activity Port	Launch Vehicle Control	Primary Living & Working Area Experiment Laboratory Stowage

FIGURE 1.2.1-2 SKYLAB - FUNCTION OF MODULES

- o Instrument Unit (IU) - Provides SL-1 launch vehicle control.
- o Orbital Workshop (OWS) - Provides the primary living and working accommodations for the crew, experiment laboratory accommodations, stowage for supplies and approximately one-half of the Skylab electrical power.

In accordance with the Skylab design goals, MDAC-W converted the S-IVB/IB Stage 212 into the OWS. A profile of the module is shown in Figure 1.2.1-3. First, all Apollo S-IVB propulsion and related systems were deleted to provide a structural house. The house, in turn, was then furnished to meet the requirements imposed on the module. The converted hydrogen tank became a 10,000 cubic foot (283 meters^3) habitation area embodying a crews quarters for sleeping, food, water and waste management systems, and areas for recreation and experimentation. In addition to the crews quarters, an area forward of it provides extensive space for additional experimentation and storage of supplies. The intent was to outfit the OWS in its Skylab role to accommodate three crewmen for missions of 28, 56 and 56 days each, without resupply during the 8 month mission depicted by the mission design profile shown in Figure 1.2.1-4. Further, the S-IVB oxygen tank, a basement, was converted to a waste tank for the disposition of cluster trash as it accumulated.

Externally on the OWS, the following subsystems were installed.

- o Meteoroid Shield - To increase the probability of no pressure loss equal to or greater than 0.995 from the habitation area.
- o Solar Array - To provide electrical power to the AM power distribution and control system.
- o Thruster Attitude Control - To provide primary attitude control through the ATM control moment gyroscopes (CMG) spin-up and backup/supplemental attitude control for CMG desaturation, for maneuvers and docking transients.

Miscellaneous equipment for subsystems installed in both the forward and aft skirts of OWS will be discussed in Section 2.2.

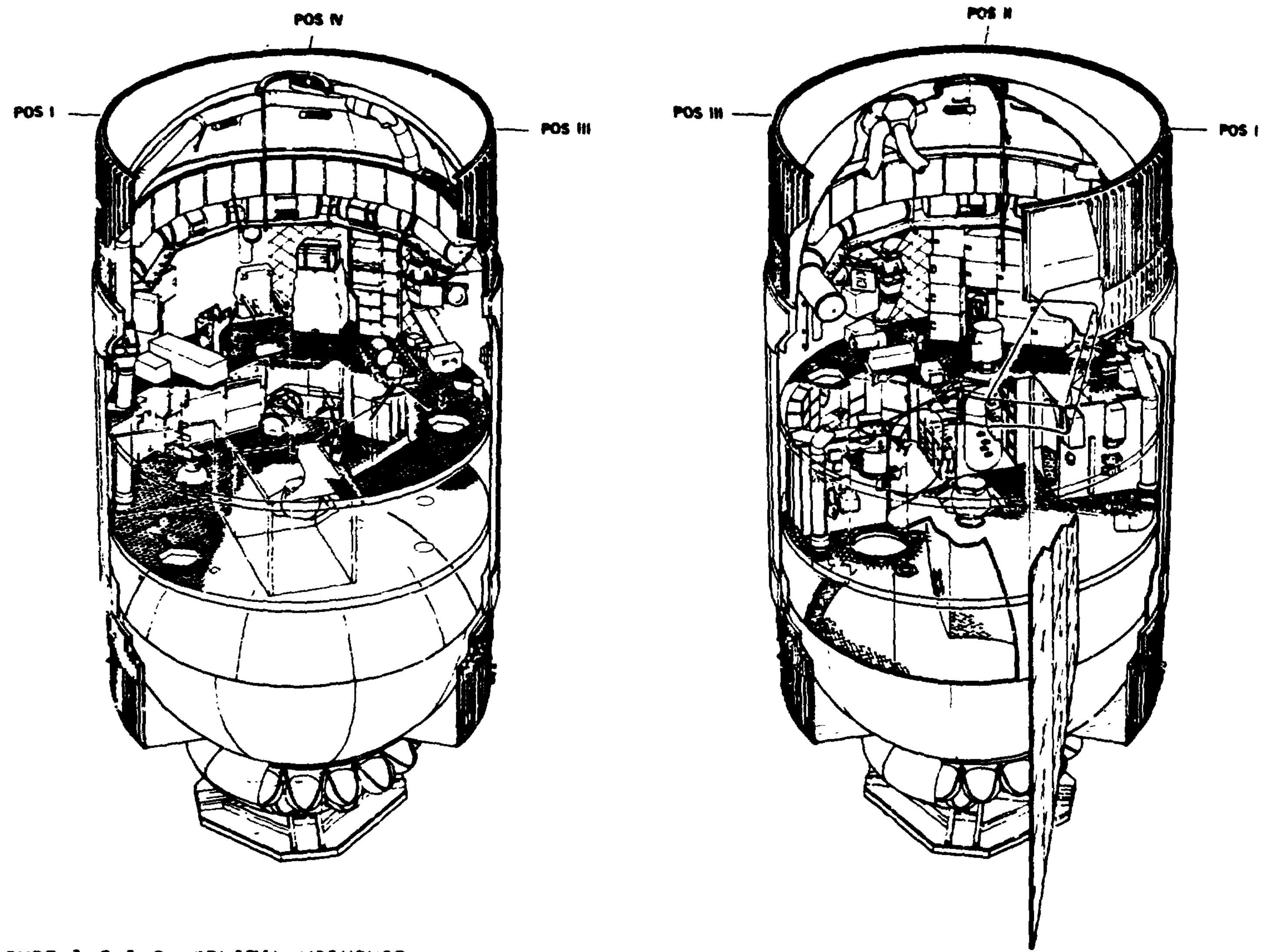


FIGURE 1.2.1-3 ORBITAL WORKSHOP

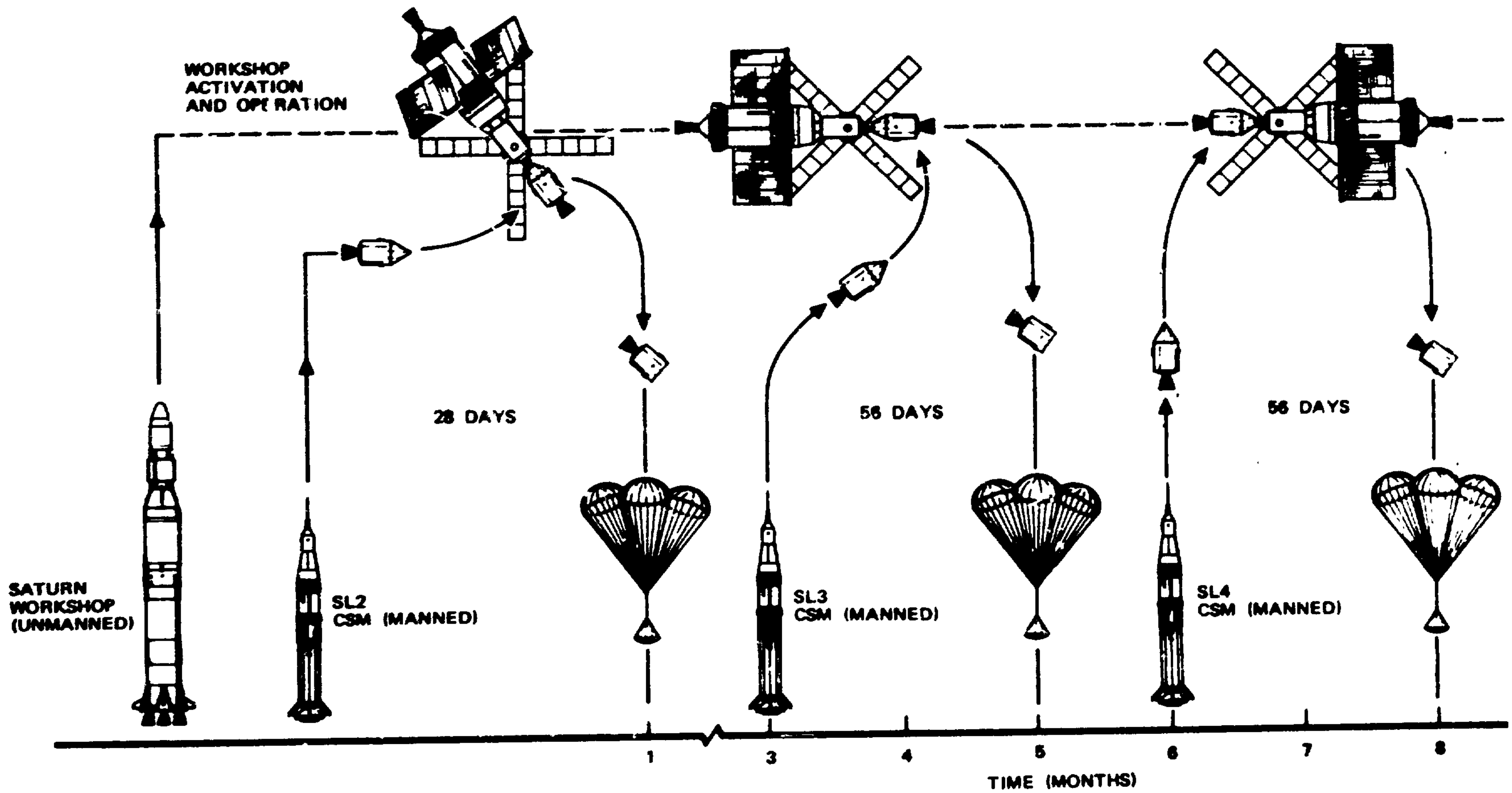


FIGURE 1.2.1-4 MISSION DESIGN PROFILE

1.2.2 Mission Results

1.2.2.1 SL-1 Mission - SL-1 was launched from Launch Complex 39A at KSC on 14 May 1973 (134:17:30:00.589 Greenwich mean time (GMT)). The SL-1 launch phase was nominal until approximately 63 seconds ground elapsed time (GET), at which time the OWS meteoroid shield external temperatures went off scale followed by the loss of the three meteoroid shield secured indications. The noted instrumentation was the first indication of the loss of the OWS meteoroid shield. During the same time period, SAS Wing 2 secured indication was lost, indicating that SAS Wing 2 was no longer secured to the OWS. SAS Wing 2 was subsequently separated from the OWS, at approximately 593 seconds, due to retrorocket plume impingement forces.

SL-1 was inserted into a 433.8 by 431.5 kilometer (KM) orbit. Habitation area venting and waste tank venting was nominal. Jettison of the radiator protective shield occurred at 17:39:57.14 GMT and was nominal. SAS beam and wing deployment commands were issued at 18:11:05.73 seconds GMT and 18:22:05.00 seconds GMT, respectively; however deployment was not nominal. SAS Wing 2 was lost at approximately 593 seconds, as previously described, and SAS Wing 1 was prevented from deploying by the remnants of the OWS meteoroid shield. In an attempt to deploy SAS Wing 1, the SAS beam fairing and SAS wing section backup commands (Airlock Module Digital Command System) were transmitted, but deployment was unsuccessful. SAS Wing 1 was finally deployed by the SL-1/SL-2 crew during extravehicular activities (EVA) on day of year (DOY) 158.

1.2.2.2 SL-2 Mission - SL-2 was launched from Launch Complex 39B at KSC on 25 May 1973 (145:13:00:00.50 GMT) which was ten days later than originally planned. The ten-day launch slip was required to assess the Skylab thermal and electrical environment due to the loss of the meteoroid shield and SAS Wing 2 and to develop hardware and workarounds required to provide a habitable environment for the SL-1/SL-2 mission. Prior to Skylab habitation, the SL-2 crew performed a standup EVA (SEVA) at 23:52:15 GMT in an unsuccessful attempt to deploy SAS Wing 1.

Following the SEVA, SL-2 hard dock with the cluster was confirmed at 03:52:00 GMT after several unsuccessful attempts. OWS activation of the OWS was modified to allow for the deployment of the JSC parasol (146:21:52:00:00 to 147:01:30:00 GMT) from the solar SAL at which time the OWS interior temperatures started to drop. As previously noted, SAS Wing 1 was deployed on DOY 158 during EVA and the wing sections reached full deployment after three revolutions. Skylab deactivation on DOY 173 was nominal. Command Module (CM) splashdown occurred on MD 29 (173:13:49:49 GMT).

Fifty-five experiments and 9 subsystem/operational detailed test objectives had been planned for the SL-1/SL-2 mission. Of these, data were obtained on 46 experiments and 9 subsystem/operational detailed test objectives. Those experiments cancelled or having low performance were generally those of low priority.

1.2.2.3 SL-3 Mission - SL-3 was launched from Launch Complex 39B at KSC on 28 July 1973 (209:11:10:50.30 GMT). The CSM docked to the cluster at 209:19:38:50 GMT and subsequently performed a normal SWS activation. On DOY 218 the crew went EVA to install ATM film, deploy S149 and deploy the twin-pole sun shield. A second EVA on DOY 236 was performed to install the rate gyro package and change out the ATM film. During third EVA on DOY 265 ATM film, S149 and S230 sample retrieval were accomplished. Cluster deactivation was nominal and was terminated with CM splashdown on 25 September 1973.

Forty-four experiments, 14 science demonstrations, 11 student investigations, and 8 subsystem/operational detailed test objectives had been planned for the mission. The science demonstrations were carried as candidates to be performed at crew option. All planned objectives were not completed, but data were obtained on all but two of the planned objectives. Only six of the original 14 science demonstrations were worked into the schedule by the crew. In addition, data were obtained on 12 experiments, two science demonstrations, and eight special tests which had not been planned, but which were requested of and approved by the Flight Management Team for performance during the mission. A student investigation was added also, but failed when a performance was attempted.

1.2.2.4 SL-4 Mission - SL-4 was launched from Complex 39B at KSC on 16 November 1973 (320:14:01:23.4 GMT). CSM docking to the cluster was successful after the third attempt and it was followed by a normal workshop activation. The significant OWS system changes made for the SL-4 mission were (1) The M092 experiment vent was vented to the waste tank for non-propulsive venting to conserve Thruster Attitude Control System (TACS) gas, and (2) additional consumables (towels, urine bags, food, etc.) were flown up for an extended mission of 84 days.

Four EVA's were conducted during the SL-4 mission on DOY's 326, 359, and 363 of 1973 and DOY 34 of 1974. SL-4 splashdown occurred on 8 February 1974 completing an 84 day mission.

For SL-4, 56 experiments (including hydrogen alpha telescope and earth visual observations), 26 science demonstrations, 13 student investigations, and 15 subsystem/operational detailed test objectives were planned. In addition, plans were made to observe the comet Kohoutek using hardware from 6 of the on-board corollary experiments and 6 Apollo Telescope Mount experiments. The science demonstrations were classified as candidates to be performed at the crew's option. All planned objectives were not completed, but data were obtained on all except 3. The crew was able to schedule only 11 of the planned 26 science demonstrations. In addition, data were obtained on 5 additional subsystem/operational detailed test objectives. These objectives were not planned before SL-4 launch, but were requested during the mission and approved by the Flight Management Team.

1.2.2.5 Mission Summary - The OWS meteoroid shield failure on SL-1 caused a ten day delay in the launch of SL-2. Further, corrective action taken due to the anomaly introduced previously unplanned crew tasks during SL-2 and SL-3. During the early days of SL-2, experiment operations were somewhat reduced because of high temperatures in the workshop and the limited electrical power available. However, after deployment of the

JSC parasol sun-shield, enabling temperature reduction and the deployment of SAS-1 providing approximately a 50 percent increase in power, experiment activities were resumed generally as planned.

Figures 1.2.1-4 and 1.2.2.5-1 show a comparison between the Mission Design Profile and the Mission Actual Profile. The overall mission life for Skylab planned for 8 months actually became 9 months. The crew occupancy periods planned for a total of 140 days ended up being 171 days, a significant 22 percent increase in crew flight time.

Table 1.2.2.5-1 shows a comparison between the calendar date, the day of year and mission day, for the overall Skylab mission life.

Table 1.2.2.5-2 tabulates experiment activity in the OWS for each mission and reflects the percentage completion.

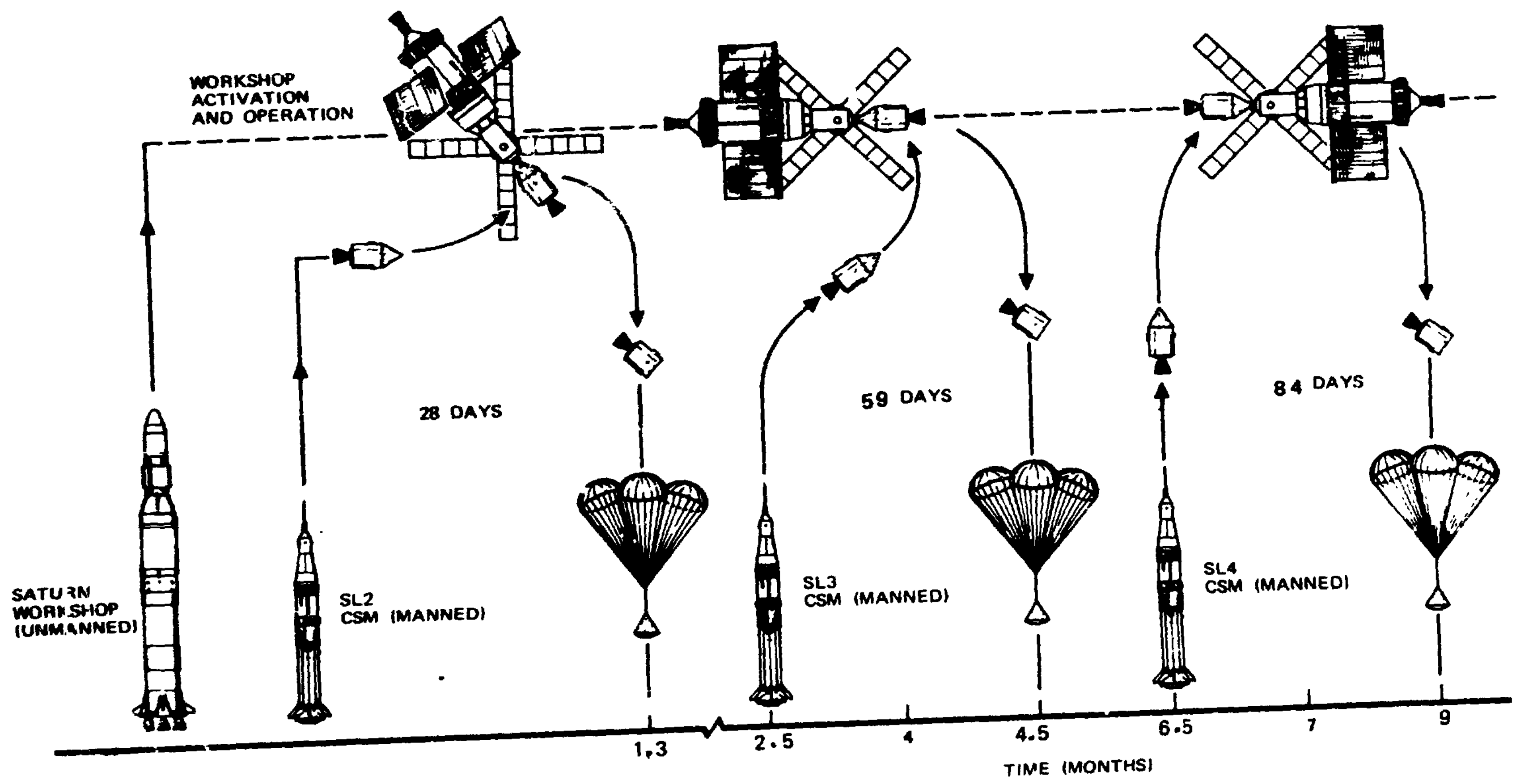


FIGURE 1.2.2.5-1 MISSION ACTUAL PROFILE

Table 1.2.2.5-1

SKYLAB MISSIONS - CALENDAR DATE/DAY OF YEAR/MISSION DAY

CALENDAR DATE	DAY OF YEAR	MISSION DAY	CALENDAR DATE	DAY OF YEAR	MISSION DAY	CALENDAR DATE	DAY OF YEAR	MISSION DAY
* 05-15-73	135	-10	07-01-73	182	--	08-17-73	229	21
05-16-73	136	-09	07-02-73	183	--	08-18-73	230	22
05-17-73	137	-08	07-03-73	184	--	08-19-73	231	23
05-18-73	138	-07	07-04-73	185	--	08-20-73	232	24
05-19-73	139	-06	07-05-73	186	--	08-21-73	233	25
05-20-73	140	-05	07-06-73	187	--	08-22-73	234	26
05-21-73	141	-04	07-07-73	188	--	08-23-73	235	27
05-22-73	142	-03	07-08-73	189	--	08-24-73	236	28
05-23-73	143	-02	07-09-73	190	--	08-25-73	237	29
05-24-73	144	-01	07-10-73	191	--	08-26-73	238	30
		SL-2	07-11-73	192	--	08-27-73	239	31
05-25-73	145	01	07-12-73	193	--	08-28-73	240	32
05-26-73	146	02	07-13-73	194	--	08-29-73	241	33
05-27-73	147	03	07-14-73	195	--	08-30-73	242	34
05-28-73	148	04	07-15-73	196	--	08-31-73	243	35
05-29-73	149	05	07-16-73	197	--	09-01-73	244	36
05-30-73	150	06	07-17-73	198	--	09-02-73	245	37
05-31-73	151	07	07-18-73	199	--	09-03-73	246	38
06-01-73	152	08	07-19-73	200	--	09-04-73	247	39
06-02-73	153	09	07-20-73	201	--	09-05-73	248	40
06-03-73	154	10	07-21-73	202	--	09-06-73	249	41
06-04-73	155	11	07-22-73	203	--	09-07-73	250	42
06-05-73	156	12	07-23-73	204	--	09-08-73	251	43
06-06-73	157	13	07-24-73	205	--	09-09-73	252	44
06-07-73	158	14	07-25-73	206	--	09-10-73	253	45
06-08-73	159	15	07-26-73	207	--	09-11-73	254	46
06-09-73	160	16	07-27-73	208	--	09-12-73	255	47
06-10-73	161	17			SL-3	09-13-73	256	48
06-11-73	162	18	07-28-73	209	01	09-14-73	257	49
06-12-73	163	19	07-29-73	210	02	09-15-73	258	50
06-13-73	164	20	07-30-73	211	03	09-16-73	259	51
06-14-73	165	21	07-31-73	212	04	09-17-73	260	52
06-15-73	166	22	08-01-73	213	05	09-18-73	261	53
06-16-73	167	23	08-02-73	214	06	09-19-73	262	54
06-17-73	168	24	08-03-73	215	07	09-20-73	263	55
06-18-73	169	25	08-04-73	216	08	09-21-73	264	56
06-19-73	170	26	08-05-73	217	09	09-22-73	265	57
06-20-73	171	27	08-06-73	218	10	09-23-73	266	58
06-21-73	172	28	08-07-73	219	11	09-24-73	267	59
06-22-73	173	29	08-08-73	220	12	09-25-73	268	60
06-23-73	174	--	08-09-73	221	13	09-26-73	269	--
06-24-73	175	--	08-10-73	222	14	09-27-73	270	--
06-25-73	176	--	08-11-73	223	15	09-28-73	271	--
06-26-73	177	--	08-12-73	224	16	09-29-73	272	--
06-27-73	178	--	08-13-73	225	17	09-30-73	273	--
06-28-73	179	--	08-14-73	226	18	10-01-73	274	--
06-29-73	180	--	08-15-73	227	19	10-02-73	275	--
06-30-73	181	--	08-16-73	228				

*SL-1 Launch was on 05-14-73 (DOY 134)

Table 1.2.2.5-1

SKYLAB MISSIONS - CALENDAR DATE/DAY OF YEAR/MISSION DAY (Continued)

CALENDAR DATE	DAY OF YEAR	MISSION DAY	CALENDAR DATE	DAY OF YEAR	MISSION DAY	CALENDAR DATE	DAY OF YEAR	MISSION DAY
10-03-73	276	--	11-17-73	321	02	01-02-74	02	48
10-04-73	277	--	11-18-73	322	03	01-03-74	03	49
10-05-73	278	--	11-19-73	323	04	01-04-74	04	50
10-06-73	279	--	11-20-73	324	05	01-05-74	05	51
10-07-73	280	--	11-21-73	325	06	01-06-74	06	52
10-08-73	281	--	11-22-73	326	07	01-07-74	07	53
10-09-73	282	--	11-23-73	327	08	01-08-74	08	54
10-10-73	283	--	11-24-73	328	09	01-09-74	09	55
10-11-73	284	--	11-25-73	329	10	01-10-74	10	56
10-12-73	285	--	11-26-73	330	11	01-11-74	11	57
10-13-73	286	--	11-27-73	331	12	01-12-74	12	58
10-14-73	287	--	11-28-73	332	13	01-13-74	13	59
10-15-73	288	--	11-29-73	333	14	01-14-74	14	60
10-16-73	289	--	11-30-73	334	15	01-15-74	15	61
10-17-73	290	--	12-01-73	335	16	01-16-74	16	62
10-18-73	291	--	12-02-73	336	17	01-17-74	17	63
10-19-73	292	--	12-03-73	337	18	01-18-74	18	64
10-20-73	293	--	12-04-73	338	19	01-19-74	19	65
10-21-73	294	--	12-05-73	339	20	01-20-74	20	66
10-22-73	295	--	12-06-73	340	21	01-21-74	21	67
10-23-73	296	--	12-07-73	341	22	01-22-74	22	68
10-24-73	297	--	12-08-73	342	23	01-23-74	23	69
10-25-73	298	--	12-09-73	343	24	01-24-74	24	70
10-26-73	299	--	12-10-73	344	25	01-25-74	25	71
10-27-73	300	--	12-11-73	345	26	01-26-74	26	72
10-28-73	301	--	12-12-73	346	27	01-27-74	27	73
10-29-73	302	--	12-13-73	347	28	01-28-74	28	74
10-30-73	303	--	12-14-73	348	29	01-29-74	29	75
10-31-73	304	--	12-15-73	349	30	01-30-74	30	76
11-01-73	305	--	12-16-73	350	31	01-31-74	31	77
11-02-73	306	--	12-17-73	351	32	02-01-74	32	78
11-03-73	307	--	12-18-73	352	33	02-02-74	33	79
11-04-73	308	--	12-19-73	353	34	02-03-74	34	80
11-05-73	309	--	12-20-73	354	35	02-04-74	35	81
11-06-73	310	--	12-21-73	355	36	02-05-74	36	82
11-07-73	311	--	12-22-73	356	37	02-06-74	37	83
11-08-73	312	--	12-23-73	357	38	02-07-74	38	84
11-09-73	313	--	12-24-73	358	39	02-08-74	39	85
11-10-73	314	--	12-25-73	359	40			
11-11-73	315	--	12-26-73	360	41			
11-12-73	316	--	12-27-73	361	42			
11-13-73	317	--	12-28-73	362	43			
11-14-73	318	--	12-29-73	363	44			
11-15-73	319	--	12-30-73	364	45			
		SL-4	12-31-73	365	46			
11-16-73	320	01	01-01-74	01	47			

TABLE 1.2.2.5-2

OWS EXPERIMENT ACTIVITY

EXPERIMENT	SL-2			SL-3			SL-4		
	PLANNED	ACTUAL	PERCENT	PLANNED	ACTUAL	PERCENT	PLANNED	ACTUAL	PERCENT
M092 Inflight Lower Body Negative Pressure	24	22	91	51	50	98	78	67	86
M093 Inflight Vectorcardiogram	24	18	75	51	49	96	78	63	81
M131 Human Vestibular Function	19	13	68	21	24	114	30	27	90
M133 Sleep Monitoring	15	13	87	21	20	95	8	18	225
M171 Metabolic Activity	15	15	100	24	28	117	36	36	100
M487 Habitability Crew Quarters	11	11	100	18	18	100	21	21	100
M509 Astronaut Maneuvering Unit	-	-	-	4	6	150	8	5	63
M516 Crew Activities/Maintenance	-	-	-	7	5	71	7	3	43
S019 UV Stellar Astronomy	8	4	50	12	27	225	14	13	93
S020 UV/X-Ray Solar Photography	-	-	-	-	-	-	1	3	300
S063 UV Airglow Horizon Photography	-	-	-	12	12	100	7	7	100
S073 Gegenschein/Zodiacal Light	11	11	100	30	6	20	36	15	42
S149 Particle Collection	1	1	100	2	2	100	1	1	100
S183 UV Panorama	9	3	33	12	14	117	23	18	78
S190B Earth Terrain Camera	9	6	67	22	29	132	50	38	76
S201 Extreme UV Electronographic Camera	-	-	-	-	-	-	12	10	83
S228 Trans-Uranic Cosmic Rays	1	1	100	1	1	100	3	3	100
S019K UV Stellar Astronomy - Kohoutek	-	-	-	-	-	-	20	13	65
S063K UV Airglow Horizon Photography - Kohoutek	-	-	-	-	-	-	22	14	64
S073K Gegenschein/Zodiacal Light - Kohoutek	-	-	-	-	-	-	0	1	-

TABLE 1.2.2.5-2 (Continued)

OWS EXPERIMENT ACTIVITY

EXPERIMENT		SL-2			SL-3			SL-4		
		PLANNED	ACTUAL	PERCENT	PLANNED	ACTUAL	PERCENT	PLANNED	ACTUAL	PERCENT
S183K	UV Panorama - Kohoutek	-	-	-	-	-	-	13	6	46
S201K	Comet Kohoutek Photometric Photography	-	-	-	-	-	-	15	14	93
T025K	Coronograph Contamination Measurements - Kohoutek	-	-	-	-	-	-	2	2	100
T002	Manual Navigation Sightings	-	-	-	34	31	91	34	16	47
T003	Aerosol Analysis	10	7	70	20	19	95	26	30	115
T013	Crew Vehicle Disturbances	-	-	-	1	1	100	-	-	-
T020	Foot Controlled Maneuvering Unit	-	-	-	3	3	100	3	2	67
T025	Coronograph Contamination Measurement	-	-	-	-	-		1	1	100
T053	Earth Laser Beacon Assessment	-	-	-	-	-		46	20	44
ED31	Bacteria and spores	2	2	100	-	-	-	2	2	100
ED32	In Vitro Immunology	-	-	-	1	1	100	-	-	-
ED41	Motor Sensory Performance	-	-	-	-	-	-	3	3	100
ED52	Web Formation	-	-	-	3	3	100	-	-	-
ED61/62	Plant Growth/Plant Phototropism	-	-	-	-	-	-	2	2	100
ED63	Cytoplastic Streaming	-	-	-	3	1	33	3	2	67
ED72	Capillary Studies	-	-	-	-	-	-	3	1	33
ED74	Mass Measurement	-	-	-	4	4	100	-	-	-
ED76	Neutron Analysis	2	2	100	-	-	-	2	2	100
ED78	Liquid Motion	-	-	-	1	0	0	-	-	-

SECTION 2 - SYSTEM DESIGN AND PERFORMANCE

2.1 GENERAL

2.1.1 Design Philosophy

We believe some 30 to 35 years before Skylab's OWS became an orbiting reality, the notion was discussed by Dr. Wernher van Braun and some of his very close associates in Peenemunde, Germany. The idea was to convert an orbiting, spent rocket stage into a "house" for man to visit, a workshop to work in, a laboratory in which to conduct experiments. A proposal to study the use of Saturn SIV as a Manned Space Laboratory was documented in November 1962 by the Douglas Aircraft Company. The Space Laboratory and a Gemini capsule with two crewmen were to be boosted from the Atlantic Missile Range (AMR) by an S-1 launch vehicle. Following separation from the booster, the SIV propulsion system was to insert itself and Gemini into a 250 nautical mile (463.3 kilometer) orbit. After SIV hydrogen depletion, the 4200 ft^3 (118.9 $meters^3$) tank would be purged, then filled with stored atmosphere and conditioned for crew entry. Figure 2.1.1-1 shows this early concept.

A number of different concepts for workshop were envisioned by study groups; some involving new hardware. However, the Apollo Extensions Support study established that system equipment and module components developed for Apollo, with logical modification, were more than adequate to support the development of an orbiting workshop.

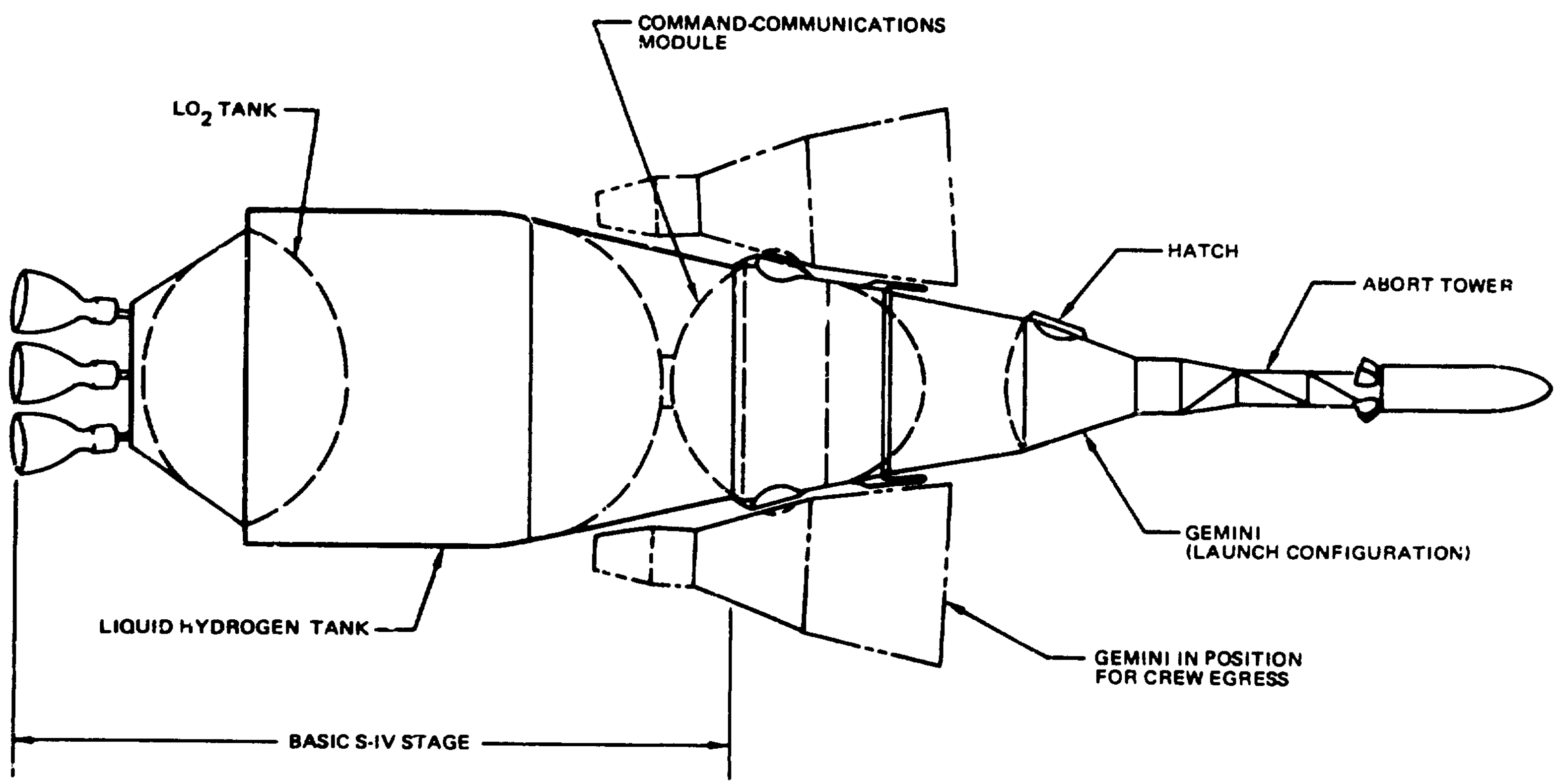


Figure 2.1.1-1 S-IV-Gemini Space Laboratory Concept

From the very first thoughts of a space workshop and continuing through the evolution studies, the utilization of previously developed hardware was significant in the thinking. The Orbital Workshop (OWS) was not to be the product of a number of development flight articles, each more sophisticated than its predecessor as was planned for Apollo, but rather it was to have its development foundation in the Apollo Program.

Throughout the growth period of the OWS, every concept, every new innovation was reviewed and evaluated in terms of design simplicity and low cost. There is inherent high level hardware reliability in design simplicity as is the case for inherent low cost. However, having infinitely higher regard in the minds of all was to design safety into the OWS for the crews and assurance for mission accomplishment. All systems affecting crew safety and mission accomplishment are redundant. Further, the OWS is designed fireproof and offers no hazard from off-gassing and toxicity.

Following is a tabulation of the general OWS design principles:

A. No single failure in any system will cause or require abort or compromise crew safety.

B. During an abort, no single failure in a subsystem or component shall compromise crew safety.

C. Mission/Safety Critical Items whose failure could adversely affect crew safety or result in not achieving primary mission objectives was documented in terms of:

- o Justification for retention in design
- o Efforts to preclude the occurrence of the critical failure or to minimize the consequences of the failure
- o Provisions for inflight maintainability
- o Description of contingency procedures
- o Consideration of other factors.

D. No single failure point in the Emergency and Warning System will cause failure to indicate an emergency or warning or cause a false emergency or warning.

E. Sharp corners and edges that could cause laceration or puncture wounds were eliminated.

2.1.2 <u>Wet to Dry Evolution</u>

The SIV-Gemini space laboratory proposal previously mentioned provided the first conceptual definition of a "wet" workshop; "wet" meaning that SIV was a propulsion stage until orbited, rendered safe (passivated) and conditioned for crew occupancy. Following the Apollo Extensions support study, there was much talk of using an S-IVB/IB to carry a Command and Service Module (CSM) with sun-study telescopes into orbit. The telescope assembly for deployment by the crew to the service module for operation was the initial Apollo Telescope Mount (ATM) concept. Through further Apollo Applications

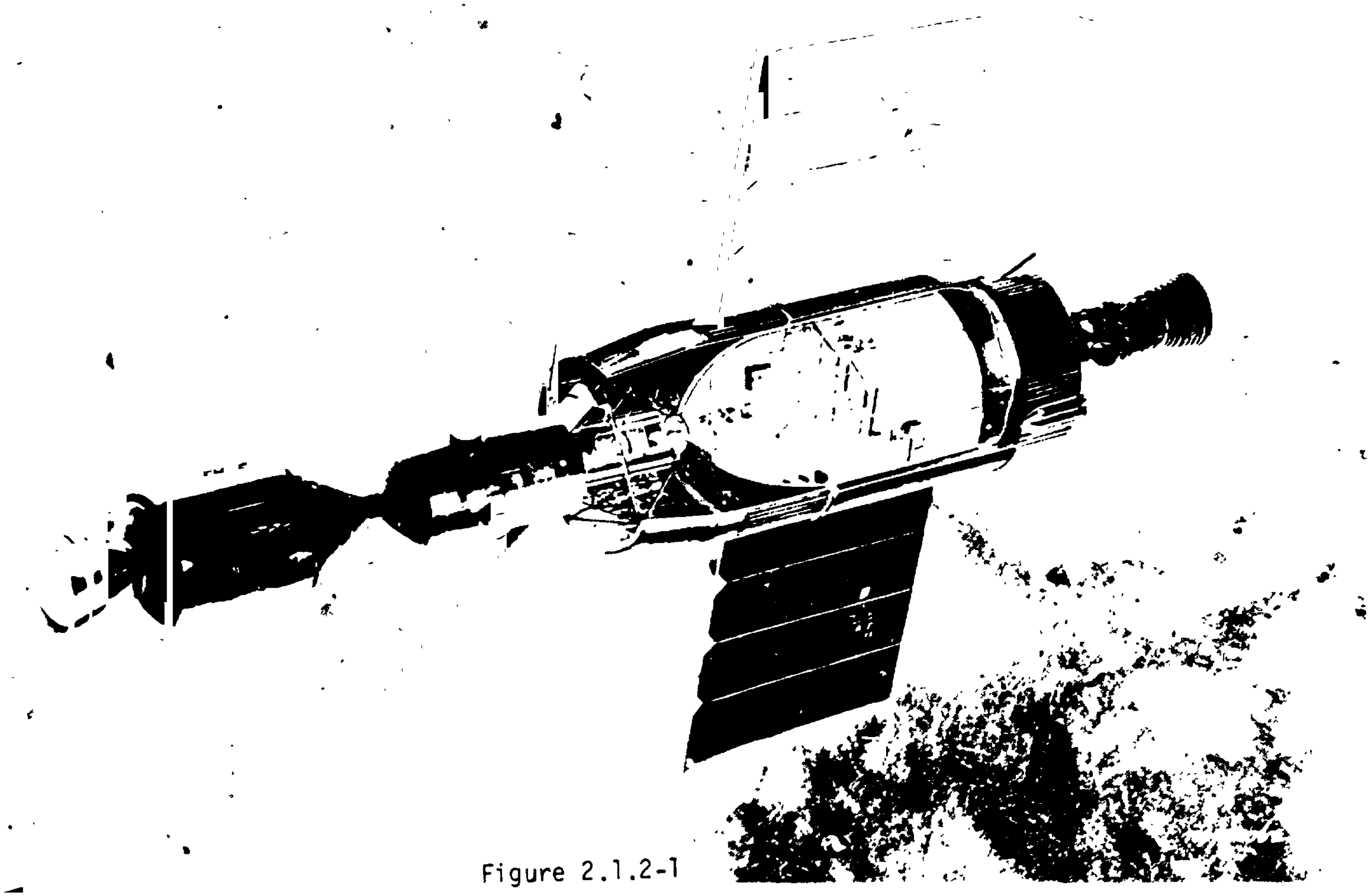


Figure 2.1.2-1

NASA SATURN S-IVB ORBITAL WORKSHOP

studies, the first concept for the space station to have hardware modification and development started was a "cluster" consisting of an Airlock Module (AM), Multiple Docking Adapter (MDA) and OWS, "wet" workshop as shown in Figure 2.1.2-1 in-orbit with docked CSM.

Modifications of the S-IVB into the wet workshop could not in any way compromise the S-IVB function for its basic Apollo mission role. This, of course, together with the requirement for materials compatibility with LH_2, greatly reduced the capability for pre-installation of equipment in the modified S-IVB. In turn, this required extensive activation tasks on the part of the flight crew to ready their laboratory for work. In retrospect, the enormity of this effort (i.e., the de-installation of MDA launch stowed equipment, its transfer to and re-installation in OWS, checkout and operation activation, together with several days of interruption in the collection of sequential medical data on the crew) makes it somewhat difficult to totally recapture the rationale to justify development of the concept. Perhaps that rationale is obscured by what we recognize as the magnificence of the "dry" workshop, its pre-installed systems/equipment, and the way they have performed for Skylab. But, the "wet" workshop experiment was a logical step in the acquisition of Skylab and would have provided a very fruitful missi .

Figure 2.1.2-2 depicts the launch configuration for "wet" workshop, the first to be mocked-up and delivered to Marshall Space Flight Center (MSFC). Its accommodations were minimal, consisting principally of "house" structure, rooms, ceiling, floor, insulation and ventilation ducting.

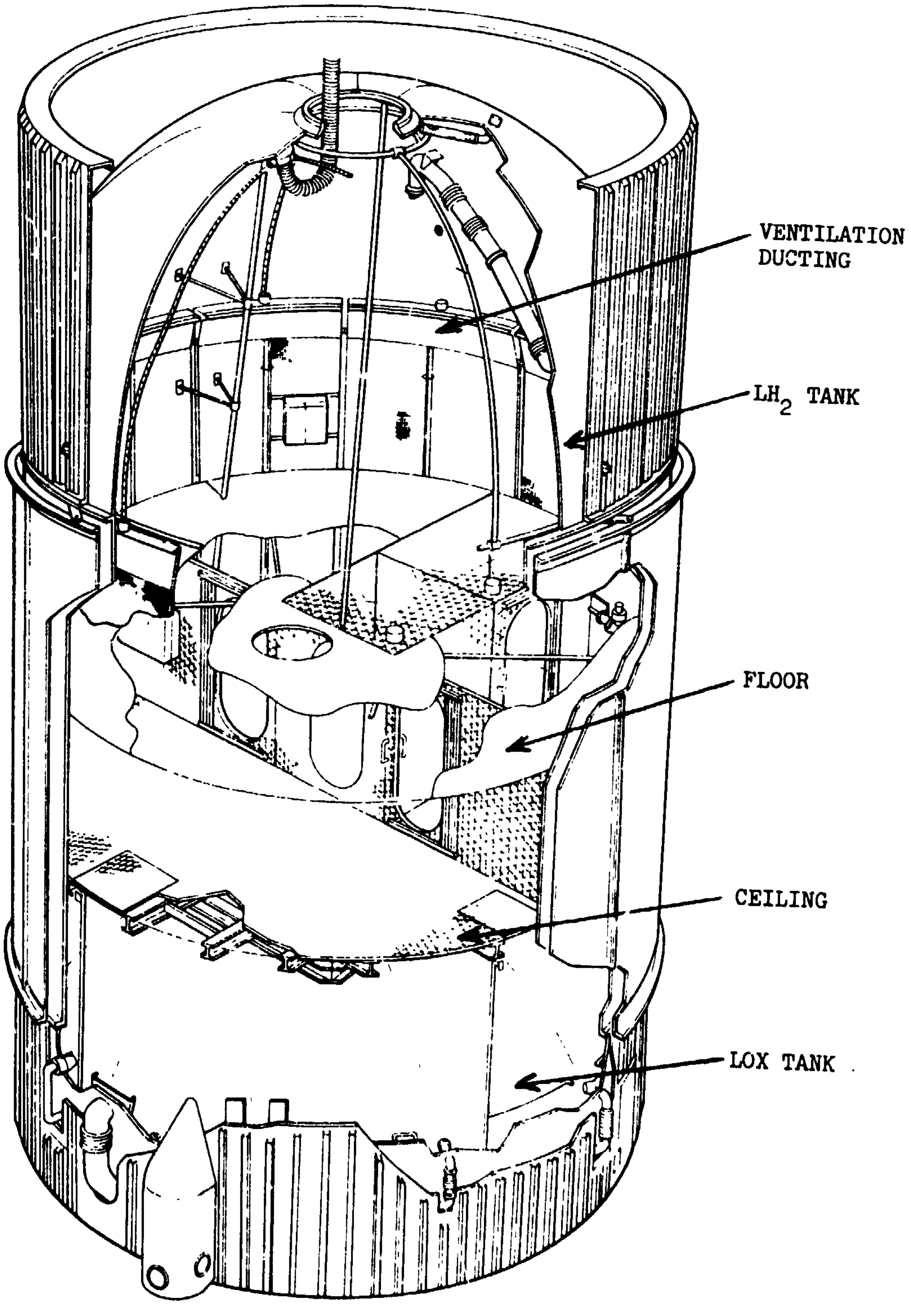


Figure 2.1.2-2 WET WORKSHOP LAUNCH CONFIGURATION

By now, the notion to modify a lunar module ascent stage (LM) into a control center for an ATM and launch them together on a Saturn 1B for docking to the "Wet" Workshop was being discussed. The rendezvous and docking of these vehicles was considered to be effected in an unmanned mode. This required the development of a major new operational capability; unmanned rendezvous and docking. It was recognized, however, that if the workshop launch stack could accommodate an ATM, it would greatly simplify the program by elimination of extensive programmatic and technical requirements.

The capability to reduce program cost and complexity by eliminating the ATM as a free flying module, the ability to rapidly activate the workshop by having systems/equipment pre-installed and checked out prior to launch, and the capability to significantly expand the mission potential with the weight margins offered by a Saturn V launch as compared to a 1B launch supplied rationale for a conversion of the Workshop from "Wet" to "Dry". The successful moon landing of July 1969, changes in funding and the reduction in planned moon visits made Saturn V vehicles available for Apollo Applications, thus, the next logical step in Workshop development, the conversion from "Wet" to "Dry". This also allowed the use of the CSM with minimum modification as purely a crew logistics vehicle; it would not be required to provide logistics support from the expanded capability of the space station. Plans were formulated for the launch of two Saturn V workshops with ATM's to be visited up to seven times; however, this was re-defined in 1970 to the program we know today and renamed Skylab.

By far the most affected module in the Skylab cluster by the "wet" to "dry" conversion was the OWS. The requirement to preserve the S-IVB as a propulsive rocket stage no longer existed. Pre-installation of systems/equipment brought about three significant philosophical changes in the design of OWS. First, the weight allowance, an increase of nearly 150,000 pounds (68,000 kilograms) for the cluster, could be used to reduce the criticality of weight in systems design. This permitted more rugged and redundant designs with greater inherent safety and expected operational life. Secondly, the enhanced capability produced a philosophical shift in design from the space station as an experiment to the position that Skylab was a major operational step in space exploration; that the medical and scientific/technology activities stood alone on the experimental basis. Last, the total mission life time, and therefore the experiment program, was greatly expanded and enriched by the availability of the ample weight margins.

2.1.3 Overall Test Program

A. Test Program - The DAC-56697, Saturn I Workshop (SIW) Test Plan of 17 September 1968, for the "wet concept," outlined a test program which would verify flight readiness and demonstrate that workshop hardware would in no way jeopardize the performance and reliability of the basic S-IVB flight vehicle. The plan encompassed Engineering Laboratory tests of components and subsystems, special system tests, acceptance firing program at the Sacramento Test Center (STC), prelaunch checkout program at Kennedy Space Center (KSC) and flight test planning and evaluation. The requirements within the plan were combined to form a complete program so that the accumulated test results could be utilized for the final verification of launch configuration of the workshop.

The component and subsystem testing program guidelines and constraints were based on a one-of-a-kind experimental vehicle concept, with primary emphasis placed on testing of hardware whose failure could adversely impair crew safety and hardware whose failure could result in failure to achieve primary mission objectives.

The following stage checkout and testing guidelines were established in order to reduce ground support test equipment requirements and to ensure that equipment verification tests were accomplished on hardware in a near-launch configuration as possible:

1/ Acceptance Firing (STC)

- o Install SIW in Beta test stand and perform essentially booster stage checkout plus meteoroid shield deployment.
- o Airlock simulator not to be used.

2/ Launch Preparations (KSC)

- o Install SIW in low bay for Experiment Verification Tests (EVT).
- o Conduct EVT with other cluster interfaces as required.
- o Perform plugs-in and plugs-out leak tests.
- o Electromagnetic Compatibility (EMC) tests were to be performed during EVT utilizing turn-on and lock philosophy.
- o Checkout in stack as required to ensure integrated booster and workshop system operation.

B. Test Program Changes "Wet" to "Dry" - Testing effort expended prior to the decision to change the OWS concept from "wet" to "dry" included testing that was common to both the "wet" and "dry" concept and that which was peculiar to only the "wet" concept.

1/ Component and Subsystem Testing - The component test program at the time of conversion from "wet" to "dry" consisted of 71 line items, of which 24 were "wet" peculiar and 7 had been completed. The completed items were primarily development tests necessary to evaluate various tank penetration sealing devices, floor and wall attachments and a major test program on the LH_2 insulation system. The LH_2 insulation tests consisted of coupon testing to determine outgassing characteristics of the hydrogen which would enter the insulation foam cells during the LH_2 loading at STC and at KSC. After determining from the coupon tests that the insulation system could be satisfactorily passivated, an 8 foot (2.4 meter) diameter tank test was conducted to determine the conductivity of the insulation system and to determine the passivation time required prior to habitation. The conductivity test was conducted at STC consisting of an LH_2 loading and boil off test. The passivation test was conducted at MSFC. The tank was installed in a vacuum chamber loaded with LH_2, subjected to pressure profile simulating launch to orbit conditions. During the passivation and during habitation modes, gas analyses were made to assure that passivation was satisfactory. These data were to be used in support of the mission operation procedures.

2/ System Testing - The change from wet to dry concept had a significant impact on the system level testing; however, the decision was made early enough in the program that the expended resources consisted of planning effort only.

Major program changes were the elimination of the acceptance firing program at STC and the SIW no longer required propulsion system verification checks at Huntington Beach and at KSC.

C. Overall Program (Dry) - The overall test program objectives and guidelines for dry concept were basically the same as for wet concept. The major difference being in the realignment of specific tests to match the dry concept hardware. The test program that was performed consisted of the following tests:

- o Component and Subsystem (Development, Qualification, and Production Acceptance).
- o Special Design Support and Verification.
- o Spacecraft System and Integration.
- o Mission Support.

D. Component and Subsystem Tests - Development, Qualification and Production Acceptance testing were accomplished at the hardware level of assembly. Development tests were conducted on prototype or pre-production configurations depending on the particular phase of design evaluation at the time of initiation. This type of testing was accomplished to optimize hardware configuration and to identify areas of marginal design performance. These tests were also utilized to demonstrate fulfillment of design objectives. Qualification tests were performed on production hardware to demonstrate that design and production methods would result in a product which fulfilled the design requirements. Production acceptance tests were performed on all deliverable items to ensure that production methods and quality control produced an article which satisfied the design intent.

The development and qualification program consisted of 154 line items. These tests are identified in Section 5 of this report.

E. Special Design Support and Verification Tests - Design Support and Verification Tests are categorized as special tests since they do not meet the requirements of the standard OWS Development or Qualification Test Program. These tests were usually conducted at government facilities and/or subcontractor facilities and required MDAC-W hardware, software and technical support personnel. There were 27 of these tests conducted. Zero-gravity and neutral buoyancy test programs were included in this category.

F. Spacecraft System Tests - Post-manufacturing checkout of the OWS-1 was accomplished in the Huntington Beach Vehicle Checkout Laboratory during the period 6 November 1971 through 16 August 1972. The objective of this activity was to (a) provide an OWS checked out and calibrated to an extent consistent with the ambient 1-G environment, and (b) provide an OWS acceptable for planned, integrated cluster system testing at the Kennedy Space Center. Checkout was performed utilizing flight hardware within the constraints of hardware availability. Detail test requirements, acceptance criteria and operational constraints were provided in the 1B83429, OWS-1 Test and Checkout Requirements, Specifications and Criteria.

Checkout was initiated with the start of Continuity/Compatibility testing and continued through completion of the All Systems Test, EMC, and residual subsystem retests. During this checkout period, all subsystems, Crew Compartment Fit and Function, and the AST and EMC tests were performed. Thousands of elapsed hours of manufacturing work were accomplished in parallel. No major problems were encountered. All items associated with open work being transferred to KSC were noted in Section 4 of the MDC G3078B, OWS Predelivery Turnover Review Report, Huntington Beach.

G. Integrated Vehicle Testing (KSC) - There were three primary areas of operations at KSC:

- o Operations and Checkout Building (O&C) - General office area and Acceptance Checkout Equipment (ACE) rooms located on the third floor.
- o Vertical Assembly Building (VAB) - Office area, receiving inspection, equipment storage, assembly and test site; MDAC OWS-1 occupied some of the low bay area, High Bay 2, 20A and B, 24B, 25B, and 26B of Towers A and B.
- o Launch Complex 39A - OWS-1 Launch Site.

All pre-flight preparations and testing were conducted in accordance with the Pre-Flight Operations Procedures (POP).

All High Bay 2 facilities, GSE, electrical, mechanical, and fluid systems were verified and certified as functional, clean, and mercury free prior to vehicle hookup.

The OWS-1 arrived at KSC on 22 September 1972 and was inspected and moved to the VAB on 23 September 1972. Integrated Vehicle Testing (KSC) consisted of verification of each subsystem, system verification and final system test and launch. Electrical and mechanical closeout was started 2 April 1973 and completed 13 April 1973. All test problems were satisfactorily resolved prior to launch. Section 5 of this report lists in detail each test conducted and their anomalies and resolutions.

H. Mission Support Tests - Real time Mission Support was provided by MDAC-W during mission simulations KSC prelaunch activity and Skylab missions through the OWS Mission Support Room (MSR).

The MSR was the focal point of OWS mission support activities and special tests utilizing the OWS Backup and development and qualification hardware. Special tests were initiated by MSR action item or direct request from the NASA or MDAC-W authorized personnel. All requests were in the form of a Mission Support Test Request (MSTR) and were reviewed and approved by a MSR Room Captain, VCL Chief Program Engineer, and the NASA Senior Checkout Representative.

Special tests were performed to simulate certain mission problems and test solutions prior to transmitting action item responses to the Huntsville Operations Support Center (HOSC).

There were 31 tests performed utilizing the OWS Backup and 64 Laboratory tests from SL-1 roll-out through SL-4 mission completion.

2.1.4 Final Configuration Discussion

2.1.4.1 Structural System - The OWS structural system (Figure 2.1.4.1-1) is a modified S-IVB stage and consists of a forward skirt, propellant tanks, aft skirt, thrust structure and a main tunnel. The skirts and main tunnel serve the same function for the OWS that they did for the S-IVB, i.e., to carry structural loads and accommodate externally mounted equipment, plumbing/wiring. The thrust structure has no J2 engine thrust loads to transmit, but otherwise it was used similarly to the S-IVB usage to accommodate installation of additional equipment and integration hardware peculiar to the OWS.

Modification of the S-IVB propellant tanks for the OWS were more extensive. A larger, reusable entry hatch replaced the S-IVB hatch in the forward dome of the LH_2 tank. A side panel was added to the LH_2 tank for ground access and to provide entry into the tank for modifications, installations and checkout. Three other

ORBITAL WORKSHOP
TANK ASSEMBLY, SKIRTS AND INTERSTAGE

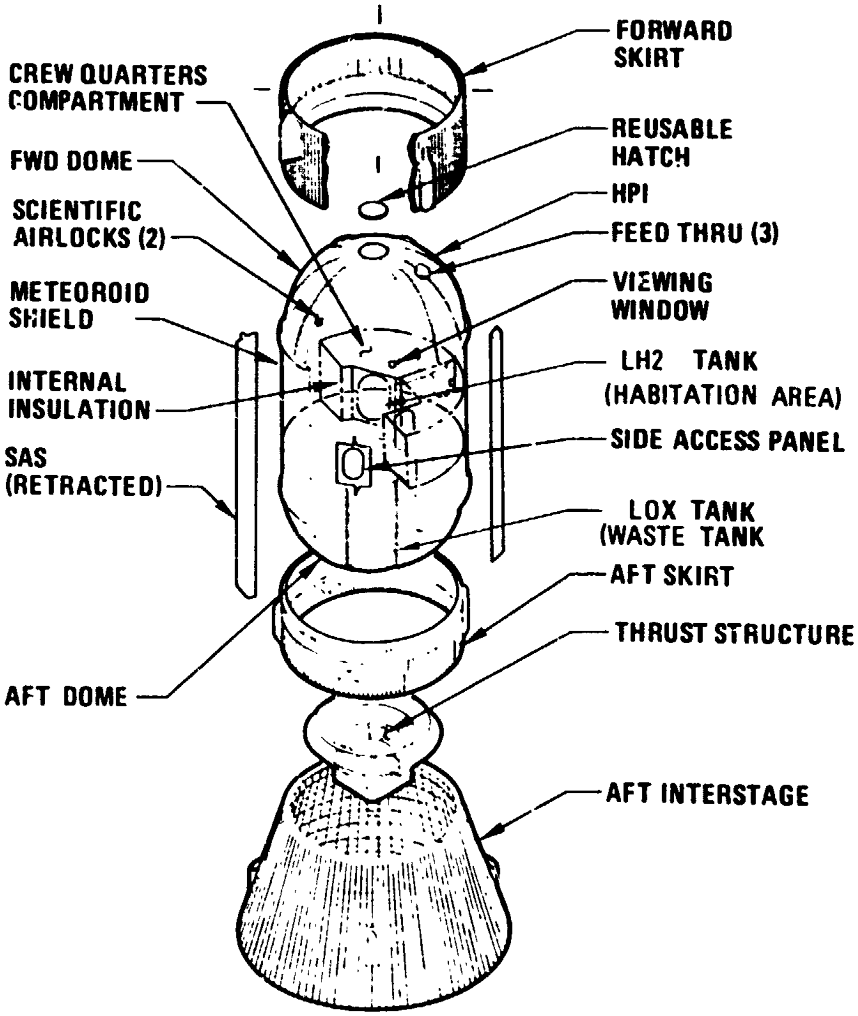


FIGURE 2.1.4.1-1

apertures were included to provide an orbital viewing window and to accommodate two scientific airlocks which provided the capability to deploy experiments external to Skylab.

Internally, the LH_2 tank modification consisted first of fully "papering" the polyurethane tank wall insulation with aluminum foil to fireproof the habitation area. A pair of grid floors enclosing the crew quarters were installed and crew quarters consisting of a wardroom, waste management and sleep compartments and a medical experiment compartment were included.

The S-IVB LOX tank was converted to a waste tank for the disposition of Skylab trash. The tank was compartmented with screens; one compartment used to collect liquid waste which was non-propulsively vented overboard. The common bulkhead between the habitation area and the waste tank was reworked at the center for the installation of a trash lock through which trash was passed by the Skylab crews.

2.1.4.2 Meteoroid Shield System - A shield for the OWS habitation area protection against meteoroid penetration is afforded (Figure 2.1.4.2-1). The probability against [illegible]sure loss from penetration is equal to or greater than 0.995. The [illegible]ield is made from aluminum sheet and is pretensioned against the tank wall for launch and ascent. It is released on orbit by ordnance serverance of tie-down straps and is designed to deploy to a standoff distance from the tank wall of 5 in.(12.7 cm). The deployment is accomplished by energy stored in torsion bar springs installed at the forward and aft skirts. On deployment, the shield is designed to envelop the habitation area and through use of thermal coatings provides the passive thermal control for the Workshop.

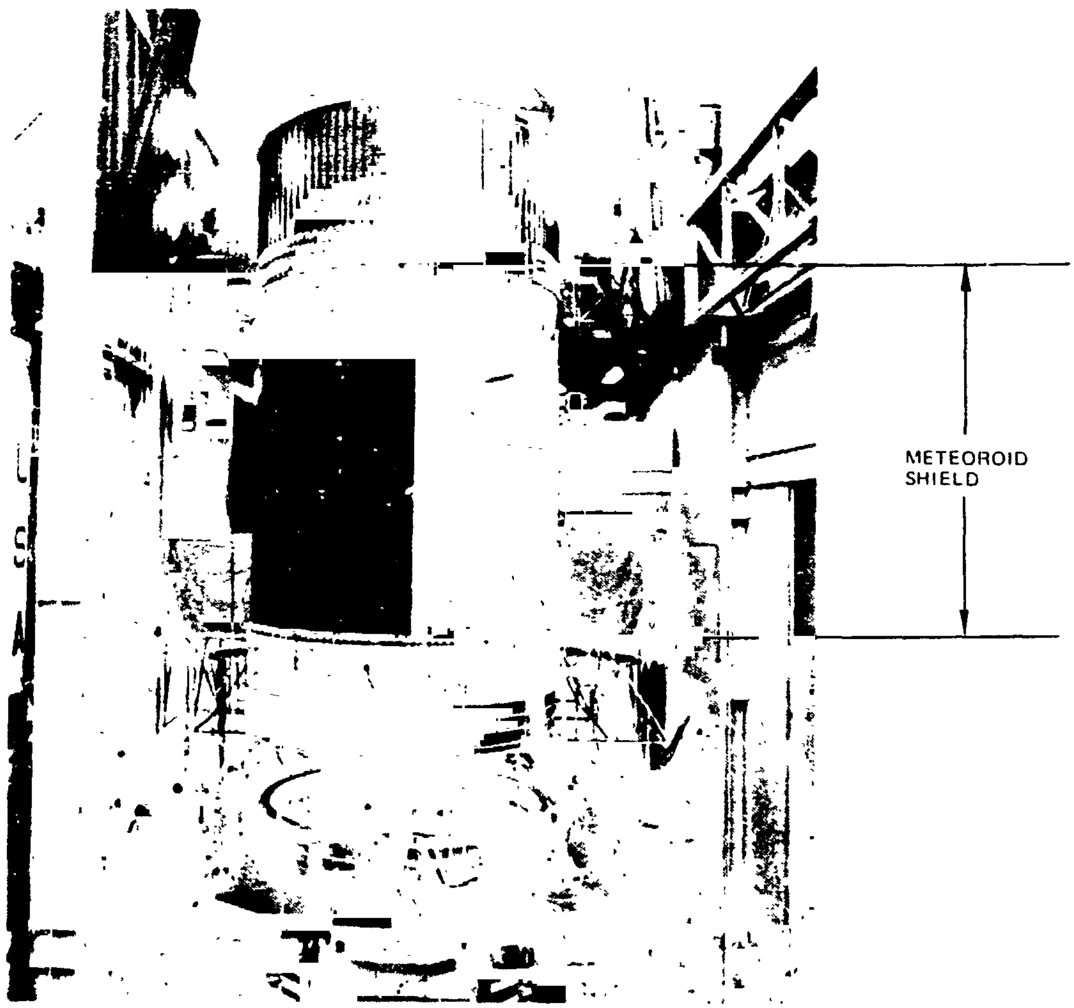


FIGURE 2.1.4.2-1 METEOROID SHIELD

2.1.4.3 Environmental/Thermal Control Subsystem (E/TCS) - The E/TCS design is based upon passive thermal control of the OWS environment with augmentation by convective heating and cooling of the atmosphere during manned phases and radiative heating of the internal structure during unmanned phases. The E/TCS is thus made up of two basic subsystems: an active thermal control subsystem including ventilation and a passive thermal control subsystem.

The passive thermal control subsystem consists of optical property control of the OWS interior and exterior surfaces, high performance insulation (HPI) on the forward dome, polyurethane insulation lining the inside of the OWS pressure shell and heat pipes attached to structural penetrations of the interior insulation. The exterior surface finishes and the HPI blanket control the net energy balance between the OWS and the external space environment. The heat transfer rates from the habitation area to the meteoroid shield, and from the forward and aft dome areas, are regulated by surface finish control. Also, the interior habitation area wall temperatures are made more uniform with optical property control of these surfaces and with the heat pipes.

The active thermal control subsystem provides continuous control of the OWS internal environment during periods of astronaut habitation. The cabin gas temperature is controlled by cabin gas heat exchangers in the airlock module (AM) and by convective heaters in the three VCS ducts. Reconstituted air from the AM is mixed with recirculated air in the OWS. Prior to habitation, radiant heaters maintain temperatures above the minimum levels that satisfy food and film storage requirements. Figure 2.1.4.3-1 shows the OWS active system components.

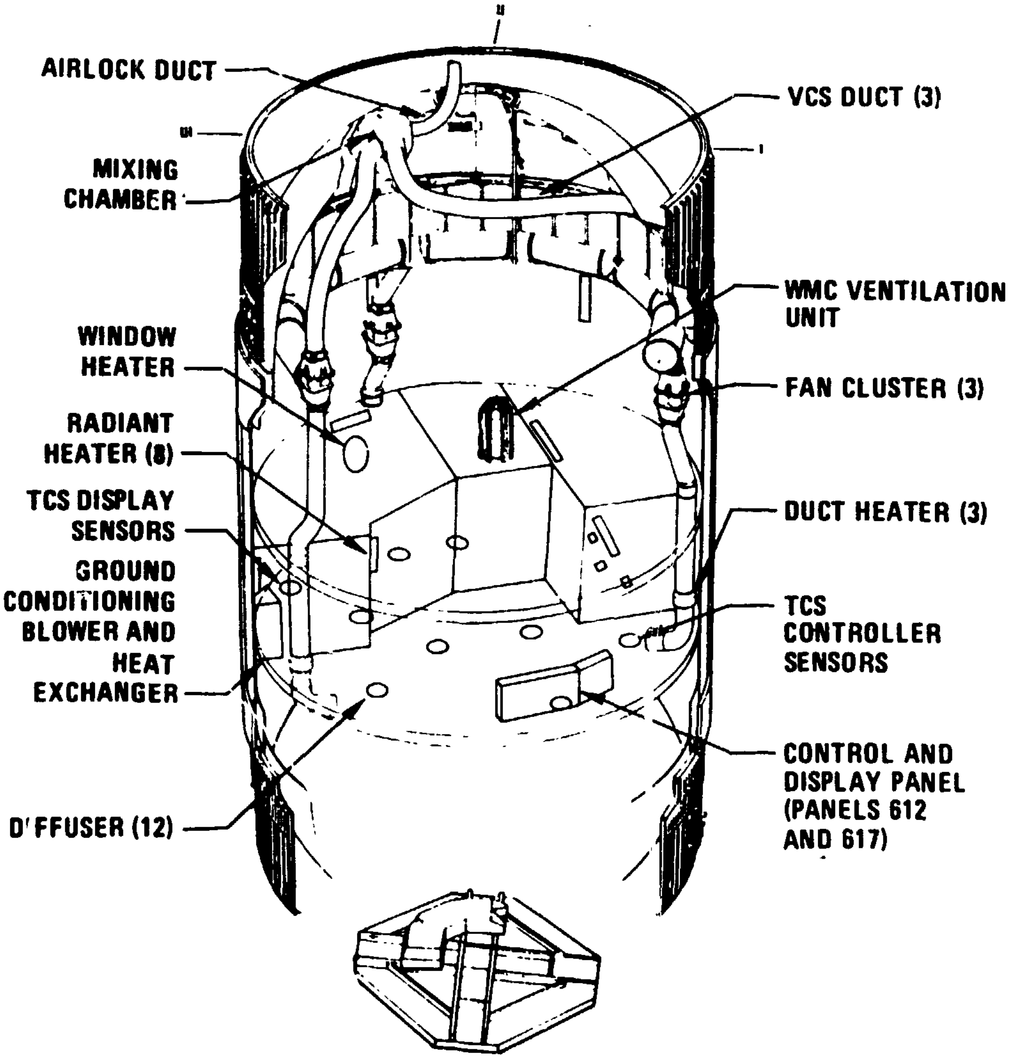


FIGURE 2.1.4.3-1 ACTIVE THERMAL CONTROL SYSTEM

2.1.4.4 Thruster Attitude Control System (TACS) - For most of the eight-month long Skylab mission, the primary source of attitude control is the three control moment gyros (CMG's) which provide the pointing accuracy and stability necessary for many Skylab astronomical and earth resources experiments, and which maintain the solar inertial attitude necessary for the Skylab solar arrays. A propulsive attitude control system (ACS) is needed to provide control during CMG spinup (the first ten hours of the mission), to handle docking transients and large maneuvers beyond the capability of the CMG's, to desaturate the CMG's when necessary, and to provide a contingency capability in case of CMG failure. This system designated TACS (thruster attitude control system) provides over 81,000 lb/sec (360,000 N-sec) of impulse. A high thrust level of 50 lb (222 N) is required at the start of the mission for separation transients, a 20 lb (90 N) thrust minimum is required for each of the three dockings with Apollo command modules, and a 10 lb (45 N) minimum was specified for the rest of the mission.

The system is a blow-down system using gaseous nitrogen as the propellant. The plumbing system is fully brazed. Figure 2.1.4.4-1 shows the location of TACS equipment on OWS. Two modules of three thrusters each, 180° apart on the OWS aft skirt utilize quad-redundant valves for each thruster.

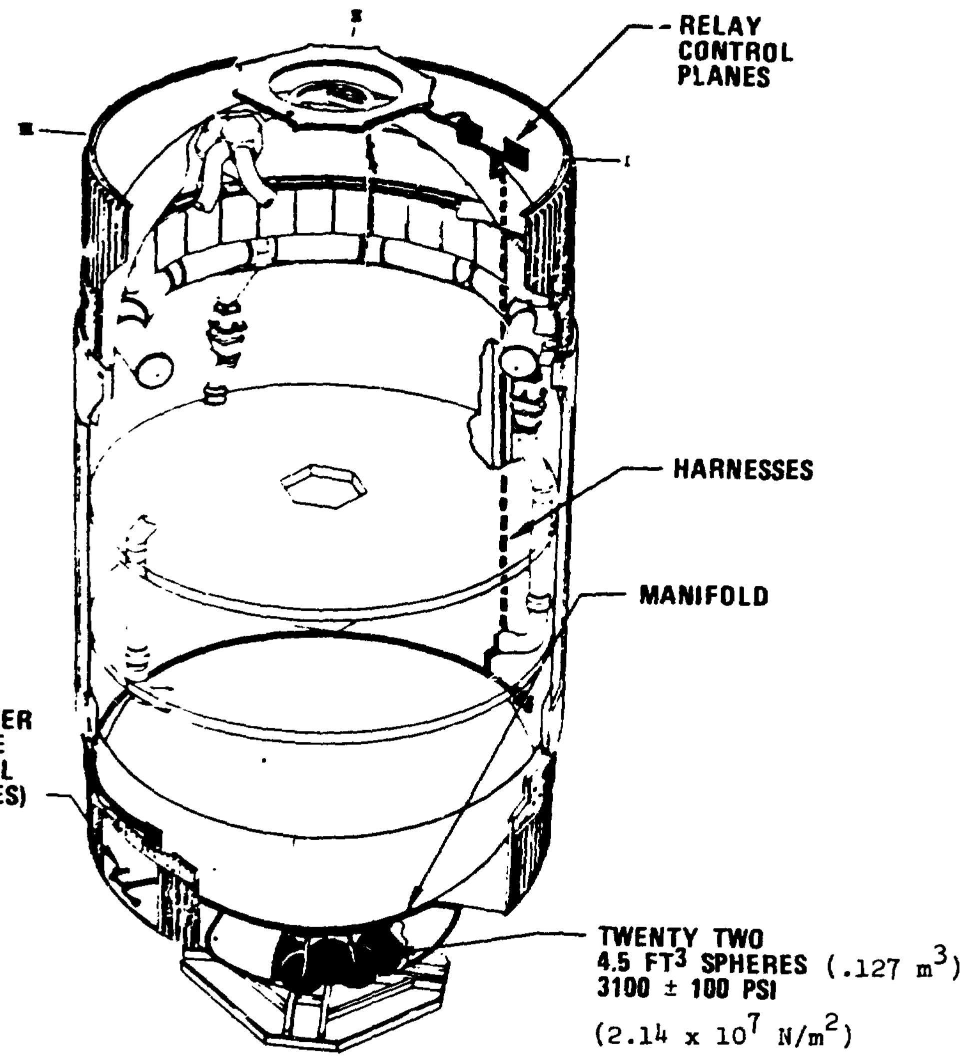
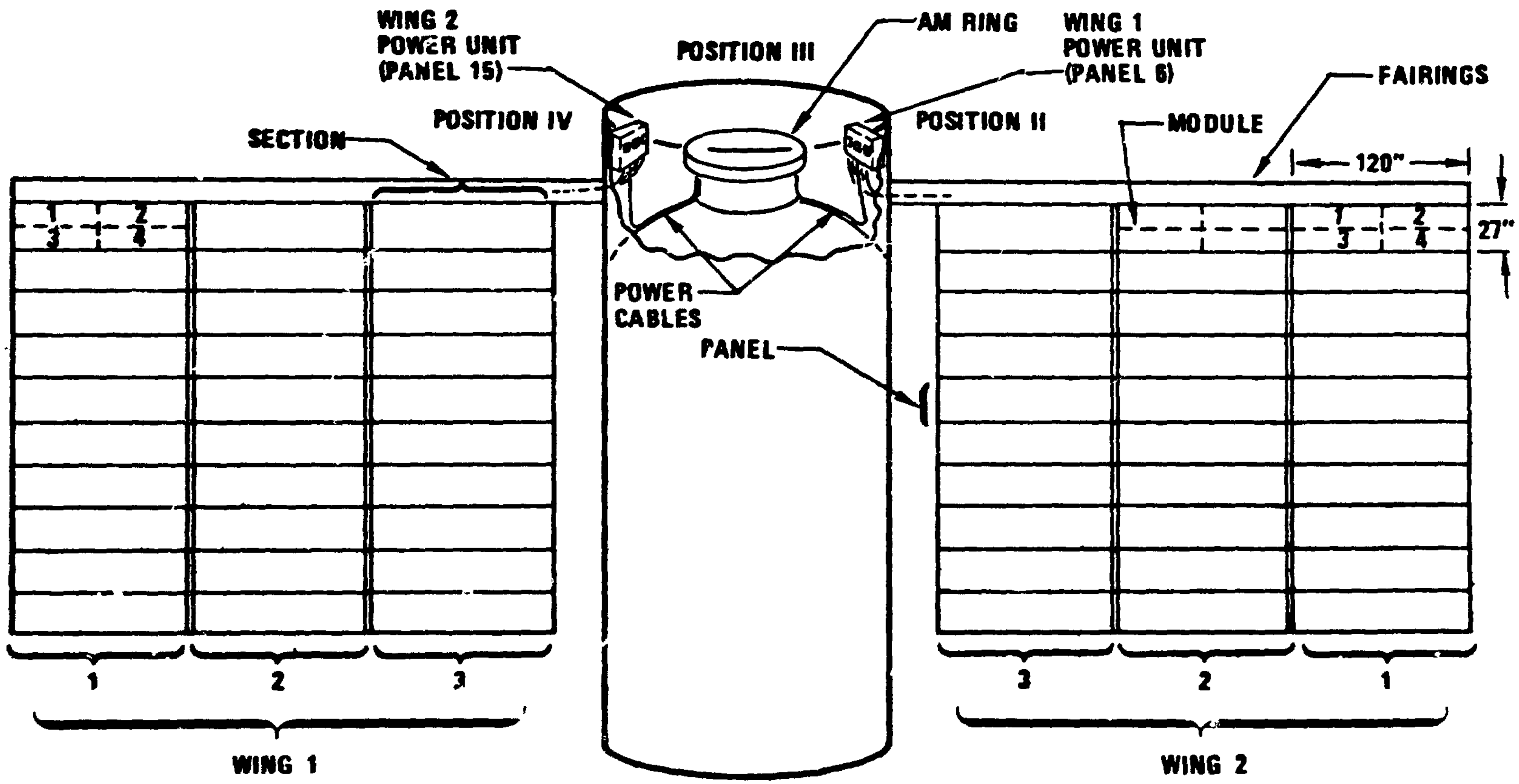


FIGURE 2.1.4.4-1 THRUSTER ATTITUDE CONTROL SYSTEM

2.1.4.5 Solar Array System (SAS) - The Solar Array System (SAS) for OWS is made up of two wings, each consisting of a beam fairing and three wing sections. Each section contains ten identical active solar panels for a total of 30 panels per wing or 60 for the complete system. The system supplies electrical power to the AM for distribution to equipment requiring power. SAS provides an average of 10,496 watts between 51 and 125 volts during the sunlit portion of each orbit.

For launch and ascent of SL-1 the SAS beam fairings housing the array are stowed snugly against the OWS meteoroid shield/tank structure. A GN_2 ground purge is introduced into the beam fairings to insure an atmosphere environment around the stowed array of 50 percent relative humidity or less. During launch the beam fairings are vented to preclude over pressurization of the structural fairings.

After insertion of SL-1 in orbit, an ordnance severence system releases the SAS beam fairings for deployment. The deployment is accomplished with a viscously damped spring actuator. Subsequently, the wing sections are released and deployed from the beam fairing by similar systems. The beam fairings and wing sections are mechanically latched in the deployed positions. Figure 2.1.4.5-1 shows the deployed SAS and related equipment relative to the OWS tank structure.



NOTES:
1. (616) SOLAR CELLS MAKE UP A MODULE
2. (4) MODULES MAKE UP A PANEL
3. (10) PANELS MAKE UP A WING SECTION
4. (3) WING SECTIONS PLUS BEAM FAIRING MAKE UP A WING

FIGURE 2.1.4.5-1 SOLAR ARRAY SYSTEM

2.1.4.6 Electrical Power Distribution System (EPDS) (See Figure 2.1.4.6-1) - The EPDS provides the means for power distribution from the AM to all OWS loads. Power is distributed externally to Thruster Attitude Control System (TACS), Instrumentation, etc., and through OWS feed throughs to redundant busses routed to an electrical power and control console. In turn, the power is routed from the console to systems/equipment and experiments internal to OWS. The console in conjunction with remote control panels contains switches, circuit breakers, and indicators to permit crew control of power distribution to end items. The EPDS receives 25.5 to 30 vdc from the AM and supplies 24 to 30 vdc to the end items. Wiring to end items is electrically protected with circuit breakers and physically protected from damage and fire by metallic trough type "conduits."

2.1.4.7 Illumination System - An illumination system in OWS is provided to allow for normal and emergency crew activities and experiment operations. The system consists of general illumination lighting, initial entry and emergency lighting and auxiliary lighting.

For general illumination, there are 42 floodlights, 18 in the forward compartment with 8 on the forward dome and 10 on the forward walls, 4 in the wardroom, 3 in the waste management compartment, 3 in the sleep compartment, and 14 in the experiment area. For redundancy, one-half the lights in each area are on Bus #1 and the remainder on Bus #2.

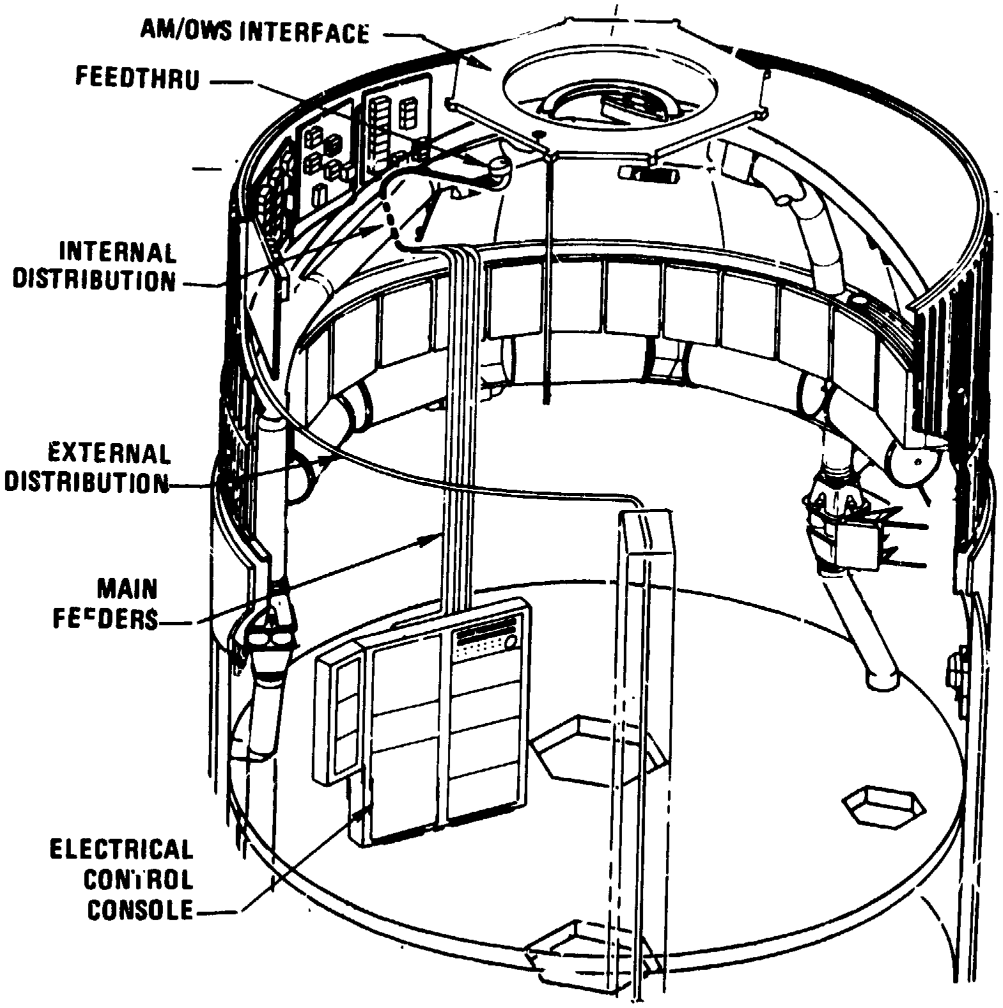


FIGURE 2.1.4.6-1 ELECTRICAL POWER DISTRIBUTION SYSTEM

For initial crew entry into OWS and emergency, a lighting system is provided to control 8 of the 18 lights in the forward compartment. These floodlights will illuminate, regardless of the position of their remote or integral light switch. The initial entry lighting is controlled by a single switch in the aft compartment of the Airlock Module and the emergency lighting is enabled by the simultaneous failure of both OWS busses which automatically supplies emergency power to the initial entry and emergency light system.

Two portable, hi intensity lights, each containing 4 permanently installed fluorescent lamps, are supplied for special illumination. Figure 2.1.4.7-1 shows the light arrangement of the OWS illumination system.

2.1.4.8 Communication, Data Acquisition and Command System - The OWS communications system provides capability for audio communication between Skylab crewmen and between the crew and ground control. It also provides accommodations for video transmission from Skylab to ground control and the acquisition of bio-medical data on the crewmen. Ten GFP Speak Intercom Assemblies (SIA's) located throughout OWS comprise the principal hardware of the system. The SIA's utilize two channels, either of which can be connected to a crewman's communication umbilical. Further, they include the capability for push-to-talk, push-to-transmit and voice tape record selection by a crewman. Each SIA also includes an audio device for caution and warning tones. Figure 2.1.4.8-1 shows the communication system arrangement.

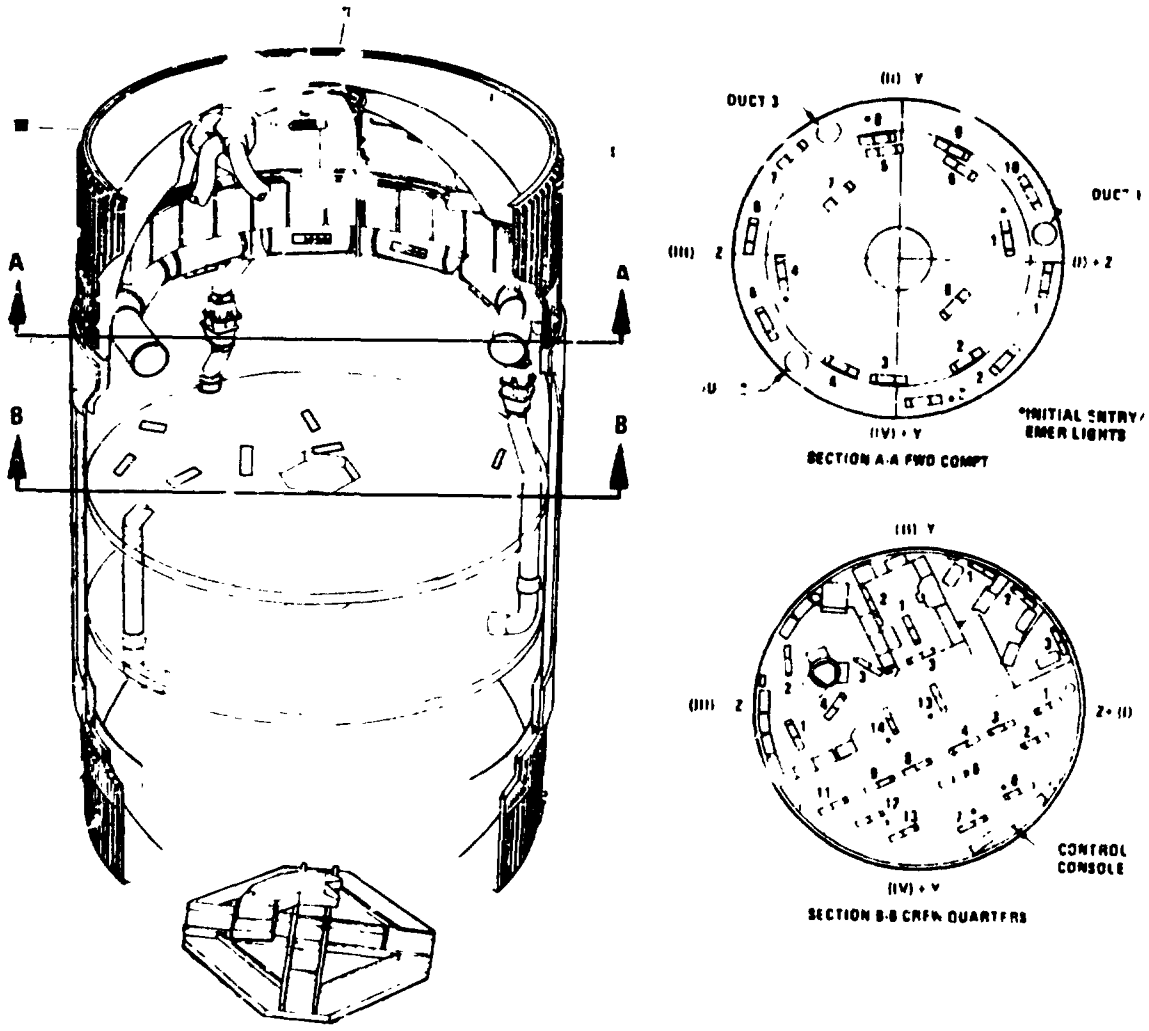


FIGURE 2.1.4.7-1 ILLUMINATION SYSTEM

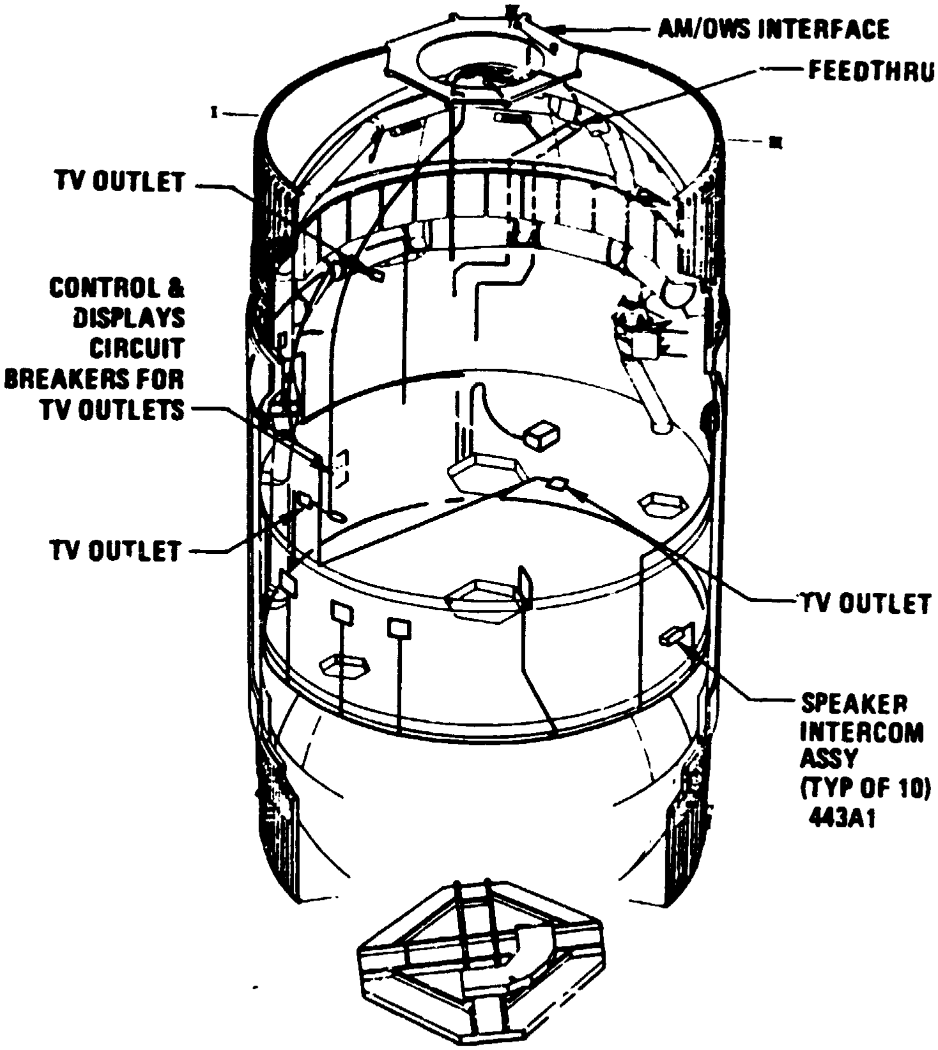


FIGURE 2.1.4.8-1 COMMUNICATIONS SYSTEM

The OWS Data Acquisition System consists of a portion of the Saturn Workshop Pulse Code Modulation Telemetry System, on-board displays and ground checkout support measurements. Low level and high level multiplexers, signal conditioning equipment and decoders are located in the forward skirt of the OWS. Signal conditioning equipment for transducers installed aft on OWS are mounted in the aft skirt (see Figure 2.1.4.8-2).

The OWS Command System provides automatic command capability for the first 7.5 hours of the mission. This is for control of tank pressures, thruster attitude control, solar array, meteoroid shield and refrigeration system radiator shield deployment, the activation of the refrigeration system, and certain AM/ATM/MDA functions. The design utilizes the S-IVB mainline switch selector, which receives command input logic from the IU. The AM Digital Command System serves as backup. Figure 2.1.4.8-3 shows the system installation on OWS.

2.1.4.9 Caution and Warning System (see Figure 2.1.4.9-1) - The caution and warning system for OWS is an integral part of the system for Skylab. The system provides visual displays and audible tones when selected parameters reach out-of-tolerance conditions. The parameters selected are those which could result in jeopardizing the crew, compromising mission objectives, or, if not responded to in time, could result in the loss of a system. The monitored parameters are categorized as Caution, Warning or Emergency parameters. The

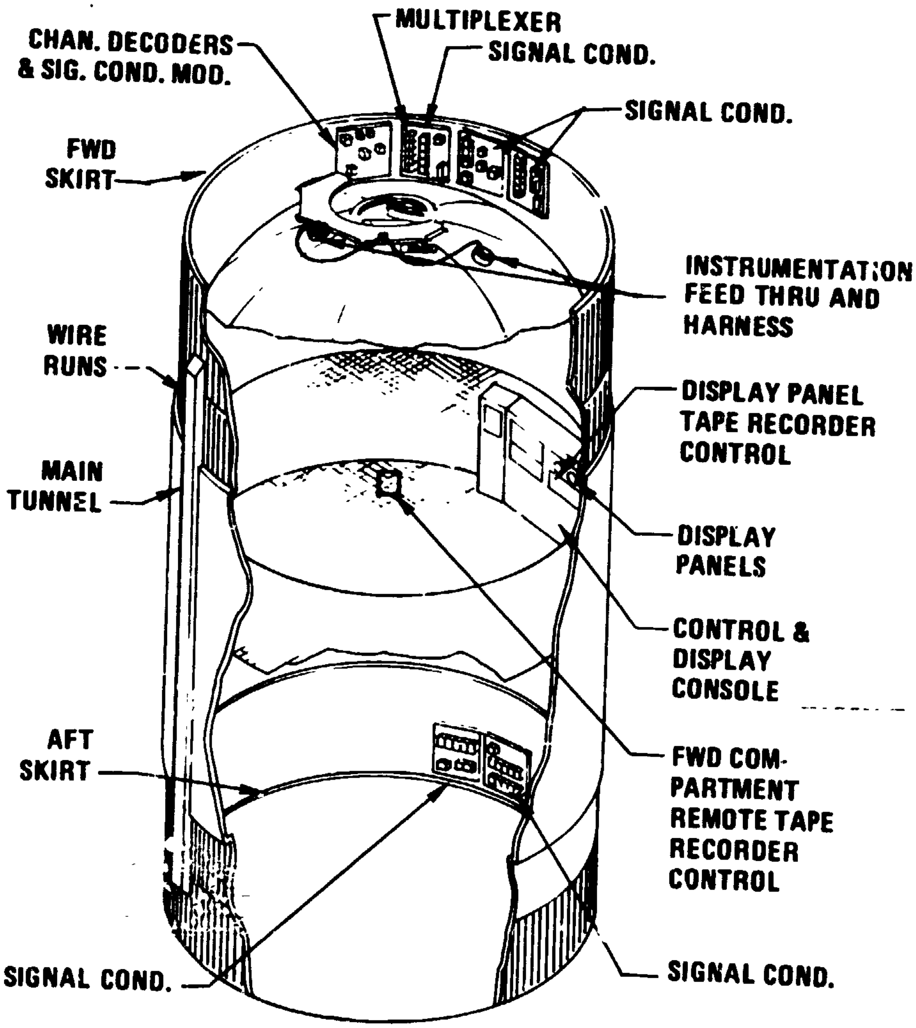


FIGURE 2.1.4.8-2 DATA ACQUISITION SYSTEM

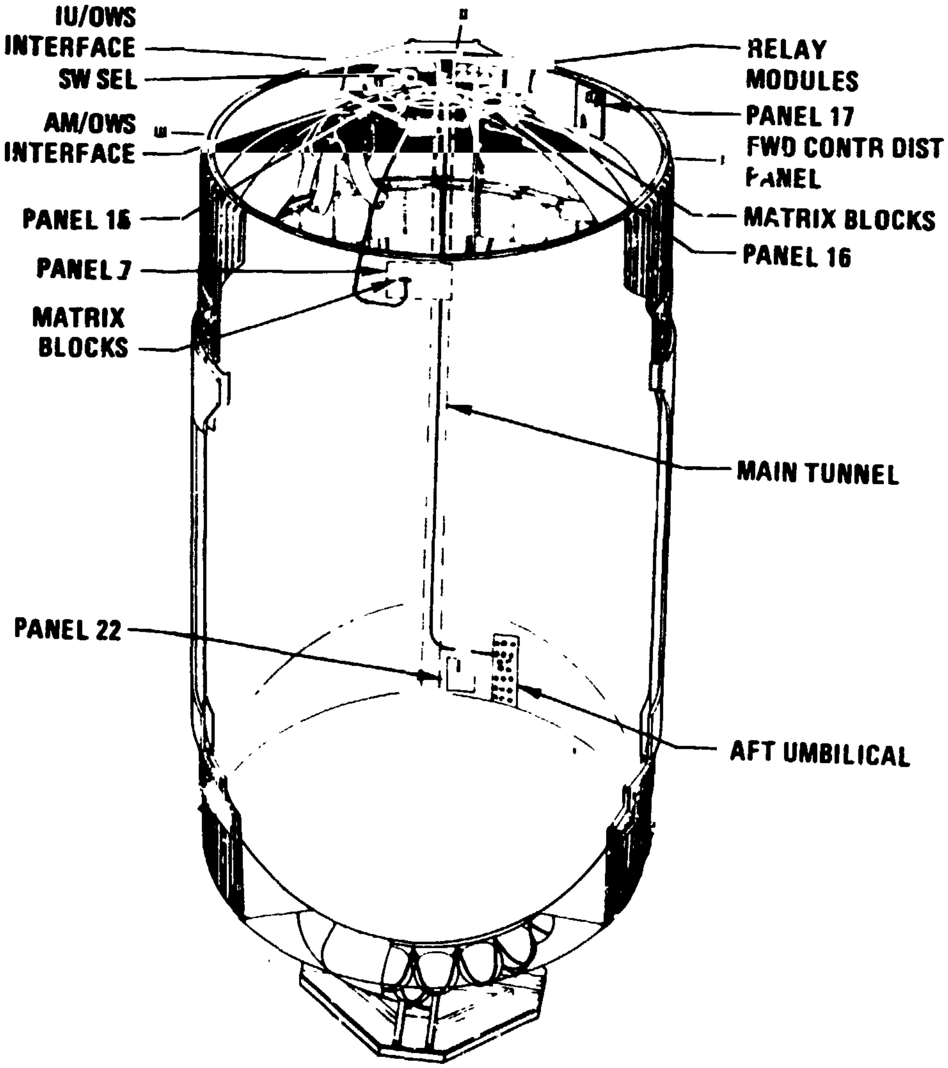


FIGURE 2.1.4.8-3 ELECTRICAL COMMAND SYSTEM

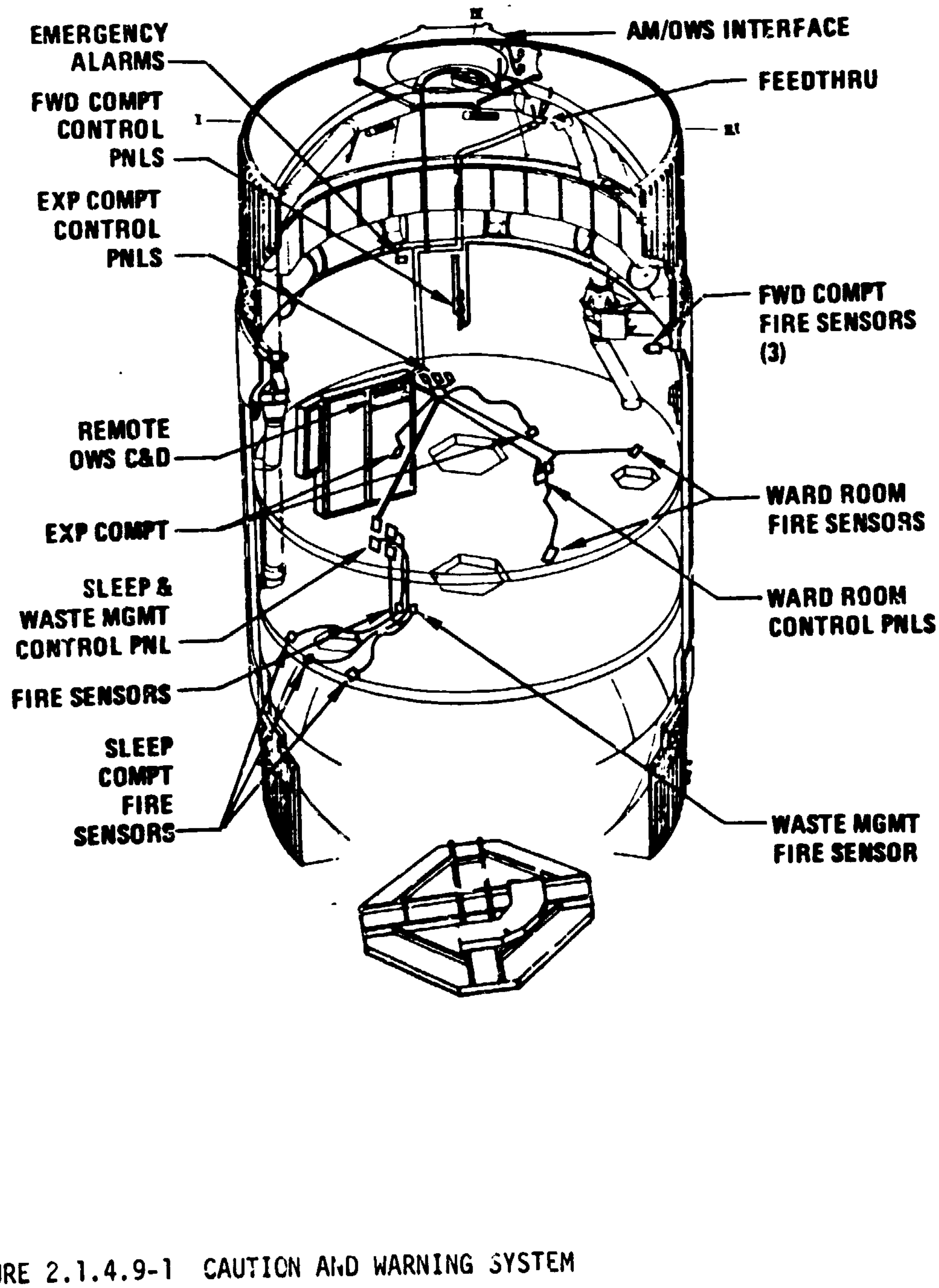


FIGURE 2.1.4.9-1 CAUTION AND WARNING SYSTEM

system is monitored in the Airlock Module; the OWS providing selected redundant displays for crew observance while in the experiment compartment. The OWS caution and warning panel is primarily a repeater station displaying the condition of selected cluster parameters. Six emergency, 2 caution and 2 warning parameters are displayed.

2.1.4.10 Habitability Support System - The OWS Habitability Support System consists of the following subsystems.

A. Waste Management System (WMS) (see Figures 2.1.4.10-1 and 2.1.4.10-2) - The waste management collection module houses the equipment used to collect feces and urine. Feces is collected in a bag using airflow into the bag to simulate gravity. The air enters the bag, passes through a hydrophobic filter and subsequently through an odor filter and blower and exhausted into the Waste Management Compartment (WMC). Urine is collected in a receiver and hose similar to an aircraft relief tube. A centrifugal separator separates the air from the urine. Air passes through the same odor control filter and blower as does the feces collection air and the urine is pumped by the separator into a four liter storage bag. In order to obtain samples to be returned for the medical experiment, the feces is vacuum dried in a waste processor and a urine sample of 120 ml is extracted from the storage bag and then placed in a freezer for storage. A vacuum cleaner is included in the waste management equipment. The same blower as used in the collection module

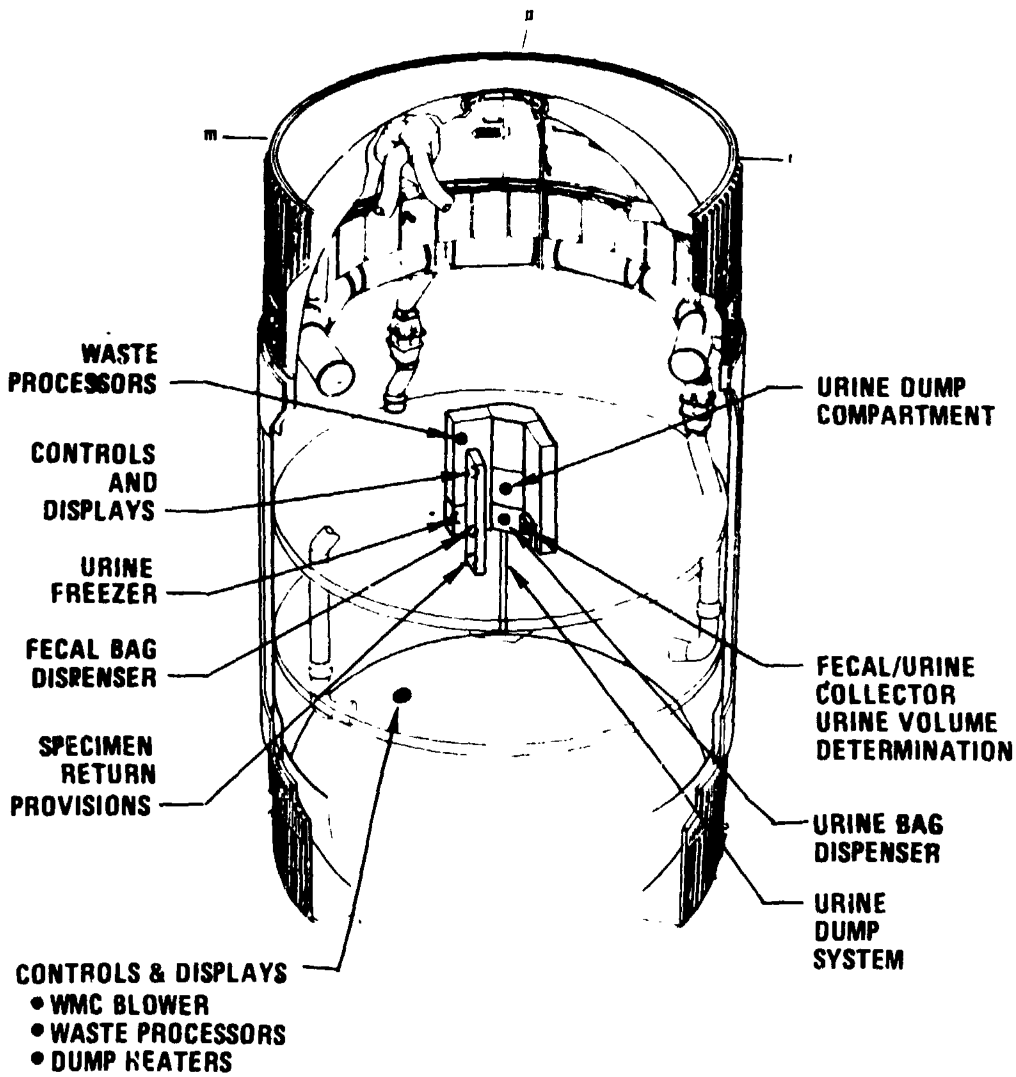


FIGURE 2.1.4.10-1 WASTE MANAGEMENT SYSTEM

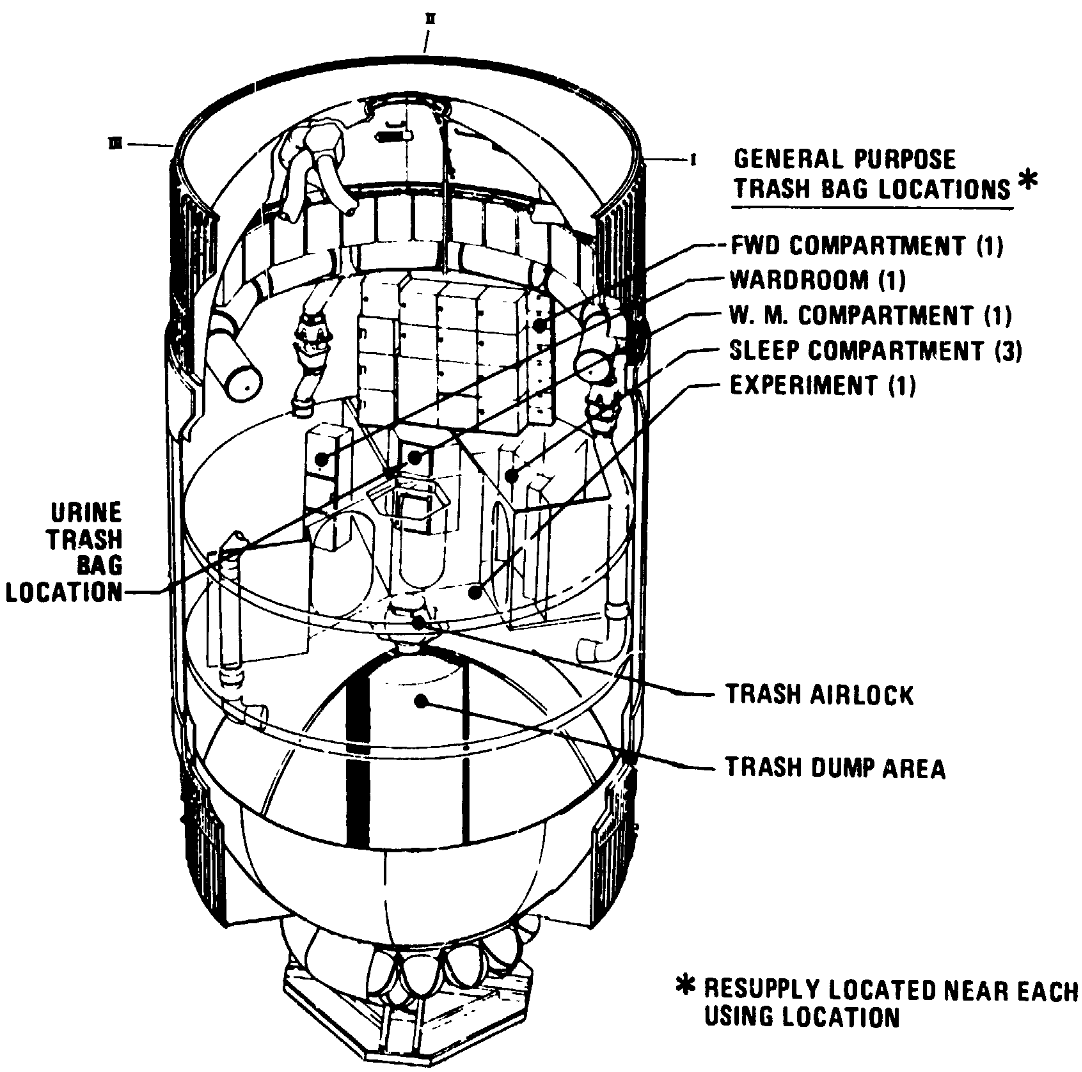


FIGURE 2.1.4.10-2 WASTE MANAGEMENT SYSTEM TRASH DISPOSAL

is used for suction. The vacuum cleaner uses a bag similar in operation to the fecal bag. The trash airlock is used to dispose of trash from the cabin into the waste tank. Trash is placed in a standard disposal bag; placed into the airlock and after closing the lid, the trash is ejected into the waste tank by a sizzor mechanism.

B. Water Management System (see Figure 2.1.4.10-3) - Water is stored in ten 600 pound (2721 kilogram) capacity stainless steel tanks. The tanks contain an integral stainless steel expulsion bellows, fill and drain ports, iodine and sample ports, level indicators and shutoff valves. The water is transferred by Teflon lined hoses to the wardroom for drinking water and to the WMC for personal hygiene water. In both compartments, the water is heated to the desired temperature. There is also a chiller in the WMC to supply chilled water for drinking. The hot water in the wardroom is used for food reconstitution and dispensers are available for both hot and chilled water. The water in each water storage tank is initially purified by using iodine as a biocide. The purity is maintained by periodically injecting iodine in the water. A portable water tank with a 26 pound (118 kilogram) capacity is provided for contingency water supply and also to support the water network fill and flush during activation.

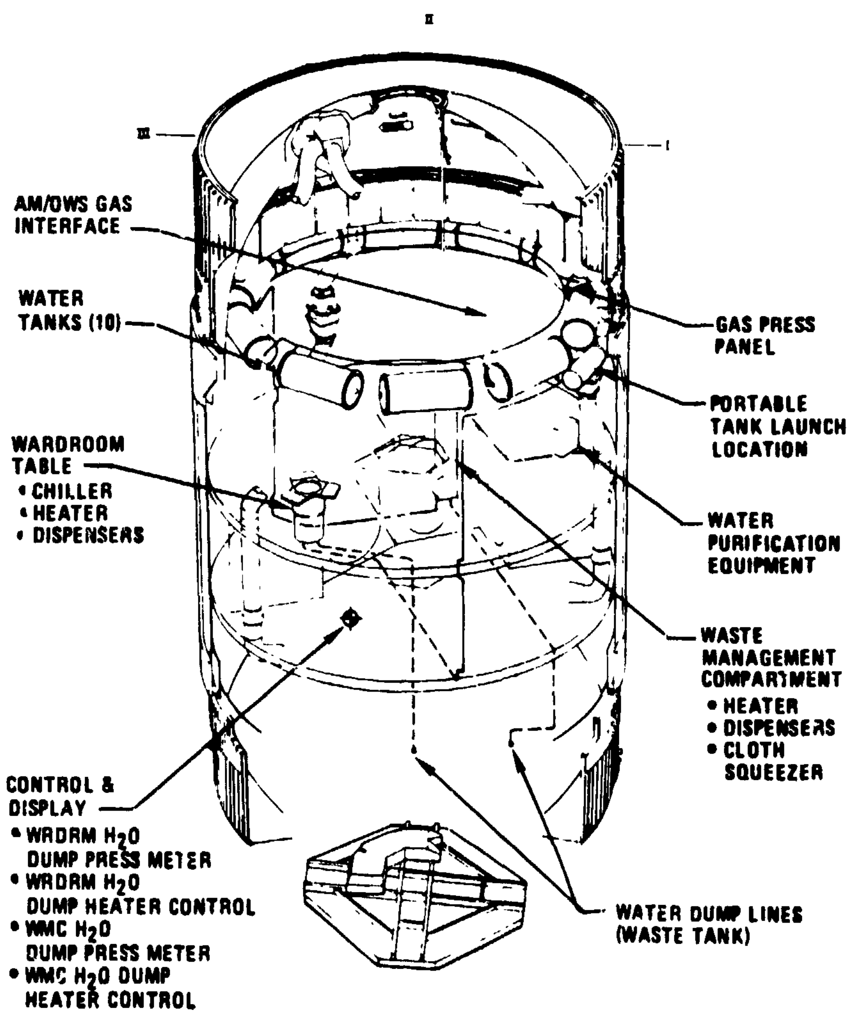


FIGURE 2.1.4.10-3 WATER SYSTEM

C. Personal Hygiene System (see Figure 2.1.4.10-4) - Personal hygiene equipment is provided for the maintenance of health and personal cleanliness. A personal hygiene module is provided to store supplies required by the crewmen. Dispensers for utility tissues, wash cloths, towels, and chemically treated cotton pads are also provided. The capability to dry wash cloths and towels is available.

D. Body Cleansing System (see Figure 2.1.4.10-4) - Body cleansing is accomplished both by the shower and by sponging with washcloths. A washcloth squeezer is provided. The shower contains an enclosure with a continuous airflow as a gravity substitute for moving water from the crewmen. A water bottle is filled from the WMC water dispenser and attached to the ceiling at the shower location. The water remaining after the shower is vacuumed and passed through a centrifugal air/liquid separator. The air is then filtered and pumped through a blower into the cabin.

E. Food Management System (see Figure 2.1.4.10-5) - The food management subsystem consisted of equipment and supplies required for the storage operation and consumption of foods. Food was stored in food boxes, galley trays, food freezers and a food chiller. A galley, components of the food table, food trays are provided for operation and preparation of the meals. Hot and chilled water are provided to reconstitute the dry food and chilled drinks. Food cans and beverage packs are grouped in menu form in food overcans. A heater tray is available to heat

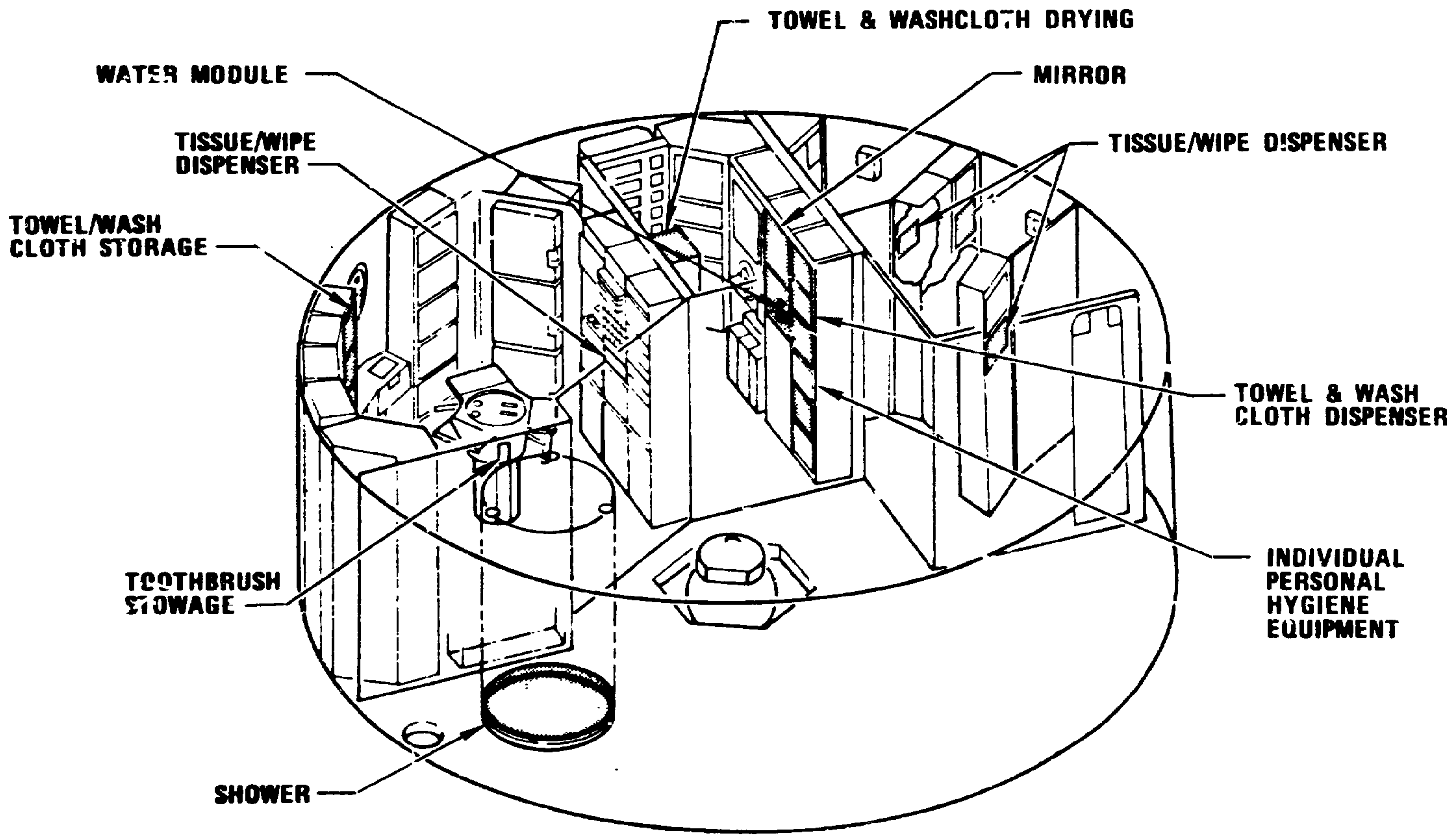


FIGURE 2.1.4.10-4 PERSONAL HYGIENE AND BODY CLEANSING SYSTEM

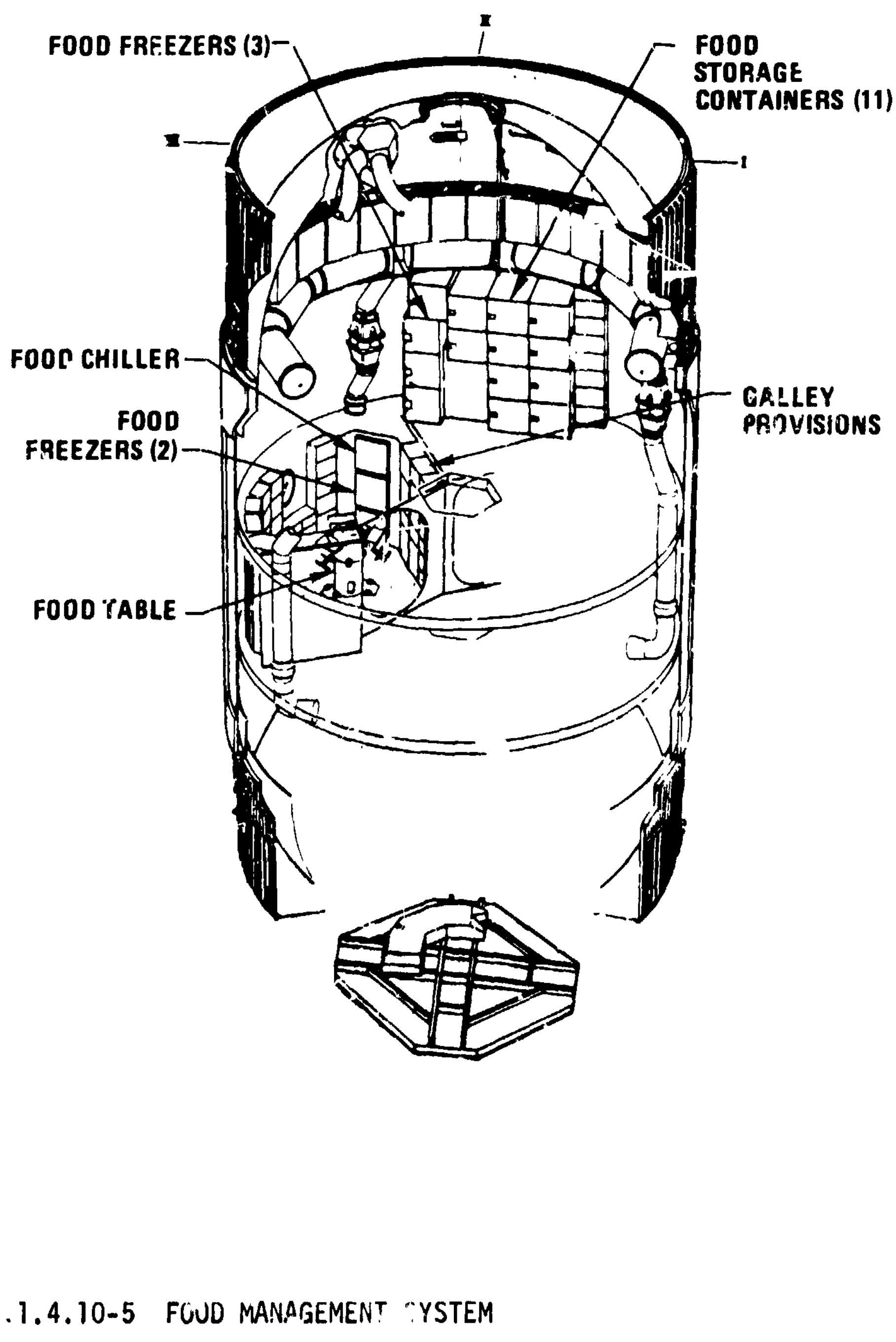


FIGURE 2.1.4.10-5 FOOD MANAGEMENT SYSTEM

the food during preparation of the meal.

F. Sleep Support System (see Figure 2.1.4.10-6) - Sleep restraints are provided for each crewman. They provided thermal comfort and body restraining capability. The sleep restraints are mounted on frames in the sleep compartment.

G. Suit Drying System - The suit drying equipment consisting of a blower, hoses and desiccant bags is provided to remove moisture from inside the pressure suits after each suited operation. Pressure suits are dried at three (3) suit drying stations located in the OWS forward compartment. Drying is accomplished by installing a suit in the drying station which consists of the PGA portable foot restraints (attached to the forward compartment floor) and a hanger strap which suspends the suit between the floor and the water ring foot restraints. The blower unit forces drying air through a hose and into the suit. Moisture is dried by the air and collected by the desiccant bags. The desiccant bags are subsequently dried in the WMC waste processor.

H. Refrigeration System (see Figure 2.1.4.10-7) - The OWS refrigeration system is a low temperature thermal control system utilizing Coolanol -15 as the refrigerant in a closed loop circuit. Heat is dissipated through a ground heat exchanger cooled by G.S.E. during prelaunch operations and by a radiator, externally mounted at the aft end of OWS, for orbital operations. The system provides food freezers and chillers for food and water in support of habitability and urine freezers and chillers

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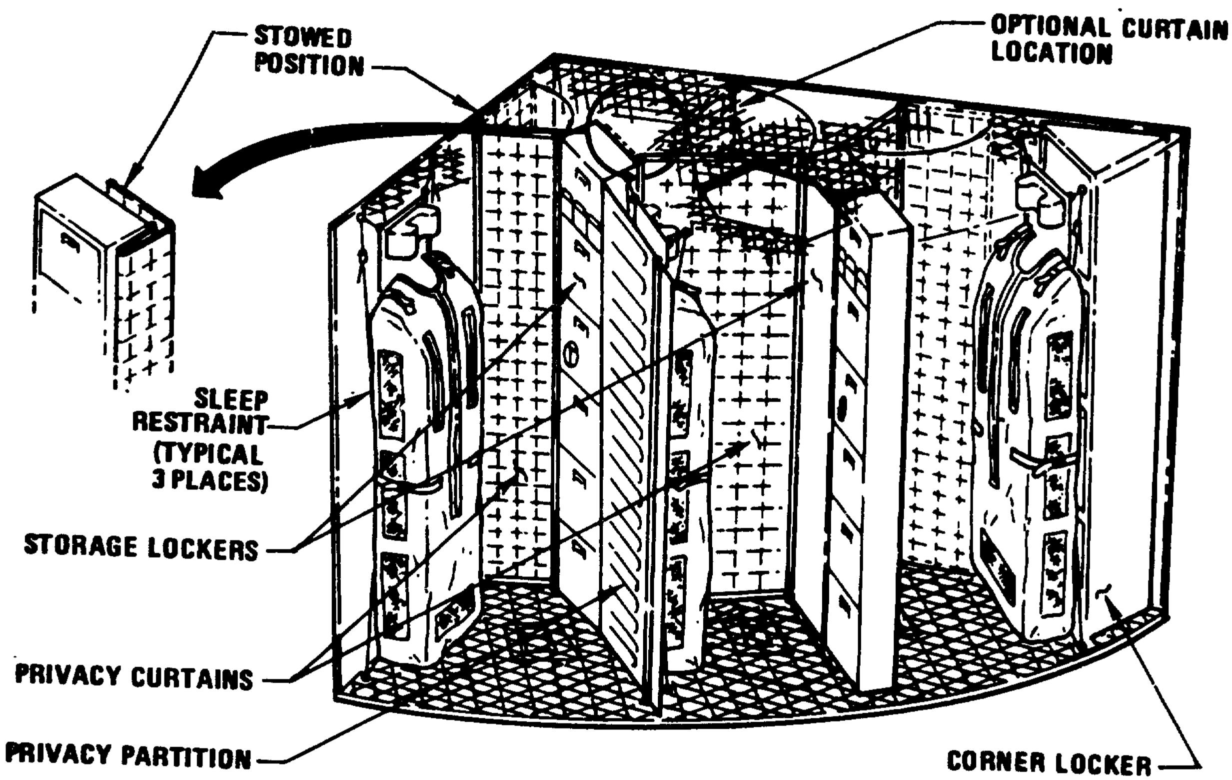


FIGURE 2.1.4.10-6 SLEEP SUPPORT SYSTEM

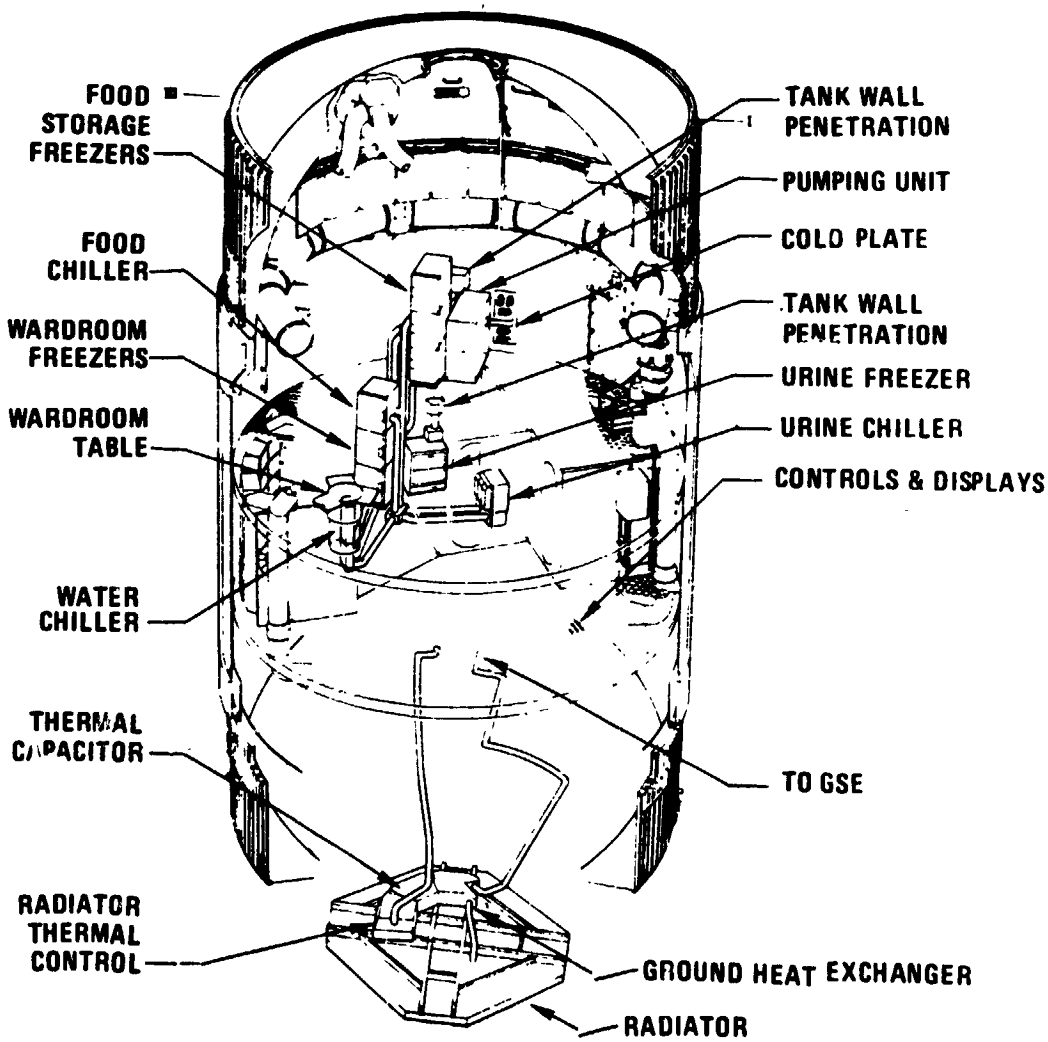


FIGURE 2.1.4.10-7 REFRIGERATION SYSTEM

in support of the bio-medical experiment. The system has dual coolant loops and redundant components to provide reliability. It controls temperatures through a range of +42°F to -20°F (5.6°C to -29°C).

I. Atmosphere Control System - The OWS which is pressurized to 26 psia (179kN/m^2) with N_2 in both the crew habitation area and waste tank for launch is vented after orbit insertion; the waste tank to vacuum and the habitation area to 1.35 psia (9.3kN/m^2) partial pressure N_2. The habitation area is then repressurized to 5 psia (34.5 kN/m^2) with O_2 to provide the desired breathing atmosphere. Section 2.1.4.3, Environmental/Thermal Control, discussed both the passive and active systems of control over cabin atmosphere temperatures. The active being provided by heat exchangers in the Airlock Module and convective heaters in the three OWS ventilation ducts. The circulated cabin gas is reconstituted in the Airlock Module. The ventilation ducts, each with a circulation fan cluster, route reconstituted air to a plenum chamber aft of the aft floor in OWS for diffusion through floor diffusers into the cabin.

2.1.4.11 Stowage System - Stowage capability for provisions is included throughout the OWS (see Figure 2.1.4.11-1). Twenty-five standardized stowage containers in the forward dome and 16 standard stowage lockers located in the various areas accommodate general provisions such as clothing, sleeping restraints, urine collection bags, etc.

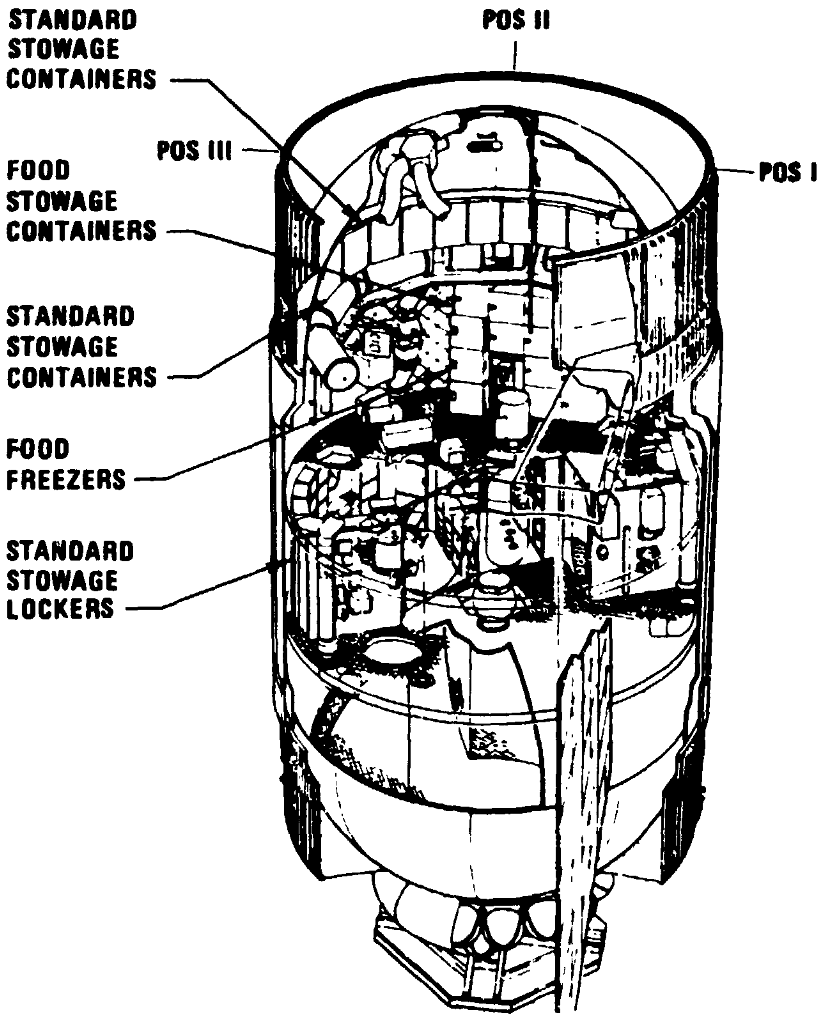


FIGURE 2.1.4.11-1 STOWAGE SYSTEM

For ambient food storage, 11 containers in the forward compartment and 2 galley cabinets are provided. Five food freezers, 3 in the forward compartment and 2 in the wardroom are installed. A refrigerator for perishable food is located in the wardroom and a urine freezer is included in the waste management compartment. The total stowage capability of the 210 containers on board is 580 ft^3 (16.4 M^3).

2.1.4.12 Experiment Accommodations - For OWS experiments, hardware accommodations necessary to integrate experiment equipment and perform the experiments are provided. These consist of structural attachments, electrical cabling, pressurization and vacuum plumbing, and stowage restraints. A pair of scientific airlocks, anti-solar and solar, are installed in the cylindrical tank walls of the habitation area in the forward compartment to provide visual and physical access outside for experiments requiring it. Figure 2.1.4.12-1 shows the vacuum provisions for the waste management system; the vacuum access is through the waste tank to utilize the non-propulsive venting system of the waste tank. Figure 2.1.4.12-2 shows the vacuum provisions to accommodate the metabolic activity and lower body negative pressure experiments.

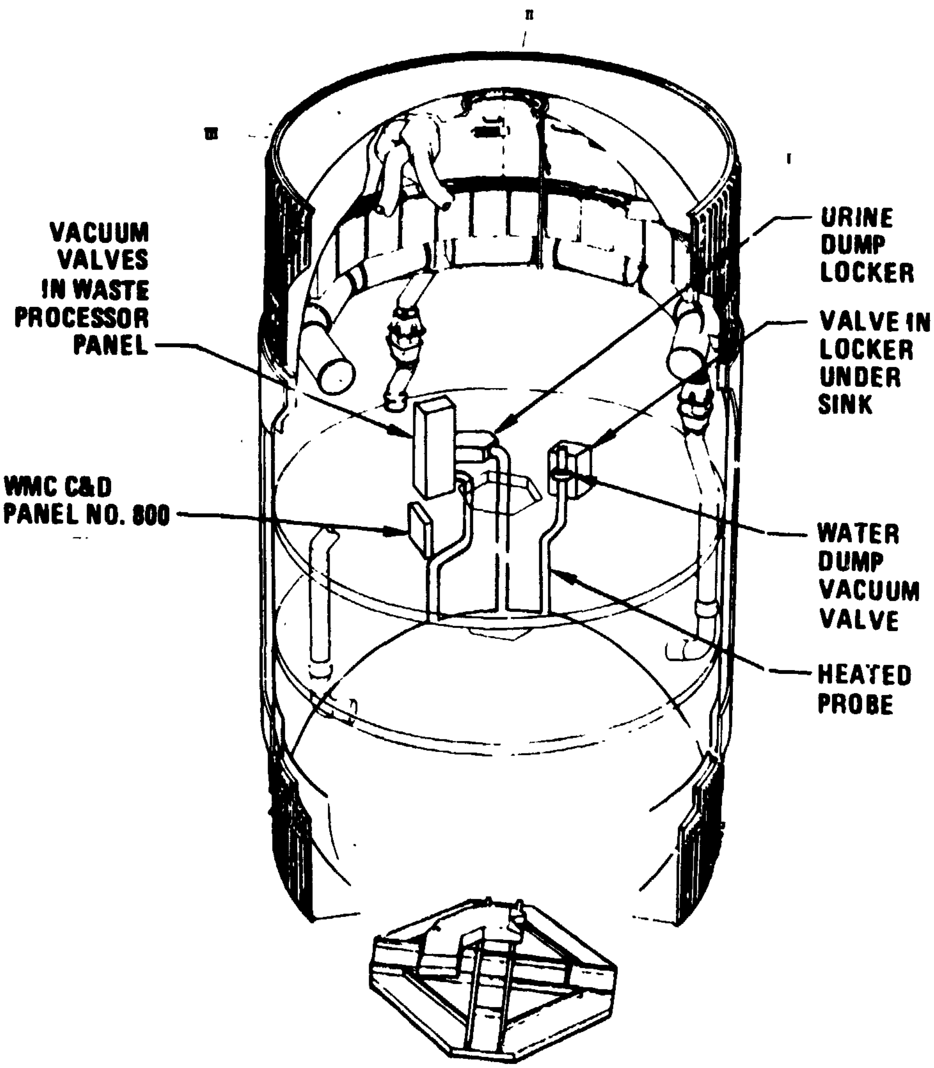


FIGURE 2.1.4.12-1 WASTE MANAGEMENT VACUUM PROVISIONS

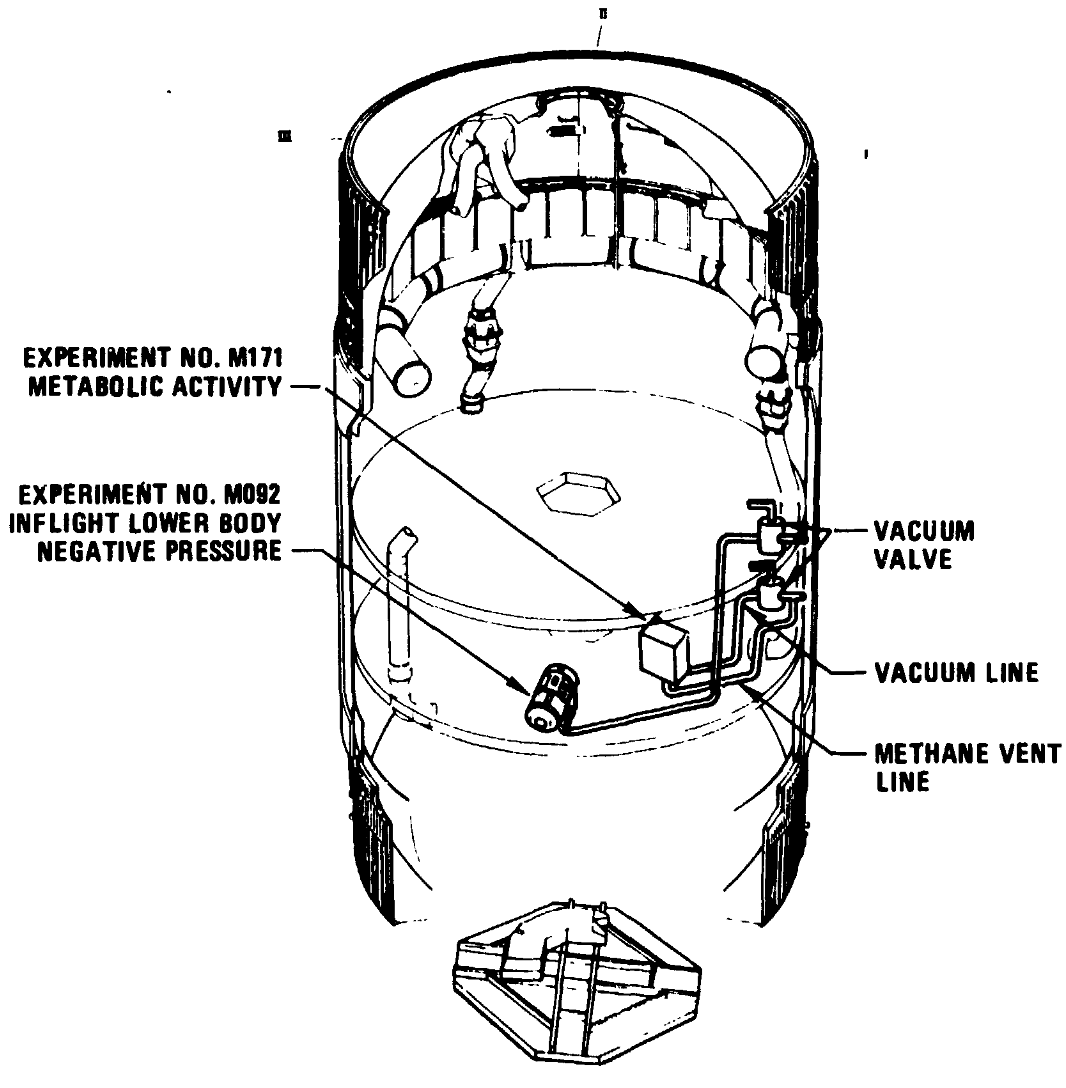


FIGURE 2.1.4.12-2 EXPERIMENT VACUUM PROVISIONS

2.1.5 Mission Performance

The launch of SL-1 occurred on 14 May 1973. Inadequate venting of pressure in the auxiliary tunnel portion of the meteoroid shield caused the shield to be torn away from OWS during early ascent. As the shield was ripped away, the structural tie-downs for Solar Array System (SAS) beam fairing 2 were severed. The unsecured SAS wing was separated from OWS by impingement forces from the retrorocket plumes at payload staging. Further, remnants of the torn shield remaining in the vicinity of SAS beam fairing 1 prevented the normal programmed orbital deployment of the array. This array, however, successfully deployed when the SL-2 crew, through Extra Vehicular Activity (EVA), broke away the deployment restraints.

The loss of the meteoroid shield caused a reduction in the probability of no pressure loss from the OWS habitation area from 0.995 during a minimum period of eight consecutive months to 0.985 during a minimum period of fifty-six consecutive days. The passive thermal protection afforded OWS by the shield was lost.

SL-2 launch scheduled for 15 May 1973 was delayed ten days during which time contingency actions to rectify the anomalous condition of SL-1 were planned and tested. Over this ten day period, the goldized OWS exterior was exposed to direct solar input. Internal temperature and temperatures of the ambient food and film rose. The internal temperature rise was rapid for approximately 1-1/2 days after orbit insertion. The change rate decreased consistent with the orbital attitude of the spacecraft which was generally Solar Inertial (S.I.)/50° pitch effected to provide temperature control. During this period, the floor food containers and OWS internal temperature was approximately 125°F (52°C), ambient food rack containers approximately 130°F (54°C), and the film vault approximately 120°F (48°C).

Also due to the loss of the meteroid shield, OWS external and internal surface temperatures approached 300 and 200°F (148°C and 93°C), respectively. These elevated temperatures caused the internal insulation to be subject to outgassing of hydrocarbon products. For this reason, the OWS atmosphere was vented five times prior to habitation. It was felt that some portion of the insulation had become debonded; however, inspection by the SL-2 crew determined the degree to be negligible. The loss of the meteoroid shield impaired the passive thermal control system; however, after the deployment of the JSC parasol, the OWS insulation system, i.e., aluminized mylar high performance insulation (HPI), external on the forward dome and the polyurethane foam internally, provided a habitable structure for the duration of the Skylab missions.

Generally, OWS systems performed in accordance with design expectations. For SL-1/SL-2 they functioned in the primary mode of operation; no backup capability was utilized.

The high temperatures encountered early in the SL-1/SL-2 mission, prior to deployment of the JSC parasol, which provided a shade for the sun side of the OWS Habitation Area, did generate some unique difficulties which were not really unexpected. The five sensors in sleep compartment 2 gave three false alarms due to increased sensitivity resulting from high temperatures. Water tank 1 iodine content was low after iodine inspection. This was expected based on test data. Ventilation duct 1 flow meter failed, perhaps as a result of high temperatures. These were minor problems that did not degrade the mission.

Several operational problems developed that were solved without mission degradation by using modified procedures; e.g., disposal of large quantities of items through the trash airlock, leakage of habitation atmosphere through the trash airlock due to the operating handle being in a wrong position and fogging of the wardroom window. The more significant problems encountered are identified below by OWS subsystem.

A. Refrigeration - On day of year (DOY) 173, following SL-1/SL-2 deactivation, data indicated simultaneous refrigerant flow through the radiator and by-pass leg of the system primary loop. The by-pass valve was cycled by flight controllers which improved the loop performance. Notwithstanding the split-flow degraded mode of operation, adequate temperature control throughout the system was provided by the primary loop.

   The coldest food freezer temperature dropped below the Contract End Item (CEI) specification limit of -20°F (-28°C) on DOY 271 and did so several times during the storage period between SL-3 splashdown and SL-4 launch. This is attributed to cabin temperatures dipping to the low 60°F's (15°C's). The freezer specification limit was for system design, the food being capable of withstanding much colder temperature. The food, in fact, was stored onboard OWS at -40°F (-40°C) at KSC. Therefore, the freezer low temperatures between -20°F and -25°F (-28°C and -31°C) had no deleterious effect on the food.

B. Electrical - Thermal analyses indicated that due to a worst case beta angle during the SL-1 mission, mounting bases for some components experienced temperatures as much as 20°F (11.1°C) higher than design maximum. However, laboratory examination of the components involved revealed that no problems would be encountered.

C. Instrumentation and Communications - The low level "B" multiplexer commenced intermittent operation on DOY 215. Mission data was not impacted, however, since we provided alternate sources for all data measurements. Thus, on-board troubleshooting of the malfunction by the Skylab crews was never recommended.

The SL-1 launch anomaly which caused the loss of SAS beam fairing 2 destroyed 50 percent of the OWS electrical power generation capability. This in no way jeopardized Skylab power systems and though some experiment activity was curtailed due to contingency power management early in SL-2, the total planned experiment activity for the program was exceeded. Due to design margin and reasonable conservatism, SAS wing 1 provided 58 percent of the total SAS power requirements and showed little detectable degradation in power output throughout the Skylab missions.

An OWS subsystem, the meteoroid shield, in part intended to provide passive thermal protection for the OWS failed during launch.

However, another OWS system, Thruster Attitude Control System (TACS), was used to maneuver the spacecraft to a more favorable attitude for thermal control, thus, substantially contributed to "salvaging SL-1."

OWS TACS successfully fulfilled all vehicle control demands imposed on it throughout the SL-1/SL-2 mission. It was the primary attitude control system for the SL-1 payload following SII separation until the Control Moment Gyro's (CMG's) were sufficiently spun up to permit transfer to Apollo Telescope Mount (ATM)/CMG control. Following transfer to CMG control, the TACS continued to function as a supplemental system to correct large attitude error rates and provide momentum relief to the CMG's. Further, TACS impulse consumption significantly exceeded the predictions for a nominal mission profile. The excessive usage is attributed to SL-1/SL-2 mission anomalies delineated as follows:

- "ATM/CMG" switchover occurred 10 hours later than scheduled due to CMG anomalies.
- Loss of the meteoroid shield imposed several "unplanned" attitude maneuvers for Skylab thermal conditioning.
- Excessive CMG "reset [illegible] ings" were performed while maintaining unusual vehicle attitude.

- Large vehicle perturbations were associated with Stand-up Extra Vehicular Activity (SEVA) and EVA to deploy SAS wing 1.
- Several unsuccessful "hard dockings" were attempted by the SL-2 CSM.

The TACS total usuable impulse at liftoff was approximately 80,000 lb-sec (356,000 N-sec). Impulse consumption to DOY 174 was approximately 45,000 lb-sec (200,000 N-sec), compared to the maximum predicted usage of about 16,000 lb-sec (71,000 N-sec). This extensive use of TACS impulse presented no concern for the completion of the Skylab missions since, after arresting the anomolous condition of SL-1/SL-2, the impulse usage was nominal. No detectable system leakage was observed from a series of periodic mass calculations throughout the completion of the mission profile.

SL-3 was launched 28 July 1973. After CSM docking, Skylab was activated; the activation of OWS was normal. The MSFC twin-pole sunshade was EVA deployed by the crew on DOY 219 as a backup for the JSC parasol, thus re-establishing the desired thermal control of OWS. OWS systems continued to perform in the primary mode. SL-3 splashdown occurred on September 25, 1973, completing the 59 day mission.

SL-4 launch occurred 16 November 1973. The mission stay time was increased from the planned 56 day mission to 84 days. The CSM was docked to the Orbiting Assembly on the third docking attempt and SWS was activated. OWS activation was normal. All OWS systems performed as required for SL-4 plus meeting the increased demands of the longer mission notably well at the end of the mission. SL-4 splashdown was on 8 February 1974 completing the Skylab program missions.

2.2 SYSTEMS

2.2.1 Structural System

2.2.1.1 Forward and Aft Skirt Structure

The OWS forward and aft skirts were SIVB/Saturn V 515 Vehicle structural Assemblies modified to meet Skylab requirements.

A. Design Requirements

1/ Withstand vehicle body bending, shear, and axial loads and transfer these loads between adjacent shell structures with no yielding that would impair function at design yield load and no failure or instability of structure at design ultimate load.

On new hardware or modified existing hardware when it could be expedient to preclude testing use an ultimate factor of safety of 3.00 and a yield factor of 2.00. Where capability could be verified by test or extensive analysis use an ultimate factor of safety of 1.40 for manned condition and an ultimate factor of safety of 1.25 for unmanned.

2/ Provide structural support for mounting electrical/electronic and other functional equipment to withstand prelaunch, launch, boost and orbital environments.

3/ Provide structural attachment and support structure for SAS wing assemblies capable of transferring load from SAS to skirt during liftoff, boost and orbit operations.

4/ Provide thermal shield to prevent excessive heat transfer between habitation area and skirts.

5/ Provide aerodynamic shaped external tunnels to accommodate cables and tubing from and between forward and aft skirts.

6/ Provide umbilical installation structural support for ground to vehicle connections required during prelaunch operations.

7/ Provide support for TACS nozzles and control valves on the aft skirt.

8/ Provide separation joint for inflight separation of the SII stage from OWS at aft skirt-to-aft interstage interface.

B. System Description - The forward skirt, as defined by Drawing 1B77164 (Figure 2.2.1.1-1), was a modified SIVB/V cylindrical shell 260 in. (660.4 cm) in diameter and 122 in. (309.9 cm) long. The basic cylinder was a semimonocoque aluminum alloy structure of sheet-metal skin, external longitudinal stringers, and internal ring frames. The forward interface flange of the cylinder provided for a bolt, field-joint type connection to the IU in compliance with 13M50202 physical ICD, 13M07002 IU functional ICD, and 13M03002 procedural ICD. The skin of the forward skirt was 0.032 in. (.812 mm) thick, 7075-T6 clad sheet. The external longitudinal stringers were made from 7075-T6 hat-section extrusions. The frame at the aft end of the cylinder was a formed 7075-T6 angle extrusion. It had an outward flange which provided for the bolted connection to the mating flange of the tank cylinder. The frame at the forward end was built up of formed 7075-T6 sheet metal and extruded parts. Two of the three intermediate frames were constructed of formed 7075-T6 sheet, while the aft intermediate frame was formed from a large 7075-T6 hat-section extrusion.

Inflight venting of the volume enclosed in the forward skirt, FAS, and the payload shroud was accomplished by four pairs of existing SIVB/V elongated holes spaced approximately 90° apart around the circumference of the forward skirt. Each pair of holes in adjacent stringer bays had an actual area of 37.5 in^2 (241.95 cm^2) such that

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
FORWARD SKIRT

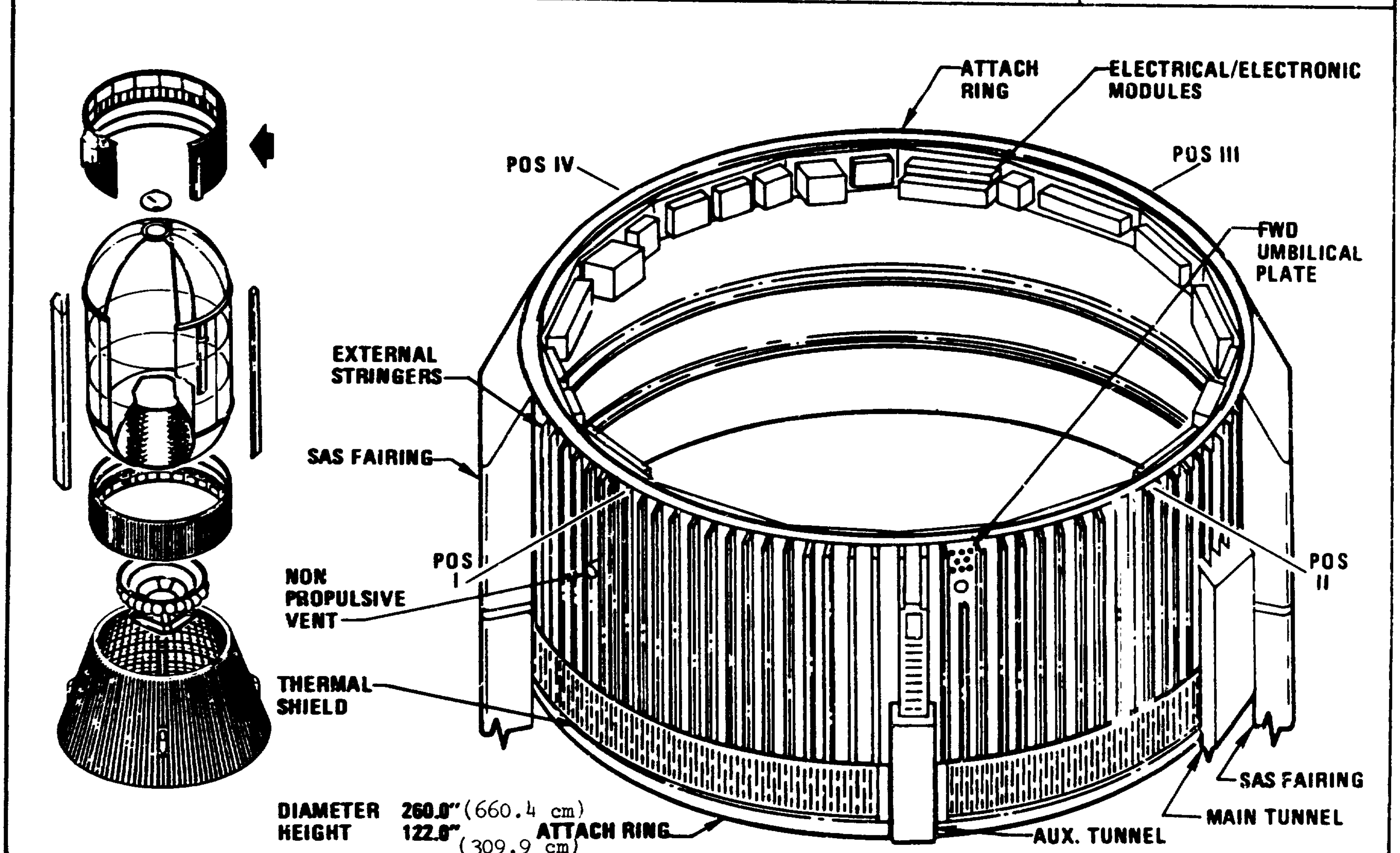


Figure 2.2.1.1-1

the total vent area was 150 in^2 (967.8 cm^2). Also, each pair of openings had internally mounted rain shields to restrict water from entering the interior of the forward skirt.

Th forward skirt umbilical structure was identical to the S-IVB/V configuration, except for the closing of two unused holes in the umbilical panel. The installation consisted of a sheet metal panel which replaced the skin in a structurally-reinforced area at the forward end of the skirt, just aft of the IU interface. It occupied the width of two stringer bays and was approximately 28 in. (712 cm) in length. The 1A72749 umbilical panel was fabricated from 0.100 in. (.254 cm) thick 7075-T6 aluminum sheet. The panel was formed flat in the central area where the holes for the connectors are located. It was riveted to the skirt basic structure, replacing an equivalent portion of the skin and the end of one stringer. The rigidity of the installed panel was increased by internal horizontal stiffeners riveted to the panel. The skirt structure surrounding the umbilical installation was strengthened by additional internal ribs and intercostals.

Electrical/electronic equipment was mounted identical to SIVB/V on panels which in turn were attached through vibration isolators to internal structural support skirt intercostals. The panels were of the same honeycomb design as used on the S-IVB/IB/V stages. Laminate fiberglass epoxy face sheets, 0.032 in. (.812 mm) thick, were bonded to a 1.125 in. (2.86 cm) thick, glass fabric reinforced phenolic honeycomb core. Glass fabric, 0.063 in. (1.60 mm) thick, volan-finish, "C" channels were attached at the forward and aft ends of the panel. Delron inserts were installed in the panel for mounting the electrical components.

These panels were mounted on two or three isolators, depending on the size of the panel, and weight of the electrical components. The isolators, in turn, mounted to the forward skirt structure. The isolators of the same design as used on S-IVB/IB/V stages consisted of an elastometer molded in between two aluminum extrusions with mounting studs (Figure 2.2.1.1-2).

Some of these panels were thermally insulated on the inboard side and some on the outboard side with a low-emissivity shroud. These shrouds consisted of two to four layers of goldized Kapton (metalized polyimide film) with a polyester net fabric between each layer of goldized Kapton. The shrouds were assembled to the panel by velcro or tie strings (Figure 2.2.1.1-3).

The SAS attachment fittings were incorporated into the 1B77164-1 forward skirt as integral parts of the structure and cannot otherwise be identified by a single drawing number. Design changes to the forward skirt included the addition of sheet metal channels to strengthen the frames at Stations 3161.6 and 3193.6, machined intercostals between the two frames, and sheet metal ribs, all of which served as internal backup structure at the SAS forward fairing attach points. Externally mounted fittings were installed at these same locations to receive the concentrated loads and transfer them to the backup structure. Further aft on the forward skirt, just forward of the tank joint, externally mounted fittings were located. The ordnance tie-down/release assemblies were attached to these fittings. A major design change to the basic

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
FORWARD SKIRT - PANEL INSTALLATION

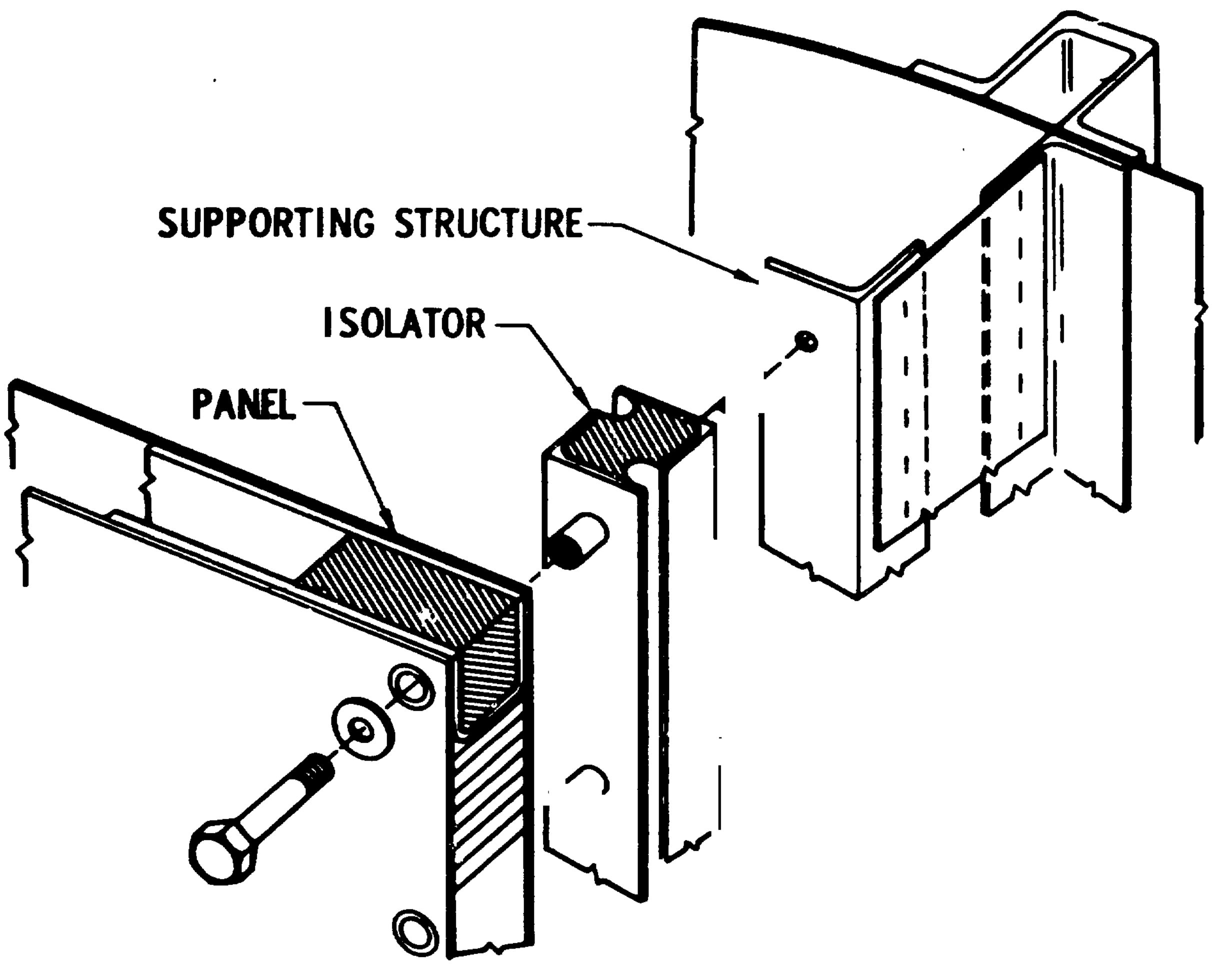


Figure 2.2.1.1-2

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
AFT SKIRT - PANEL INSTALLATIONS

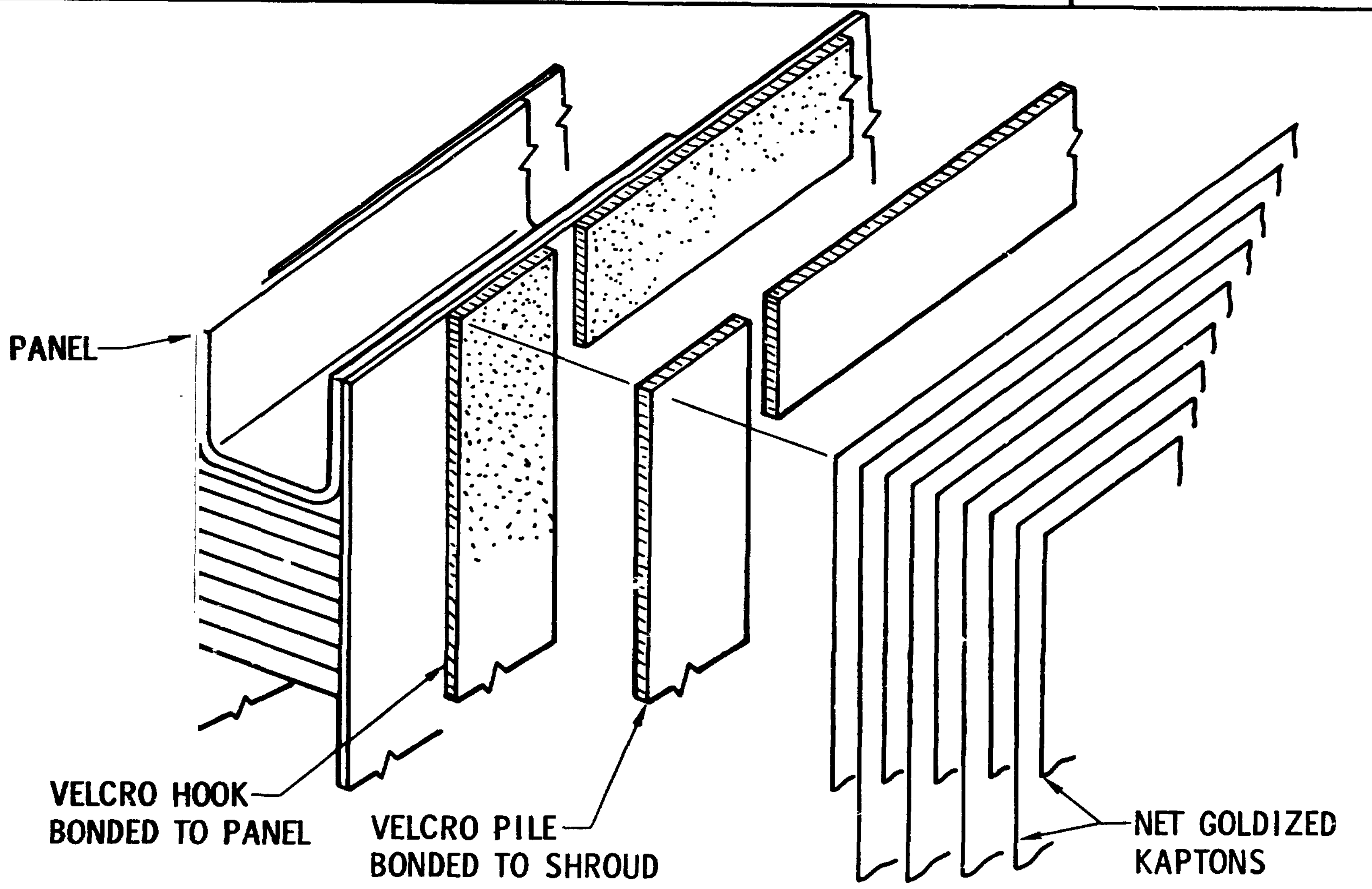


Figure 2.2.1.1-3

structure of the forward skirt was the replacement of the sheet metal skin with a chemically milled panel at each of the two SAS locations. The thickness of the chemically milled panel ranged from 0.032 to 0.122 in. (.812 to 2.84 mm), providing increased strength and stiffness in the areas subjected to SAS loads.

The thermal shield installation, Drawing 1B79502, was added to the structure which extended forward from the tank joint to the first intermediate forward skirt frame, a length of about 31 in. (78.74 cm). It consisted of panels which formed a continuous band around the circumference of the forward skirt, except where interrupted by the SAS wing assemblies. The shield was made up of beaded aluminum sheet-metal panels which were attached to fiberglass hat section supports mounted circumferentially on the forward skirt structure at 4 stations 6.78 in. (17.22 cm) apart. The hat-section supports were riveted to the outer face of the forward skirt stringers, and the beaded shield panels were riveted to the outer face of the fiberglass supports. The shield was thus located radially about 1.5 in. (3.81 cm) outward of the forward skirt skin.

The area immediately forward of the tank joint for a length of approximately 7.0 in. (17.78 cm) was not covered by the aluminum panels. To provide removable panels for access, and also to minimize the amount of heat transferred to the shield by conduction and radiation from the tank joint flanges, fiberglass panels were used for the thermal shield in this area. At the forward end of the beaded aluminum portion of the shield, sloping panels of fiberglass were installed between each pair of stringers to provide a ramp-like transition between the forward skirt skin and the surface of the thermal shield.

The OWS aft skirt, a modified S-IVB/V aft skirt (Figure 2.2.1.1-4), was a cylindrical structure, 260 in. (660.4 cm) in diameter and 85.5 in. (217.17 cm) long. The typical skin was .040 in. (1.016 mm) thick, 7075-T6 clad sheet. A typical stringer was a 1.375 in. (3.493 cm) high, 7075-T6 aluminum extrusion hat section. The stringers were machined to reduce weight in the intermediate bays where the full section was not required for strength.

The ring frames in the aft skirt consisted of two I section frames made from 7075-T6 aluminum extrusion tees, with 7075-T6 clad sheet web, and one 7075-T6 aluminum hat section extrusion frame. (See Figure 2.2.1.1-5).

The forward interface angle was machined from a 7075-T6 aluminum extrusion and bolted to the habitation area tank attach angle. The aft interface angle was a machined angle which was part of the separation joint.

The aft skirt umbilical was identical to the S-IVB/V aft umbilical except for minor changes to accommodate revised connectors for the OWS system.

Two connector locations had been modified for the OWS refrigeration lines. Connector locations not used were plugged. These included the holes for the LOX fill and drain system, LH_2 fill and drain system, and four fluid connectors. Two pivot points on the aft interstage and three umbilical carrier support fittings supported the umbilical carrier and transferred the carrier loads to the OWS structure.

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
AFT SKIRT

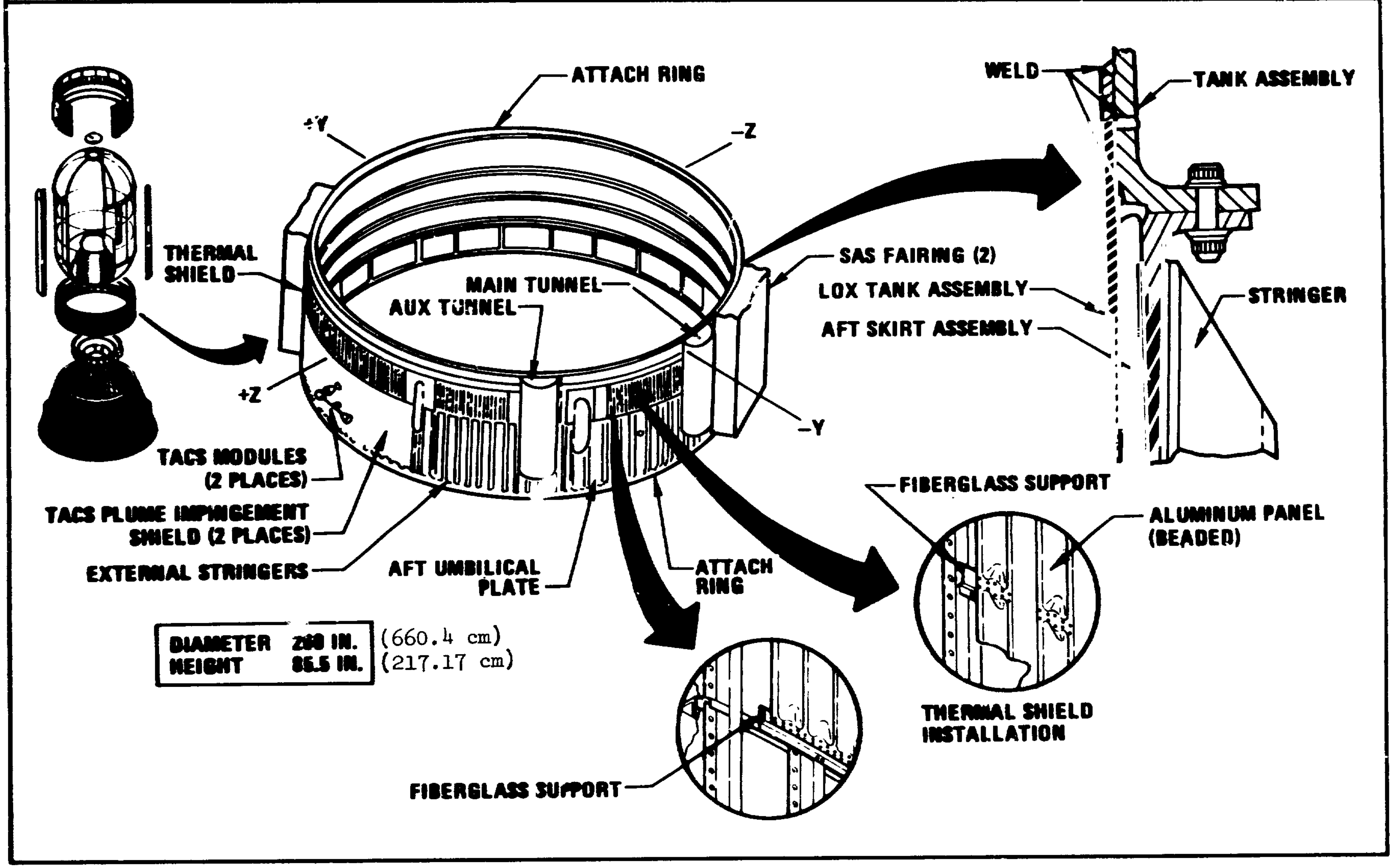


Figure 2.2.1.1-4

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
TYPICAL AFT SKIRT SECTION

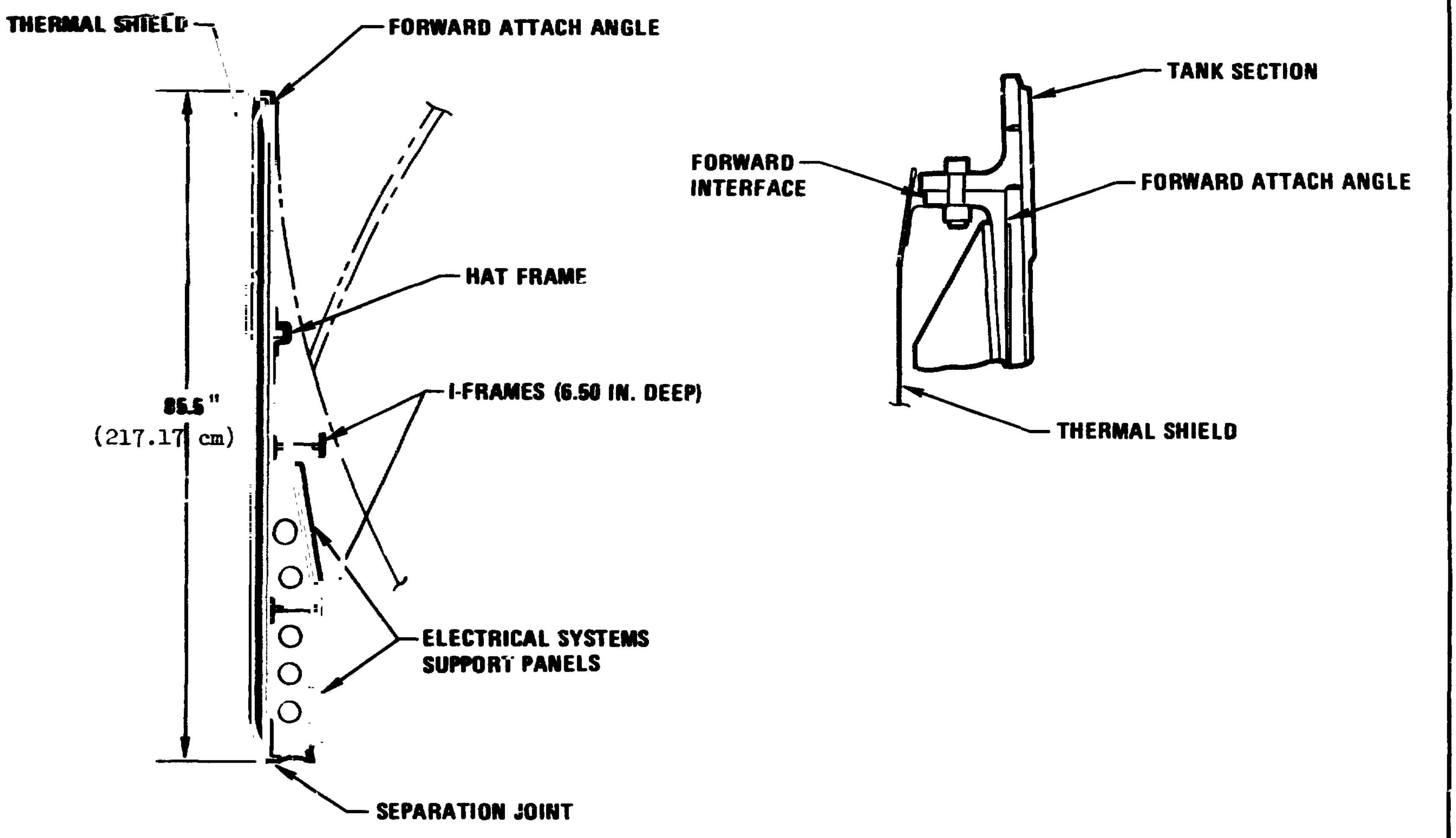


Figure 2.2.1.1-5

The umbilical plate (1A72896-501) was machined from a 3 in. (7.62 cm) x 39 in. (99.06 cm) x 48 in. (121.92 cm) 7075-T651 aluminum plate with pockets for the connectors and ribs for stiffness to carry the loads from the umbilical carrier and to help transfer the axial load from the habitation area section to the aft interstage. The umbilical installation replaced ten typical stringers with five H-section stringers with a heavier section and four short hat sections and H-section stringers. The short stringers reduced hard-point loading on the tank flange and transferred the load from the forward interface into the umbilical plate and full length H-section stringers. (See Figure 2.2.1.1-6.)

The aft skirt thermal shiedl was installed 1.75 in. (4.445 cm) radially from the skirt skin on the exterior of the skirt. It extended 33 in. (83.82 cm) aft from the aft skirt-to-tank interface.

The thermal shield was made up of beaded 7075-T6 clad sheets 0.020 in. (.508 mm) thick and 0.060 in. (1.524 mm) thick fiberglass panels. The shield was supported by four 0.060 in. (1.524 mm) thick fiberglass hat sections, and a 0.060 in. (1.524 mm) thick fiberglass angle attached to the forward interface angle of the aft skirt (see Figure 2.2.1.1-7). The fiberglass panels extended 0.382 in. (.970 cm) forward of the aft skirt-to-tank interface to provide the required shadow for the interface flanges.

The aft skirt thermal shield was sealed except for drain holes to prevent damage and contamination of the gold foil liner on the inner surface of the shield.

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
AFT SKIRT - UMBILICAL INSTALLATION

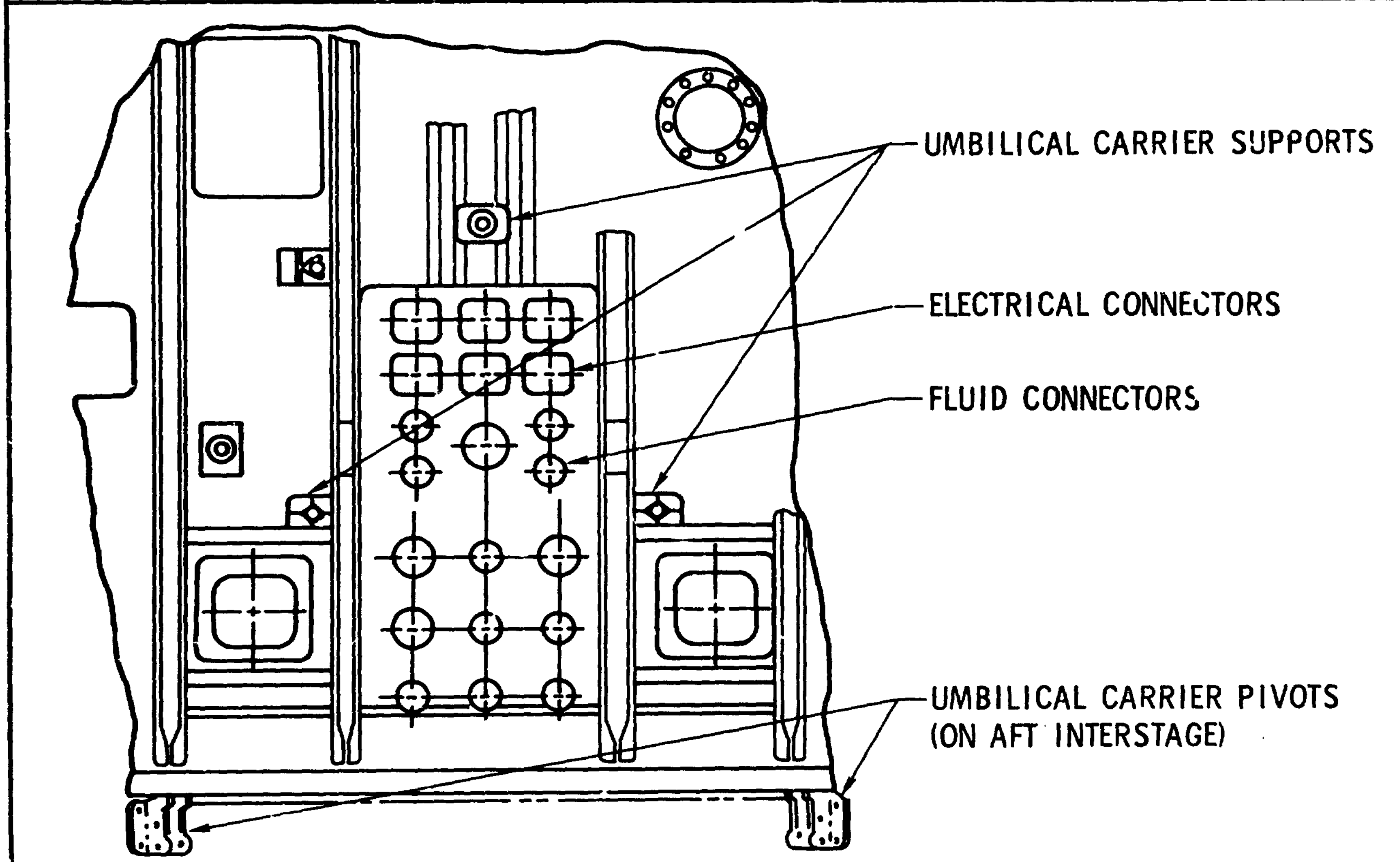


Figure 2.2.1.1-6

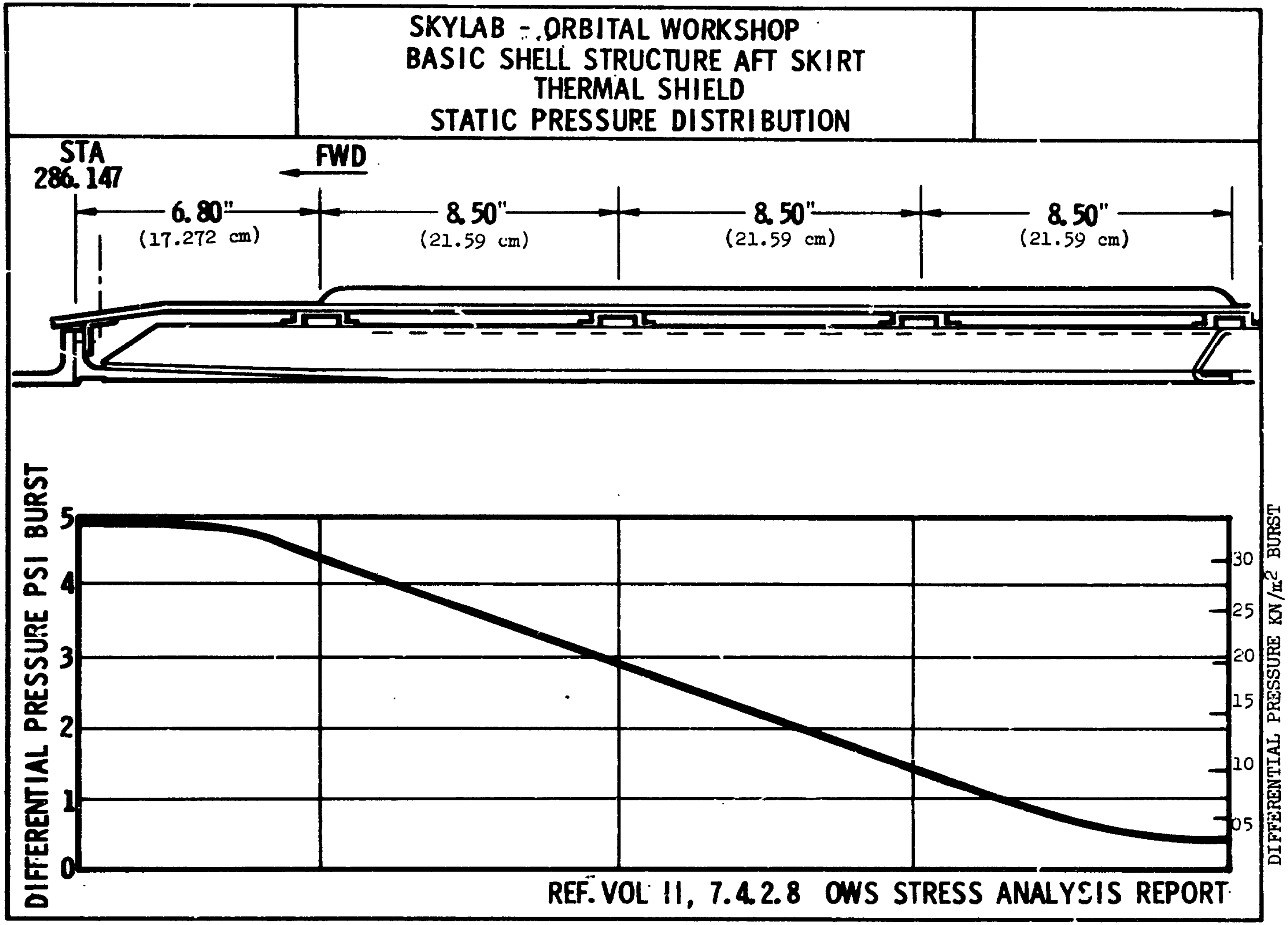


Figure 2.2.1.1-7

The description of the equipment mounting panels and support structure for the aft skirt was identical to that of the forward skirt except for the TACS valves and nozzles.

The TACS valves and nozzles were mounted on fiberglass honeycomb panels approximately 1.25 x 17.5 x 18.0 in. (3.175 x 44.45 x 45.75 cm). These panels were of the same design as the electrical equipment panels. The panels were mounted on Aeroflex cable-type isolators (six per panel) which consisted of aluminum cable retainer bars and stainless steel cable. The tangential thruster nozzles were supported by steel fittings mounted to the panel, and the radial thruster nozzle is supported by aluminum fittings mounted to the panel (Figures 2.2.1.1-8 and 2.2.1.1-9).

Each protrusion through the skirt skin is sealed with a coated nylon-cloth heat-sealable boot (1B44621, 1B88396). The boot is clamped around each tube and support fitting with an S0985-07D CRES band clamp.

The aft skirt SAS attach fittings were installed on the S-IVB/V aft skirt as a part of the conversion to OWS configuration. Three machined fittings were used to attach each SAS beam fairing to the aft skirt. One fitting replaced a stringer and carried the axial stringer loads as well as the SAS loads from two attach points. Two separate fittings were used at the other two SAS attach points. These fittings were installed between existing stringers and pick up existing attach locations on the stringer. No internal modifications were required to support the SAS attach fittings.

The OWS separation joint was identical to the S-IVB/V separation joint (1A83216-503). The joint consisted of two 7075-T6 aluminum

SKYLAB - ORBITAL WORKSHOP
TACS NOZZLES PANEL INSTALLATION

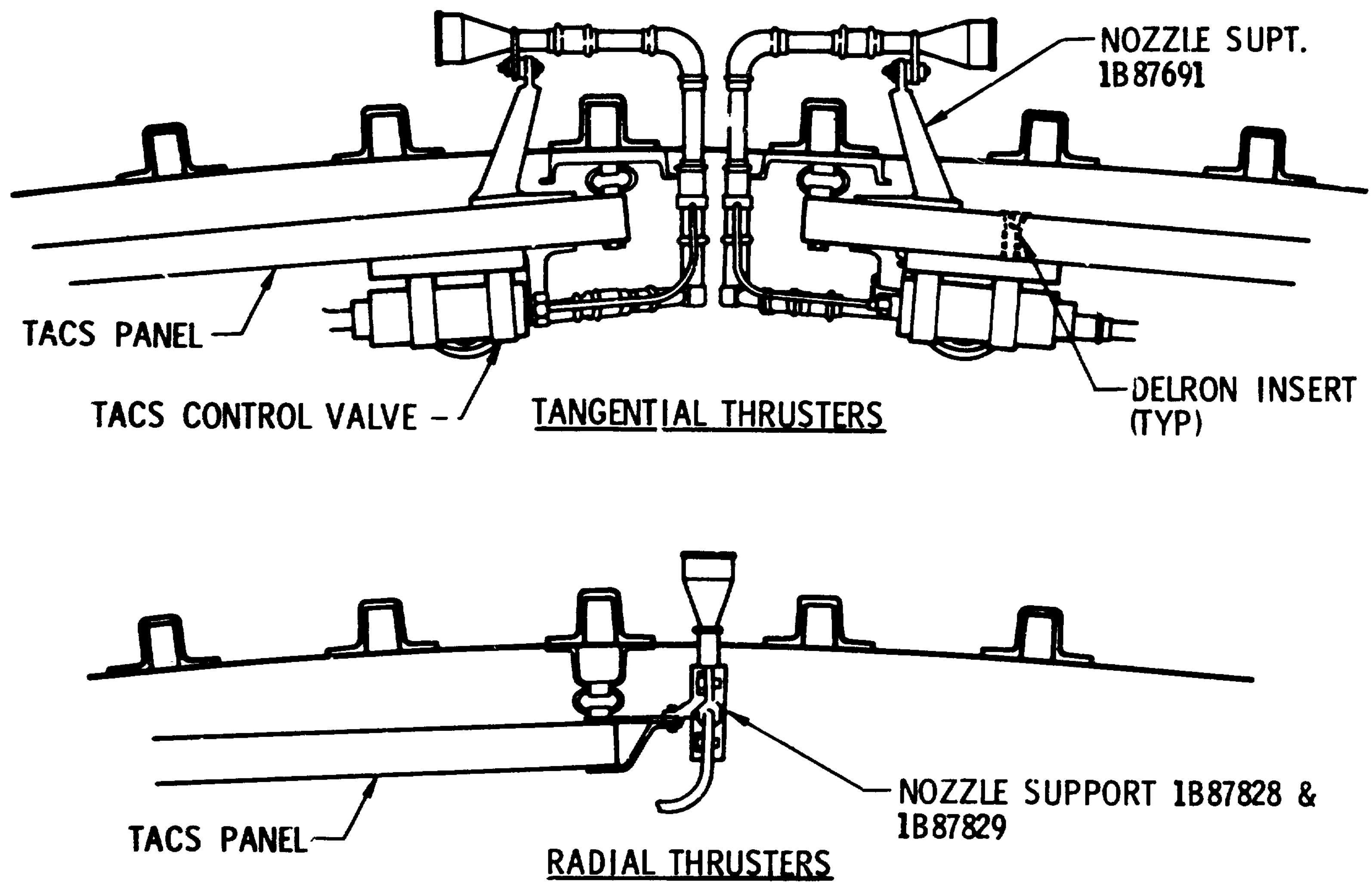


Figure 2.2.1.1-8

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
AFT SKIRT - TACS NOZZLE, MOUNTING PROVISIONS

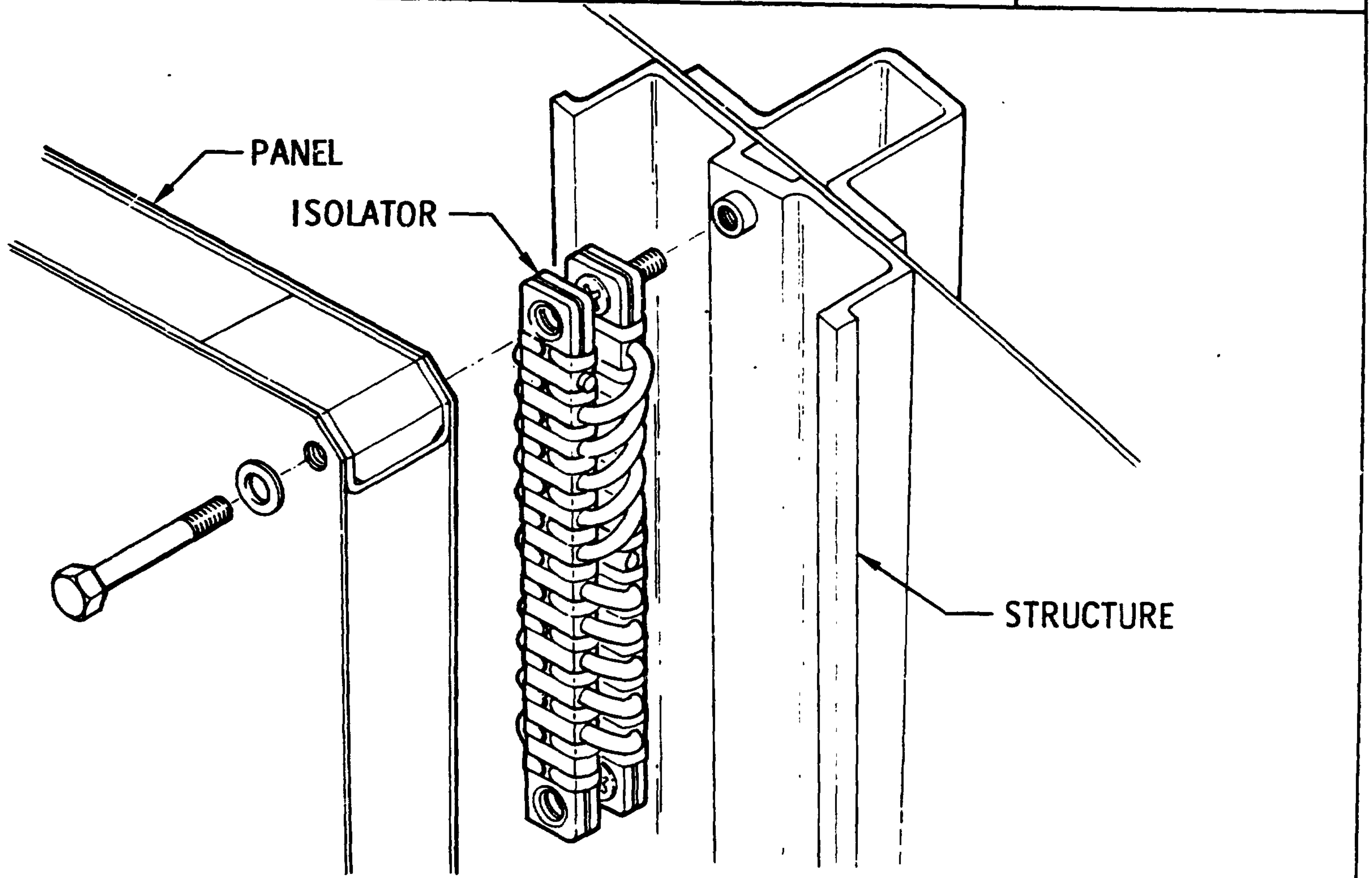


Figure 2.2.1.1-9

extrusion angle machined to form an interlocking joint. The upper angle had a groove in its horizontal leg, 0.176 in. (.447 cm) deep x 0.957 in. (2.43 cm) wide. The lower angle had a mating protrusion which fitted into the groove to transfer shear loads from the skirt to the interstage. The vertical leg of the lower angle was machined to form a 0.050 in. (1.27 mm) thick band which is severed at separation. A groove was machined in the vertical leg of the upper angle to form a cavity between the lower and upper angles to contain the redundant mild detonating fuses which provided the energy required to sever the joint. (See Figure 2.2.1.1-10.)

Two aerodynamically shaped external tunnels extended from the forward skirt over the cylindrical habitation tank to the aft skirt. One of these, the auxiliary tunnel, was an integral part of the meteoroid shield and it will be described in that section of this report. The other, the main tunnel was identical to the SIVB/V tunnel being approximately 5 in. (12.7 cm) high and 24 in. (60.96 cm) wide and made up of several similar sections for the extended length. The sections were stiffened at joints by channel section frames where they were attached through slotted holes. The sections were further stiffened by intermediate "J" section frames. Those covers in the skirt areas were bolted to skirt structure and the covers in the tank section were bolted to clips which mechanically attached to star-shaped doublers bonded to the tank surface. The tunnel was sealed and vented identically to SIVB/V. The tunnel installation was flight proven on previous Saturn flights.

ORBITAL WORKSHOP SEPARATION JOINT

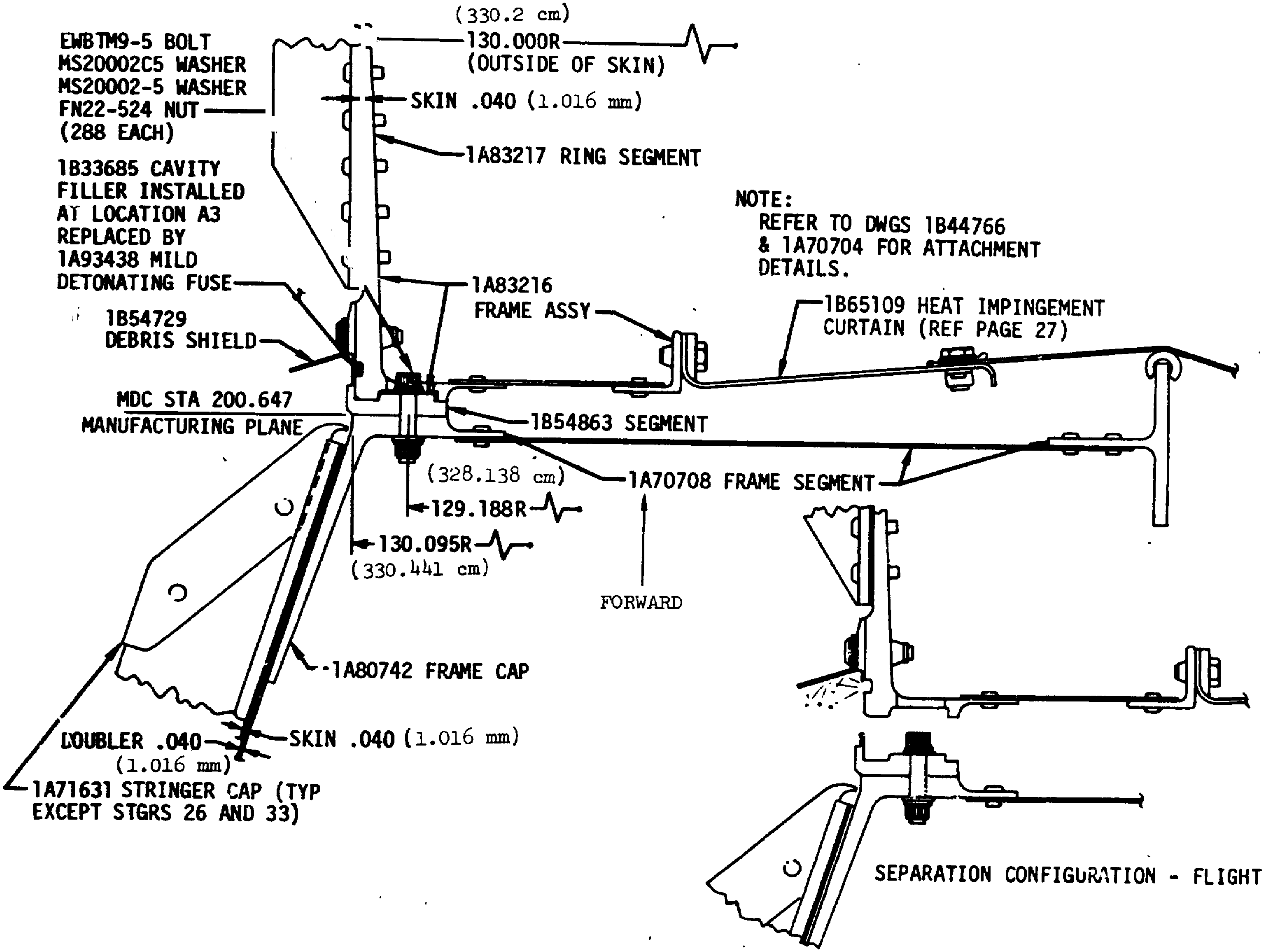


Figure 2.2.1.1-10

C. Testing - The basic structures of the OWS forward and aft skirts were tested as SIVB/V structures for critical design load conditions. The structural capability of the modifications made to the skirts for accommodation of OWS changes and additions was verified by analysis and related testing. A high confidence level in the analysis of the untested modifications for OWS skirts was attained as a result of SIVB/V test data (Reference MDAC Technical Memoranda A3-860-KBBC-TM-96 for the forward skirt and A3-860-KBBC-TM-101 for the aft skirt).

Another test performed on OWS skirt structures was a development test on the thermal shield to verify its ability to withstand the acoustical environment of launch. Test results showed no damage from the most critical acoustic loading. (Ref. MDAC Document No. TM-137). Some tests on other hardware provided test data relative to the OWS skirts, such as the SAS attach fittings which were vibration tested on the qualification test of the SAS wing assembly. Similarly on the SAS forward fairing test, data was obtained for fairing attachment to the forward skirt.

Qualification tests were conducted on each panel/isolator configuration for SIVB/V stages including dynamic loading such as sine sweep, random vibration and shock as well as static failure tests with satisfactory results. Production acceptance and lot verification testing was also conducted on isolators received from the vendor. The panels were qualified by similarity.

D. Mission Results - The forward and aft skirts performed satisfactorily throughout the flight and the mission. Structural performance showed no area of concern as loads experienced during boost were the same level of less than the skirts had experineced on AS-512 and AS-511. The anomaly that occurred at approximately 63 seconds resulting in structural failure and release of OWS meteoroid shield introduced higher than design loads into SAS Wing No. 2 tie down fittings causing premature fracture and partial deployment. At approximately 593 seconds SAS Wing No. 2 was torn from its hinges as a result if impingement forces of the S-II retroplume. Pitcures taken during post SL-2 flyaround showed little structural damage to the skirts from these failures. Any noticeable damage to the skirts did not affect structural shell performance during the ascent phase.

E. Conclusions and Recommendations - The modified S-IVB/V forward and aft skirts for OWS were proven structures from the Saturn Program and were readily adaptable to Skylab vehicle application. The anomalies and unscheduled EVA's performed on SL-2 mission did reveal that the skirt structures did not inherently provide adequate built-in restraints for the astronauts. Real time methods and restraints were devised to accomplish EVA tasks. On new programs, space vehicle design should consider the requirement of provisions for external EVA restraints. The restraints could be built-in or adaptive but should not compromise the primary mission requirements.

The option of using conventional factors of safety accompanied by normal analyses and testing versus the use of more conservative factors of safety without testing or where otherwise cost effective is recommended as good practice if weight is not critical.

F. Development History - The forward and aft skirts, being existing SIVB structures required little development for OWS since their function was almost identical to Saturn vehicles. Modifications to accommodate new systems, thermal insulation, and SIVB deletions were the only development changes required for OWS.

2.2.1.2 Workshop/Waste Tank Structure

A. Design Requirements

1/ Basic Habitation Area Structure - The basic functional requirements of the habitation area structure were as follows:

a. A habitable environment with crew quarters, provisions, and consumables.

b. A capability for experiment installation and storage prior to launch.

c. Disposal of waste materials.

d. Installation of scientific airlocks.

e. Capability to withstand induced and natural environments.

The structure for the habitation area was originally a Saturn S-IVB tank. The liquid hydrogen tank was to be modified to become the habitation area tank, and the liquid oxygen tank was to be modified to become the waste tank. All tank penetration or openings not used were to be sealed. The forward dome opening was to be modified to contain a forward entry hatch and to interface with the airlock module. The cylindrical portion of the habitation tank was to be reworked to provide a side access panel for ground access, two scientific airlocks, and a wardroom window. Openings in the common bulkhead were to be made for trash airlock and liquid dump probes.

The interior of the habitation tank was to be divided into compartments by floors and walls to provide crew quarters. The interior structure was required to have the capability of supporting equipment for loads specified in DAC 56612B and DAC 56620C.

The habitation area structure was to be designed to withstand all external and internal loads which occur during ground and water transportation, handling, prelaunch, launch, and on-orbit conditions specification, as specified in documents DAC 56612B and DAC 56620C. General design requirements were specified in Contract End Item CP2080J1C. The cylindrical portion of the habitation area tank was to be capable of transferring loads between the forward and aft skirts.

Total allowable leakage (including the entry hatch adapter ring-to-ring seal) was to be less than 5 lbs (2.27kg) mass per day when the OWS was pressurized to the on-orbit pressure environment of 5.0 psia (.345 x 10^5 N/m^2) nominal. Flammability and offgassing requirements were defined in 1B79102.

Since the habitation area tank was originally built for the S-IVB Program, one of the design goals in modifying the tankage for the OWS Program was that any reduction in pressure capability would be minimized. Consequently, the original burst pressure capabilities of 62.0 psid (4.27 x 10^5 N/m^2) for cylindrical shell, 55.8 psid (3.84 x 10^5 N/m^2) for the forward dome and +41.0 psid (2.83 x 10^5 N/m^2) and -26.3 psid (1.81 x 10^5 N/m^2) for the common bulkhead (positive differential pressure indicates waste tank pressure is greater than habitation area pressure) were considered when incorporating provisions for the access panel, viewing window, scientific airlocks (SAL's), and other penetrations through the tank walls.

The habitation tank was to be capable of sustaining local loading in various areas induced by equipment and structure supported by the cylinder wall. Included were loads at the two peripheral floor supports and one peripheral water storage support caused by boost accelerations.

Numerous components were to be either bolted to the tank wall or mounted to the wall by disc attachment with nut plates bonded to the tank wall insulation. The tank wall was to be capable of sustaining local loading from vibration of the components and from external acoustic sound pressure levels impinging on the tank wall.

The common bulkhead was to be capable of sustaining normal operating limit differential pressures ranging from -10.0 psid ($-.689 \times 10^5$ N/m^2) (waste tank pressure is less than habitation area tank pressure) to +17.0 psid (1.17×10^5 N/m^2) (waste tank pressure is greater than habitation area tank pressure). In addition, a malfunction condition was defined by the common bulkhead as a case where the waste tank or habitation area tank vent valve failed open at liftoff. Under these circumstances, the maximum limit differential pressures were -20.5 psid (1.413×10^5 N/m^2) and +26.0 psid (1.79×10^5 N/m^2). Several penetrations were to be made and equipment was to be mounted to the common bulkhead. The bulkhead was to be capable of sustaining loads from this equipment.

For launch conditions, ultimate and yield factors of safety of 1.25 and 1.10, respectively, were required for all flight loads except random vibration, or on internal pressure when it relieves buckling load. An ultimate factor of safety of 1.40 was used for vibration loads. The relieving effect of pressure was included by subtracting axial tension due to minimum limit differential pressure for the limit compression shell load times 1.25.

For a malfunction condition (an unplanned event not causing mission abort), the ultimate and yield factors of safety were both 1.00.

For loadings during the manned phase of the mission, ultimate and yield factors of safety of 1.40 and 1.10, respectively, were required. Basic design criteria for all conditions were contained in DAC 56612B.

2/ Crew Accommodations - Major portions of the OWS interior were to be designed to accommodate crew requirements per Contract End Item Specification CP2080J1C. Curtains or doors were to be provided for the wardroom opening, waste management compartment opening and each sleep compartment opening to give the crewmen privacy. Astronaut aids were required throughout the OWS to aid the crewmen in moving to and from work stations and for providing restraint when accomplishing on-orbit tasks. Some of the astronaut aids were portable to permit restraint of the crewman or his equipment at a convenient location. The astronaut aids and restraints required and their locations were as follows:

a. Forward access hatch handrails
b. Forward dome handrails
c. Center handrail
d. Experiment area circumferential handrail
e. Forward compartment vertical handrails
f. Waste management compartment ceiling handrail
g. Experiment area handholds
h. Sleep compartment handhold
i. Waste management compartment handhold
j. Portable handholds
k. Portable tether attach brackets
l. Portable foot restraints
    1. Constant wear portable foot restraints
    2. Pressure suit portable foot restraints
m. Adjustable length tethers (2)

The OWS open grid floors were to be designed for mating with the triangular shoes and providing a foot restraint. A 42 in. (106.68 cm) opening in the center of the ceiling was required to permit free passage of an unsuited crewman. Emergency egress openings were required in the ceiling over the wardroom and sleep compartment.

Closeouts (Barriers) were to be provided for the permanent lockers in the wardroom and sleep compartment to allow the required airflow and to prevent passage of food and loose particles greater than 0.2 in. (.508 cm) (maximum) in diameter. Closeouts in the experiment and forward compartment areas were required to prevent passage of particles greater than 0.4 in. (1.106 cm) (maximum). The waste management compartment was to be sealed to prevent particulate migration to adjacent compartments.

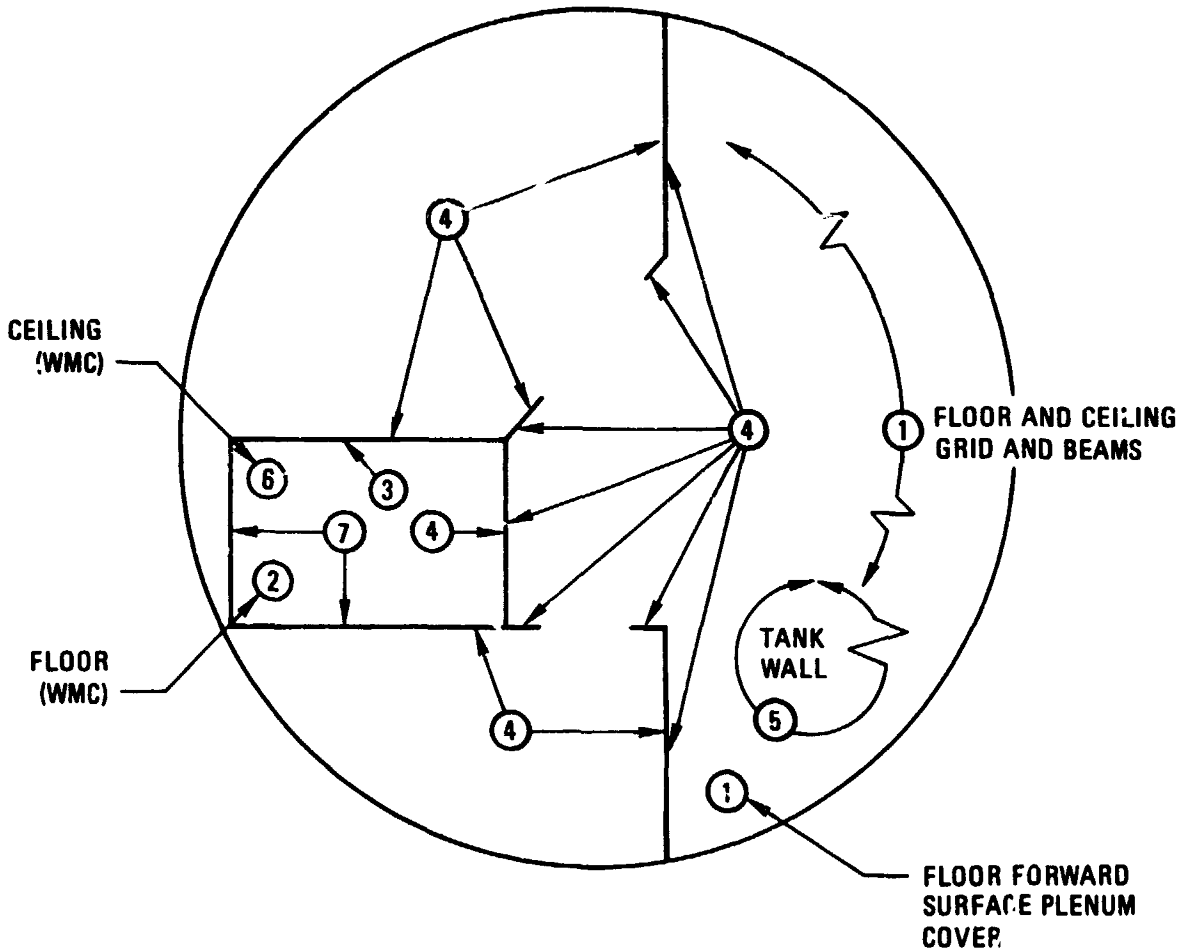
The color scheme for the interior of the OWS was specified in Contract End Item Specification CP2080J1C. The crew quarters walls, floor and ceiling were to be colored as shown in Figure 2.2.1.2-1. The forward tank wall color listed in Table 2.2.1.2-1 was to be extended throughout the cylindrical and forward dome sections forward of the aft face of the aft floor.

There are numerous other designs required to provide crew accommodations. In this report, these other designs will be discussed as a part of the system in which they function.

B. System Description

1/ Basic Habitation Area Tank - The habitation area tank (1A39303) consisted of a forward dome (1B64442), tank cylinder (1A39306), and an aft dome assembly (1B39309). The forward and aft domes are 260 in. (660.4 cm) diameter hemispheres of 2014-T651 aluminum. The cylinder was 260 in. (660.4 cm) in diameter by 268.6 in. (682.24 cm) long and machined from 2014-T651 aluminum with integral machined ribs to form 45 degree "waffle" pattern for stiffening. The common bulkhead was a 260 in. (660.4 cm) diameter partial hemisphere with a forward and aft face of 2104-T651 aluminum, separated by 1.75 in. (4.445 cm) of 3/16 in. (.475 cm) hexcell fiberglass honeycomb bonded to each face. It was attached to the aft dome and separated the habitatation area from the waste tank. (See Figure 2.2.1.2-2.)

VIEW LOOKING FORWARD AT MDAC-WD STATION 350.663



COLOR KEY

1 YELLOW
2 RED-TAN
3 LIGHT BLUE
4 GOLD
5 LIGHT YELLOW
6 CLEAR
7 STANDARD STORAGE LOCKERS

Figure 2.2.1.2-1 ORBITAL WORKSHOP INTERNAL COLOR REQUIREMENTS

TABLE 2.2.1.2-1
ORBITAL WORKSHOP INTERNAL COLOR REQUIREMENTS

Item	Color
Anemostats	Yellow
WMC Lockers, Surfaces Visible from within the Wardroom	Light Blue
Fecal/Urine Collector Unit	Clear
Waste Processor	Clear
Wardroom Soft Door	Natural
Wardroom Lockers, Surfaces Visible from within the Wardroom	Gold, Red-tan, Off White (Micatex with Kel-F 800 topcoat) or Lamanar X500
Wardroom Table	Off White (TFE) and Clear
Wardroom Table Restraints	Natural
Mobility Aids	Dark Blue
Electrical Panels, Mounting Structures	Dark Brown
Electrical Panel, Faces	Off White (Micatex with Kel-F 800 topcoat) or Lamanar X500
Light Housing Assemblies (except for Reflector surfaces)	Clear
Sleep Compartment Lockers, Surfaces visible from within the sleep compartment	Gold
Sleep Restraints	Natural
Sleep Compartment Privacy Curtains	Natural
Sleep Compartment Privacy Partition	Gold

SKYLAB - ORBITAL WORKSHOP
BASIC HABITATION AREA TANK STRUCTURE

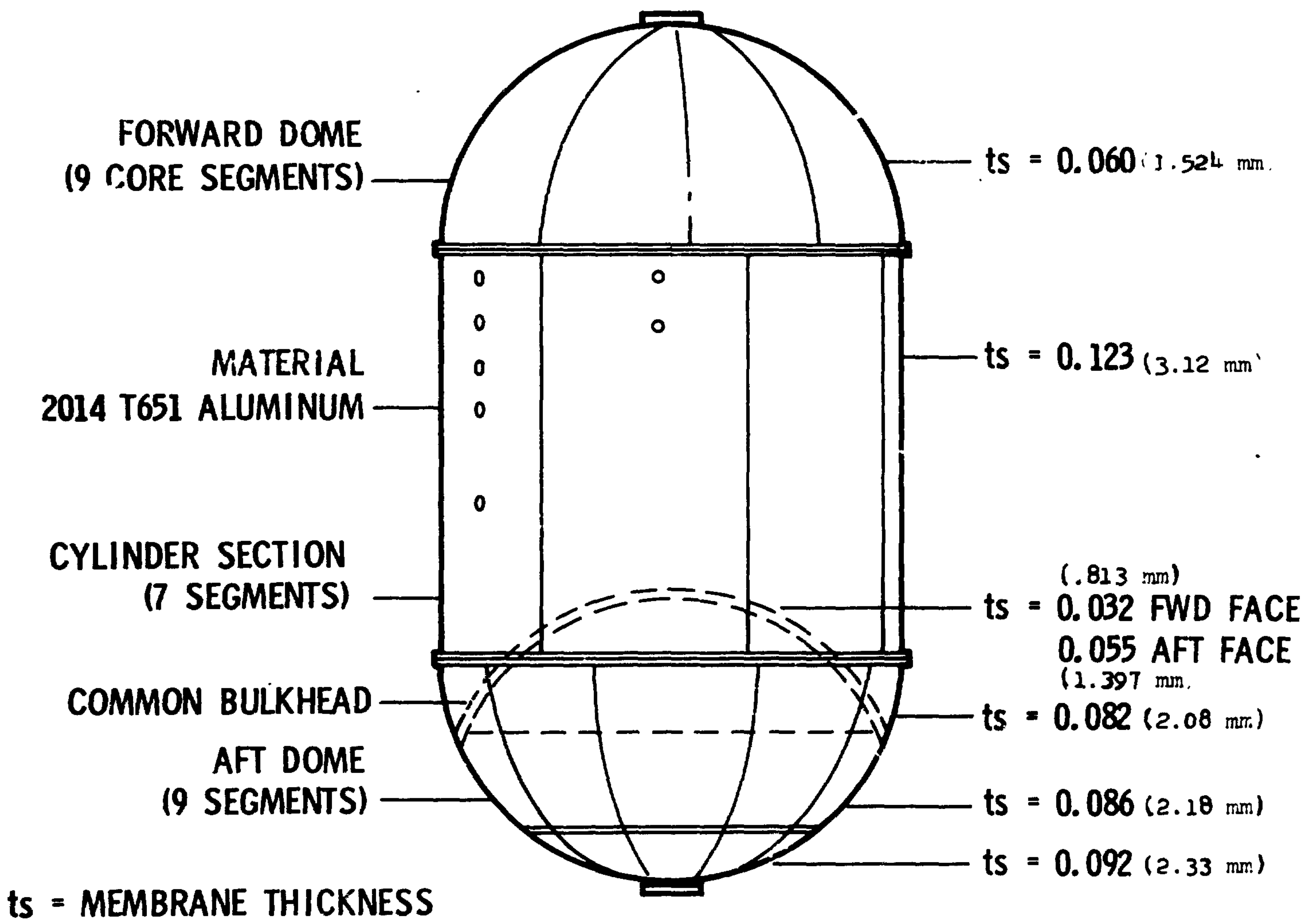


Figure 2.2.1.2-2

The habitation area walls (forward dome, cylinder, and the aft dome) were insulated with 3D polyurethane foam tile per 1A39314. The forward face of the common bulkhead was insulated with 1D polyurethane foam per 1B82223. 1D polyurethane foam was installed over the 3D foam on the aft dome and the lower 20 in. (50.8 cm) of the cylinder. To satisfy the flammability requirements of 1B78110, a flame retardant coating of 0.003 in. (.076 mm) aluminum foil was bonded over the insulation in the forward dome and cylinder per 1B81577. The aft dome and common bulkhead insulation was covered with 0.005 in. (.127 mm) aluminum foil. The foil on the forward dome and cylinder was coated with colored teflon to satisfy interior color requirements.

For attachment of crew quarter floor supports, water bottle supports, ACS fan supports, equipment supports, and electrical bonding strap attach, holes were drilled in the waffle rib intersections to a maximum depth of 0.625 in. (1.587 cm). This hole was then tapped to provide threads for the bolts or studs. (See Figure 2.2.1.2-3.)

To support equipment, wiring, conduits and experiments, phenolic discs with a nutplate attached were bonded to the insulation with adhesive. Two layers of fiberglass liner were bonded over the discs. (See Figure 2.2.1.2-4) There were approximately 1450 discs bonded to the habitation area tank wall.

Items contributing toward passive thermal control were as follows:

- o The optical property control of the interior and exterior surfaces was regulated by the use of paint, coated films, and coated tape. Also, included in the system was the use of high performance insulation (HPI) over the forward dome area.
- o A third control was the use of complete circumferential thermal shields, covering a longitudinal portion of the forward and aft skirts.
- o The inside of the OWS pressure shell was lined with reinforced foam insulatio which also contributed to the passive thermal control system.

SKYLAB - ORBITAL WORKSHOP
CYLINDER RIB INTERSECTION ATTACH

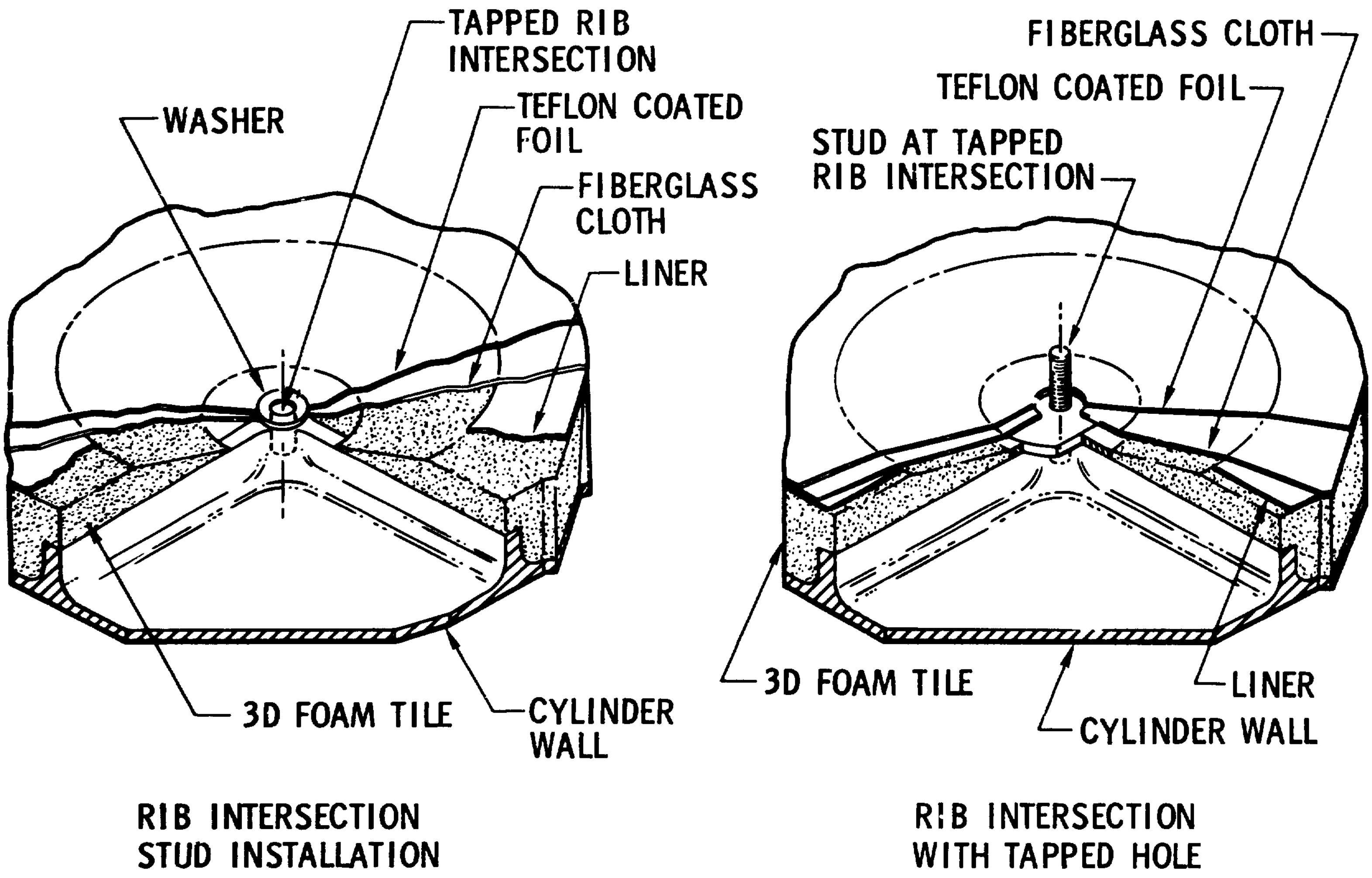


Figure 2.2.1.2-3

ORBITAL WORKSHOP
SUPPORT
BONDED EXPERIMENT DISC INSTALLATION

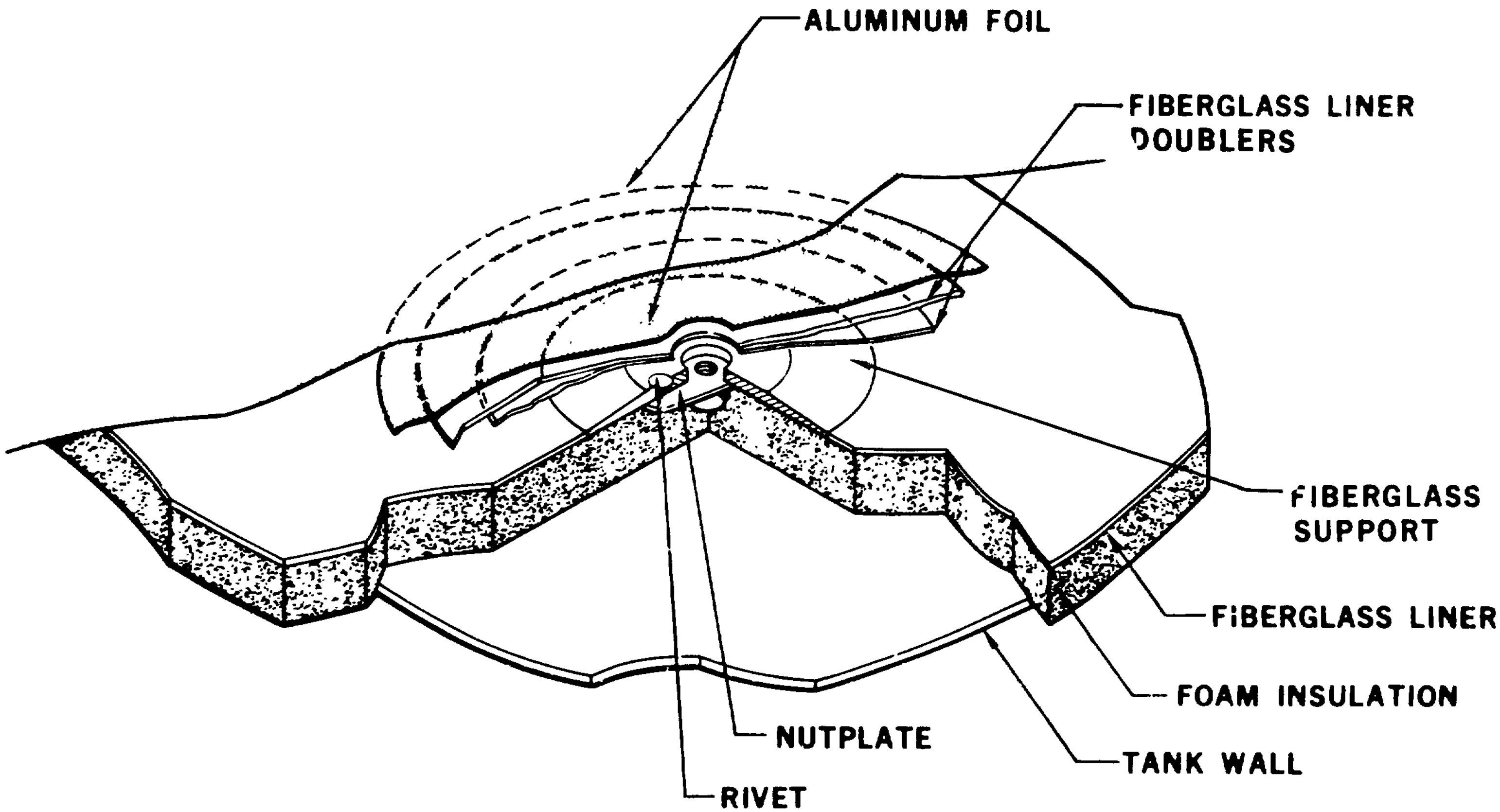


Figure 2.2.1.2-4

A typical HPI installation is shown in Figure 2.2.1.2-5. The preinstalled HPI was 10 layers of gold-coated polyimide film, 0.0005 in. (.0127 mm) thick, and 9 layers of separator material consisting of 0.007 in. (.1778 mm) thick polyester (Dacron) mesh fabric. On each side of the composite material was a layer of closely woven polyester fabric (Dacron). The two cover sheets were reinforced around the edges and at 12 in. (30.48 cm) intervals with doubler strips (Dacron) bonded with epoxy resin to the cover sheet. The composite panel was held together with a unique plastic fastener. These fasteners were low in conductivity, lightweight, commercially available, and easy to install.

The upper HPI panels were installed over the dome and then covered with a grid that supports the wiring. The lower pre-installed HPI panels were bonded with adhesive tabs to the lower portion of the forward dome prior to the forward skirt attachment.

The major portion of the forward dome was covered with 18 gore-shaped HPI panels prior to vehicle rollout. These panels were made up of 48 layers of reflector film, 47 layers of separator material, and two cover sheets. The reflector films were 0.00015 in. (.0038 mm) thick Mylar, coated on both faces with aluminum and perforated with 0.055 in. (1.397 mm) diameter holes at 3/8 in. (.952 cm) intervals. The separator sheets were 0.007 in. (.1778 mm) thick Dacron net material. This material was a loose weave, light-weight Dacron cloth, and was the structural component of the HPI panels. Increased structural capability was achieved by impregnating doublers with epoxy resin and combining it with the cover sheet for form a laminate material.

ORBITAL WORKSHOP
HIGH PERFORMANCE INSULATION

PROTECTIVE COVER

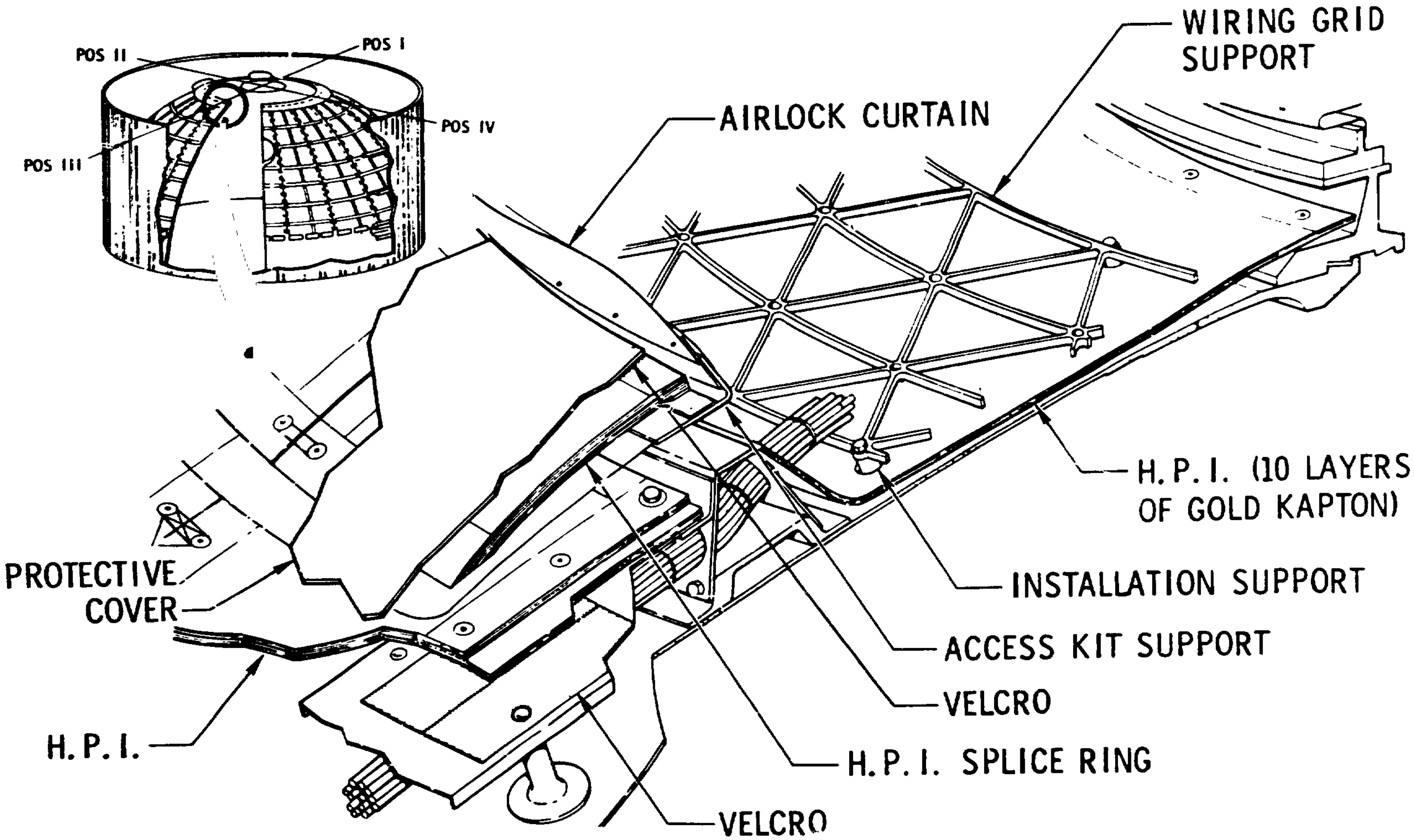


Figure 2.2.1.2-5

The cover protected the HPI from liquid spillage and falling tools and retained the gas purge. The forward dome HPI was covered with Hypalon-coated Nylon cloth. The cover was made up of six segments and extended from the access kit support ring down to the forward skirt hat section, at Station 585. Each segment was made from three pieces of material sewn together. The covers incorporated built-in protrusions to cover the electrical feedthroughs. The covers had a stainless steel thread sewn into the fabric in a 6 in. (15.24 cm) matrix to electrically ground the fabric to the vehicle. The HPI was installed with Velcro attachments at the aft and forward edges of the segment. Plastic zippers were used for the radial splices. The cover included plastic fittings over which a boot was laced where wire bundles penetrated.

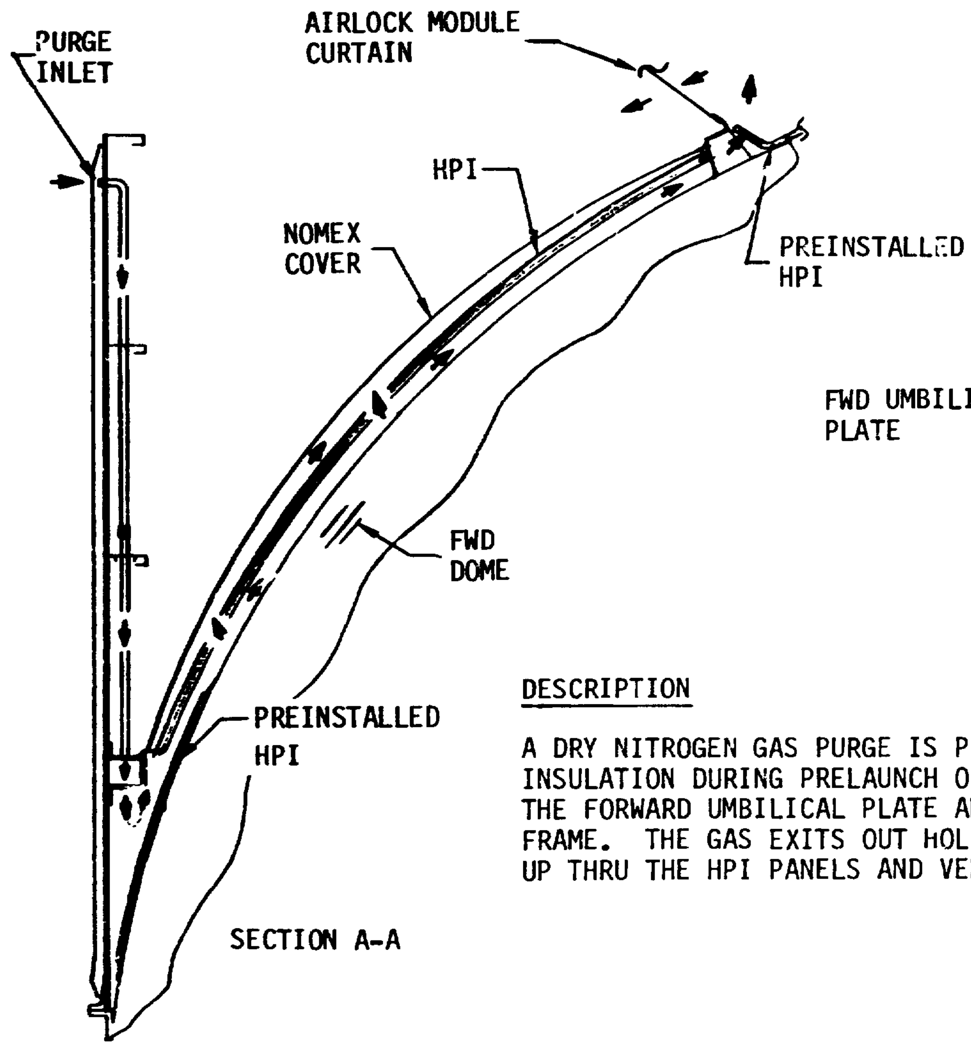
A dry nitrogen gas purge conditioned the high performance insulation during prelaunch operations. (See Figure 2.2.1.2-6).

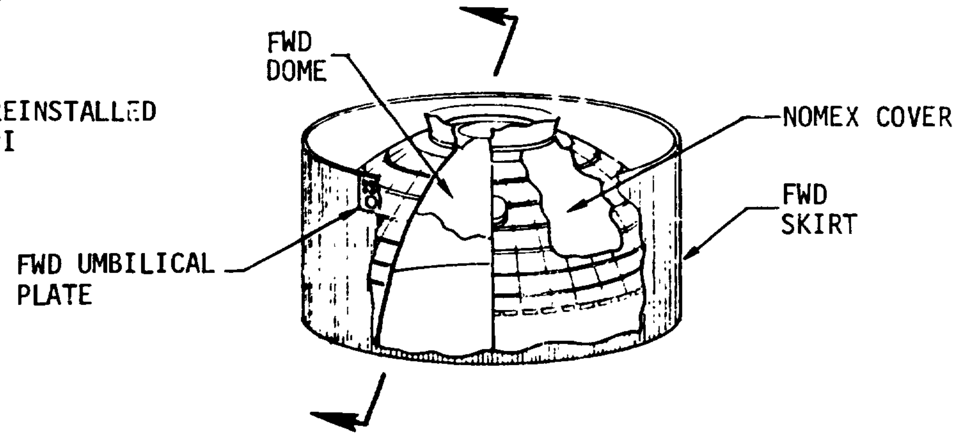
The external surface of the cylinder was covered with gold-coated pressure-sensitive Polyimide tape, except in the area under the main (fixed) tunnel. The inner surface of the main tunnel was also covered with the gold tape.

The side access panel was designed to permit entry into the OWS during manufacture and pre-launch in order to support installation and assembly, service, checkout, and provisioning. (See Figure 2.2.1.2-7). Entry on the launch pad was not scheduled. However, the design accounted for an unscheduled or contingency entry.

ORBITAL WORKSHOP

HIGH PERFORMANCE INSULATION PURGE SYSTEM





DESCRIPTION

A DRY NITROGEN GAS PURGE IS PROVIDED TO CONDITION THE HIGH PERFORMANCE INSULATION DURING PRELAUNCH OPERATIONS. THE PURGE INLET IS LOCATED ON THE FORWARD UMBILICAL PLATE AND IS PIPED DOWN TO THE FORWARD SKIRT HAT FRAME. THE GAS EXITS OUT HOLES AT THE BOTTOM OF THE FRAME FILTERING UP THRU THE HPI PANELS AND VENTS OUT THROUGH THE AIRLOCK MODULE CURTAIN.

Figure 2.2.1.2-6

SKYLAB - ORBITAL WORKSHOP
SIDE ACCESS PANEL HABITATION AREA

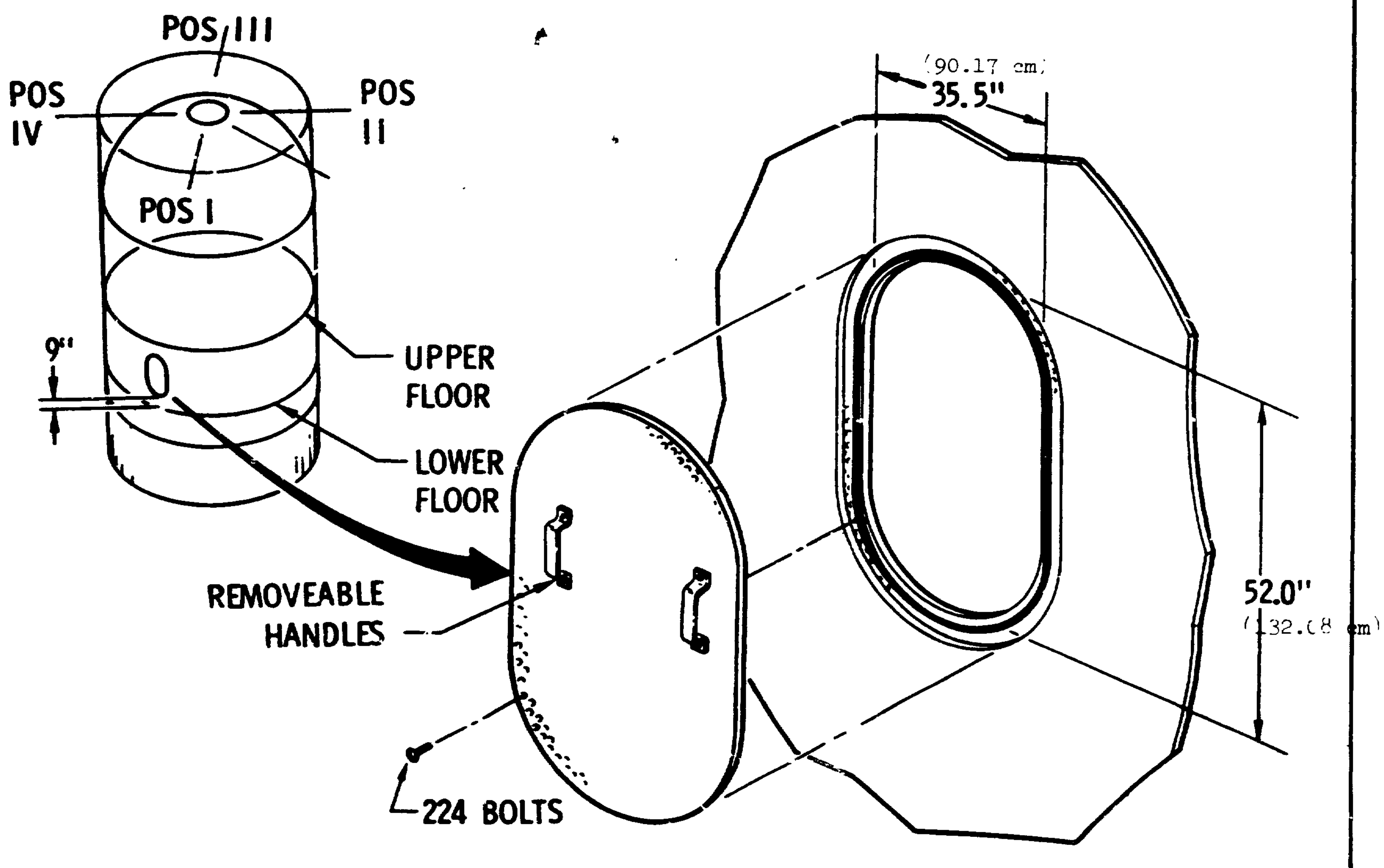


Figure 2.2.1.2-7

Existing and modified openings were all sealed or had the capability to be closed and sealed. Penetrations in the forward dome included two electrical feed-throughs and a controls feed-through. An electrical connector fitting was installed on each of the three feed-throughs and was attached with a Marman conoseal and bolted-on coupling. Pressurization or purging ports were plugged with a MC238-C4 plug and MC252 seal.

Penetrations in the cylinder were required for two scientific airlocks, viewing window, access panel, LBNP and metabolic analyzer, six helium ports and a refrigeration feed-through. Existing penetrations, such as helium ports, were closed per 1B78902 using a Marman blind flange, conoseal gasket, and coupling. A bead of silicone sealant provided an additional seal. Of those ports which were sealed, four were covered with 1 in. (2.54 cm) thick 3D foam insulation and then covered with a layer of fiberglass cloth, a layer of 0.005 in. (.127 mm) thick aluminum foil, and one layer of teflon-coated foil. One of the blind flanges had a tube welded into it to provide for water line access. One sealed port had provision for a TV coax connector which was installed into a hole in the blind flange. Insulation was provided by a 1B89166-1 box assembly made of 6061-T6 aluminum and filled with a flexible polyurethane foam. A sixth helium port was closed by a refrigeration line which was attached by using a Marman conoseal gasket and coupling. An aluminum tube filled with flexible polyurethane foam provided insulation for the port and tubing.

Utilizing the existing provision for a cylinder port that had a local tank skin thickness of 0.213 in. (.541 cm), a 6.37 in. (16.179 cm) diameter hole was cut in the cylinder wall. A 1B69683-1 fitting was installed using lock bolts and a silicone sealant. A blind flange, with provisions for refrigeration lines 1B80868, closed the port using a Marman blind flange, conoseal gasket and coupling.

Penetrations in the aft dome were closed per 1B79334-1. (Aft dome closures are shown in Figure 2.2.1.2-8).

The waste tank vent ports were installed 180° from each other, 51.75 in. (131.445 cm) aft of the dome cylinder joint, and 18 degrees from position I toward position II and from position II toward IV. The 1B87786 vent port fitting was made of 6061-T651 aluminum and attached to the aft dome-waste tank wall with adhesive and lock bolts per 1B87877-1. The lockbolts and edge of the fitting were sealed with a silicone sealant and the vent system was sealed with an O-ring. The vent system was attached by bolts and supported by the vent port fitting. It extended through a hole in the aft skirt. Two tapered rings provided for vent system directional adjustment.

2/ <u>Forward Entry Hatch</u>

The forward entry hatch was located at the apex of the forward dome and interfaced with the airlock module (See Figure 2.2.1.2-9). The hatch functioned as a structural part, carrying pressure loads during Skylab boost. It allowed entry into the OWS on-orbit and acted as the aft airlock hatch during EVA's. Operation of the hatch was performed manually with one hand. Two functions were performed with the hatch handle - pressure equalization and

SKYLAB - ORBITAL WORKSHOP
AFT DOME PORT CLOSURES

AFT DOME SEALED PORTS (6 PLACES), LH_2 FILL LINE, LH_2 AND LO_2 CHILL RETURN, LO_2 INSTRUMENTATION PROBE, SVE WIRE LEAD OUT, AND PU PROBE. EACH PORT HAS A DOUBLE SEAL WITH A CONOSEAL GASKET AND A BEAD OF SILICONE SEALANT.

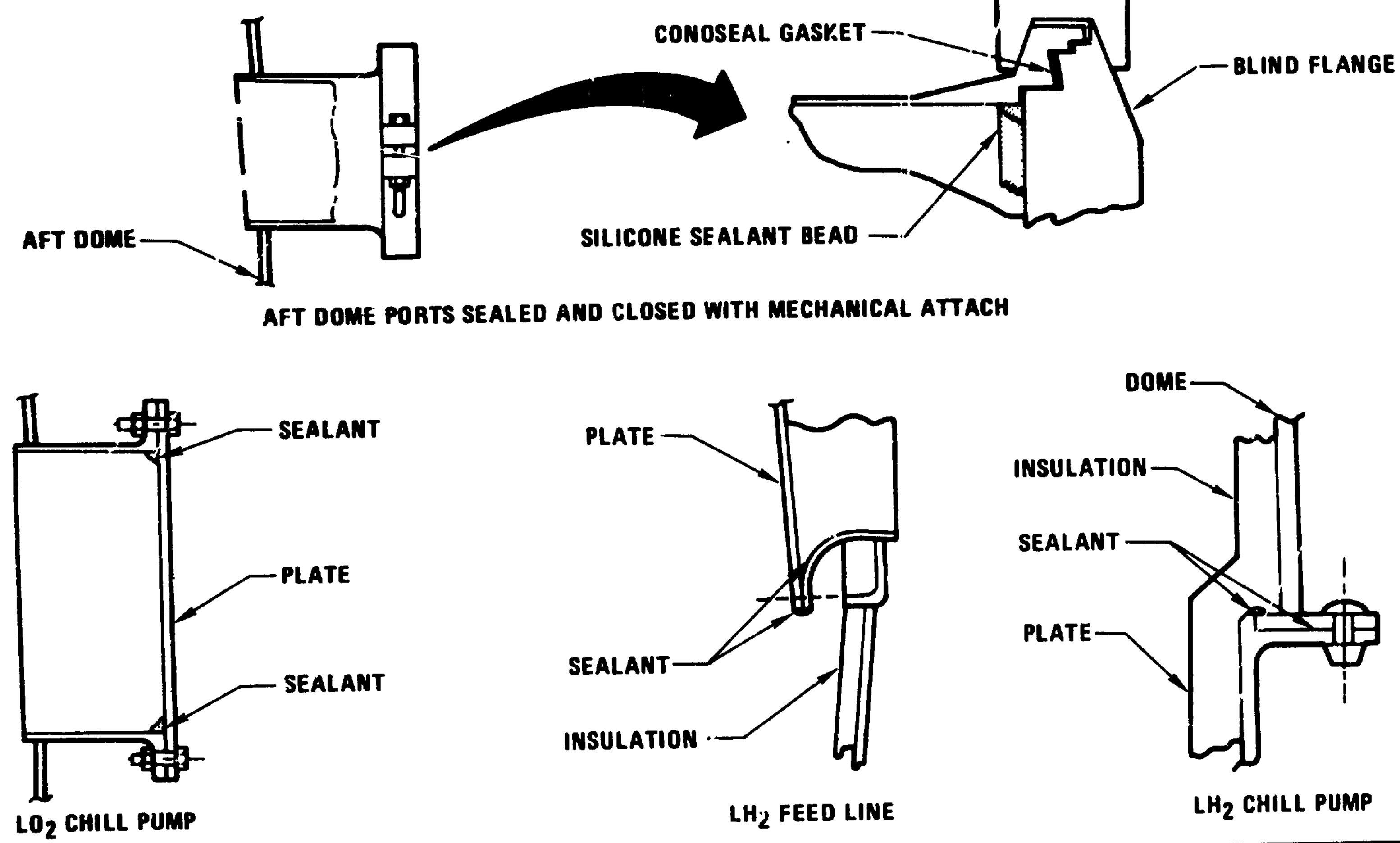
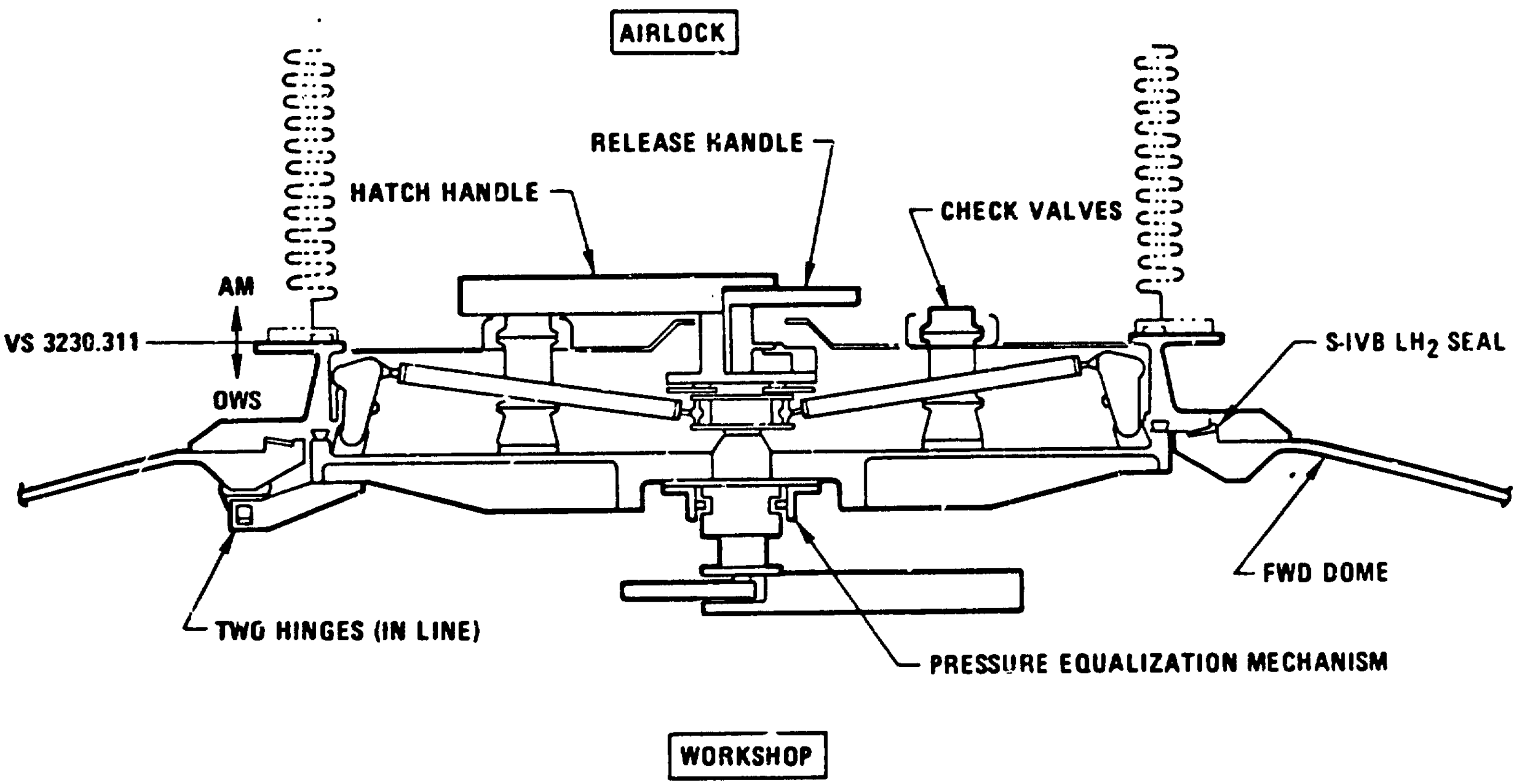


Figure 2.2.1.2-8

SKYLAB - ORBITAL WORKSHOP
HABITATION AREA TANK
FORWARD ENTRY HATCH



latching/unlatching. The release handle locked and unlocked the hatch handle to prevent inadvertent hatch operation. Pressure equalization was required to obtain a zero pressure differential across the hatch prior to unlatching. When opened, the hatch was swung on hinges into the habitation area until it locked into tis stowage position on the forward dome wall.

To prevent buckling of the OWS forward dome, redundant check valves in the hatch would crack at a pressure differential of 0.2 psid (1.378 x 10^5 N/m2) when AM pressure was greater than OWS pressure.

3/ Wardroom Window - The viewing window design provided protection from meteoroid impact, internal equipment impact and solar ultra-violet and infra-red radiation, heating capability for defogging, a metal cover to resist boost pressures, and a shade to prevent light intrusion at the crew's option.

The viewing window (See Figure 2.2.1.2-10) was installed from the inside of the OWS and, being flat, did not extend beyond the tank wall. The window glazings carried a small differential pressure during boost, and the cavity between the glazings was vented down by the crew during initial OWS activation. Tank shell loads were carried around the window cutout by the window reinforcing ring to which the window assembly was attached. An 18 in. (45.72 cm) diameter clear view opening was provided.

An electro-conductive coating, applied to one glazing of the assembly provided heating capability to prevent fogging of the glass. This heater was designed to maintain the inner surface of the window within the range of 55°F (12.8°C) to 105°F (40.6°C).

SKYLAB - ORBITAL WORKSHOP
VIEWING WINDOW ASSEMBLY INSTALLATION

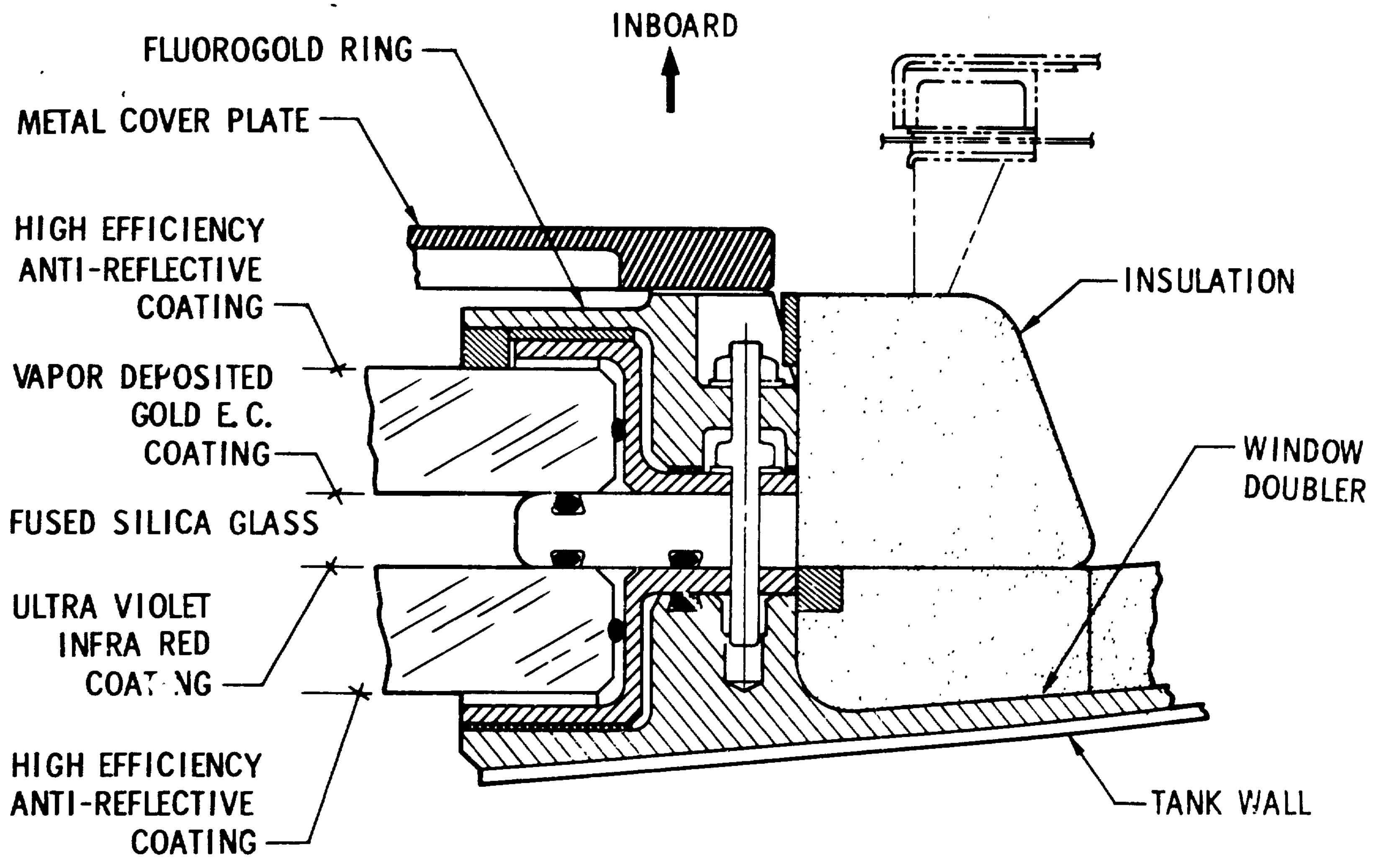


Figure 2.2.1.2-10

Silicone and Viton rubber seals, cushions, and spacers used throughout the window design were cured at high temperatures and low pressures to prevent contamination due to outgassing.

4/ Scientific Airlocks

The basic scientific airlock (SAL) system consisted of a number of separate experiment packages and two scientific airlocks. One SAL was mounted on the solar side and one mounted to the anti-solar side of the spacecraft. Each SAL experiment was enclosed in a pressure shell having an external flange which mated with and sealed against the inboard side of the airlock. A metal plate window was also provided for each airlock and was sealed to the airlock in the same manner.

The SAL was a pressure vessel sealed against the external environment by a translatable outer door. The inboard face had an opening which could be sealed by an experiment or window. The outer door was closed whenever an experiment or window was installed or removed, to isolate the cabin from the exterior environment. After installation, the pressure vessel of the experiment became part of the airlock pressure vessel and the outer door could be opened and the experiment deployed outside the spacecraft. The internal pressure of the airlock was equalized to either the cabin pressure or the external environment by a vent, or depressurization valve.

The two airlocks (1B82361-1 and -501) were identical except for thermal coatings. Figure 2.2.1.2-11 shows the basic airlock with the vacuum fitting at the top, an experiment (or window) latching handle on the right hand side, a pressure gage at the center, and a crank for opening the outer door at the lower left. The outer door waffle pattern is seen through the opening on the sealing surface on the inboard side of the airlock.

SKYLAB - ORBITAL WORKSHOP
SCIENTIFIC AIRLOCK

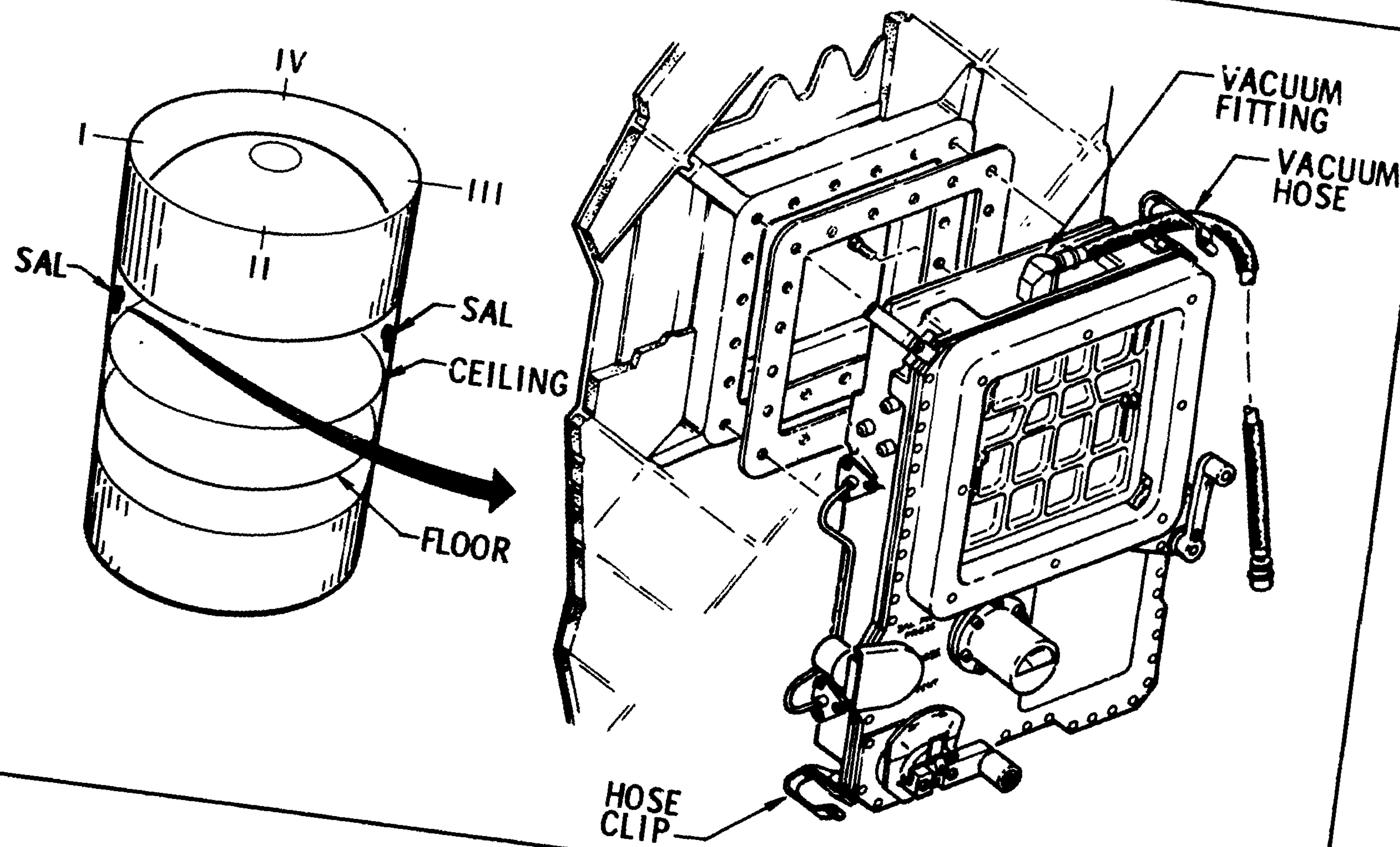


Figure 2.2.1.2-11

An indicator mounted above the crank showed the door position relative to fully open and fully closed. A lock was provided to prevent accidental rotation of the crank. The valve on the left hand side of the airlock permitted venting the airlock cavity either overboard or to the cabin. The experiment/window latching mechanism was enclosed in the square tubular structure; roller dogs emerged through openings and engaged the flange of the experiment/window, forcing the rubber seals on the flange against the airlock sealing surface.

The vacuum hose (1B86940-1) was attached to an experiment through a quick disconnect fitting; the other end was attached to the vacuum fitting at the top of the airlock. This allowed evacuation of the experiment or of a film cannister.

The windows, 1B86327-1 (solar) and -501 (anti-solar), were installed in window containers 1B83341-1 for launch. On-orbit, the launch restraint bolts were removed and the window installed in the airlock. It remained there at all times, except when an experiment was installed, at which time it was returned to the window container and kept there until the experiment was removed.

The depressurization coupling, 1B90775-1, was mounted on the side of each window container. It mated with the male quick disconnect on an experiment or film cannister and was used to repressurize the container to cabin atmosphere.

5/ Water Container Support Structure

The location of the water containers was important from a flight dynamics and control standpoint in that it was necessary to minimize the mass moment of inertia of the Skylab cluster in order to maintain the capabilities of the control moment gyros. This situation made it desirable to locate the heavy water containers as close to the cluster center of gravity as possible. Since the cluster center of gravity was forward of the OWS, the control moment gyro consideration dictated an extreme forward location within the OWS interior (See Figure 2.2.1.2-12).

The water container support structure was designed to transfer all loads from ten water storage containers and 25 stowage containers to the habitaiton area tank cylindrical shell. The structure consisted of two basic elements. First, the circular ring structure provided the actual support for the ten individual water containers. Second, the conical load distribution ring provided load transfer from the circular ring to the habitation area tank shell.

The circular ring was designed to react applied loads so that primarily shear loads were transmitted to the shell. Container loads in the vehicle thrust direction tended to cause the circular ring cross-section to roll. However, due to its high circumferential stiffness, little rotation occurred. Therefore, there was negligible radial deflection of the joint between the circular and conical rings, and the load transmitted to the shell was shear. Container loads in the lateral directions (radial or tangential) resulted in a sinusoidal shear distribution in the circumferential plane, with peak shears occurring 90 degrees from the applied load direction. The

SKYLAB -ORBITAL WORKSHOP
WATER STORAGE CONTAINER INSTALLATION

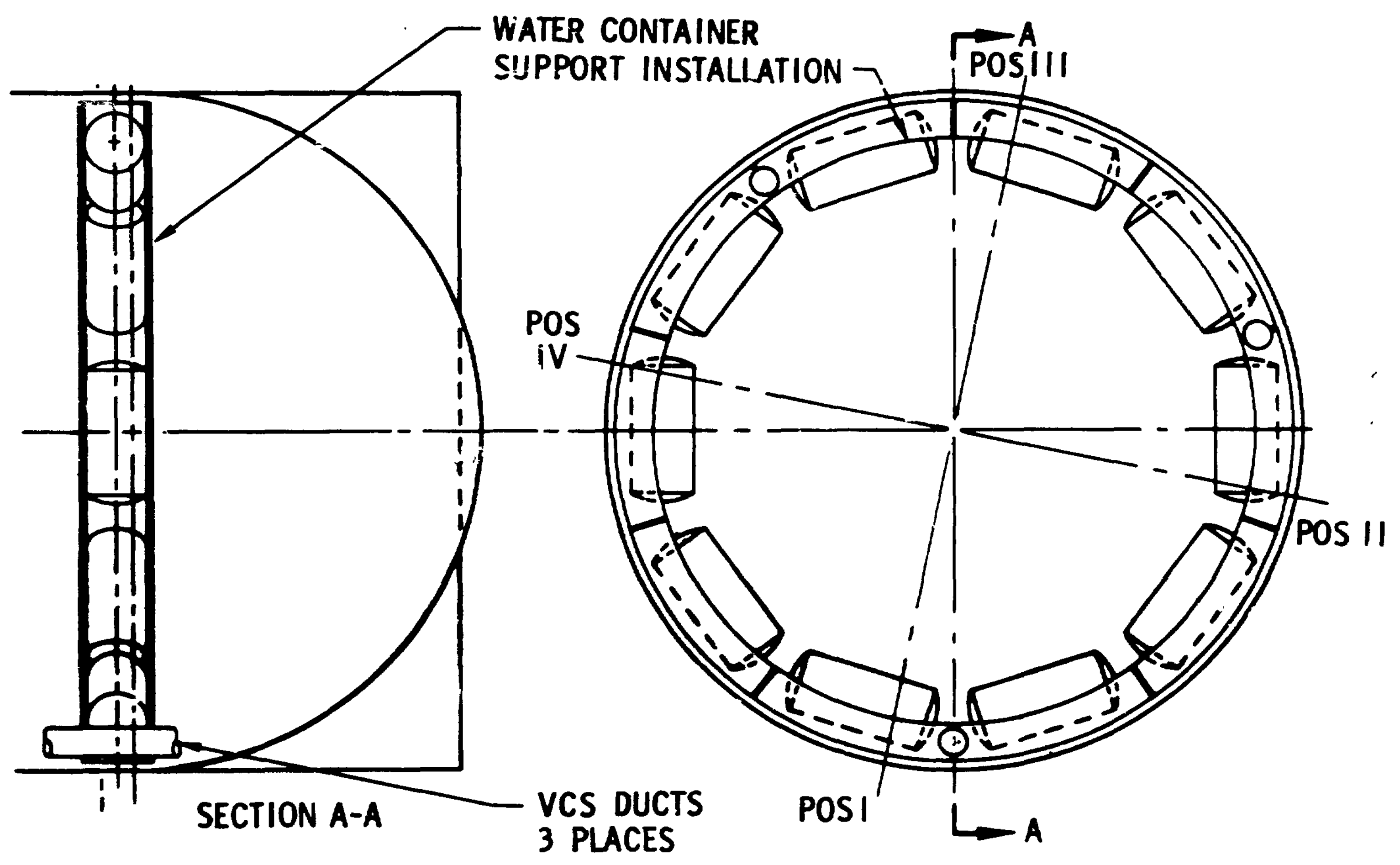


FIGURE 2.2.1.2-12

conical ring was designed to transmit container loads to the shell while minimizing radial restraint of shell growth due to internal pressurization.

The 1B77173 water container support structure circular ring consisted of two parallel cap and web frames joined by a vertical corrugated web. Within this channel-like cross section, bulkheads were properly spaced to accept the installation of the water containers.

The circular ring structure was made of ten units called frame segments. There were two basic configurations. One configuration, 1B79598, had a constant cross section while the other, 1B79633, was modified at one end to provide an opening for a VCS duct to pass through. It took four 1B79598 frame segments and six 1B79603 frame segments spliced together to make the complete ring.

The upper frame of each frame segment consisted of two extruded angle caps riveted to a beaded sheet-metal web. The lower frame was similar to the upper except that the outboard cap was a modified tee with one leg machined at a skewed angle. This leg was the interface for the conical ring.

The 1B79613 conical ring was composed of thin aluminum sheet sections spliced together to form a continuous ring assembly. The conical nomenclature was derived from the fact that one edge of the ring is attached to the habitation area tank shell, while the other edge is attached to the smaller diameter circular ring (See Figure 2.2.1.2-13).

SKYLAB - ORBITAL WORKSHOP
1/10 SEGMENT OF WATER CONTAINER SUPPORT INSTALLATION

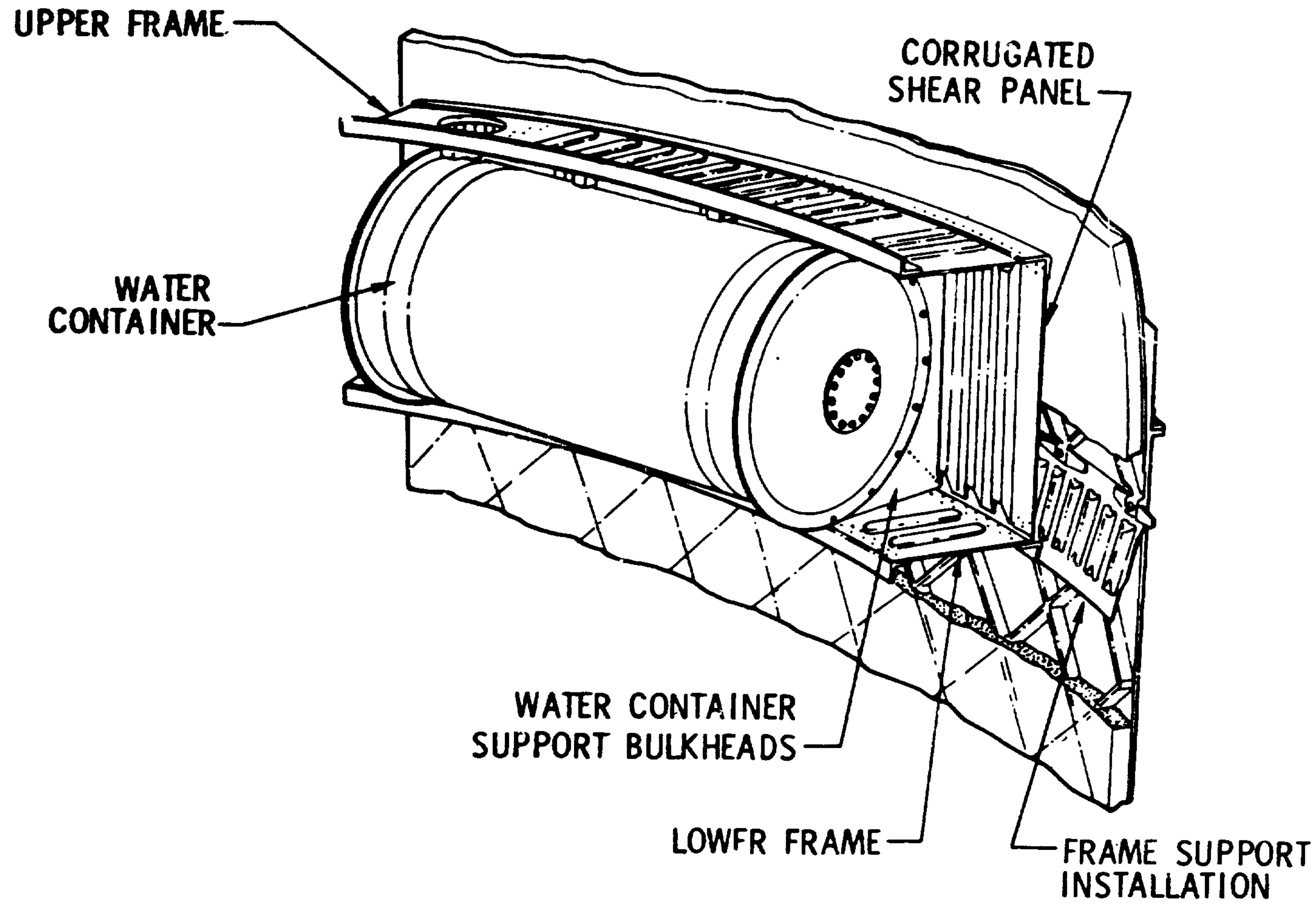


Figure 2.2.1.2-13

Two intercostals per container support bulkhead were installed to react container axial loads. The intercostals were aligned at approximately 45 degrees with respect to the water container centerline and extended from the container support bulkhead to the frame stabilization bulkhead. The intercostals provided additional stiffening to the support bulkheads to provide a more uniform load distribution on the water container support flange.

The upper and lower frames were joined on the outboard edge by a vertical corrugated sheet-metal web with Hi-Lok fasteners. Two machined bulkheads per water container were installed parallel to each other. In addition to the container support bulkheads, there were two intermediate machined bulkheads required per frame segment to maintain support structure cross-sectional stability. The conical ring was composed of 21 beaded sheet sections spliced together to form a continuous ring assembly. The inboard flange of the ring was attached to the shell with bolts on approximately 13.5 in.(34.29 cm)centers. The bolt spacing corresponded with the intersection of the waffle-pattern integral tank skin stiffeners. This intersection was selected to take advantage of the load distribution capabilities in that the conical ring tended to chord between attachments. As the chording occurred, hoop tension in the ring was reduced, thus, effectively increasing the ring flexibility. Beading the sloped portion of the ring in the meridional direction offered little restraint to expansion while maintaining the capability of taking compressive loads from vehicle dynamic transients. In order to prevent condensation on the tank wall and conical support ring on the cold side of the OWS, a series of heat-pipe thermal conductors were attached by supports which use the same tank wall attachments as the conical ring.

6/ Stowage Container Support Structure

The 1B85461 stowage container support structure was a circular ring structure attached to the forward frame of the 1B77173 water container support structure. The circular ring consisted of a cap and web horizontal frame, vertical and canted webs, and 26 vertical machined bulkheads spaced around the ring between containers. Machined toe fittings attached to each bulkhead to provide the interface with the water container support structure.

The support structure was located in a horizontal circular plane in the forward experiment compartment adjacent to the forward dome. Since the support structure was added to the program at a late date, minimum impact on manufacturing, as well as optimum load distribution, was accomplished by attaching the stowage container support structure to the water container support structure upper frame outboard cap at 26 discrete points. These points corresponded to the bulkhead locations and pick up existing fasteners in the frame cap.

The stowage container support structure was designed to transfer all loads from 25 stowage containers to the water container support structure. The structure was composed of five basic elements. The elements are a horizontal circular cap and web frame vertical bulkheads, vertical webs, inclined webs, and bulkhead end fittings (See Figure 2.2.1.2-14). The circular frame was designed to react radial load inputs from the containers. The bulkheads were designed to react vehicle thrust direction loading and to shear out circular frame reactions. The vertical and inclined webs were designed to carry lateral loading through shear. All container loading was

SKYLAB - ORBITAL WORKSHOP

WATER CONTAINER TYPICAL CROSS-SECTION

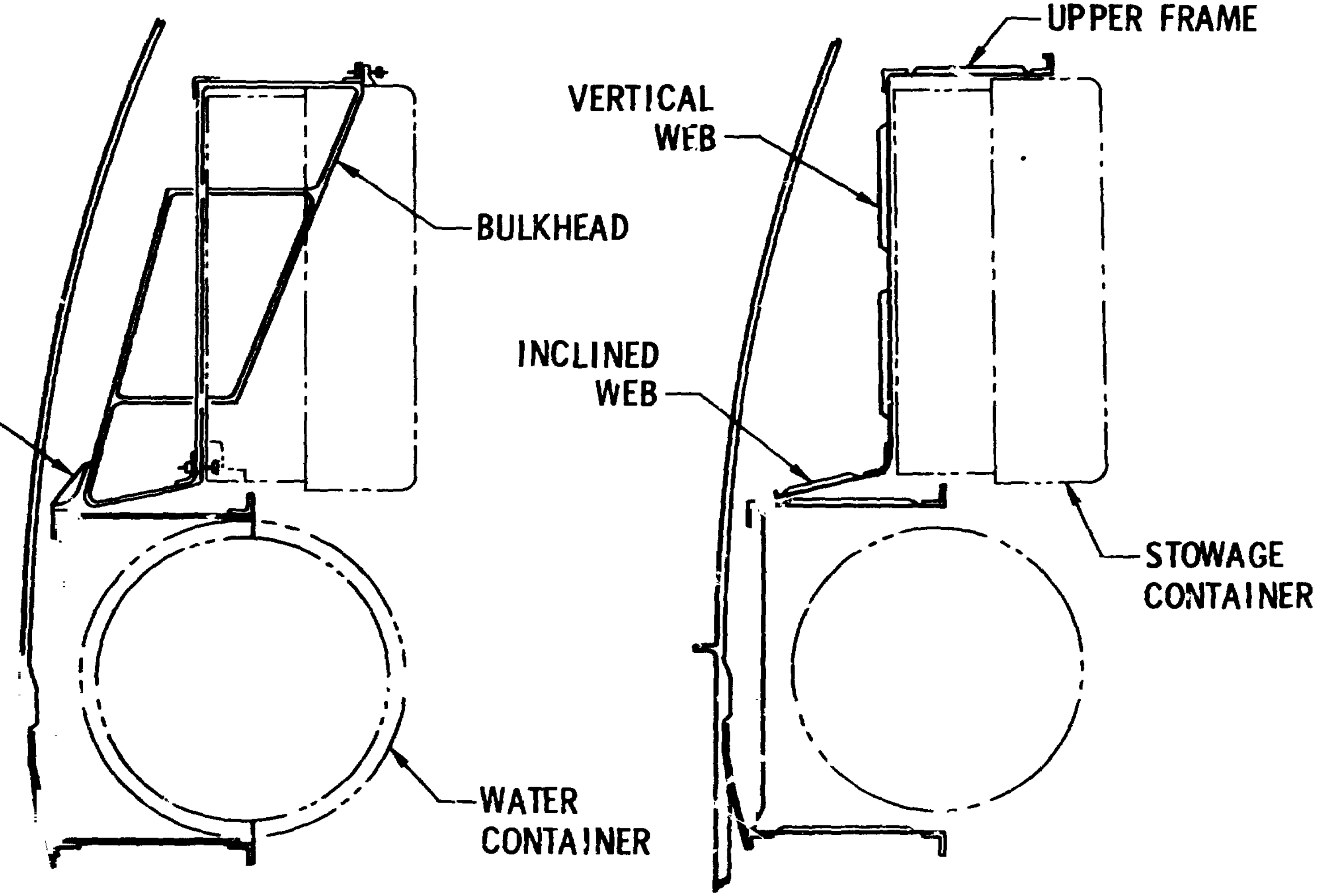


Figure 2.2.1.2-14

eventually dumped into the outboard cap of the upper frame of the water container support structure through the bulkheads and bulkhead end fittings. The loads transferred were shear and tension or compression.

7/ Crew Quarters Structure

The two floors together with the compartment walls were a structural system designed to provide living and work quarters for the crew and to provide support area for equipment and experiments. Most of the equipment and experiments were mounted on the forward surface of the forward floor. The loads from all equipment installations were distributed from floor to floor through the compartment walls. The walls acted as tension, compression, and shear members between the floors. These loads were then transferred to the cylindrical tank wall through the support cones at the periphery of each floor. These cones acted as tension members for aft loads, as compression members for forward loads, and as shear members for radial loads. The cones were also designed to accommodate the growth of the habitation area tank cylinder due to internal pressurization. The compartment walls divided the crew quarters into four compartments identified as the wardroom, experiment, waste management, and sleep compartments. See Figure 2.2.1.2-15 for compartment layout.

The crew quarters internal walls consisted of four double grid panel walls and four single d panel walls. The structure for the double grid walls was built from 3 in. (7.62 cm) wide formed and extruded aluminum channels. The horizontal channels at the top and bottom were bolted to the forward and aft floors to form an integral load-carrying structure. The intermediate horizontal, vertical, and diagonal channels divided the large grid panels into smaller panels to prevent

SKYLAB - ORBITAL WORKSHOP

"EW QUARTERS INSTALLATIONS

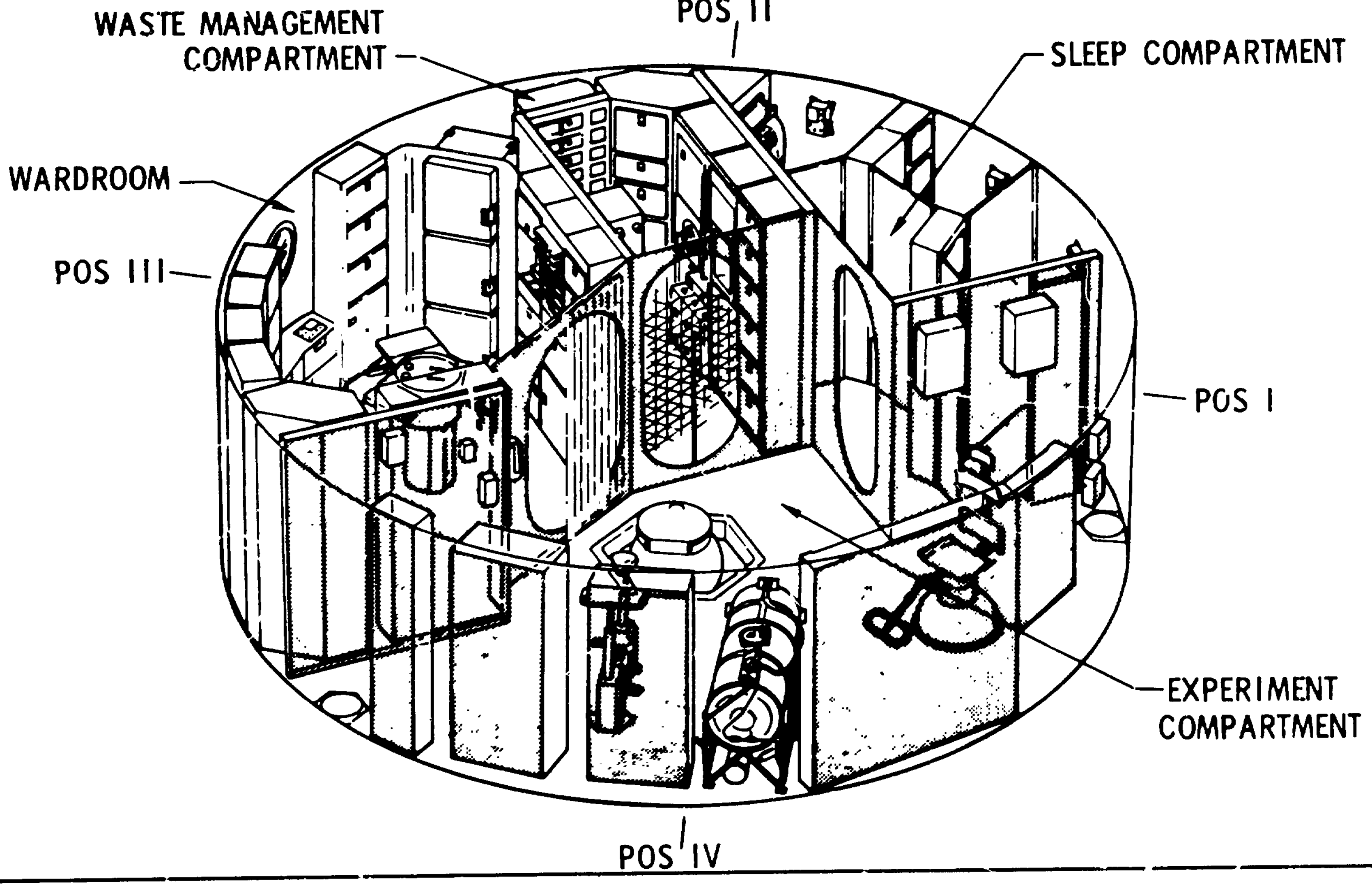


Figure 2.2.1.2-15

buckling under shear loads. The edges were attached to the ends of the double grid walls. The walls had three door openings. The opening to the wardroom was covered by a soft, slide curtain. The door to the waste management compartment was a two piece folding door of an integral skin grid construction. It folded in the center with a full-length hinge and was guided by struts running in a track at the top and bottom. The strut track follower had a built-in emergency breakaway feature. This feature was in the form of a ball and socket track follower held in position in the strut by a ball plunger detent. The resistance of this detent was set such that a load of 50 to 105 lbs (22.65 to 47.65 kg) at the center of the door would disengage the followers from the tracks and allow the door to open without folding.

The forward compartment floor consisted of an eight inch beam and intercostal structure sandwiched between open triangular pattern with intermediate intercostals where required, to carry concentrated loads from floor mounted equipment such as the film vault, food containers, experiments, etc. Figure 2.2.1.2-16 shows a typical floor arrangement with crew quarters walls. Where peak loads occurred, bathtub fittings were added to beams and intercostals to provide maximum load carrying capability.

The floor gird was machined from half inch aluminum plate to 0.400 in. (1.016 cm) thickness and into a 4.2 in. (10.668 cm) equilateral triangluar open grid pattern with holes at each of the intersections as shown on Figure 2.2.1.2-17. This open pattern allowed the atmosphere to be circulated through the habitation area and provided a mobility aid for

SKYLAB - ORBITAL WORKSHOP
CREW QUARTERS
STRUCTURAL ARRANGEMENT

*20.32 cm
**10.668 cm

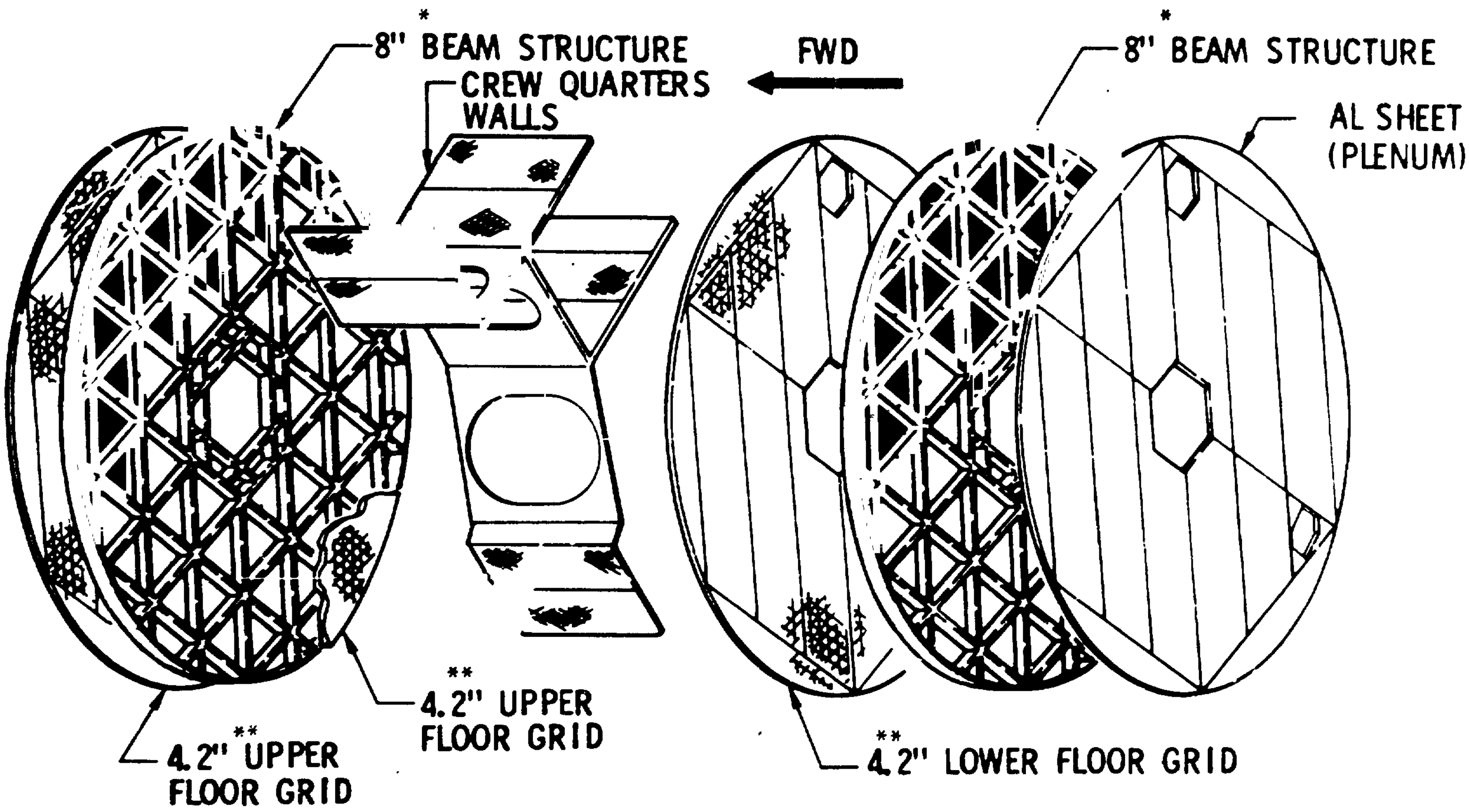
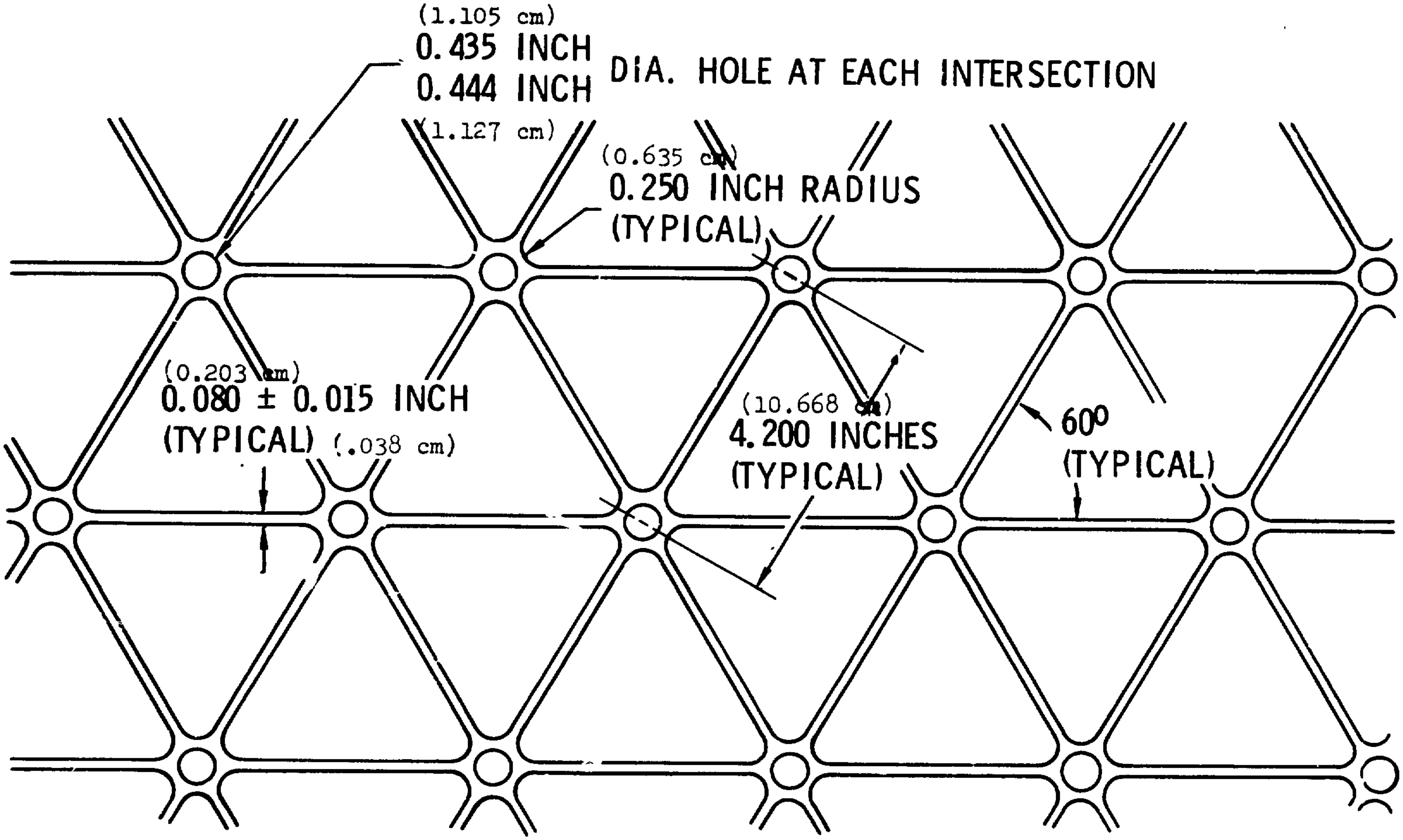


Figure 2.2.1.2-16

ORBITAL WORKSHOP
FLOOR GRID PATTERN



MADE FROM HALF-INCH ALUMINUM PLATE, MACHINED TO 0.400 INCH THICKNESS

Figure 2.2.1.2-17

the crewmen and a mounting surface for the various types of portable restraints and mobility aids. The aft surface of the upper floor had a 0.025 in. (.635 mm) thick aluminum sheet attached to the grid in the area of the waste management compartment to seal it from particulate migration. One 42 in. (106.68 cm) hexagonal opening in the center of the floor provided ingress to or egress from the crew quarters. Two smaller openings in the floor, above the sleep compartment and wardroom provided emergency egress from the crew quarters.

The crew quarters floor was located 6.5 ft. (1.98 m) aft of the bottom of the forward compartment floor. It was similar to the forward compartment floor but had a 0.040 in. (1.016 mm) thick aluminum sheet on the aft surface in place of the grid panels. This formed a plenum for the air circulation system. The floor had three openings similar to the forward compartment floor. The 42 in. (106.68 cm) hexagonal opening in the center provided access to the trash airlock and the two smaller openings (opposite the emergency egress openings in the forward compartment floor) provided access to the aft compartment. These two openings were covered to maintain a closed plenum. The forward surface of the floor had a 0.025 in. (.635 mm) thick aluminum sheet attached to the grid in the area of the waste management compartment to seal it from particulate migration.

The 1B77172 and 1B79177 conical rings were designed to transmit vehicle axial acceleration by tension and compression and lateral acceleration by shear. In addition, the conical ring was designed with low radial stiffness in order to minimize restraint due to shell growth. The conical support ring was composed of 21 beaded sheet sections spliced together to form a continuous ring assembly. The inboard flange of the ring was attached to the lower frame outboard cap with Hi-Loks. The outboard flange of the ring was attached to the lower frame outboard cap with Hi-Loks. The outboard flange was attached to the shell with bolts on approximately 13.5 in. (34.29 cm) centers. The bolt spacing corresponded with the intersection of the waffle pattern integral tank skin-stiffeners. This intersection was selected to take advantage of the load distribution capabilities of the integral stiffeners. The large bolt spacing benefited tank expansion in that the conical ring tended to chord between attachments. As the chording occurred, hoop tension in the ring was reduced, thus, effectively increasing the ring flexibility. Figure 2.2.1.2-18 shows the installation to the tank wall. A series of heat-pipe thermal conductors were attached by supports attaching both floor conical rings to the tank wall. The purpose of the heat-pipes was to prevent condensation on the installation by transferring heat from the solar (hot) side of the vehicle to the anti-solar (cold) side.

8/ Main Tunnel Structure

The OWS main tunnel was attached to the external cylindrical tank wall by angles bonded to the tank wall. The tunnel extended from the forward skirt along the tank to the aft skirt, a total distance of approximately 35 ft. (10.668 m). The tunnel configuration was developed on S-IVB and was 24 in. (60.96 cm) wide by 5 in. (12.7 cm) high. Internal stiffening frames were spaced every 9 to 10 in. (22.86 to 25.4 cm). The tunnel covered wiring and tubing that connected systems in the aft skirt with forward skirt components. Wire bundles and tubes were supported at intervals of less than 12 in (30.48 cm) by Nylafil standoffs bonded to the external tank wall.

.KYLAB - ORBITAL WORKSHOP
FLOOR STRUCTURE

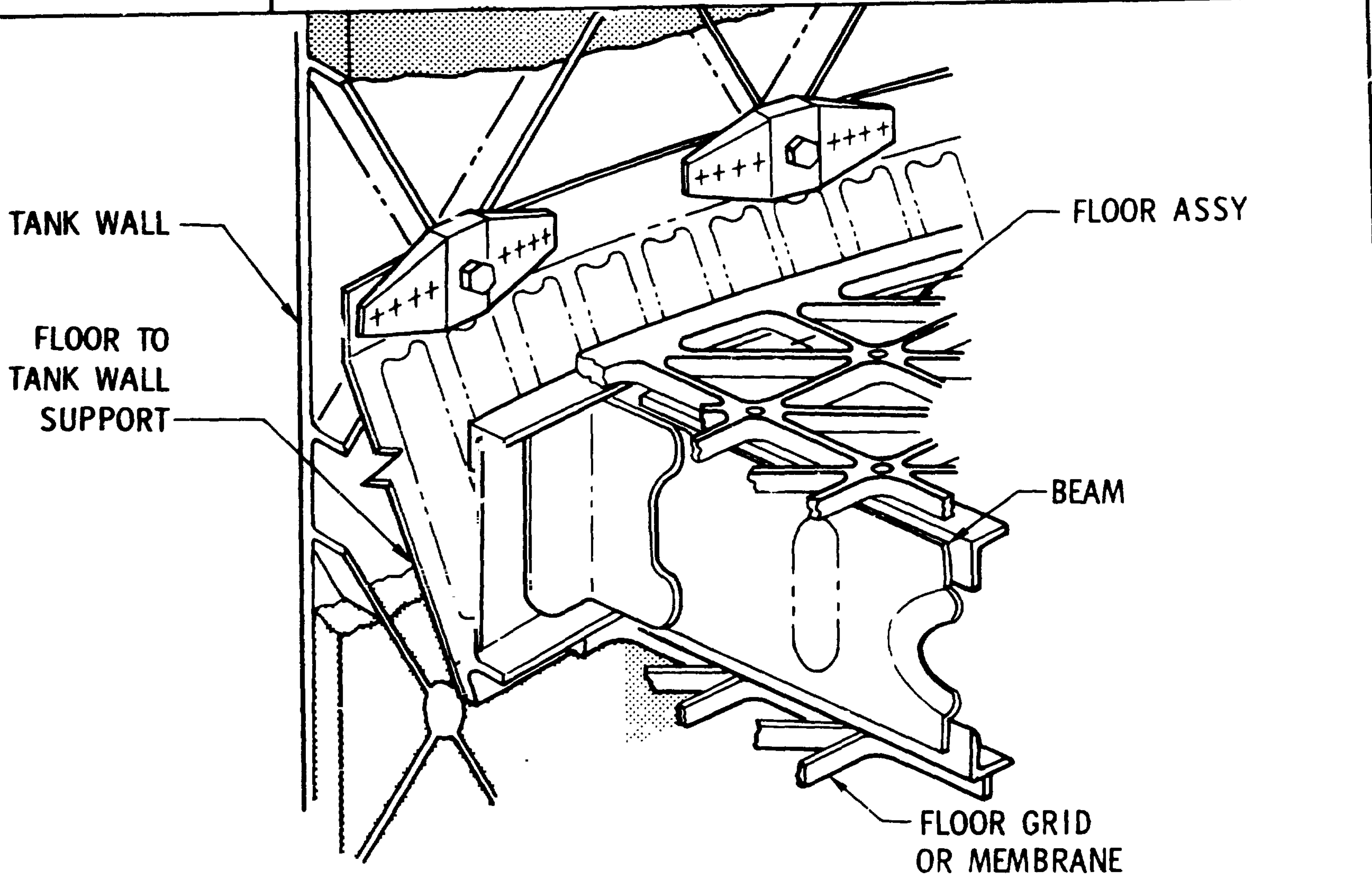


Figure 2.2.1.2-18

9/ Waste Tank Structure - The OWS waste tank was converted from the existing LOX tank of the S-IVB/V stage. It was readily accessible from the habitation area and provided 2,233 ft^3 (63.216 m^3) of volume for trash disposal. Solid waste materials entered the waste tank through a trash airlock assembly which penetrated the common bulkhead. Liquid waste entered the waste tank through liquid dump probes penetrating the common bulkhead. Non-propulsive vents were added to the waste tank to allow for venting in space. To prevent contamination of the outside of the OWS, a series of 10 micron filter screens were added to the waste tank to prevent large particles from leaving the waste tank through the non-propulsive vents.

The waste tank (1A39307) was composed of two domes welded together to form a truncated sphere. The domes were designed with a nominal 130 in. (330.2 cm) radius and the common dome was truncated to a 125 in. (317.5 cm) radius. The maximum height between the two domes was 178.162 in. (452.53 cm).

The aft dome (1B63286) was a 260 in. (660.4 cm) diameter hemisphere, made up of nine segments of a sphere. The dome was made from 2014-T651 aluminum and had a minimum thickness of 0.082 in. (2.08 mm). The aft dome was welded to the cylinder as shown in Figure 2.2.1.2-19.

The common bulkhead (1A39309) was a 260 in. (660.4 cm) diameter partial hemisphere with a forward and aft face of 2014-T651 aluminum, separated by 1.75 in. (4.445 cm) of 3/16 in. (.475 cm) fiberglass honeycomb bonded to each face. The common bulkhead was welded to the aft dome with two circumferential welds, one to each facing sheet. The "Y" joint at the aft dome intersection was also lockbolted.

ORBITAL WORKSHOP WASTE TANK

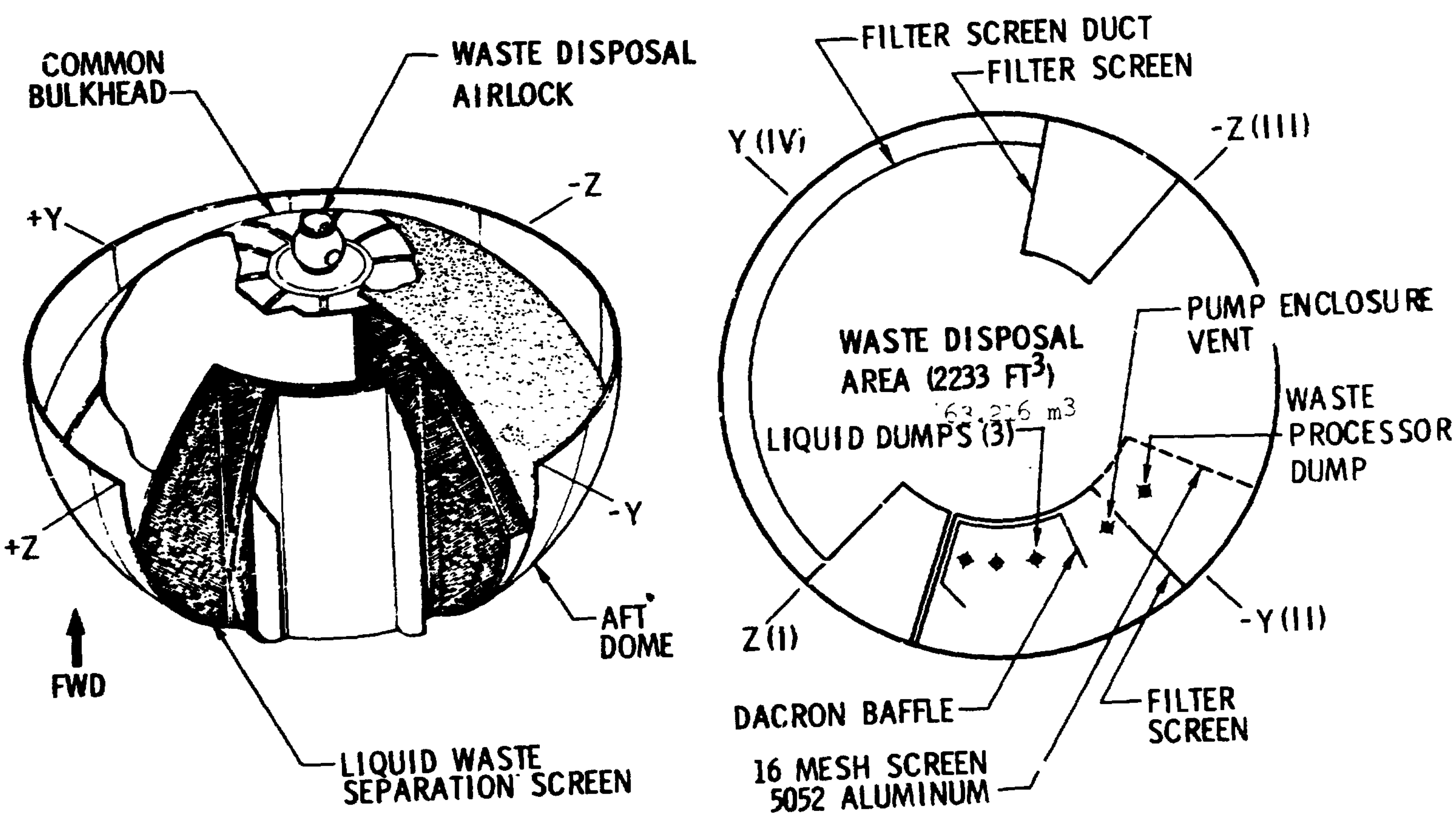


Figure 2.2.1.2-19

The trash screen formed compartments within the waste tank. Two screens formed cylindrical segments on a 65 in. (165.1 cm) radius running between the common bulkhead and the aft dome. The cylindrical screens were joined by side curtains which ran radically outboard from the cylindrical screen. The screens were supported by a system of longerons running between the aft dome and the common bulkhead. The two NPV vent screens were connected by a 12 in. (30.48 cm) radius half torus which ran around the waste tank at the intersection of the aft dome and the common bulkhead to equalize pressure between the two NPV's.. (See Figure 2.2.1.2-19.)

The trash screen (1B79330) was constructed of fine diameter wire woven in a twilled dutch weave. The wire was 304L stainless steel wire. There were 325 wires, 0.0014 in. (.0355 mm) in diameter in the wrap direction and 2,300 wires, 0.0010 in. (.0254 mm) in the diameter in the chute direction. A section of the trash screen was constructed using 16 x 16 mesh 5056 aluminum screen with a wire diameter of 0.018 in. (457 mm). There were two layers of 325 x 2,300 mesh filter screens separating the NPV compartment and the liquid dump compartment. These screens were separated by aluminum spacers and stainless steel bolts spaced 12 in. (30.48 cm) on center. A 1-1/2 in. (3.81 cm) band of aluminized tape ran around the periphery of the screen segments.

The trash screen support structure (1B79330) consisted of a series of angles spanning between the common bulkhead and the aft dome located on a 65 in. (165.1 cm) radius. These angles were 2-1/2 in. (6.35 cm) x 2-1/2 in. (6.35 cm) x 3/16 in. (.476 cm) 7075-T6 extrusions. They were supported by 2024-T4 tee slips (1B84444) bonded to the waste tank domes with polyurethane adhesive. The angles were bolted to the tee clips with 3/16 in. (.476 cm) diameter stainless steel bolts and were designed with expansion joints so that the tank was not loaded when the waste tank expanded due to pressurization in the ascent mode. The screen was also supported by tee clips bonded to the domes with polyurethane adhesive. The screen was bolted to the tie clips with 3/16 in. (.476 cm) diameter stainless steel bolts.

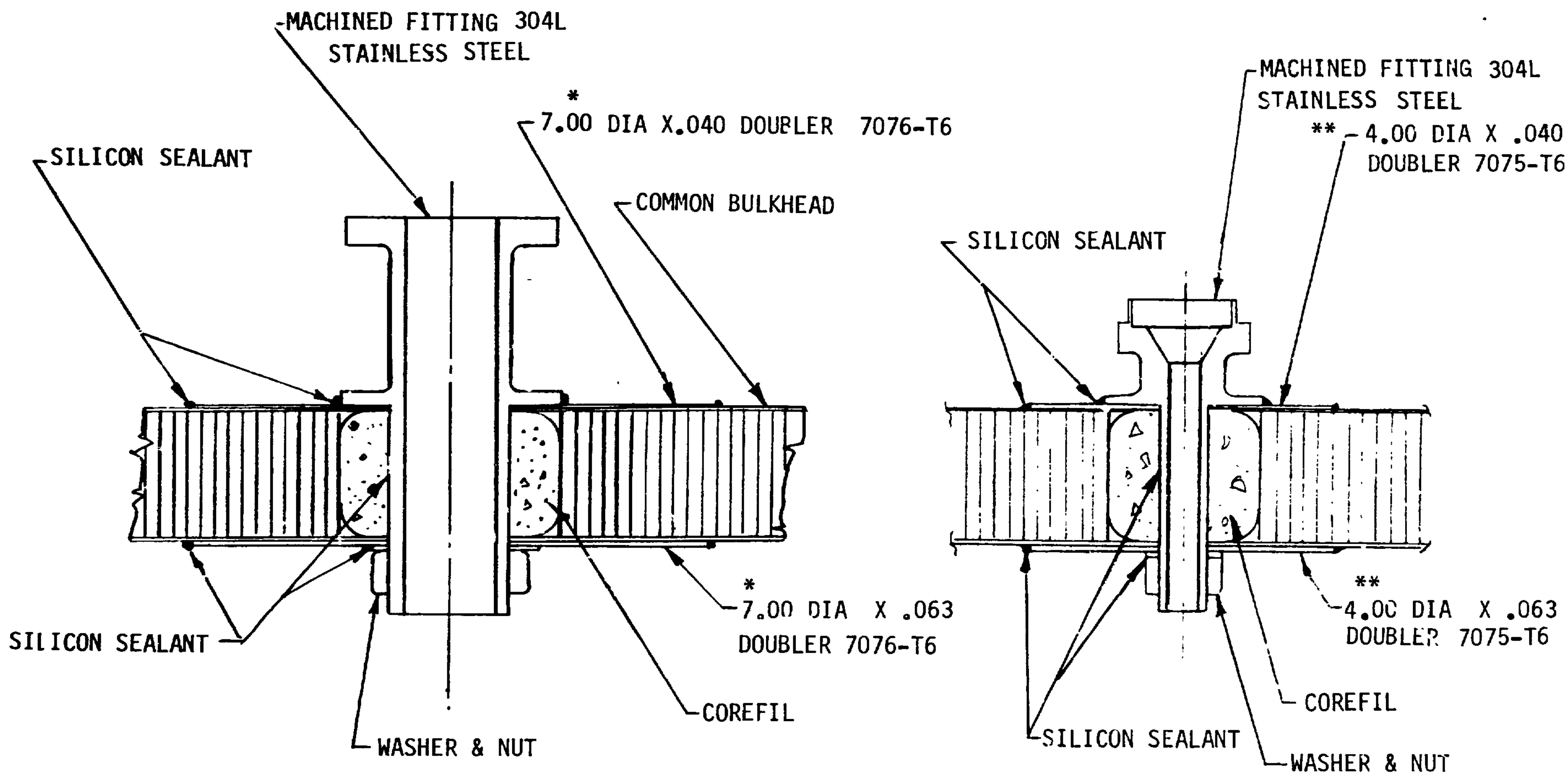
The half toroid vent duct was supported by 1 in. (2.54 cm) a 5/8 in. (1.587 cm) x 1/32 in. (.762 mm) aluminum angle arches located at 15-degree intervals around the toroid (Figure 2.2.1.2-19). The arches were attached to the domes with 2 in. (5.08 cm) x 2 in. (5.08 cm) x 1 in. (2.54 cm) x 0.094 in. (.2387 cm) tee clips that had been bonded to the domes with polyurethane adhesive. The arches were attached to the tee clips with 3/16 in. (.476 cm) diameter stainless steel bolts.

A 6 in. (15.24 cm) x 2-1/2 in. (6.35 cm) x 0.032 in. (.812 mm) deflector was attached to the common bulkhead with tee clips to prevent direct impingement of urine on the wardroom water dump probe.

The penetrations through the waste tank were all designed using a fitting to feed through the skin. The areas adjacent to the openings in the basic tank structure were all reinforced with appropriate doublers to lower stresses due to discontinuities. The penetrations were doubly sealed with O-rings and a sealant. (See Figure 2.2.1.2-20.)

The liquid and vacuum dump probe installation through the common bulkhead consisted of a 14.5 in. (36.83 cm) diameter hole for the trash disposal airlock fitting, three 5/8 in. (1.588 cm) diameter hoels for the liquid dump fittings, a 1-5/8 in. (4.128 cm) diameter hole for the waste processor dump fitting, and a 1-5/8 in. (4.128 cm) diameter hole for the refrigeration pump vacuum dump fitting. The dump probes were defined on Drawing 1B79329 The non-propulsive vents penetrated the aft dome in two places at station 234,437, 30 degrees from Position I and IV.

ORBITAL WORKSHOP
PENETRATION DETAILS-COMMON BULKHEAD



*17.78 cm DIA x 1.016 mm DOUBLER
**10.16 cm DIA x 1.016 mm DOUBLER

WASTE PROCESSOR FEED THRU

LIQUID DUMP FEEDTHRU

Figure 2.2.1.2-20

The trash airlock feedthrough penetrated the center of the common bulkhead with a two piece, machined fitting which was bolted to the common bulkhead with stainless steel bolts. A 13.81 in. (35.077 cm) I.D. ring (1B79308) penetrated through the common bulkhead and supported the trash airlock. A flange was machined into the ring to pick up attachments through the forward skin on the common bulkhead. A support ring, 1B79307 was machined from a 7075-T651 aluminum plate and bolted to the 1B79308 ring and the aft skin of the common bulkhead. The cut-out in the common bulkhead was reinforced by two 31-3/4 in. (80.65 cm) O.D. diameter doublers, 1B79329-9 and 1B79329-11. The doublers were bonded to the common bulkhead skins with ployurethane adhesive. The installation was sealed with silicone sealant.

The liquid dump fittings (1B79305) were designed similar to a flanged bolt. The fittings were machined from a 304L stainless steel bar and designed with a flange that rested against the forward skin of the common bulkhead.

A stainless steel nut was threaded on the fitting shaft and torqued against the aft side of the common bulkhead. The cut-out in the common bulkhead was reinforced on each side by 4 in.

(10.16 cm) O.D. doublers (1B79325 and 1B79322) bonded to the common bulkhead with polyurethane adhesive. The installation was sealed around the doubler periphery with silicone sealant.

The two vacuum dump fittings (1B79306) were similar to the liquid dump flanged bolt fitting. The cutout in the common bulkhead was reinforced by two 7 in. (17.78 cm) diameter doublers, 1B79326 and 1B79331, one on each side of the common bulkhead. The doublers were bonded to the bulkhead and sealed in the same manner as the liquid probes.

The two non-propulsive vents (1B87877) were designed to allow the waste tank to vent through vent ducts. A machined fitting (1B87786) reinforced the aft dome skin and supported the non-propulsive vent duct. The 6061-T651 fitting was designed with a flange that acted as a doubler to reinforce the aft dome. The 1B87786 fitting was attached to the aft dome by means of polyurethane adhesive and 3/16 in. (.476 cm) diameter lockbolts. The installation was sealed with silicone sealant.

The LOX instrument probe, P.U. probe, SVE wire lead-out port, LOX helium heater, and LOX chill return penetrations were sealed with an Aeroquip 59021 blind flange and ST152 conoseal gasket coupling. A secondary seal of the joint was achieved with a fillet bead of silicone sealant.

10/ Crew Accommodations - Closeouts (barriers) in the wardroom and sleep compartment under stationary lockers were installed to prevent particles and articles greater than 0.2 in. (.51 cm) diameter from entering inaccessible areas. All other closeouts were sized to prevent particles greater than 0.4 in.(1.0 cm) in diameter from entering inaccessible area. An area was considered accessible if a person could see into it with the aid of an inspection mirror and if an object could be retrieved from it by hand, vacuum cleaner, or retrieval aid tool. Also, areas beneath movable experiment boxes and under fixed hardware mounted into open grid with a reach distance less than 8 in. (20.3 cm) measured horizontally from two adjacent grid openings were considered accessbile.

The WMC was sealed to prevent water, urine, hair clippings, fecal matter, etc., from leaving the compartment or being lodged in inaccessible spaces within the compartment. Sealing was accomplished with fluorocarbon rubber cove strip attached to all equipment cabinets and stowage lockers at their intersection with floor, wall, and ceiling panels (Figure 2.2.1.2-21).

The following types of closeouts were installed at other locations in the OWS.

- o Rubber and aluminum foil patches for closing lightening holes in floor intercostals.
- o Formed floor-to-ceiling aluminum panels between lockers, partitions, freezer, entertainment locker, etc. (Figure 2.2.1.2-22).
- o Rubber floor to ceiling cove strips between lockers, partitions, consoles, and the tank wall (Figure 2.2.1.2-22).

ORBITAL WORKSHOP
WMC CLOSEOUT AT FLOOR

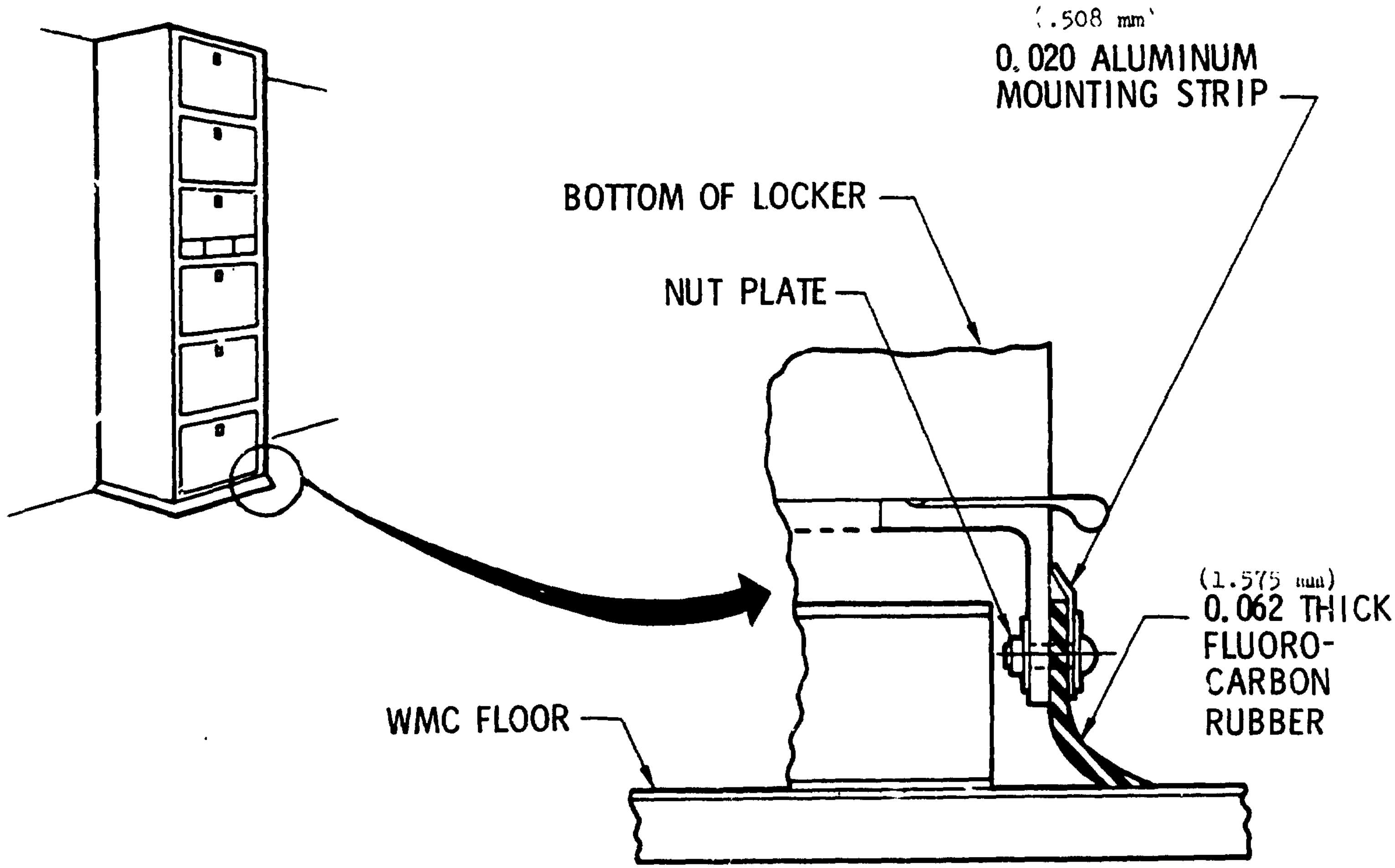


Figure 2.2.1.2-21

ORBITAL WORKSHOP
CLOSEOUTS BETWEEN FLOOR & CEILING
(17 PLACES)

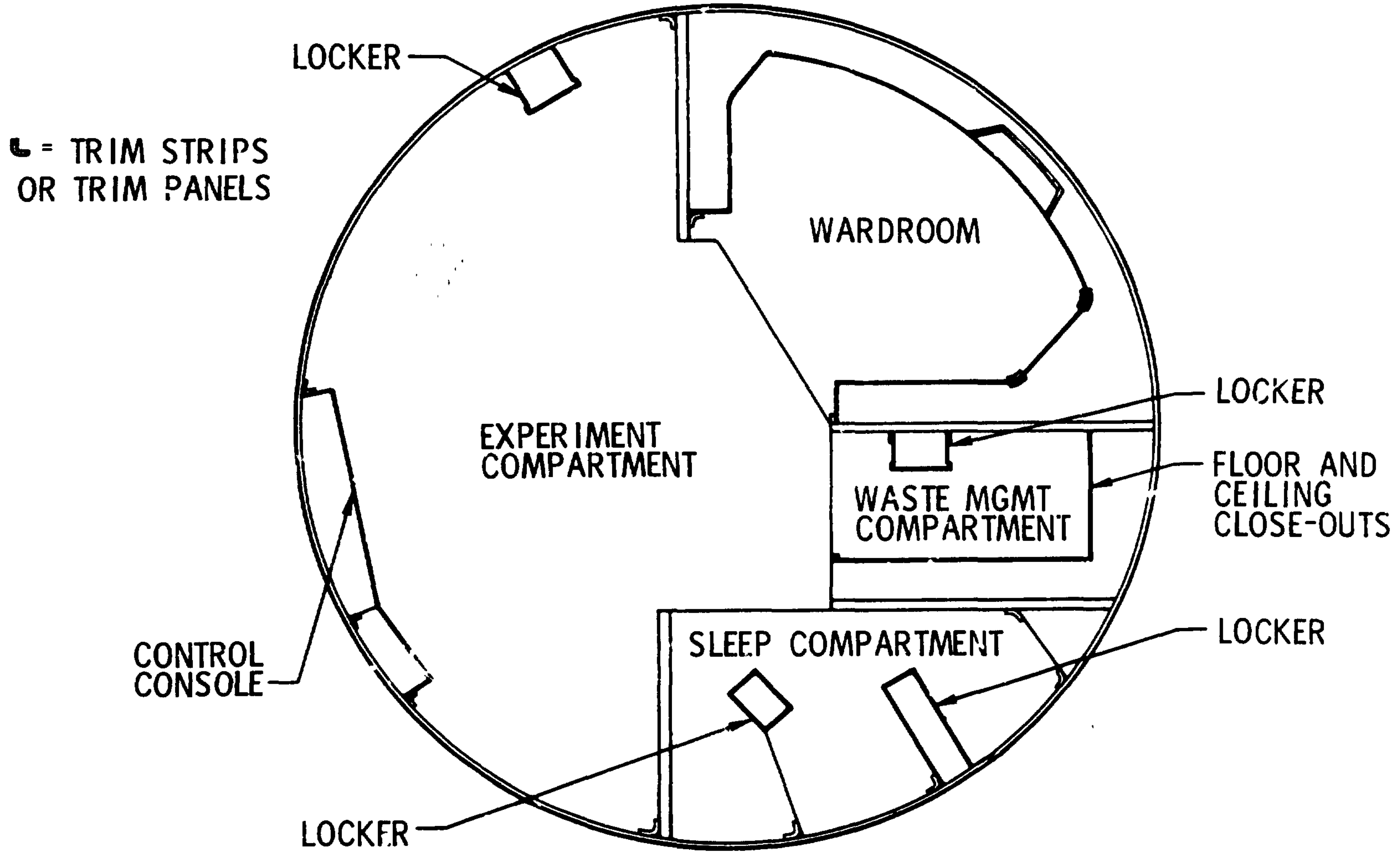


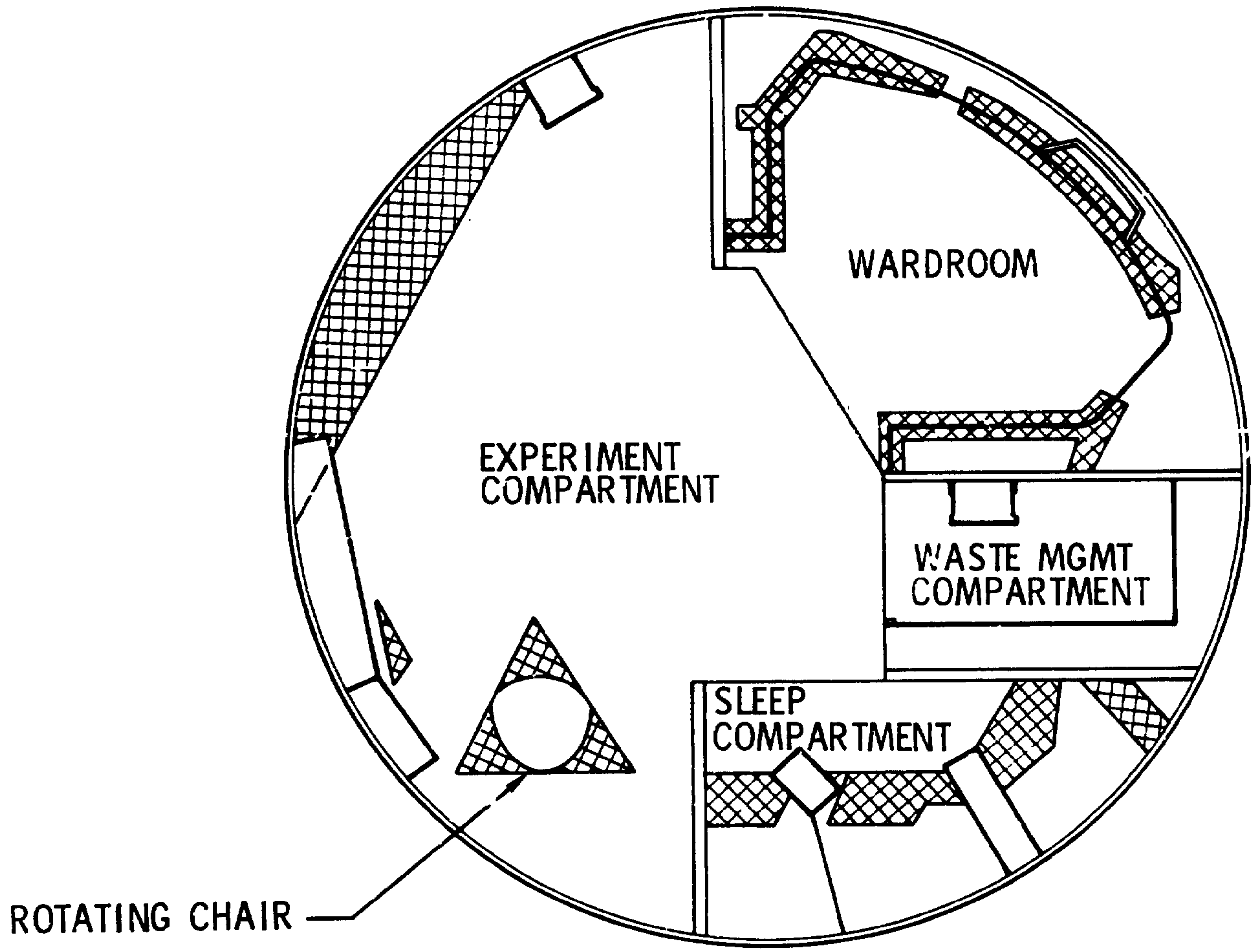
Figure 2.2.1.2-22

- Aluminum rubber trim strips on faces of lockers at floor and ceiling intersections.
- Aluminum panels on aft floor grid adjacent to lockers and equipment (Figure 2.2.1.2-23).
- Perforated aluminum panels on aft ceiling and forward floor adjacent to lockers and equipment (Figure 2.2.1.2-24).
- Aluminum wire screen above ground thermal conditioning unit in wardroom. (Screen openings were greater than .2 in. (.508 cm) due to airflow restrictions).
- Perforated aluminum strips above and below water containers (Figure 2.2.1.2-25).
- Aluminum wire screen and rubber cove strips outboard of food container rack to tank wall.

The wardroom curtain was a single accordian-pleated fabric panel held taut between floor and ceiling tracks. (Figure 2.2.1.2-26). One vertical edge was permanently attached to the wardroom wall and the other was stiffened with a full length aluminum tube which served as the hand-hold for operating the curtain. In its stowed or folded position, it was retained by 3 loop straps. A single strap on the tube was snapped to the wardroom wall to hold it in the fully deployed position. The curtain fabric was TFE coated beta glass with a sewn on matrix of stainless steel yarn for grounding electrostatic charges. Glass-filled TFE ball-sliders were used in aluminum tracks to reduce friction and preclude binding. A break-away feature was added by attaching the curtain to the sliders with snaps and to the vertical tube with velcro.

The interior of the OWS and equipment surfaces which were normally visible to the crew were finished in a color scheme

ORBITAL WORKSHOP CLOSEOUTS ON AFT FLOOR



ORBITAL WORKSHOP FORWARD FLOOR CLOSEOUTS

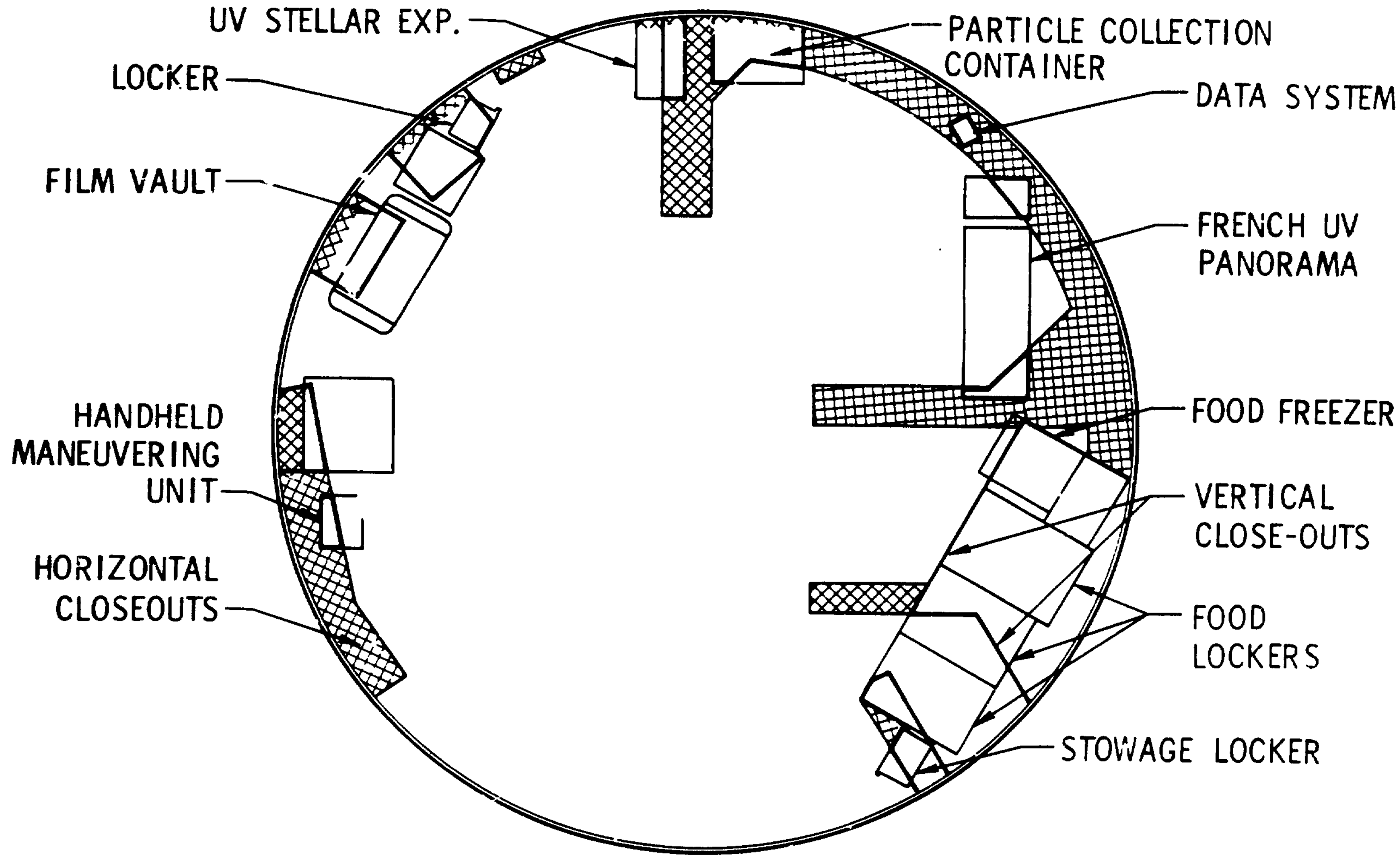


Figure 2.2.1.2-24

ORBITAL WORKSHOP
WATER TANK CLOSEOUTS

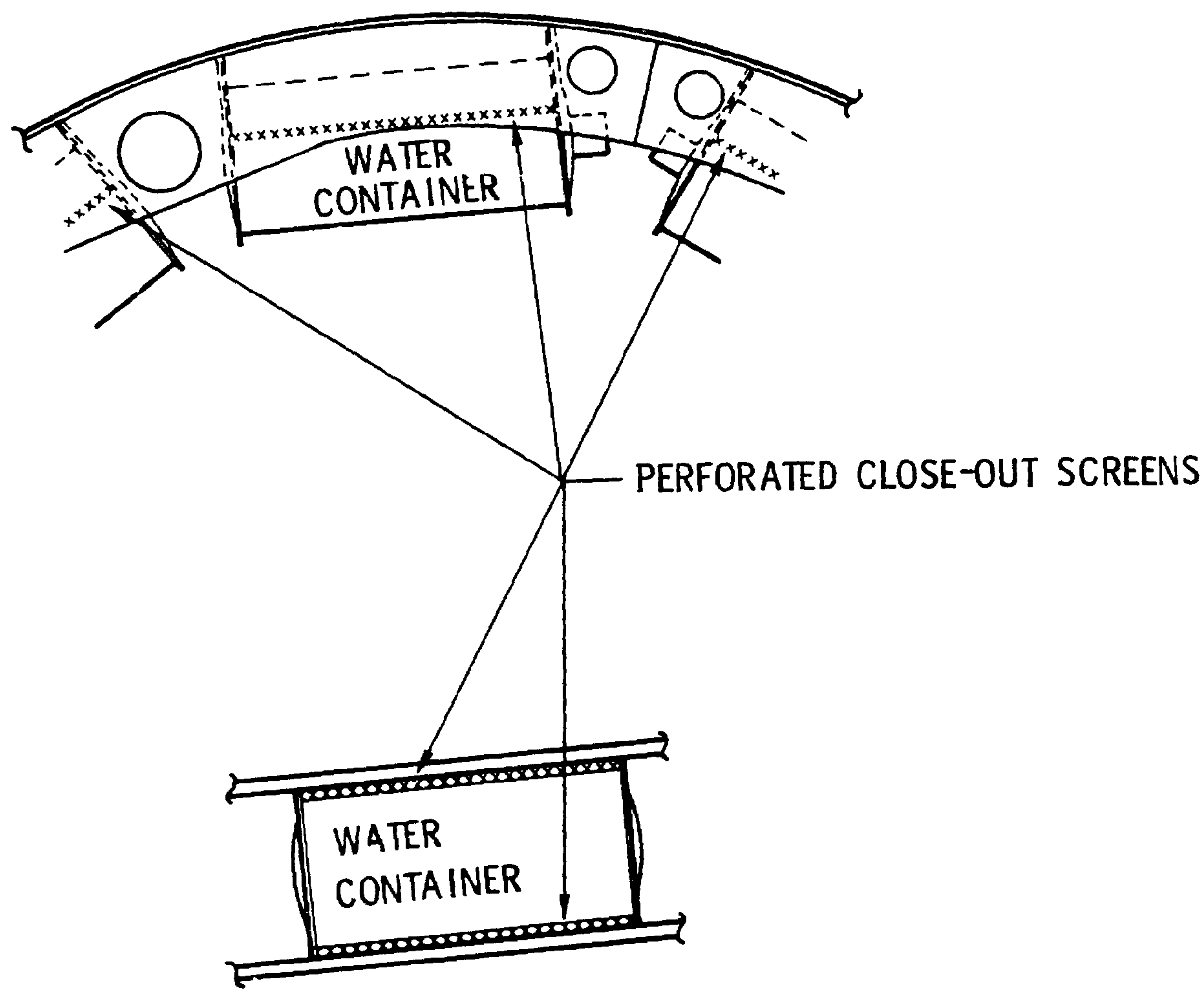


Figure 2.2.1.2-25

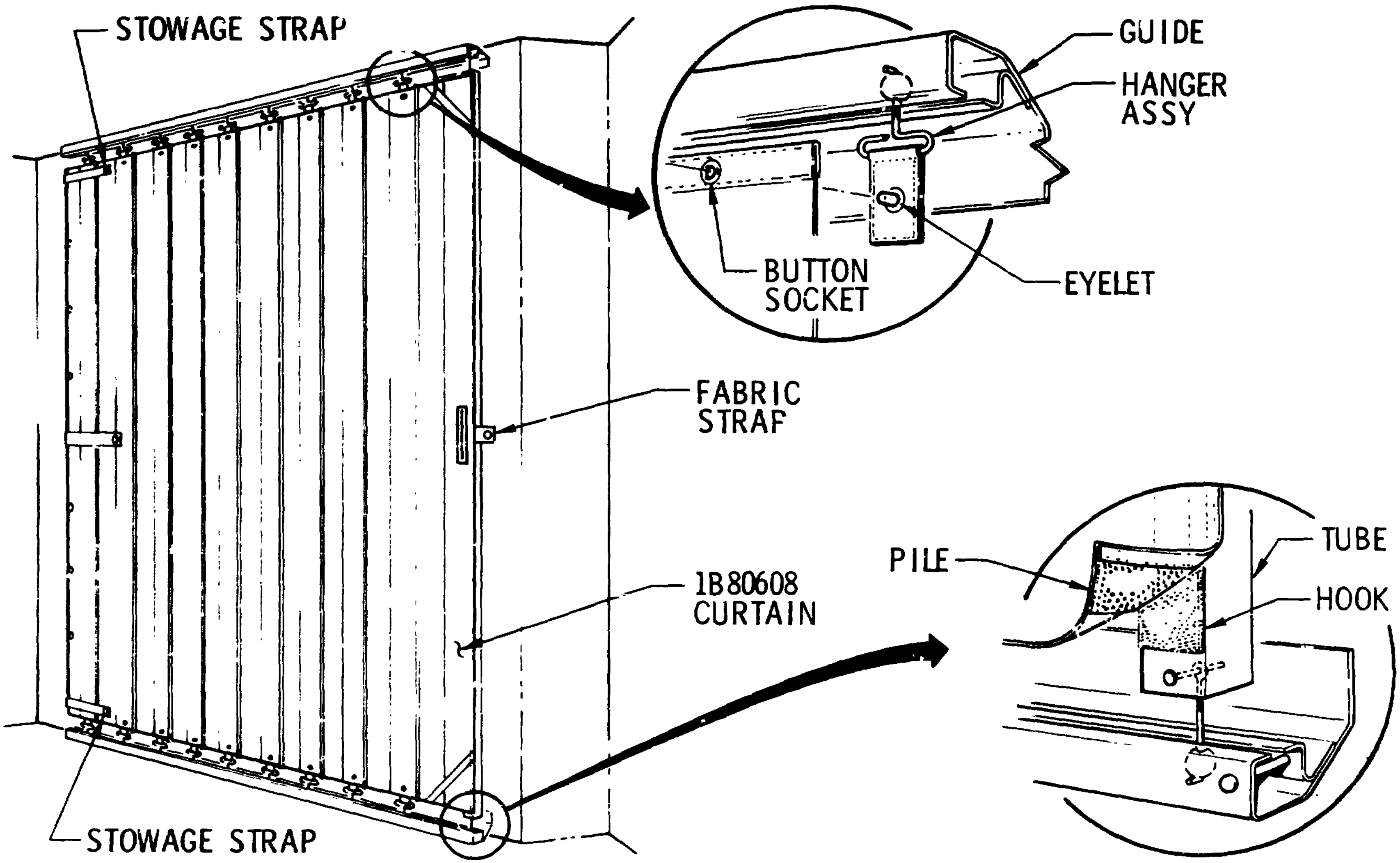


Figure 2.2.1.2-26. Wardroom Entry Curtain

FIGURE 2.2.1.2-26

designed to enhance both crew comfort and interior illumination while meeting thermal, toxicity and durability requirements. Most materials were used in their natural finishes, e.g., stainless steel and fabrics, or integral color finishes as in the case of aluminum anodize. Since the optical reflectance of these metallic finishes was relatively low, off-white paint was applied to stowage locker faces in the wardroom and forward dome areas to improve internal illumination and on the electrical control panels to provide adequate legibility of control markings.

The crews conducted habitation activities, work tasks, and experimentation in a zero-G environment which necessitated the use of special aids to restrain the crewman and loose equipment. Grid, foot restraints, handrails, handholds, tethering devices, thigh restraints, sleep restraints, and waste management restraints were supplied for restraining the crewmen and aiding him in moving about the OWS. Some portable restraints were provided in addition to those permanently attached to structure.

Grid was used in the OWS as structural members but also served as a restraint for the crewman and was provided in three forms: open grid, closed grid, and modified grid. Open grid was located in the crew quarters area and provided an open triangular area large enough to permit the crewman to grasp the grid as a hand restraint or as a mobility aid to assist him in traversing a surface (Figure 2.2.1.2-27). Closed grid, used mainly on the crew quarters partitions, could not be grasped because it contained an integral skin cover. Modified grid, located on platforms in the forward OWS area, contained triangular open areas and other geometrical openings which were not arranged in the symetrical grid pattern. All three types of grid permitted the installation of portable restraints into their hole and triangle pattern.

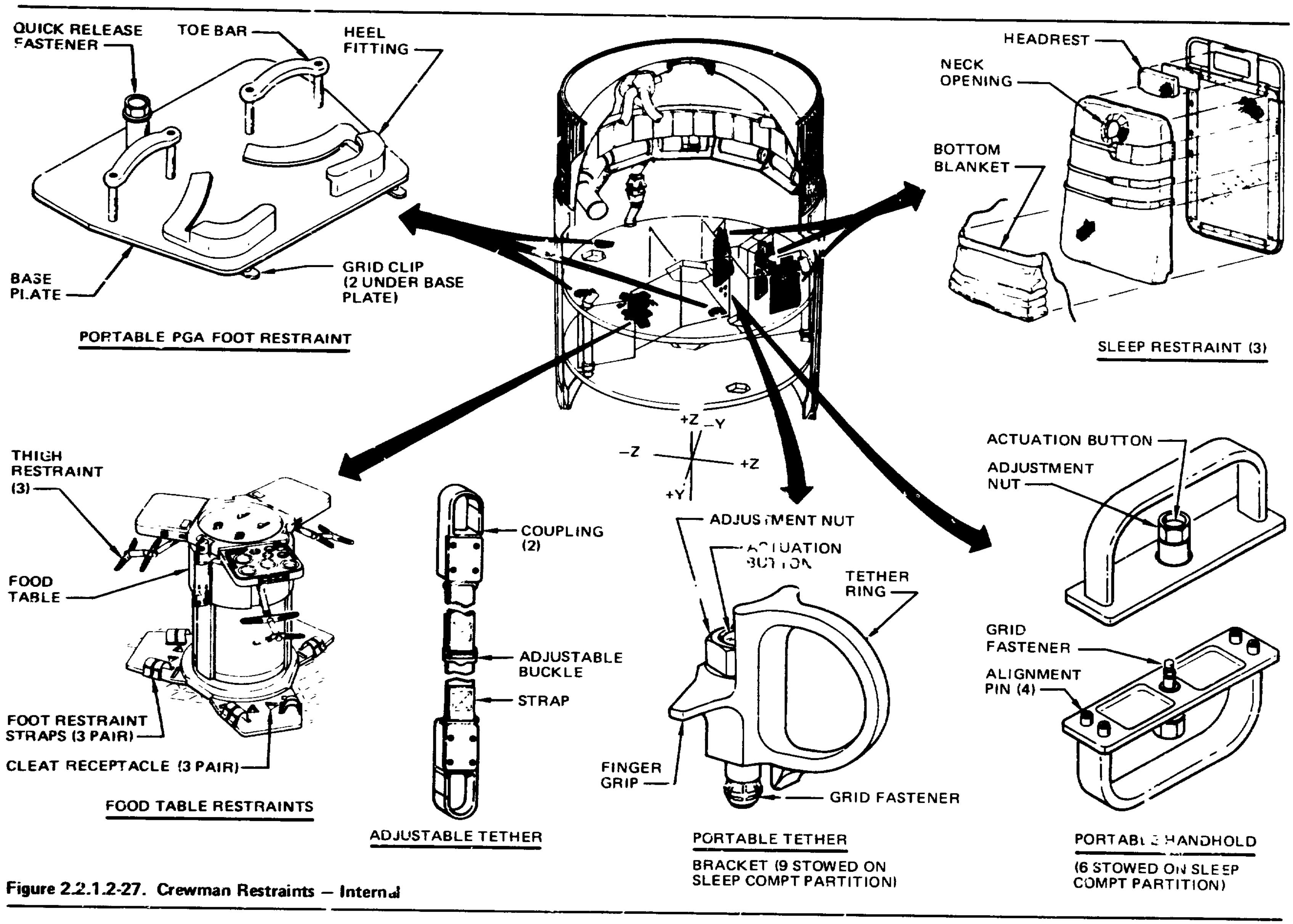


Figure 2.2.1.2-27. Crewman Restraints – Internal

FIGURE 2.2.1.2-27

Foot restraints were provided in the OWS to permit restraint of the crewman's feet. Restraint at the foot-level provided body restraint in a desired orientation to free the crewman's hands to perform two-handed work activities in the zero-G environment. These restraints were provided in fixed and portable forms to permit operational flexibility for habitation activities. Platform foot restraints, light-duty foot restraints, food table foot restraints, and a footwell composed the fixed restraints. Portable restraints consisted of triangular shoes for each crewman and three portable PGA foot restraints.

The OWS platform foot restraints, located under the water tanks in the OWS forward area, were composed of sections of modified grid platforms, rigidly mounted around the periphery of the tank wall (Figure 2.2.1.2-28). Each platform section contained: (1) the hole pattern found in grid to facilitate the use of any of the portable restraints; (2) cleat receptacles, which accept the insertion of the triangle shoe for foot restraint; and (3) open slots to permit the insertion of the bare foot for convenient foot restraint or to serve as a hand restraint. The OWS platform foot restraints were used mainly to gain access to the D400 series stowage compartments and the water tanks.

Two pairs of light-duty foot restraints were permanently located on the WMC floor, one in front of the fecal/urine collector and one in front of the handwasher (Figure 2.2.1.2-29). The foot restraints permitted waste management activities, personal hygiene functions and equipment servicing. Each of the light-duty foot restraints was fitted with two velcro-lined straps to provide an adjustment for each crewman's foot.

The base of the food table was used to restrain the crewman's

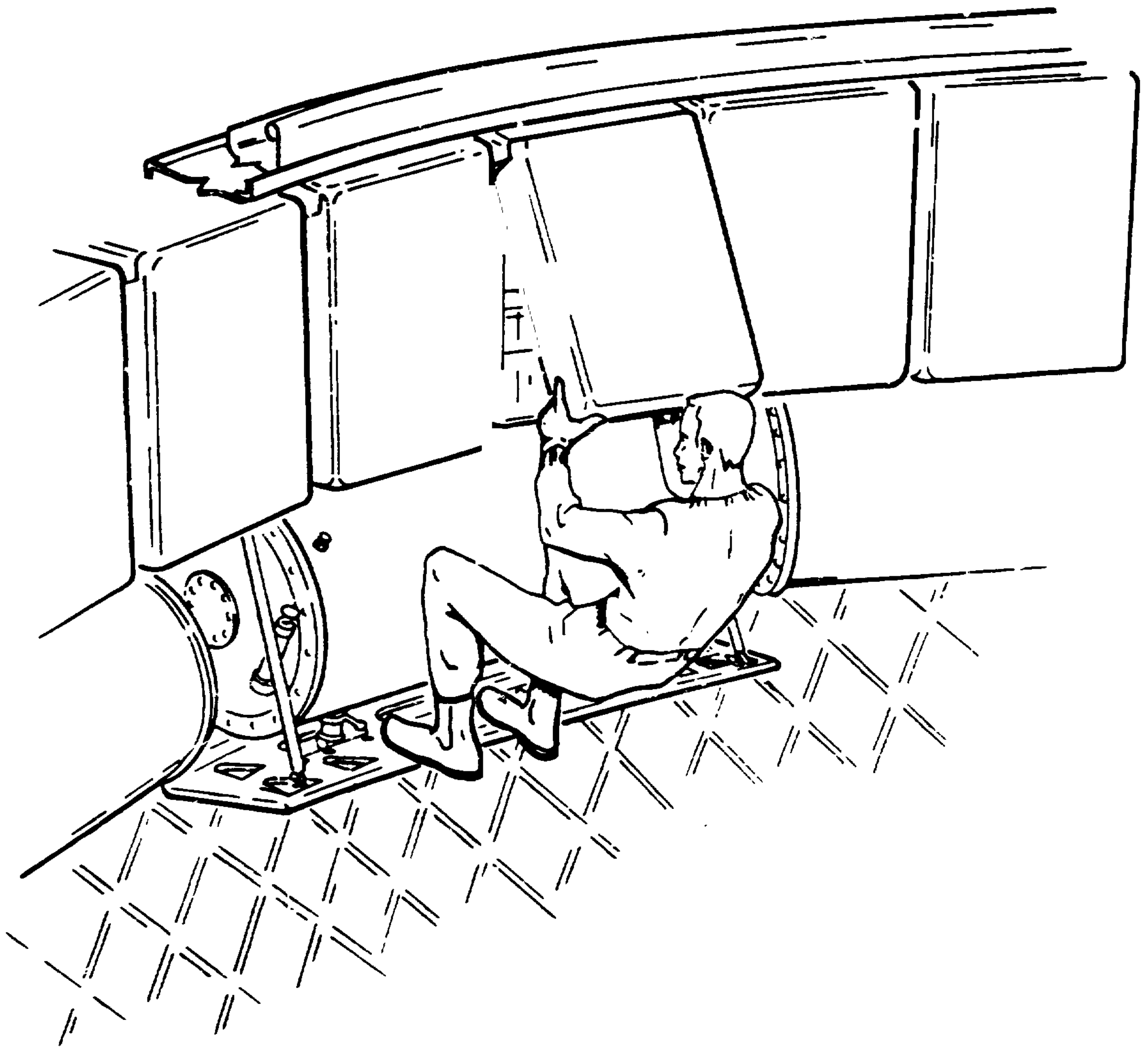


Figure 2.2.1.2-28. Astronaut Aids (Platform Foot Restraints)

FIGURE 2.2.1.2-28

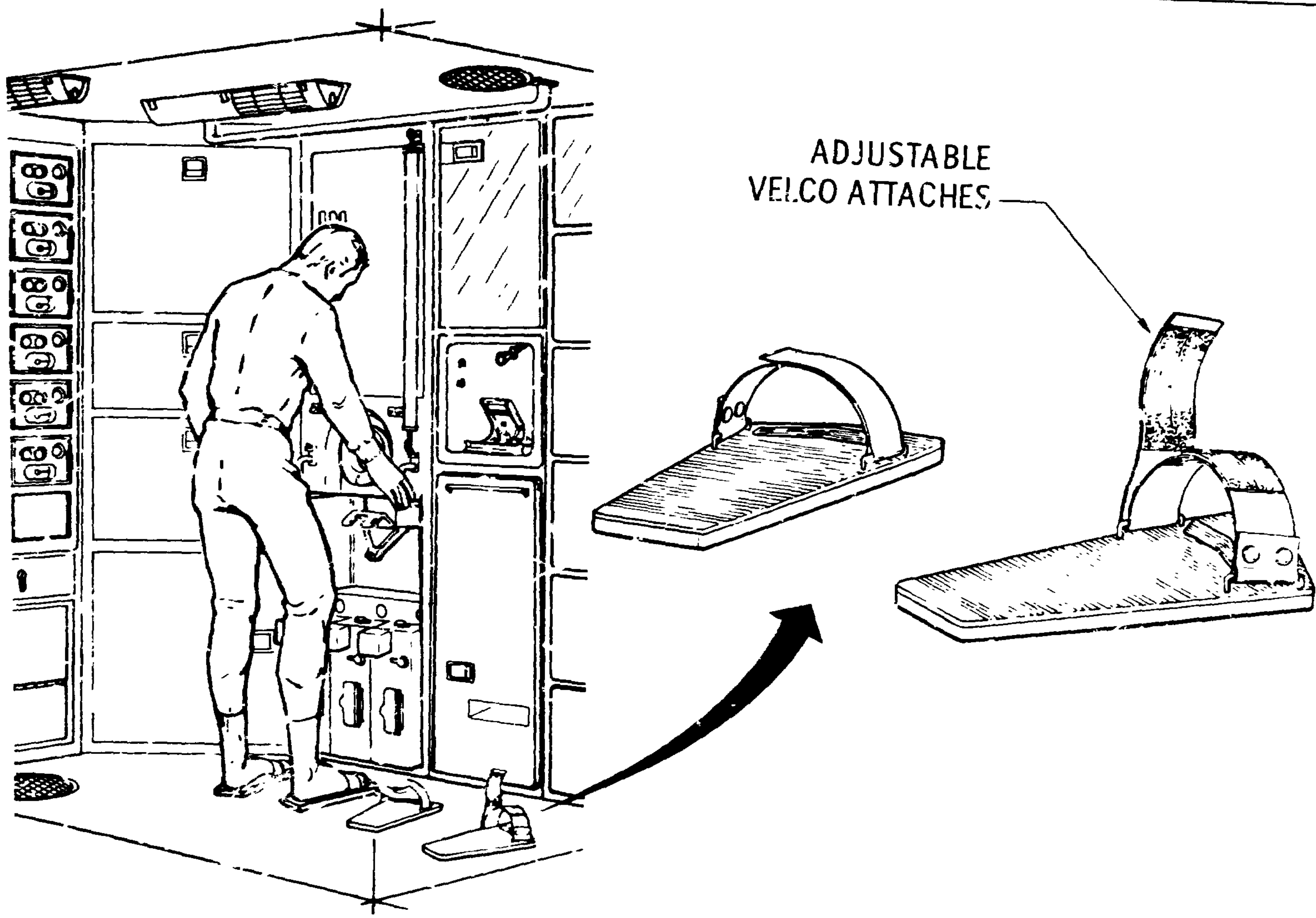


Figure 2.2.1.2-29. Astronaut Aids (Light Duty Foot Restraints)

FIGURE 2.2.1.2-29

feet while the crewmen utilized the food table. Each of the three food table's eating stations had a set of fixed foot restraints, which were composed of two adjustable foot restraint straps for bare foot restraint and two cleat receptacles for the cleats of the triangle shoes. (Figure 2.2.1.2-27). The foot restraint straps were lined with velcro to permit adjustment for each crewman's foot. In addition, at the toe-end of each foot restraint, a toe slot was provided on the floor-mounted base plate for additional stability.

The compartment door directly beneath the handwasher in the WMC contained a horizontal slot that served as a foot well to accept and retain bare feet inserted into the slot (Figure 2.2.1.2-30). The crewman occupied this position when hairbrushing and nail clipping functions were conducted directly beneath the ceiling-mounted intake of the WMC fan. In addition, this positioned the crewman in front of the mirror on stowage compartment H830 door to facilitate the accomplishment of these personal hygiene functions.

Two portable PGA foot restraints were provided for use on the OWS forward compartment floor grid as a restraint for the crewman while he donned and doffed his spacesuit or for spacesuit restraint during suit drying operations. A third PGA foot restraint provided crew restraint during experiment operations involving a suited crewman such as M509 and T020. The PGA foot restraint accepted and retained the PGA boots through use of a toe-bar and a heel fitting. Heel clips, which were an integral part of the PGA boots, engaged under the foot restraint heel fittings to provide rigid PGA boot restraint. A quick-release fastener was located at the rear of the baseplate to permit easy installation and removal of the restraint from the grid surface (Figure 2.2.1.2-27). Two grid clips fitted to the underside of the baseplate positively captured the grid surface.

C-3

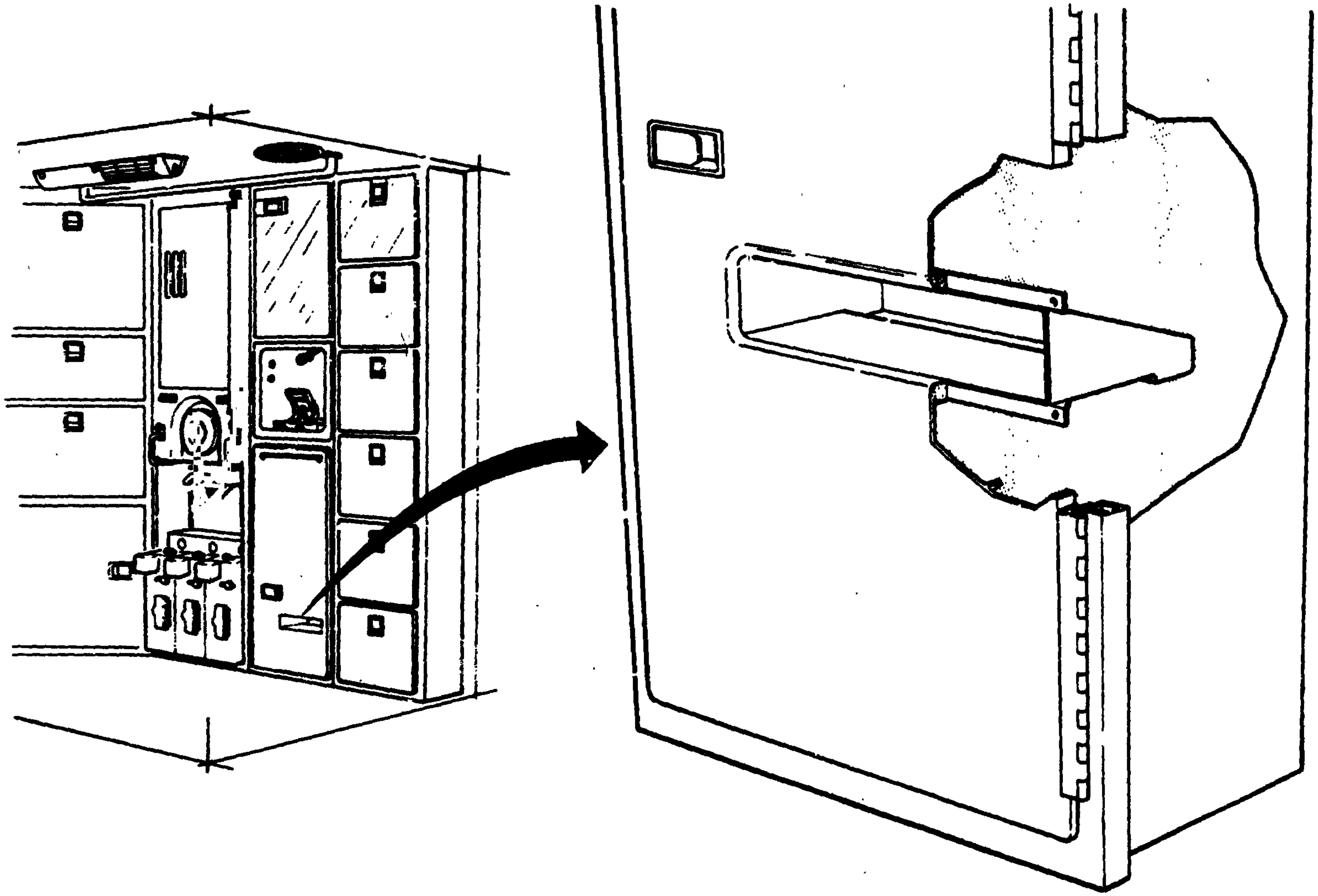


Figure 2.2.1.2-30. Footwell Restraint

FIGURE 2.2.1.2-30

One pair of custom-fitted, rigid sole, laced, high-top triangle shoes was provided to each crewman for use as a foot restraint on grid type surfaces (Figure 2.2.1.2-31). A triangular cleat with an integral engage/disengage mechanism was fitted to the sole of each shoe. The cleat had a slide adjustment from the toe to ball of the foot to a suitable position. It could then be tightened by means of a wing nut to the sole of the shoe. The cleat could be inserted into the triangular cutouts in open grid or into the cleat receptacles located on the platform foot restraints, food table restraints and also used with the ergometer. Once the cleats were inserted, the shoes were rotated heels-in to engage the cleat onto the underside of the attaching surface. The cleat was disengaged using the reverse action. The cleats' detented locks minimized the crewman's unintentional disengagement and provided tactile feedback of the lock/unlock position. Should the cleat fail, it was removable from the shoe. Replacement grid cleats were available from triangle shoes assigned to previous/subsequent mission crewmen. Shoe covering is PBI durette fabric with with beta tape laces. The sole is full-length, rigid laminate of fluorel sheet and aluminum sheet. The sole track is aluminum with a corrosion-resistant steel sliding nut. The cleat assembly is composed of corrosion-resistant steel and aluminum. In addition, three sizes of "conical cleats" were provided by NASA. The conical cleats overlapped the backside of the grid, usually at the grid node. These were also removable and replaceable. The three sizes were flown to provide on-orbit determination of the optimum configuration. The conical cleats

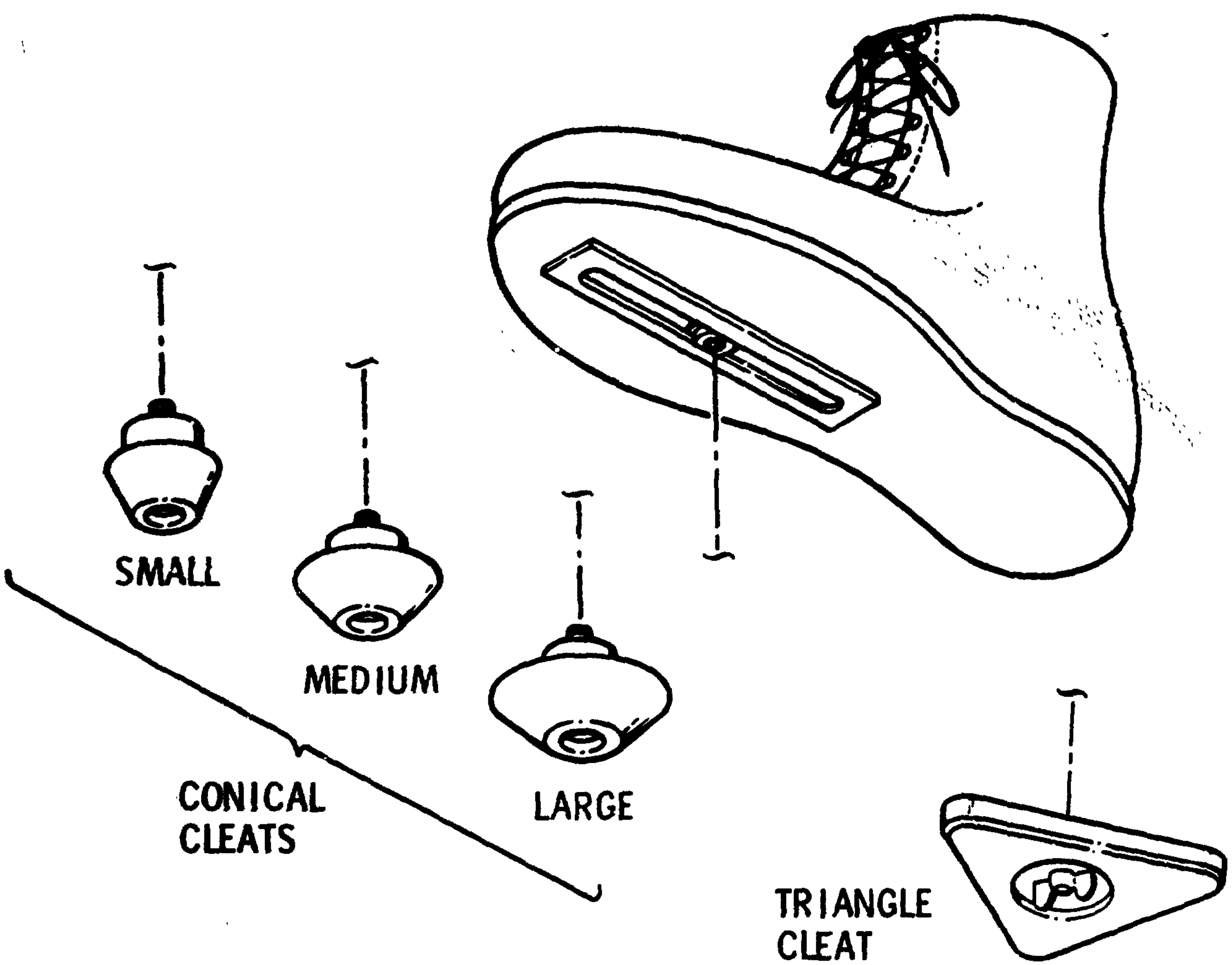


Figure 2.2.1.2-31. Triangle Shoe

were not dimensionally compatible with the triangular cutouts in the food table foot platforms.

Internal handrails were mounted throughout the OWS to permit the crewmen to move through the OWS and provided restraint while operating controls on panels, performing maintenance, gaining access to stowed items, relocating equipment and utilizing components of operational equipment (Figure 2.2.1.2-32).

Handholds were premanently fixed to structure throughout the OWS. They were located in the vicinity of operational equipment in areas where handrails were not required either because of restricted space of where there was no concern for mobility provisions (Figure 2.2.1.2-32). The fixed handholds were shorter than handrails and were mounted directly to pieces of operational equipment. Fixed handholds were used to facilitate hatch movement, for restraint while obtaining stowed items or while operating controls.

Handrails and handholds were flattened tubing with a typical cross-section of 5/8 x 1-3/8 in. (1.59 x 3.45 cm) with a 2-1/4 in. (5.7 cm) clearance above the surface for bare hand and spacesuit glove use.

A "Fireman's Pole" was provided for on-orbit installation between the OWS forward hatch and the experiment compartment egress opening in the OWS forward compartment (Figure 2.2.1.2-32).

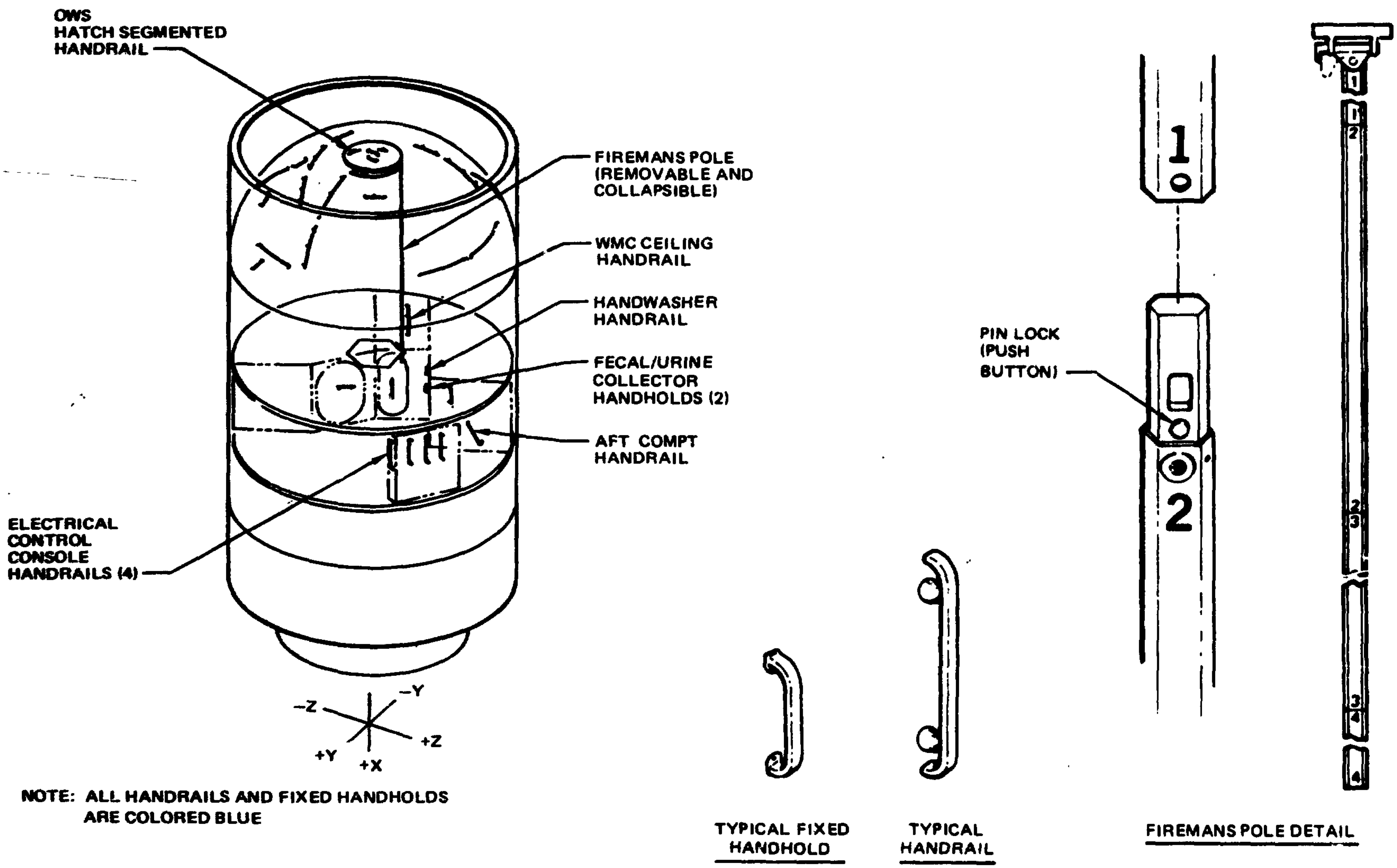


Figure 2.2.1.2-32. Crewman Fixed Hand Restraints (Internal)

The pole permitted rapid translation through the forward compartment for emergency egress from the crew quarters. Because of its rigidity, the pole was torsional stability in free space while the crewman traversed its length with equipment in hand or when trying to maintain a desired orientation. The "fireman's pole" was removable and could be rapidly broken down into four sections through use of its pin-lock joints for stowage.

Six portable handholds were supplied that attached to grid in the OWS. The portable handholds were used and could be operated with the gloved or bare hand. An actuation button was utilized to operate a quick-release grid fastener to attach or detach the unit from the circular hole pattern in the grid (Figure 2.2.1.2-27). A visual indication of positive engagement of the unit in the grid hole was provided on the handhold by viewing the position of the button in its housing. The actuation button was depressed to engage the handhold into the grid. Positive engagement was attained when the actuation button was flush with the finger grip. Four alignment pins (two at each handhold underside extremity) grasped the grid pattern and provided handhold rotational restraint and even load distribution on the grid. A hexagonal adjustment nut was located under the finger grip and was rotated by hand to tighten the handhold against the mounting surface if required.

Portable tether brackets attached to the grid provisions in the OWS and provided a convenient attach point for the LSU tether during suited IVA operations. The bracket could be operated with a gloved or bare hand. An actuation button was utilized to operate a quick-release grid fastener that attached or detached the unit from the circular hole pattern in the grid (Figure 2.2.1.2-27). Operation of the grid fastener and its adjustment nut was as described for the portable handhold.

Two adjustable tethers were provided in the OWS forward area for use as a mobility aid to supplement the "Fireman's Pole." The adjustable tether was a 20 ft (6.1 m) (maximum length) strap, which could be adjusted to a desired length through use of a buckle (Figure 2.2.1.2-27). Each end of the strap was fitted with a quick-release coupling that would connect to open grid, handrails, handholds, or convenient structure. Each coupling was spring-loaded closed to permit positive capture of the attaching structure. One adjustable tether was deployed for launch between the OWS forward compartment floor grid and a handrail on the OWS forward dome to assist the crewman in translating through the OWS forward area during OWS activation. The other tether was attached to a forward compartment wall handrail, through the frame of the condensate tank, and terminating at a handrail on the dome near the on-orbit location of the condensate tank. The tether was used to aid in controlling the condensate tank during relocation to its on-orbit position. The adjustable tethers were removed when not in use and stowed.

Three thigh restraints were located at the food table, one at each eating station. The thigh restraints were used to provide a comfortable and efficient means of stabilizing the crewman in a neutral body position while he occupied the eating station (Figure 2.2.1.2-27). The thigh restraint was frictioned hinged in two places: (1) at the table to permit the selection of the desired use elevation, for out-of-the-way stowage, and to permit the opening of the food table pedestal access doors, and (2) at the mid-point of the thigh restraint to provide the selection of the desired seating position. The thigh restraint was fitted with a slide-adjustment to permit its conformation to the size of the crewman's thighs. The thigh restraints were used in conjunction with the food table foot restraints during food management and off-duty activities.

The sleep restraints were provided for thermal comfort and body restraint while the crewman was sleeping. Each of the three sleep areas within the sleep compartment contained a vertically mounted sleep restraint (Figure 2.2.1.2-27). (The complete sleep support system is covered under Paragraph 2.2.11.6).

The fecal/urine collector in the WMC was fitted with an adjustable lap restraint, to provide whole body restraint while the crewman was seated on the fecal collector. The strap-type restraint was adjustable for comfortable restraint through use of a velcro lining. One end of the lap restraint was attached to a ring adjacent to the fecal collector. A quick-release spring slip was attached to the other end of the lap restraint and could be positioned in one of two locations: (1) to a ring adjacent to the fecal collector for use, or (2) snapped in a ring at the top of the fecal/urine collector cabinet for convenient stowage when not in use.

The lower leg restraints on dome ring lockers D424 and D430 were provided to assist two crewmen in translating, positioning, and installing the condensate holding tank at its on-orbit location on the OWS forward dome (Figure 2.2.1.2-33).

DOME RING LOCKER LOWER LEG RESTRAINTS

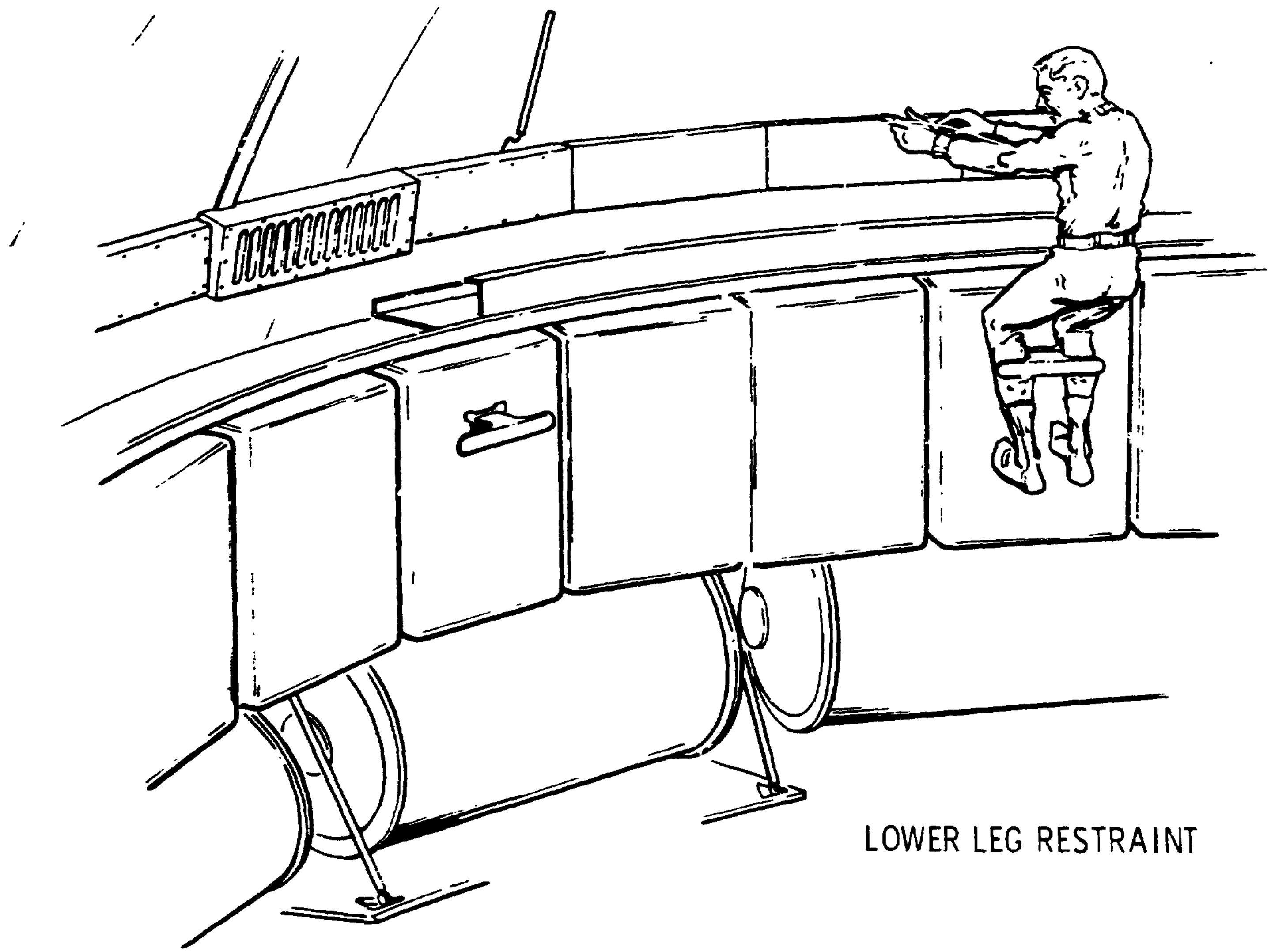


Figure 2.2.1.2-33

C. Testing

Basic Habitation Area Tank - Structural integrity of the habitation area tank was verified by the Dynamic Test Article (DTA) and the Static Test Article (STA). The full scale habitation tank for these tests was obtained from the S-IVB Program (Facilities Vehicle). The DTA test was performed at JSC, Houston, Texas. Subsequent to the post DTA test inspection, the tank was shipped to MSFC for static load testing and meteoroid shield deployment testing.

DTA testing verified structural integrity of mounting provisions for internal tank components. All components were mass simulated and associated bracketry and structures were simulated to provide the proper mass, stiffness, strength, and geometry of the flight configuration. No failures of basic tank structure or component attachments were noted at the completion of testing. The test environments simulated the maximum expected sound pressure levels during liftoff and boost and maximum transient response during launch, engine cutoff and stage separation. Sufficient data were obtained to verify the dynamic design and test criteria for tank mounted components and also verify the analytical dynamic models used to calculate dynamic loads for the crew quarters floor and water container ring assemblies.

The primary configuration change from the DTA to the STA was the installation of a simulated meteoroid shield. The shield was flight rigged for Test 3 - Ground Wind, Side Access Panel in at 99.9 Wing Loads (Ultimate), maximum compression over side access panel and Test 8 - Ground Wind, Side Access Panel Out at 99.9% Wind Loads (Ultimate -

Malfunction Condition), Maximum Compression Over Access Panel Opening. Later testing to prove the structural integrity of the meteoroid shield for pressure conditions, involved replacement of the simulated shield with a flight type meteoroid shield.

STA testing subjected the basic tank structure to ground wind loadings with the side access panel installed and removed, maximum vehicle loadings during launch and ascent, and maximum design differential pressures. See Section 2.2.2 for influence of meteoroid shield. All test requirements were successfully met, and no failures or detrimental yielding of tank structure occurred.

In addition to the DTA and STA tests, three development tests were also run on basic tank structure. The first test involved the determination of the vertical load carrying capability of the cylinder wall rib intersection. Although this test was conducted under the title,"OWS Floor Support Bolt Test," it is directly applicable to the water container support structure, TCS support structure, equipment support structure, and cylinder wall rib intersection attachment. The test specimens consisted of a single panel of cylinder wall with a centrally located waffle grid intersection. Various sized bolts were used and an aft shear load was applied to several configurations. Each specimen was tested to failure and failed by vertically shearing out at the rib intersection or excessive yielding of the intersection.

Test data were evaluated for the tapped hold specimens, which correspond with the production application, to determine a nominal shear-out load. The allowable load used in the stress analysis was 5,700 lbs (2586 kg), which was the test value corresponding with minimum design thread engagement and pre-laod. A complete written report of this test may be found in Volume X, Section 2.7.4 of the OWS Strength Analysis Report.

The second test involved determination and verification of the tensile and shear load-carrying capability of the 1B76421 1 disc and other phenolic discs that were bonded to the 3D foam insulation on the habitation area tank wall. The specimens were production disc assemblies bonded to a panel consisting of a metal plate simulating the habitation area tank wall, 3D foam insulation bonded to the plate, a disc bonded to the 3D foam, fiberglass liner and doubler bonded over the foam and disc assembly, and a coat of aluminum foil. The specimens were tested at ambient or cryogenic temperatures, since the wet workshop was still under consideration, by subjecting them to tensile and shear loads. These tests proved the feasibility of the bonded disc installation. A complete report of this test is in MDAC Technical Memorandum No. 131, dated August 27, 1968. Flight installations were acceptance tested by applying a 240 lb (109 kg) radial load to verify manufactured capability.

The third test involved evaluation of the design of the electrical feed-through installation on the forward dome. A secondary objective was to evaluate the design of the fuel tank pressurization port (vent port) installation. The test specimen (1T16368) was a partial forward dome similar to production with an electrical feed-through port and vent port installed. The test was run in the following phases: (I) Application of 27.3 psid (1.88×10^5 N/m^2) proof pressure; (1.05 x maximum boost pressure); (II) Ten pressure cycles from 0 to 12.4 psid (.855 x 10^5 N/m^2) (maximum pre-launch pressure); (III) Boost pressures [11.3 to 22.1 psid (.78 x 10^5 to 1.52 x 10^5 N/m^2)] and an acoustic environment more severe than frequencies expected during launch and ascent; (IV) Pressurization of the specimen to 36.5 psid (2.51 x 10^5 N/m^2) which is 12 percent higher than ultimate. All tests were conducted at room temperature. This test successfully demonstrated the structural integrity of the electrical feed-through installation and vent port installation. Reference Memorandum A3-250-ABC8-M-28, dated August 10, 1970.

The side access panel was tested as an integral part of the DTA/STA tank. During STA testing, the panel was successfully installed on the tank while the tank was subjected to maximum design ground winds to demonstrate on pad entry capability. The meteoroid shield was not flight rigged for this test.

Three tests were conducted with the objective of demonstrating the access panel could be installed on the OWS under any ground wind condition consistent with personnel safety. The tests consisted of preloading a bar that had the tank portion of the Milson fastener installed. A corresponding pre-drilled bar was then installed using the other half of the Milson to prove the capability of the installation for various offsets and hole size configurations.

Two separate qualification tests were performed on the OWS forward entry hatch. The initial test, line item EC-2, subjected the hatch to the full spectrum of design launch and functional loads. Simulated weights for the flight check valves were the only deviation from flight configuration on the test specimen. Two vendor parts failed during the vibration phase of this test.

The SLC3AT09 quick release pin from space lock, 14c, which held the release handle in the "lock" position, unscrewed. The pin had a two-part housing without self-locking provisions. The pin was replaced with a one-part housing pin, LW3C-T-900, from the Hartwell Company.

The press-fitted and staked race of the KBEL 3-65 rod end (Kahr Bearing Company) parted from its housing due to vibration. To prevent this failure, the vendor redesigned the race installation and was required to perform a production acceptance test on each rod end that loads the bearing in the thrust direction. To qualify the rod end and complete the EC-2 test, a component test, EC-44, was successfully performed on the redesigned rod end.

Subsequent to completion of the initial qualification test, the hatch was tested, line item EC-46, to determine its capability as an aft airlock hatch during EVA's. The same test specimen, refurbished to include redesigned rod end bearing and new o-ring seals for the periphery and handle mechanism was subjected to 100 functional cycles. Leak checks and hatch handle load measurements were within maximum design allowables during this test proving its capability as an airlock hatch. During disassembly of the hatch after the test, a bearing shaft failure was detected on one of twenty-four bearings. The failure was a result of misalignment which caused excessive loads and resulted in a check of all twenty-four flight hatch bearings. The flight hatch had been properly adjusted.

The structural integrity of the HPI was established by strength analysis and similarity to other designed HPI. The Velcro fasteners and Nylafil stand-offs had an ultimate factor of safety greater than 5.4 over limit. Structural integrity was established by acoustics test of similar R&D HPI panels under more severe environment (Ref. Report MDC G2135 dated February, 1971). This test was more severe than the conditions which were imposed on the OWS HPI panels. The test was run at -300°F (-184.4°C) and with greater vibration loads than on the OWS. The panel withstood the test except that 4 of 30 heat-sealed buttons failed, and a 0.06 in. (1.524 mm) gap appeared at the butt joint.

The corrective action was to use a one-piece panel fastener in place of the heat-sealed buttons, to fasten the panels continuously at the edges with Velcro and lacing, and to use a shiplap joint. The margin of safety was high and no additional tests were required.

All penetration seals on the OWS were acceptance tested on the flight vehicle. Two mass decay leak tests were performed, and both tests showed the equivalent on-orbit leakage would be approximately 1.0 lb (.453 kg) per day, which is much less than the design requirement of 5.0 lbs (2.268 kg) per day.

Equivalent leakage accounted for the test condition of gaseous helium at 5 psig (.345 x 10^5 N/m^2) as compared to on-orbit condition of a oxygen-nitrogen gas mixture at 5 psia (.345 x 10^5 N/m^2).

Wardroom window development tests were conducted to determine design feasibility and functional parameters for verifying that the design was adequate for the intended use and for indicating critical areas where design improvements would be required to ensure that the window assembly would be capable of successfully meeting the qualification test requirements.

Each individual glazing was subjected to pressure tests prior to being installed in the assembly. Each glazing was tested to a differential pressure of 29.4 psid (2.026 x 10^5 N/m^2) in both directions, before and after optical coatings were applied. A separate uncoated glazing was pressure-tested to failure. It burst at a differential pressure of 62 psid (4.27 x 10^5 N/m^2). Two uncoated glazings were subjected to hypervelocity tests to verify their ability to withstand meteoroid impact. Another individual uncoated glazing was tested for internal impact.

The window assembly was subjected to numerous optical and environmental tests such as transmittance, transmitted wavefront, seal leakage, low temperature, and high temperature.

Of all the tests enumerated above, only the internal impact specimen failed to meet the specification requirements. The original specification was an impact load of 2 slug-ft/sec (8.89 kg m/sec) with an instrument mass of 0.5 slugs (7.295 kg) having a 0.13 in (.330 cm) spherical radius impact tip and an impact velocity of 4 ft/sec (1.22 m).

Visual inspection of three tests indicated surface damage and cracks at each of the three impact locations. As a result of this failure, a glass-protective shield was designed. The shield was made of tempered glass so as to be able to withstand impact loading. In addition, the impact criteria was thoroughly reviewed and modified to an energy concept rather than the velocity concept that had been used. The shield was then impact-tested to meet an ultimate design impact energy of 7.8 ft-lb (10.575; joules). It failed at 63.4 ft-lb (85.957; joules).

A special window assembly, without optical coatings or electrical connectors, was made for installation in the DTA. This specimen was subjected to vibro-acoustic testing without failure.

A qualification test was conducted to verify the optical, mechanical, structural, and electrical reliability and safety performance integrity of the window assembly. Tests related to transmittance, transmitted wavefront, seal leakage, high and low temperature, and individual glazing pressure tests were conducted in an identical fashion to those described for the development test. In addition, window assembly proof pressure and ultimate pressure tests were performed. In the proof pressure test, a differential pressure of 14.7 psid (1.0134×10^5 N/m^2) was applied across the outer glazing and 11.9 psid ($.820 \times 10^5$ N/m^2) across the inner glazing. The ultimate pressure test applied 18.8 psid (1.296×10^5 N/m^2) and 13.6 psid ($.937 \times 10^5$ N/m^2), respectively. All pressures were applied in the outboard direction. All the qualification tests were conducted successfully.

An acceptance test was conducted to verify that the window assembly met the requirements imposed by the design requirements drawing prior to installation in the OWS. The acceptance test consisted of transmittance, transmitted wave-front, seal leakage, assembly proof pressure, and individual glazing pressure tests identical to those conducted in the development and/or qualification tests with one exception: i.e., the window assembly's outboard glazing, as part of its individual glazing pressure tests, was **pressure-tested to 59 psid ($4.06 \times 10^5 \times N/m^2$) in its outboard direction only.** All acceptance tests were successfully completed.

The OWS scientific airlock was subjected to a two phase development test.

Phase I

1) Determine the load-deflection characteristics of a typical experiment flange seal:
   a) Double Viton seal
   b) Single Viton seal
   c) Double Butyl seal
2) Determine airlock experiment latching handle force versus experiment flange seal deflection for Viton rubber and Butyl rubber.
3) Finally, determine from the data whether the latching mechanism, as designed by North American-Rockwell, could be made adequate for use with double Viton seals, and whether metal-to-metal contact for a good electrical bond could be achieved.

The Phase I tests showed that obtaining a good seal requires very little force, but metal-to-metal contact between flange and airlock could not be achieved by the existing method. Therefore, a program was instituted to provide a low friction type of experiment latching mechanism similar to that used on Experiment S019. The new mechanism was installed and tested as part of Phase II.

Phase II

1) Proof pressure tests at pressures which could occur during checkout, boost, and orbit.

2) Functional Test

   a) Window latching mechanism

   b) Pressure control valve function and valve cover function

   c) Door opening and closing and indicator function

   d) Pressure gage function and accuracy

   e) Hose coupling and uncoupling

3) **Leakage tests at ambient temperature, with 5-psi (.3447 x 10^5 N/m^2) differential pressure at both atmospheric pressure and 5 psi (.3447 x 10^5 N/m^2) absolute. This included component leakage where possible, and total leakage of the airlock-window-outer door combinations. A separate test of vacuum hose leakage was conducted.**

4) Vibration test of the seal and the window and window container in the launch configuration per DAC-56620B.

5) Post-vibration functional and leakage test to determine any changes caused by launch forces.

6) Thermal-vacuum tests, first simulating the effect of cyclic solar radiation on the airlock having solar thermal coatings. After this test was completed, the airlock was recoated with the anti-solar thermal coatings and retested in a simulated cyclic dark side environment. Temperature measurements were made at various points on the tank wall, airlock, and window.

7) **In the experiment evacuation through SAL using the vacuum hose, one end of the hose was plugged into a 2.4-ft^3 (.068 m^3) experiment filled with air at 5 psia (.3447 x 10^5 N/m^2), and the other end into the SAL Q.D. The SAL door was in the vent position at a 1 x 10^{-3} toor vacuum.**

8) The endurance test consisted of operating the SAL through 1,000 pressurization-depressurization cycles at 5 psi (.3447 x 10^5 N/m^2) differential pressure [14.7 to 9.7 psi (1.013 x 10^5 to .668 x 10^5 N/m^2)]. This provided 1,000 cycles on:

a) The outer door crank

b) The experiment latching mechanism

c) The vent valve

d) The pressure gage

9) Test leakage after 1,000 cycles is completed.

10) For the cantilever beam test, loads of 0 to 125 lb (56.625 kg) were applied at a moment of 60 in. (1.524 m) at 5-psid .3447 x 10^5 N/m^2) differential pressure. Leakage was measured during test.

11) For the yield pressure test, 9.9 psia (.682 x 10^5 N/m^2) were applied inside the SAL chamber. Leakage was tested at 5 psid (.3447 x 10^5 N/m^2) afterward.

12) For the ultimate pressure test, 32 psid (2.206 x 10^5 N/m^2) were applied across the outer door, and 12 psid (.827 x 10^5 N/m^2) inside the SAL chamber and across the window.

13) For the window burst test, the purpose was to determine the differential pressure at which the window will burst.

The Phase II tests showed several problem areas. In the vibration tests the following problems occurred:

1) One screw in the airlock mechanism came out, and its shim was broken.

2) One mounting screw on the window container came out and another was loosened.

3) One of the launch "retainers" for the window was broken and another was loosened.

4) A thread insert loosened.

These problems were resolved in the following manner:

1) The airlock mechanism was properly shimmed, and the screw reinstalled.

2) The lugs on the window container were remachined to make the mounting surfaces parallel to the "patches" on the tank wall segment.

3) A metal washer with a soft pad was designed to fit under each "retainer" so that the retainer could be torqued to higher values.

4) The loosened thread insert was removed, and another installed.

The SAL and window container were retested. After the window retainers were tightened to 60-inch-lb (6.78 N·m) torque, all components passed the vibration test without incident and are considered structurally adequate for launch.

In the proof and leak tests, the proof tests demonstrated that the scientific airlock could withstand all pressures and loads that it would be subjected to, including a 125-lb (56.7 kg) astronaut load exerted (at a 60 in. (1.524 m) moment arm) on an experiment mounted in the airlock, without any significant leakage. The viewing window survived pressures well in excess of any flight requirement.

The thermal-vacuum tests showed that the inside surfaces of the airlock will remain well within touch temperature limits during flight.

In the functional tests and endurance tests, examination prior to test showed that the experiment latching mechanism did not meet the electrical bonding requirement because the belleville washers providing the sealing force had been assembled incorrectly. A representative of the manufacturer reassembled the unit properly so that it would exert enough force to produce metal-to-metal contact, and agreed to make sure that this was done on all future units.

During endurance testing, several problems appeared:

1) The detent that holds the vent valve in position became sticky at approximately 72 cycles, and again at 104 cycles. At 740 cycles, the pin holding the detent cam on the valve stem sheared off. The pin was pushed back in, and the test continued to 1,000 cycles. (This

was a failure of the detent, not of the valve).

When the test was complete, a small amount of oil was applied to the detent. The stickiness immediately disappeared, and the valve operated smoothly again. Disassembly showed that the problem was galling of stainless steel on stainless steel; this caused a burr to be thrown up which interfered with smooth operation.

The flight parts had been lubricated to prevent recurrence of this phenomenon.

2) Seven of the eight roller shafts on the experiment latching mechanism showed a tendency to work out of the pivot housing. This happened continually after about 500 cycles. By design, they were to be held in place by set screw friction. Although the set screws remained tight, they did not restrain the shafts sufficie ly.

This was not a serious problem, since the roller could be pushed back into place. The vendor agreed to eliminate the problem for the flight units by machining a depression in the roller shaft so that the set screw would engage it to produce a positive lock.

The flight unit of the airlock was identical to the test unit except in the following details:

1) The "Micatex" paint on the exterior airlock surfaces and the markings were not on the test unit.
2) The vacuum hose stowage clips were not on the airlock, the hose stowage location was on the window container.
3) The windows on the flight unit include a shim designed to compensate for manufacturing tolerances and permit the proper seal preload.

A development test was performed to verify the calculated design allowables for a floor cone segment under compressive loading. The test involved compressive loading a partial conical segment in its flight configuration including a preload to account for deflection due to tank pressurization.

During testing, radial deflection was measured at three points along the meridional direction of the conical segment. Test results indicated that a graphical representation of applied load versus lateral deflection compared closely with analytical results of a test configuration which had three effective beads symmetric with respect to a tank wall attachment and considered fixed at both ends of the specimen. Yielding occurred at approximately 5,000 lbs (2268 kg) in the forward flat-sheet region of the cone segment. The beaded region became unstable at 8,775 lbs (3980 kg). Although the configuration which was tested was related to the floor system, the data were also directly applicable to the water container support structure because of similarity in design. In fact, the shorter water container conical ring would probably exhibit a higher allowable if tested.

A development strength verification test was performed on the 4.2 in. (10.67 cm) floor and wall grid panels. The panels were subjected to point, line, and bending moment loads. The purpose of the test was to determine the load carrying capability and elastic characteristics of the grid. The results of this test are documented in MDAC document, A3-860-KBBB-TM-123. A design development test was performed on the 4.2 in. (10.67 cm) floor and wall grid splice panels. In this test, the panels were subjected to point and bending moment loads. The purpose of this test was to determine the load carrying capability of the floor and wall grid splice joint. The results of this test are documented in MDAC document A3-860-KBBB-TM-124. Both tests showed that the strength of the grid design was adequate.

A comparison of resonant frequencies from the water container support structure dynamic model with the DTA test results was made in MDC G2445, "Orbital Workshop Dynamic Test Article Vibro-Acoustic Test Report - Volume II", dated October 1971. Based on this comparison, it was concluded that the analytical responses of the water bottle support system for vehicle dynamic accelerations would be conservative compared to the expected dynamic responses.

The waste tank was tested as a subsystem of the Dynamic Test Article (DTA) and the Static Test Article (STA). DTA testing was conducted to vibration criteria supplied in DAC-56620C. STA testing accounted for static and quasi-static loads and pressures.

Crew Accommodations

Development test CA-7 was performed for the purpose of developing pigmented anodic films, for the colors specified, with a thermal emittance greater than 0.80. The test resulted in the specification of anodizing processes which produced aluminum finishes capable of meeting the OWS thermal emittance, toxicity, odor and color requirements. Weathering and exposure tests caused color instability which was considered acceptable because of the limited exposure anticipated with the the OWS. (Ref: Lab Report MP 51,386).

Development Test of the Portable Handhold, Technical Memorandum No. TM 188 for Line Item CA 10, was accomplished to ascertain the structural integrity and functional reliability of the 1T10892-1 A Change, Portable Handhold. The portable handhold met the functional and load requirements of the test.

Development test of the Portable Tether Attach Fitting, Technical Memorandum No. TM 184 for Line Item CA 11, was accomplished to ascertain the structural integrity and functional reliability of the 1T10983-1, "A" Change, Portable Tether Attach Fitting. The fitting met the functional and load requirements of the test.

Initial development testing of the portable foot restraint (triangle shoe). Technical Memorandum No. TM 190 for Line Item CA-9, was accomplished to determine the functional merits and physical limitations of the portable foot restraints (triangle shoe).

It was concluded that additional design and development would be required prior to end item fabrication. The new concepts were tested in Line Item HS-1, which is contained in subsequent paragraphs of this section.

The Crew Restraints Development Test, Line Item HS-1, began as one-g and null gravity (neutrally buoyant) evalutions of various combinations of crew restraints. The test was conducted over the time period of November, 1969, through July, 1971. The test was conducted under ambient conditions and under neutrally buoyant conditions in the Neutral Buoyancy Simulator at the Marshall Space Flight Center, Huntsville, Alabama.

The purposes of the test were:

(a) To prove that the design concept of the restraints would permit comfortable performance of mission tasks.

(b) To compare various design concepts of each type of restraint (seats, PSA foot, fixed foot, sleep, portable foot and lower leg restraints) in order to select the most feasible candidate.

(c) To perform cyclic tests on the Portable Foot Restraint.

Typical test configurations are shown in Figures 2.2.1.2-34, -35 and -36.

FIGURE 2.2.1.2-34 TABLE RESTRAINTS (TRIANGLE SHOE) TEST CONFIGURATION

FIGURE 2.2.1.2-35 TABLE RESTRAINTS (FIXED FOOT RESTRAINT) TEST CONFIGURATION

All configurations of the PFR provided adequate restraint, and extremely good mobility, functional reach and comfort. The positive locking of the restraint in the "heels in" position and "heels out" position allows ease of ingressing and egressing, including rapid, emergency egress. In no case did the subjects resort to the emergency rip cord provided for additional underwater safety.

The results of the test indicated that the PFR with the flexible sole and the 15 degree indexing cleat on the toe position of the sole was superior to the other configurations. Subsequent to fabrication of a candidate restraint, one-g tests were conducted. These tests substantially validated the results obtained in the neutral buoyancy tests and, in addition indicated a need for the following modifications:

(a) The foot restraint lacings were replaced by a strap and loop fastener utilizing velcro for positioning the strap. This modification resulted in a one-handed, quick release capability.
(b) The foot restraint sole proved too flexible to provide positive restraint. Stiffening of the sole cured this difficulty.
(c) Stiffeners added to the toe and heel improved the restraint characteristics.

Subsequent NASA redirection specified an adjustable cleat position, lace tops and rigid soles.

The soft fixed foot restraint was preferred by most subjects over the hard foot restraint, because too much pressure was imposed on the insteps of the subject while performing simulated task functions.

All restraints were adequate when used in conjunction with a foot restraint.

The "H" Bar thigh restraint (Figure 2.2.1.2-37), when used with foot restraints, was preferred over the "T" bar, pelvic, zero-g chair, and the lower leg, because it offered adequate restraint without excessive restriction to body movements.

The "T" Bar restraint offered good restraint, but restricted body movements because of the lap belt around the thighs. In this configuration the subject was over-restrained.

The pelvic restraint was similar to the "T" Bar, except for the large seat pan on the pelvic restraint. This restraint also provided the subject with excessive restraint.

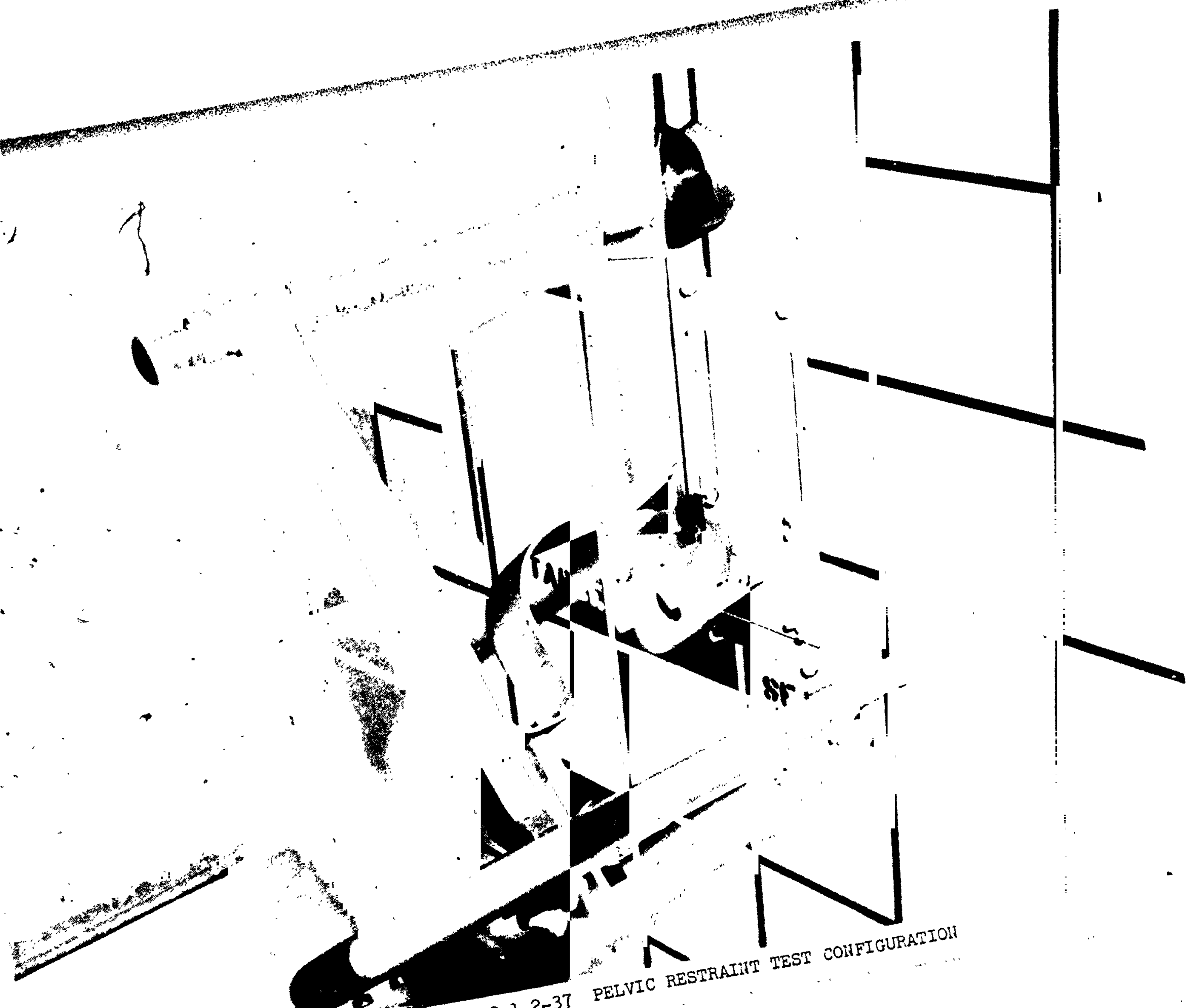


FIGURE 2.2.1.2-37 PELVIC RESTRAINT TEST CONFIGURATION

The zero-g chair, a molded seat pan with molded surfaces coming outside or over the thighs, offered adequate restraint for the larger subjects, but was very poor for the smaller subjects, since they had to exert some force to spread their legs apart to be restrained. When used with the PFR, it was ineffective, since the subject has to spread his legs to remain restrained. This tended to put his legs in a "heels out" position.

The lower leg restraint was adequate, but most subjects did not like the comfort afforded by the two points of contact (feet and knees). In this restraint the subject had to exert additional leg force to maintain proper restraint.

The results of these tests indicated that a good foot restraint was all the restraint necessary to accomplish most task functions. However, an upper body restraint which does not preclude mobility was desirable to stabilize the subject while performing long duration and leisure task functions.

The test indicated that the table height should be fixed at 40 in. (101.6 cm) and that the thigh restraint should be made adjustable.

The results of the bag type sleep restraint test indicated that the sleep restraint had adequate room and good mobility. Ingressing and egressing offered no problems. Emergency egress using the center zipper and the breakaway snaps worked well. This approach was subsequently replaced by a NASA design that provided more comfort and restraint flexibility by using PBI and Spandex materials.

The NASA "Dutch Shoe" approach to PGA foot restraints was not satisfactory. NASA replaced it with a design that firmly secured the heel to the boot.

D. Mission Results

1/ The acceptable performance of these structures in general would be their capability to withstand the critical environments of launch and ascent without failure and still perform required functions on orbit. S-IVB testing, analysis, and flight data provided sufficient confidence in the analysis and testing of modifications and new structure on OWS that practically no instrumentation was considered necessary to measure structural performance during flight. Therefore, mission results will only be reported for those structural systems where functional, measured, or unusual performance was recorded.

2/ The habitation tank structure withstood critical launch and ascent loading without any apparent damage. One accelerometer mounted on the film vault was oriented in the longitudinal direction to measure the dynamic environment of the crew quarters floor. The measured levels were well below design and test criteria.

3/ The failure of the meteoroid shield during ascent caused the OWS structural temperatures after orbital insertion to exceed the design limits of certain materials. As a result, a special test was performed at MDAC and MSFC on the bond strength of the 3D tile insulation on the tank interior. Flight data indicated a 300°F temperature on the outer skin with a calculated temperature of 200°F at the bond line of the insulation. Mission Support Test Request (MSTR) 006-1 was initiated by MDAC.

4/ A test specimen was made and temperatures simulating flight conditions were imposed on the specimen. Results showed cracks in Lefkoweld adhesive in the bond line, the progression of the cracks with time, some expansion of the foam and discoloration at the bond line. Other outgassing tests of the materials subjected to excessive temperatures were conducted to determine atmospheric conditions in OWS prior to Skylab 2 launch. Also tests for the load carrying capability of bonded phenolic discs after exposure to these high temperatures were conducted with results showing lesser capability but with adequate margins for orbital loading conditions.

5/ Although there was no leak check equipment on board to check the habitation tank leakage, the performance in this regard was much better than the 5 lb/day allocation.

6/ The Wardroom Window did not perform as expected with regards to fogging. The ice spot which formed on the inner surface of the outboard glazing prior to initial activation and the reoccurrence of ice, frost, and fog on subsequent missions was thought to be caused by diffusion of cabin air into the cavity between panes which was initially filled with dry nitrogen gas. This diffusion could be thru the seals on the glass or the vent valve for the cavity. Development tests on the window failed to produce such a fogging condition and production acceptance testing of the flight window assembly leakage was nil. Procedures, on board hardware, and a fitting flown up on SL-3 allowed evacuation of the cavity to vacuum thru the anti-solar SAL which sublimated the ice and cleared the window. The first evacuation was followed by a backfill with desiccated air thru the SAL desiccant cannister. After a period of time fogging reoccurred; therefore, subsequent evacuations left the cavity locked up with a vacuum. A fit check test of the hardware used for window evacuation, MSTR 278, was conducted in OWS 2 at MDAC prior to actual flight evacuation.

7/ The OWS entry hatch was used for a total of 9 EVA's with no known or reported problems with regards to function or leakage. During the pressurization cycles prior to habitation by the first crew (Skylab 2) the OWS pressure was bleeding into the AM. The AM/MDA pressure lagged the OWS pressure within 1378-2067 Newt/M^2 (.2-.3 psi) until 34,452 Newt/M^2 (5 psia) was reached then the pressure equalized. The first crew devised make shift flapper valves from onboard materials (mosite and tape). These were applied to the OWS side of the entry hatch check valves. During subsequent operation of the hatch, leakage was minimized.

8/ The Scientific Airlocks functioned properly thru the entire mission. The JSC parasol was installed and remained in the solar SAL after initial activation of SL-2 thru the entire mission. Experiments were continually installed and removed from the anti-solar SAL during SL-2, -3, and -4 missions. No leakage or mal-function of the hardware has been reported.

9/ Astronaut Aids/Crew Accommodations

a. Trim and Closeouts

No crew comments were specifically directed at the effectiveness of trim and closeouts. There were no reports of lost objects trapped in inaccessible places. In general, loose objects and solid particles migrated along the air circulation paths and were collected on the inlet screens on the OWS dome. Liquids are moist food particles usually adhered to the first surface they contacted. They were then wiped off or, as in the case of water droplets, left to evaporate. Wiping up spills in the Waste Management Compartment (WMC) was simplified due to the sealing afforded by the WMC trim.

b. Wardroom Curtain

The curtain was closed by the SL-3 crew to reduce wardroom illumination during some T002 manual navigation sightings. The crew implied that the curtain was satisfactory.

c. Interior Colors

A variety of crew comments were made regarding surface finishes, color selection and general light level. The dark blue color coding of handrails and the general durability of finishes were praised. Attitudes toward the choice of interior colors varied from casual acceptance to severe criticism for lack of imagination. A common crew suggestion was the use of lighter colors for the purpose of improving the overall light level.

d. Fixed Handrails and Handholds

The dome and forward compartment handholds and handrails were used occasionally for maintaining body stability and orientation. They were also used as an alternate translation aid for moving between the dome area and the forward compartment floor area when carrying equipment. In addition, the handholds near the SAL were not used, probably due to the large number of handholds inherent in the equipment in that area. It was indicated that a few additional handrails could have been added around the film vault and food boxes. The fecal collector handholds were extensively used by the crewmen to pull themselves down firmly on the fecal collector seat in order to obtain a good seal. The WMC ceiling handrail was satisfactorily utilized as intended to assist the crewmen in maintaining body stability and orientation. The handwasher handrail was used to enable the crewmen to apply the necessary forces to operate the washcloth squeezer. No handrail problems, malfunctions or failures were noted in any OWS area.

e. Light Duty Foot Restraint

The light duty foot restraints in the WMC provided adequate restraint when used as intended, i.e., with bare, stocking, or soft slippered feet. The locations of the WMC foot restraints were excellent. The crewmen attempted to use these restraints with little success while wearing the triangle shoes. The restraints were not designed to accommodate the triangle shoes. The crew adjusted the straps to their maximum length in an attempt to accommodate the triangle shoe. This accounted for subsequent problems such as the velcro separating because of inadequate overlap. It also caused the strap to interfere with the urine drawer during urine bag changeout. The straps were purposely designed to be semirigid so they could be deflected down and stay out of the way to clear the urine drawers. Over extending the strap length did not allow the strap to stay flat. Longer straps with curved metal liners were developed by JSC and flown on SL-3. This did not solve the triangle shoe incompatibility and it increased the interference with the urine drawers. Adapters designed to provide triangle shoe capability in the WMC did not fly on SL-4 due to weight limitations.

f. Fireman's Pole

The fireman's pole was rigged in place between the forward dome and the forward floor on approximately DOY 162. Prior to that time the forward compartment adjustable tether was utilized. Thereafter, the fireman's pole was preferred over the adjustable tether. The crews used the fireman's pole to good advantage during mission activation to transport large packages. Other than that, all three crews found that they didn't need the fireman's pole nor the tether. After a "learning" period they found that free soaring was preferred. The fireman's pole was removed after activation on the second and third missions. No problems, malfunctions or failures were noted during all three missions for the fireman's pole.

g. Food Table Restraints

The thigh restraints were used in various ways, depending on the individual crewman. Some used them in the nominal mode, with thighs between the cross bars and others used them with their knees above the top cross bar. Utilization of the foot restraints also varied from crewman to crewman. Several crewmen reported that the shoe cleat remained in the locked condition after removal which necessitated the crewman resetting the cleat by hand. This was caused by the triangle cutouts being slightly oversized. The strap foot restraints were adequate when used with stockings or soft shoes but had the same problems as the light duty foot restraints in the WMC when used with triangle shoes.

h. Water Tank Foot Platform

The foot platform provided excellent restraint for accessing the ring containers. Generally the crew locked both triangles into the platform but many times only one foot was engaged in the triangular cutouts. While stringing water hoses and the condenrate tank hoses, one of the crewmen laid parallel to the platform and grasped it between his knees.

i. Lower Leg Restraints

The lower leg restraints on dome ring lockers D424 and D430 were successfully used to assist two crewmen in translating, positioning, and installing the condensate holding tank to its on-orbit location on the forward dome. No problems, malfunctions or failures were noted.

j. Portable Aids

1. PGA Foot Restraints

The PGA foot restraints were rated excellent by the crews. One of the OWS PGA foot restraints was taken EVA to compliment the similar foot restraints permanently mounted outside the vehicle. The PGA foot restraints were indispensible during EVA and worked very well during suit donning and doffing, and to tie down pressure suits for the drying period. The CDR on MIssion 1 had difficulty removing his boot from the restraint during EVA but this problem did not occur on subsequent missions.

2. Portable Handholds

Two of the portable handholds were used, as intended, on the T013 force plates. The only other usage for the portable handholds was mentioned when the first mission SPT reported that the handholds were used in the vicinity of the bicycle ergometer in attempting to solve ergometer restraint problems. The crew, used the handholds inherent in the grid and most equipment. The CDR, mission 1, noted that in the process of trying to replace a malfunctioning astropin in one of the handholds with another one the second astropin was destroyed. Therefore, at the end of the mission, two of the handholds were without astropins, but replacement was not considered necessary.

3. Portable Tether Brackets

The first mission SPT reported that he was using the brackets to rig ropes with handles to hold himself down on the bicycle seat while peddling the bicycle ergometer. No other usage was mentioned. The second and third crews did not report using the portable tether brackets.

4. Adjustable Tethers

The first mission CDR reported that the "strap was sufficient for transporting them "back and forth." The PLT reported on DOY 155 that relocating the condensate tank from the launch position to its on-orbit position went very smoothly although no specific mention was made concerning the adjustable tether. The other adjustable tether was used for translation between dome and forward floor. It appeared to be satisfactory, but not as versatile as the rigid fireman's pole. The main complaint was inability to easily change direction when using the flexible adjustable tether. One adjustable tether was broken but was repaired by the crew with tape. During vehicle checkout the tethers had been proof-tested to 125 pounds. No anomalies were reported by the second or third crews. As in the case of the firemans pole, the crews found that except for activation, the strap was not really required. Free soaring was found to be preferable for locomotion.

5. Triangle Shoes

The triangle shoes were extensively used and the crews felt they were the most useful and versatile restraint they had. Each mission crew tried the conical cleats in place of the triangles. They found they preferred the triangles mainly because the conical cleats tended to slip out too easily. They preferred the detented lock of the triangle shoes. The conical cleat had a tendancy to inadvertently catch in the grid. One crewman used a triangle on one foot and a mushroom conical cleat on the other foot. Some difficulty was experienced when unlocking the triangle cleats from the table foot platform as noted earlier. The wing nuts that allowed adjustment of the position of the triangle had a tendency to loosen, especially on the ergometer. The crew found that pliers were required along with periodic retightening. The PBI material of the triangle shoe uppers proved to have insufficient abrasion resistance. The crews used their toes for direction control by selective dragging of one foot or the other. Fluorel toe caps were fabricated and flown up on SL-3. The caps provided adequate toe abrasion protection. The heels of the shoes also tended to wear out at the edges of the stiff teflon heel counters. The crews used tape to reinforce worn areas. Spare uppers were flown on SL-4.

E. Conclusions and Recommendations

1/ Review of the amount and type of testing performed for OWS No. 1 compared to the amount of analyses only, indicated a good balance. The early overall comparison of only analyses producing confidence as opposed to a higher degree of confidence due to later integration of test results into analyses as the OWS Program matured, resulted in realistic minimum margins of safety. Whenever possible, the structure was overdesigned to be able to accept changes and/or increases in varying parameters such as weights, center-of-gravities, moment arms, changing or additional payload packages, etc., that would limit the use of changed/increased payload components. This overdesign philosophy is definitely recommended for floor structure where on OWS the early floor loadings almost doubled by the time of launch. The testing philosophy was to determine the ultimate structural capability, whenever possible, to show the limitations of the structure tested. Part of this activity was the investigation of alternate designs to provide a trade-off between design, analysis, and test for technical requirements and cost effectiveness.

2/ A recommendation for better optimization of the crew quarters floor/wall structure would be to integrate container/wall design, especially compartment walls. This would provide "plug-in" capability for the containers, drawers and shelves required for stowage, similar to the lockers and support structure above the water bottle ring. Exterior tank wall "plugins" must account for environmental provision to preclude condensation as well as easy orbital removal or swing out capability for access to tank wall areas subject to meteoroid penetration. The major advantage would be the capability of off-module stowage which saved considerable time during checkout of ring mounted stowage lockers. This concept is also compatible with late changes in stowage requirements.

3/ The entire concept of "astronaut" loads should be investigated in depth to determine items that should have protective barriers (switches, circuit breakers, etc.) and those that would require the capability to withstand impact.

4/ It is recommended that the magnitude of loads and impact (energy, velocity, etc.) be standardized. The ability to "force" a mechanism or structure during checkout in "1G" would need to be included in any general load criteria as this could be the critical load condition.

5/ Various sizes of internal wrenching bolts, hex head bolts, calfax fasteners, knurled knobs, dial latches, etc., were used throughout the workshop. Standardization of fasteners used by astronauts would have minimized logistics, design differences, analysis, astronaut removal and installation procedures and tools for use in orbit. Dial latches seemed ideal for their "quick release" function but were reported (by the astronauts) as unsuitable. Some method of securing the dial portion for the unlatched position would simplify reinstallation of equipment using this method of attachment.

The use of fastener pre-load torque in excess of "finger-tight" was used as a method of precluding inadvertant loosening of attachments during checkout or boost vibration and is strongly recommended. During component vibration or accoustical testing, there was an early history of fastener problems where the fastener/nut did not remain locked. This was due to repeated use of the hardware. It is recommended that when vibration or acoustic testing components, fasteners containing locking devices which attach the components should be monitored for possible replacement before the locking feature is worn out due to repeated assembly/disassembly.

6/ The use of the bonded disc on the interior of the insulation in the habitation area proved to be technically advantageous and cost effective for installation of all types of equipment and components within the OWS. Any limitation was due to the requirement for a manufacturing freeze in the relatively early stages of S-IVB conversion to the OWS. It is recommended that on any future system requiring internal insulation consider the use of this method.

7/ Design of the floor and placement of large masses on the floors had to be carefully balanced and computer modeled. This was to preclude developing a natural frequency that would adversely drive the components and experiments mounted on the floors and walls. This was in addition to having adequate load paths and strength capability. It is concluded that both strength and vibration characteristics must be considered simultaneously.

8/ To facilitate ease of inserting a gloved hand through the floor or crew equipment attachment grid, an increase in the 4.2 in. (10.6 cm) nominal triangle is suggested. Also, crew comments suggest larger radii at corners would be more comfortable for an ungloved hand.

9/ An overall design philosophy should be formulated to define "sharp" corners and edges to provide an industry wide standard for astronaut usage. A raised sheet metal tab that snagged an inspector's white glove would have to be compared to burrs in addition to corner radii as detailed on production drawings. Any criteria would have to address the question of long duration usage with a bare hand, or a foot with a sock on it, to an intermittent contact with a shoulder or knee.

10/ Geometric relationship of noise producing equipment in close relationship to quiet areas such as sleep station, study area, etc., should be studied and avoided. If close proximity cannot be avoided, sound suppressing material should be preinstalled or provided the astronaut for a contingency usage.

11/ Crew comments indicate that they would like more and larger viewing windows. A larger wardroom window was considered feasible but the 18 inch (45.72 cm) diameter window in OWS-1 was considered a good compromise between crew needs and structural requirements. Any increase in window size would have complicated the structural system. The same rationale would apply to a bubble window or another window in close proximity to an existing window. It is recommended that design for structural discontinuities caused by the incorporation of windows and the window design requirements be established as early as possible in future programs.

12/ A study of usage of contingency tools and applicability to on-board systems could have produced the requirement for additional tools, couplings, hoses, etc., for the possibility of station keeping repairs dependent upon peculiar problems. The contingency problem/solution exercise and the resulting requirements for hardware is very difficult to assess. It is recommended that pre-flight consideration be given to the possible crew repair of all on-board functioning systems while in orbit.

13/ Two scientific airlocks (SAL's) were provided in the OWS. Based on OWS experience, the scientific airlock concept is required for experiments in space. The scientific airlock concept is quite versatile in that it provides the backup capability to perform several functions other than the original scientific intent. On future programs, it is recommended that a minimum of two scientific airlocks be included and the size of the airlock opening be increased to allow more flexibility of airlock use.

14/ The OWS habitation tank contained a 52 by 35 inch (132.08 x 88.9 cm) side access panel between the forward and aft floors for ground access. This concept for ground access is mandatory for efficient manufacturing and checkout operations. In future programs, it is recommended that trade-offs be conducted to establish the number and size of opening required for ground access.

15/ Crew Accommodations

a. Trim and Closeouts

Since the trim and closeouts perform a passive role in debris control, it is assumed that the lack of crew comments on their effectiveness implies a satisfactory performance. The need to seal inaccessible volumes from bio-waste spills is valid. With an air flow, as in the OWS, the need to close off inaccessible spaces for control of solid debris becomes a lesser necessity. In future spacecraft, consideration should be given to the elimination of inaccessible volumes during the design of the basic interior architecture, thus, avoiding the use of a multitude of add-on closeouts.

b. Wardroom Curtain

Since the only recorded use of the curtain was in conjunction with T002 sightings, the need for such a curtain seems questionable. The hood supplied with T002 was much more effective at darkening the area around the window than was the Wardroom curtain because of the open wardroom ceiling. A curtain (or door) for a Wardroom type compartment on future vehicles would depend on its unique requirements for privacy and light and sound control.

c. Internal Finishes

The attractiveness of the interior and the general illumination level could have been markedly enhanced by use of lighter value colors with a higher diffuse reflectance. This could have been achieved by more widespread use of paint finishes, if particle generation, and fire had not been such a predominate concern. The development of improved finishes which meet the stringent spacecraft requirements (specifically fire) are definitely needed for future programs.

Greater visual interest could have been achieved by use of more high-chroma and low-value accent colors, such as the dark blue used on handrails and the dark brown used on wardroom trim and the trash airlock.

d. Astronaut Aids

1. General

The Skylab Missiors demonstrated that a foot restraint combined with appropriate handholds provides a basic restraint capability for just about all tasks. Free soaring provided a safe and efficient means of locomotion. Handholds are required for pre and post soaring orientation and for light body restraint. A universal intravehicular foot restraint approach should be developed for future missions. The PGA foot restraint platform was excellent but requires a universal adapter for attaching it at various work stations.

2. Fixed Handrails

The quantity of the fixed handrails on board was adequate. The handrails were located primarily for a wet workshop concept, thus, they were located where pressure-suited activities were planned. These were initially defined and located for use in possible contingency and initial entry operations in the event that pressure-suited activities were required. On future missions the incorporation of fixed handholds into the edge of equipment, as was done on Skylab stowage lockers, should be accomplished early in the design phase.

3. Light-Duty Foot Restraints

The light duty foot restraints met their design function which was to restrain bare, stocking, or soft slipper covered feet. The attempt by the flight crew to utilize the triangle shoe with light duty foot restraint never did work because of basic incompatibility. Future missions should have universal foot restraint capability throughout the vehicle.

4. Fireman's Pole

The Fireman's Pole was a satisfactory aid for translation from the forward hatch to the experiment compartment ceiling. It was utilized extensively when the crew transferred large bulky items.

For general locomotion, the crews preferred free soaring with the pole stowed out of the way. It is recommended that such a device as the Fireman's Pole be included in future large volume space vehicles to support activation and movement of large items.

5. Food Table Restraints

The food table restraints provided an effective means for relaxed restraint while preparing and eating food as well as data management and bio-medical tasks. The triangle cutouts for the triangle shoes were inadvertently oversized and caused some interface problems. Future missions might incorporate foot restraints only or a thigh restraint with larger thigh contact area which would eliminate the need for foot restraints.

6. Water Tank Foot Restraint Platforms

The platform provided the restraint required for ring locker access and water tank operations. It demonstrated the capability of foot restraints to accommodate a wide variety of tasks.

7. Lower Leg Restraints

The lower leg restraints were satisfactorily used at ring lockers D424 and D430 to assist two crewmen in translating, positioning and installing the condensate holding tank to its on-orbit location on the forward dome. Future missions should consider this versatile restraint which is compatible with shirtsleeve and pressure-suited crewmen.

8. PGA Foot Restraints

The PGA foot restraints proved to be very effective and convenient for IVA and EVA pressure-suited operations. They also provided restraint for PGA donning, doffing and drying. The PGA foot restraints should be standard equipment for PGA operations on future missions. A universal adapter for attaching the restraint to any structure should be developed to maximize its effectiveness.

9. Portable Handholds and Tether Brackets

Neither the portable handholds nor the portable tether brackets were used significantly because of the inherent handhold and tether attachment capability of the open grid. Future missions should consider the need for these items depending on the basic floor, wall, ceiling fabrication technique utilized.

10. Adjustable Tether

Although the adjustable tether worked relatively well, it did not afford the stability available in the fireman's pole, a rigid member.

11. Triangle Shoes

The triangle shoes provided effective restraint, with minimal loss of mobility, for just about all tasks in the Orbital Workshop. The conical cleat used by some of the crewmen in place of the triangles were not as effective. This was probably due to the fact that the conical cleat required constant effort and attention by the crewmen to stay engaged in the grid. On future missions the uppers of the triangle shoes should be constructed of more durable material and should have a more convenient means of donning such as zippers.

F. Development History

1/ The OWS habitation/waste tank areas were initially conceived for manned space application from early SIVB spent stage studies conducted by MDAC for NASA in the 1964-1966 time period. This developed into what became known as the Wet Workshop with not much more than a floor with compartment walls installed prior to launch. Most systems, equipment, and plugs or closures for tank penetrations had to be brought in from the AM/MDA and installed after the SIVB tank had functioned as a propulsive stage.

2/ With the changeover to dry workshop many requirements and developments for using the SIVB tank for OWS changed. Tank penetrations were permanently plugged. Floor and compartment arrangements were designed for ground installation of equipment and systems hardware. Scientific airlocks, a viewing window, and a trash airlock were added in the tank structure. Non propulsive vents, liquid dump probes, and a screen system were added to the LOX tank to allow it to become a waste tank for liquid and solid trash. To allow access for fabrication and installation of hardware in the interior habitation area, a side access panel was added to the tank cylinder at the crew quarters level. Many other tank changes were developed to accommodate OWS systems such as the water, refrigeration, and stowage along with associated astronaut aids and crew accommodations.

3/ Crew Accommodations

a. Closeouts

The development of closeouts, to control debris and loss of small items, was initiated as part of the wet to dry OWS conversion. Additional closeouts were added as new equipment and equipment relocation generated new inaccessible areas requiring debris control.

b. Wardroom Curtain

The wardroom curtain underwent few development changes. Emergency breakaway features were converted from Velcro to snaps to provide a more durable curtain.

c. Color and Finishes

Initial finishes were all anodized to be compatible with liquid H_2 on the wet workshop. As part of conversion to a dry workshop a color scheme utilizing pigmented Micatex paint was selected. Subsequent tests demonstrated a lack of durability and some pigments were toxic. As a result, a color scheme utilizing anodized finishes was choosen and implemented. Some surfaces, such as ring container lids, were later painted with Laminar X-500 to increase the effective light levels to improve photography.

d. Astronaut Aids

The portable tether attachment brackets and portable handholds were developed along with the grid floor. They underwent minor design changes to reduce the torque loads on the grid floor.

The firemans pole was originally designed to aid in transfer of equipment into the wet workshop utilizing an equipment carrier that utilized the pole as a track. After the conversion to a dry workshop, the firemans pole was retained for crew locomotion in the forward area.

The adjustable tether for locomotion in the forward area was added relatively late in the program to provide a flexible aid for locomotion in the forward area.

The fixed handholds and handrails were an inherent part of the wet workshop. The number and locations of the handholds and handrails went through numerous changes during wet to dry conversion but remained basically firm through the rest of the development program.

The light duty foot restraints were introduced about half way through Skylab development. They were provided to give foot restraint independent of the triangle shoes. Initially, a platform with both triangle cutouts and light duty foot restraints was proposed. The triangle cutouts were deleted by NASA direction. The straps were modified to utilize flourel instead of PBI webbing straps to meet flammability requirements and minimize cleaning problems.

The thigh restraint at the wardroom table was introduced as a result of underwater development tests on the wardroom table. The initial mechanical lock adjustment joints were converted to friction joints to improve useability.

The water tank foot platform was initiated to provide restraint for accessing the ring containers and provide a mount for hoses, portable H_2O bottle and cameras.

The triangle shoes were a part of the original wet workshop with its grid floors and ceilings. The original shoes had static triangles which did not lock in place. Underwater testing resulted in the incorporation of dynamic triangles which would lock in place with torque overide. The triangle was made adjustable along the sole of the shoe to accommodate the bicycle ergometer and a laced closure replaced a velcro closure at NASA direction.

The PGA foot restraint was initially the "dutch shoes" designed for use on the Gemmini Program. Underwater testing showed that the "dutch shoes" did not provide adequate restraint. NASA developed a positive heel locking restraint which was flown on Skylab.

2.2.1.3 Aft Support Structure

A. Design Requirements

1/ Provide mounting structure for (a) TACS system nitrogen gas storage spheres with manifolding and meteoroid shielding. (b) Refrigeration system radiator assembly/thermal control panel with associated equipment and piping.

2/ Contamination sensitive elements shall be shielded from direct impingement of reaction control systems or venting.

3/ Add on system hardware must clear aft interstage at separation.

B. System Description - The aft support structure, a modified SIVB thrust structure, was a 94° cone made of 7075T6 chem. milled skins, extruded stringers, and clad sheet frames with an A356 aluminum casting at the apex. (Figure 2.2.1.3-1). Like the SIVB thrust structure, the aft structure attached to a flange on the waste tank aft dome with 96 bolts. Modifications made to the basic thrust structure included replacing two of the three frames with heavier frames and fittings to carry loads from the TACS spheres. Also the casting, formerly a SIVB J-2 support fitting, was remachined to accommodate the radiator and radiator shield installations. New hardware includes saddle-type pans to support 23 gas storage spheres installed on the conical surface of the aft structure (Figures 2.2.1.3-2, -3, -4). Each sphere was retained in its pan by four preloaded 301 CRES straps. The spheres, manufactured from Tl-6AL-4V titanium forgings, had a 25 in. (63.5 cm) outside diameter with a wall thickness of .354 in. (.899 cm). Twenty two of the spheres were manifolded together by means of peripherally mounted tubing

SKYLAB - ORBITAL WORKSHOP AFT STRUCTURE

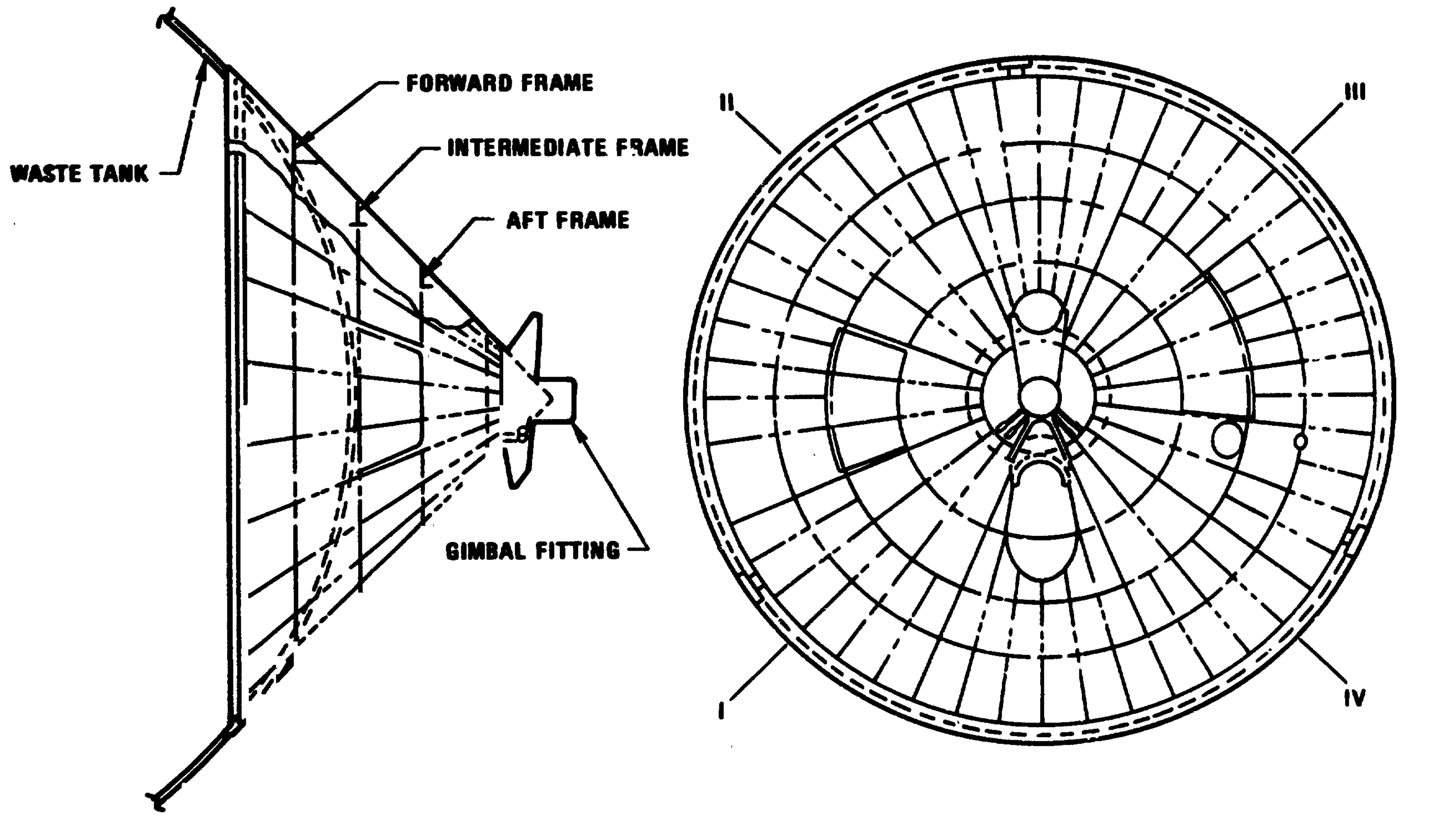


Figure 2.2.1.3-1

SKYLAB - ORBITAL WORKSHOP
TACS GAS STORAGE SPHERE INSTALLATION SUBSYSTEM

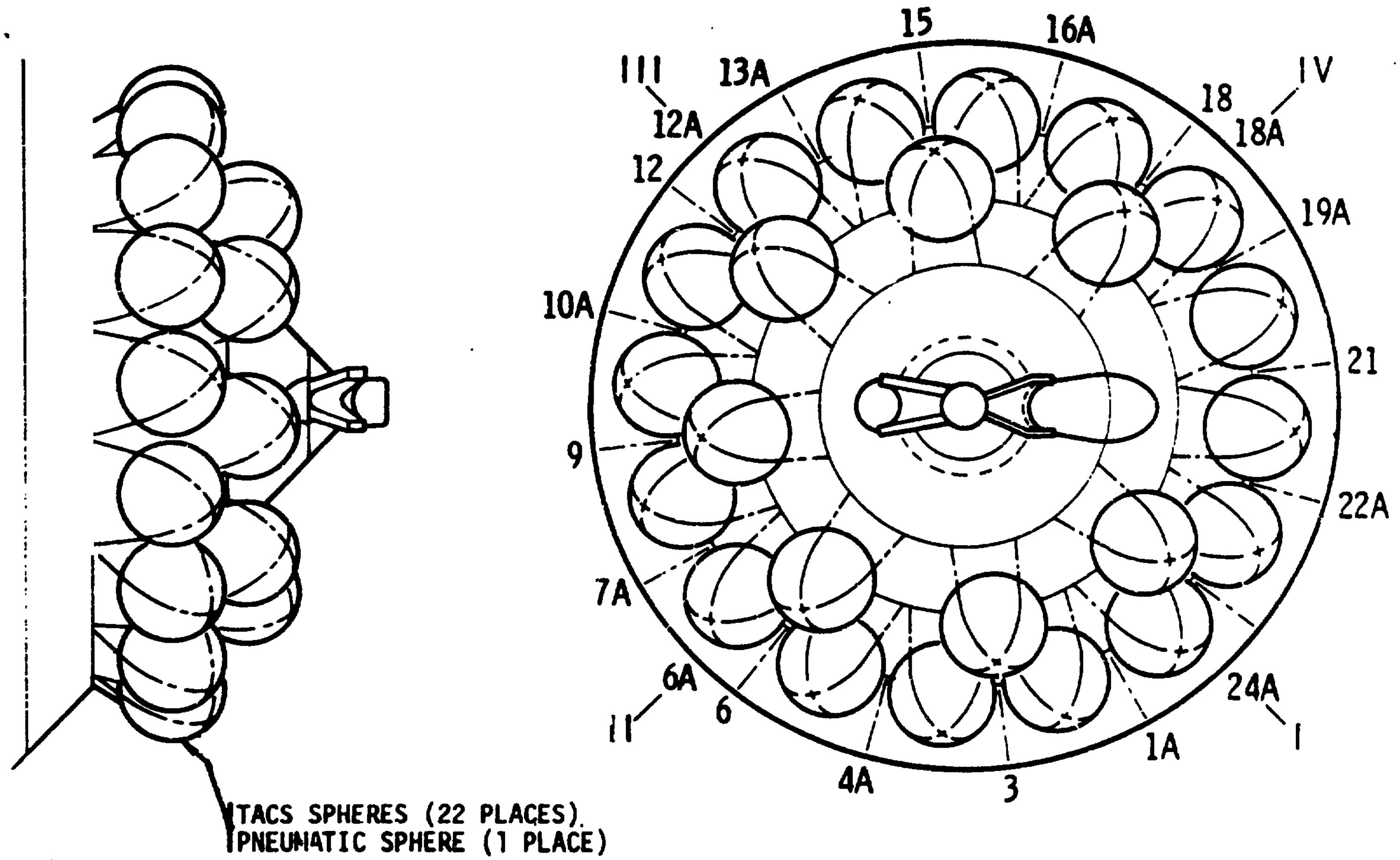


Figure 2.2.1.3-2

SKYLAB - ORBITAL WORKSHOP
TACS GAS STORAGE SPHERE INSTALLATION
SUBSYSTEM

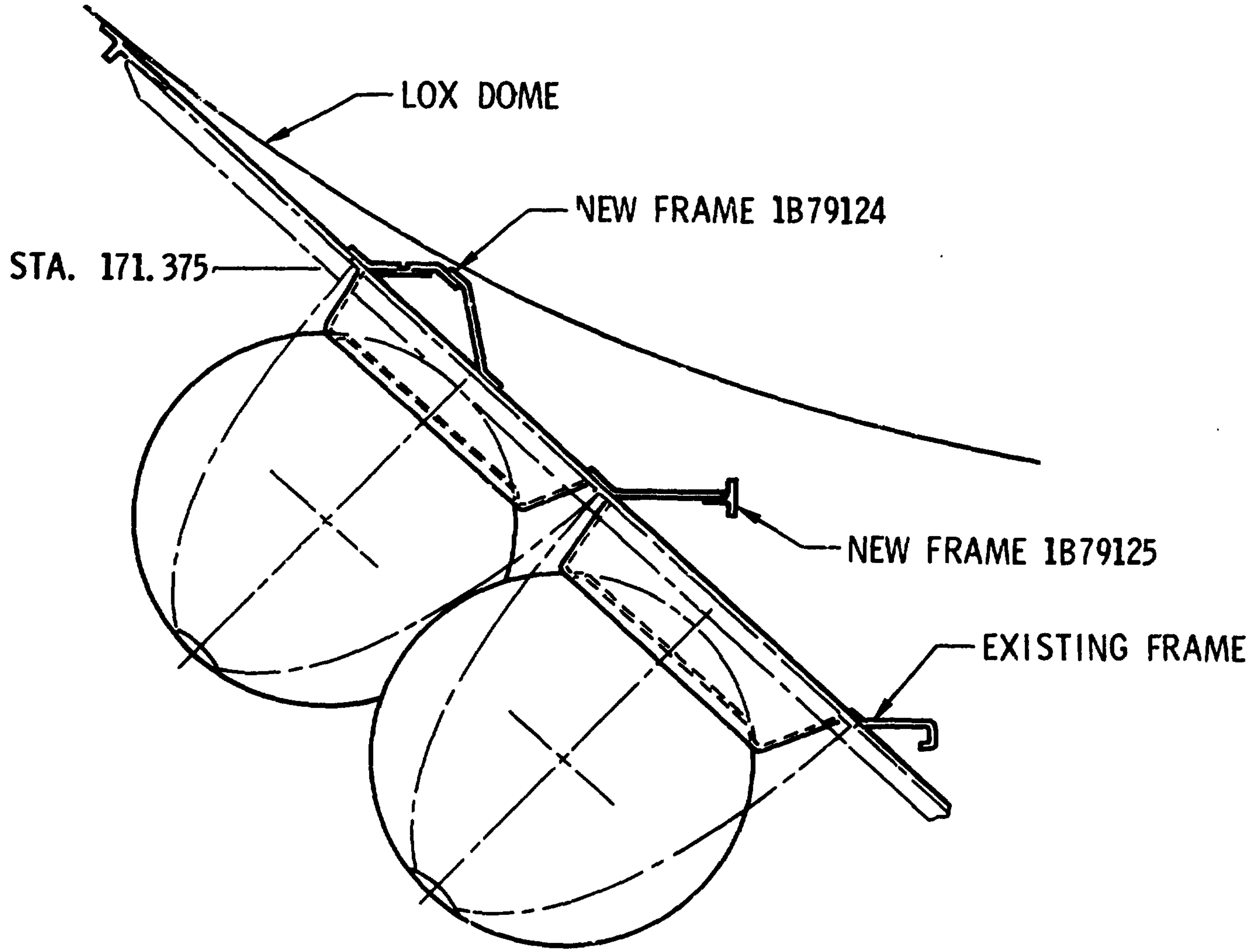


Figure 2.2.1.3.3

SKYLAB - ORBITAL WORKSHOP
TC-9 TEST SPECIMEN

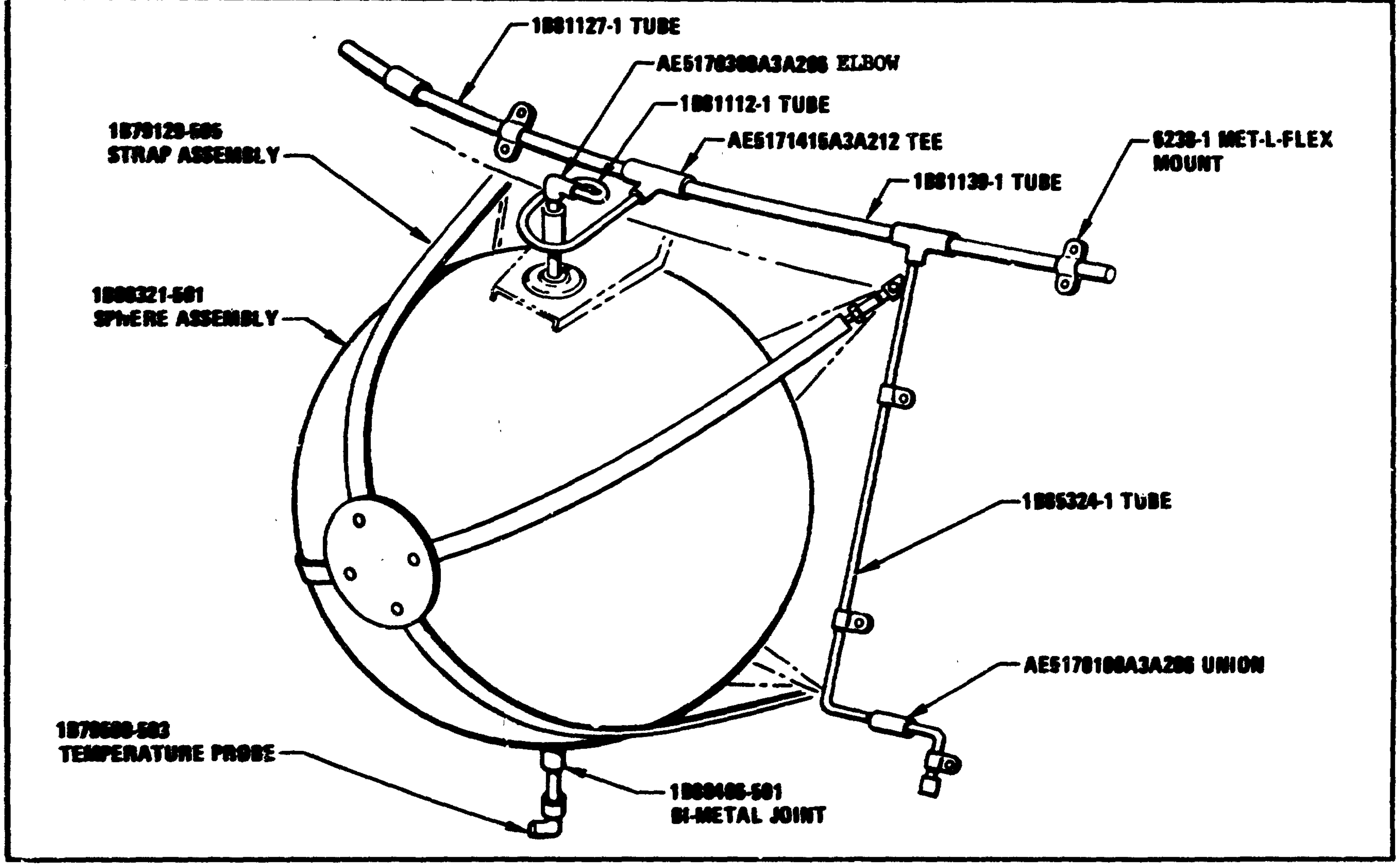


Figure 2.2.1.3-4

attached to the aft structure for the TACS system. The other sphere was used in the pneumatic control system.

A TACS sphere meteoroid shield covered the 23 spheres and manifolding. The shield, annular in shape, consisted of eight aluminum alloy identical segments connected together by extruded tees and screws.

The TACS sphere meteoroid shield system was designed for the probability of no impact in lieu of no penetration for the TACS spheres. Each segment of the shield covered a 45-degree circular section and consisted of double-wall construction separated by three 2 in. (5.08 cm) extruded frames. The outer compound curved skin was 0.050 in. (1.27 mm) 2014-T6 which was chem-milled after forming to a thickness of 0.039 in. (.99 mm) + 0.000 - 0.009 in. (.228 mm). The two lower outer skins were beaded 0.032 in. (.812 mm) 2024-T42 sheet material. The inner compound curved skin was 0.080 in. (2.03 mm) 2014-T6 sheet material. The two lower inner skins were also 0.080 in. (2.03 mm) 2024-T3 sheet material. The two intermediate "H" frames were made from 2024-T42. The lower closing frame was an extruded 2014-T6 channel. Segments were spliced together through a 0.063 in. (1.6 mm) extruded 7075-T6 tee and bolts. The segments were bolted to a 0.063 in. (1.6 mm) 2024-T4 "Z" section circular support ring which attached to the aft structure stringers by 52 extruded 7075-T6 "T" clips, Hi-loks, and huck bolts.

The TACS sphere meteoroid shield skirt was bolted to the meteoroid shield with a 0.090 in. (2.29 mm) 2024-T42 angle. The skirt (1B86008) was double-wall construction and circular in shape. It was approximately 100 in. (254 cm) in diameter and tapered from 9.00 in. (22.9 mm) to 0.50 in. (1.27 mm) deep to match the slope of the radiator. The outer skin was 0.032 in. (.812 mm) 2024-T3 and the inner skin was 0.080 in. (2.03 mm) 2024-T3. The upper cap was a 2 in. (5.08 cm) deep 2014-T62 extrusion channel; the lower cap was a 2 in. (5.08 cm) deep 0.080 in. (2.03 mm) 2024-T42 sheet metal channel (Figures 2.2.1.3-5, -6). The radiator support structure (1B80678), an irregular octagon-shaped sandwich structure with four structural supporting beams on its forward side, was attached to and stabilized by eight support struts from the aft structure (Figure 2.2.1.3-7). The sandwich structure consisted of a 5 in. (12.7 cm) thick polyurethane foam core with a forward skin 0.025 in. (.635 mm) thick by 119 in. (302.3 cm) by 119 in. (302.3 cm) made from 2024-T3 sheet. Attached to the forward skin were tapered hat-section supporting beams made from 0.063 in. (1.6 mm) 2024-T3 aluminum sheet and tapered from a 2.00 in. (5.08 cm) depth at the end to a 4.00 in. (10.2 cm) depth at the center. The aft skin, 0.025 in. (.635 mm) by 119 in. (302.3 cm) by 119 in. (302.3 cm) 6061-T6 aluminum sheet had coolant tubes seam-welded to its foam side face. The core consisted of polyurethane foam rigid blocks (9707132, Type I, Class III, density 2), with continuous fiberglass roving bonded tegether and polyurethane (9709132) foamed in place. A channel, 5.00 in. (12.7 cm) deep, 0.040 in. (1.02 mm) 2024-T3 aluminum sheet was bonded to the core around the periphery of the panel assembly. It was also riveted to the forward skin and bolted to the aft sk . The eight support tubes made from 7/8 in. (2.22 cm) O.D. by 0.065 in (1.65 mm) wall 347 CRES, had spherical rod-end bearings, with an A286 body and a type 718 nickel base alloy ball threaded into each end. One end of each tube assembly bolted into a fitting made from 2024-T351 aluminum plate, which in turn bolted to two radiator support structures beams. The other end of each tube assembly bolted to a

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TACS SPHERE METEOROID SHIELD INSTALLATION

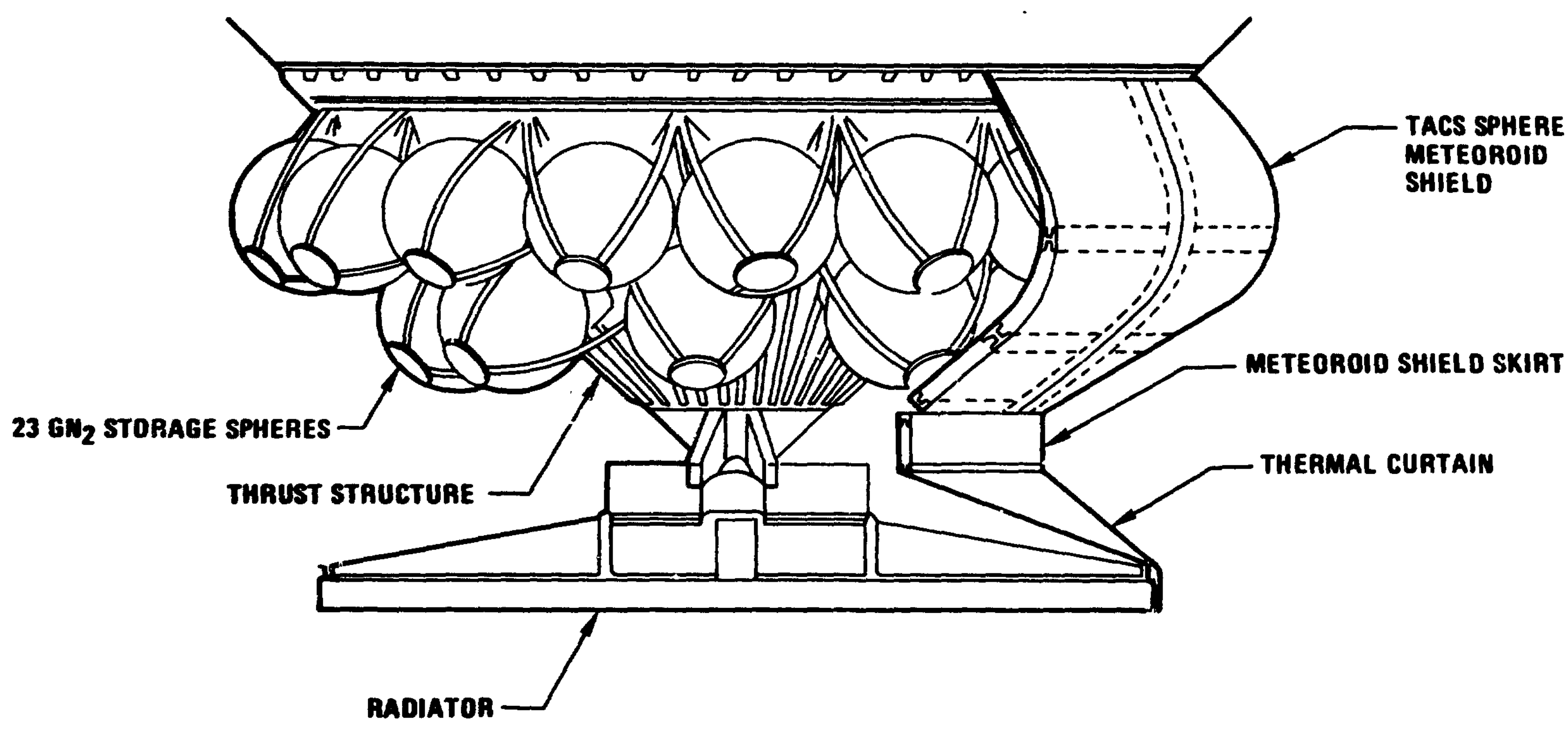


Figure 2.2.1.3-5

SKYLAB - ORBITAL WORKSHOP
TACS SPHERE METEOROID SHIELD AND SKIRT

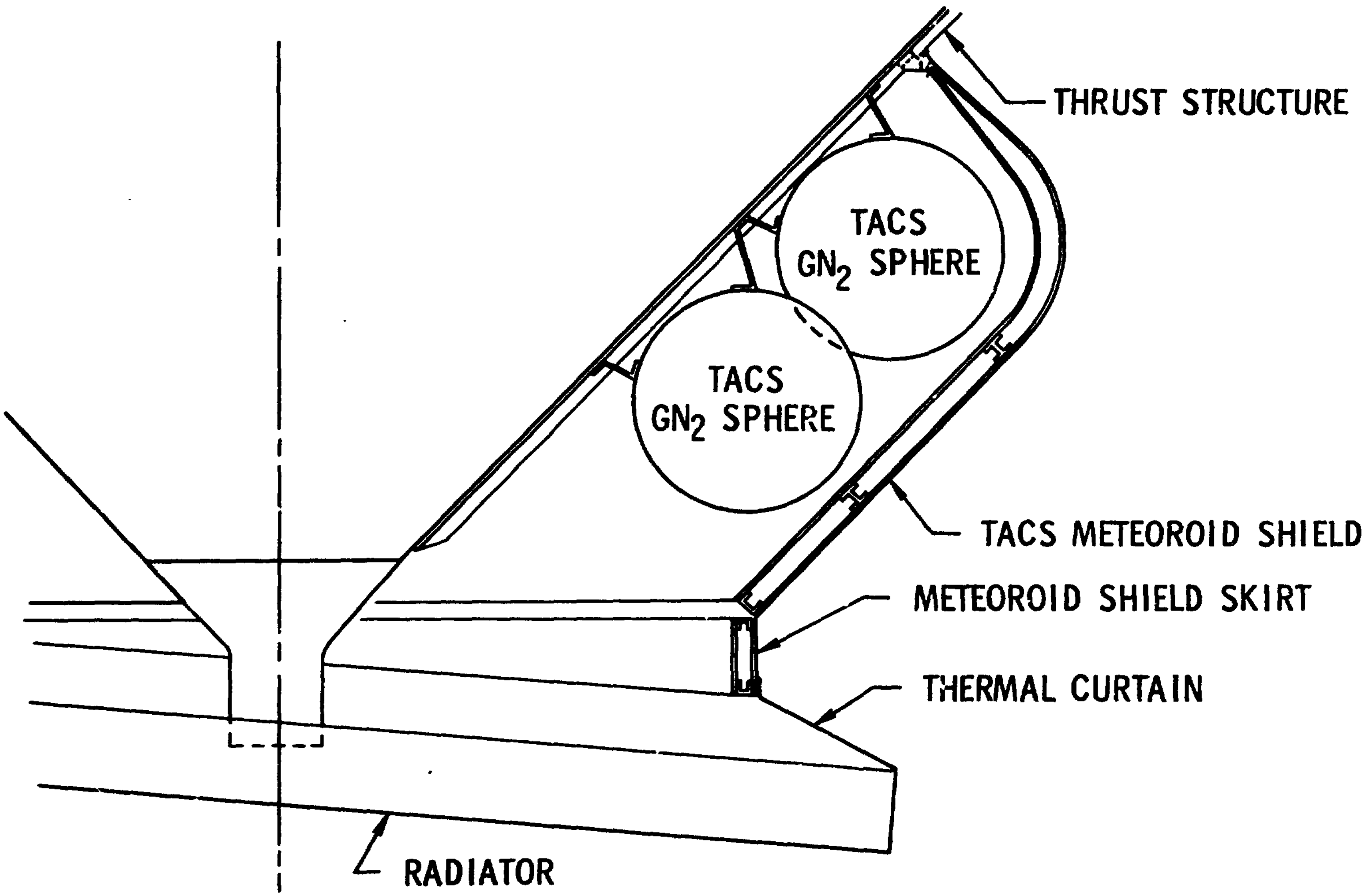


Figure 2.2.1.3-6

SKYLAB - ORBITAL WORKSHOP
AFT STRUCTURE - RADIATOR SUPPORT STRUCTURE

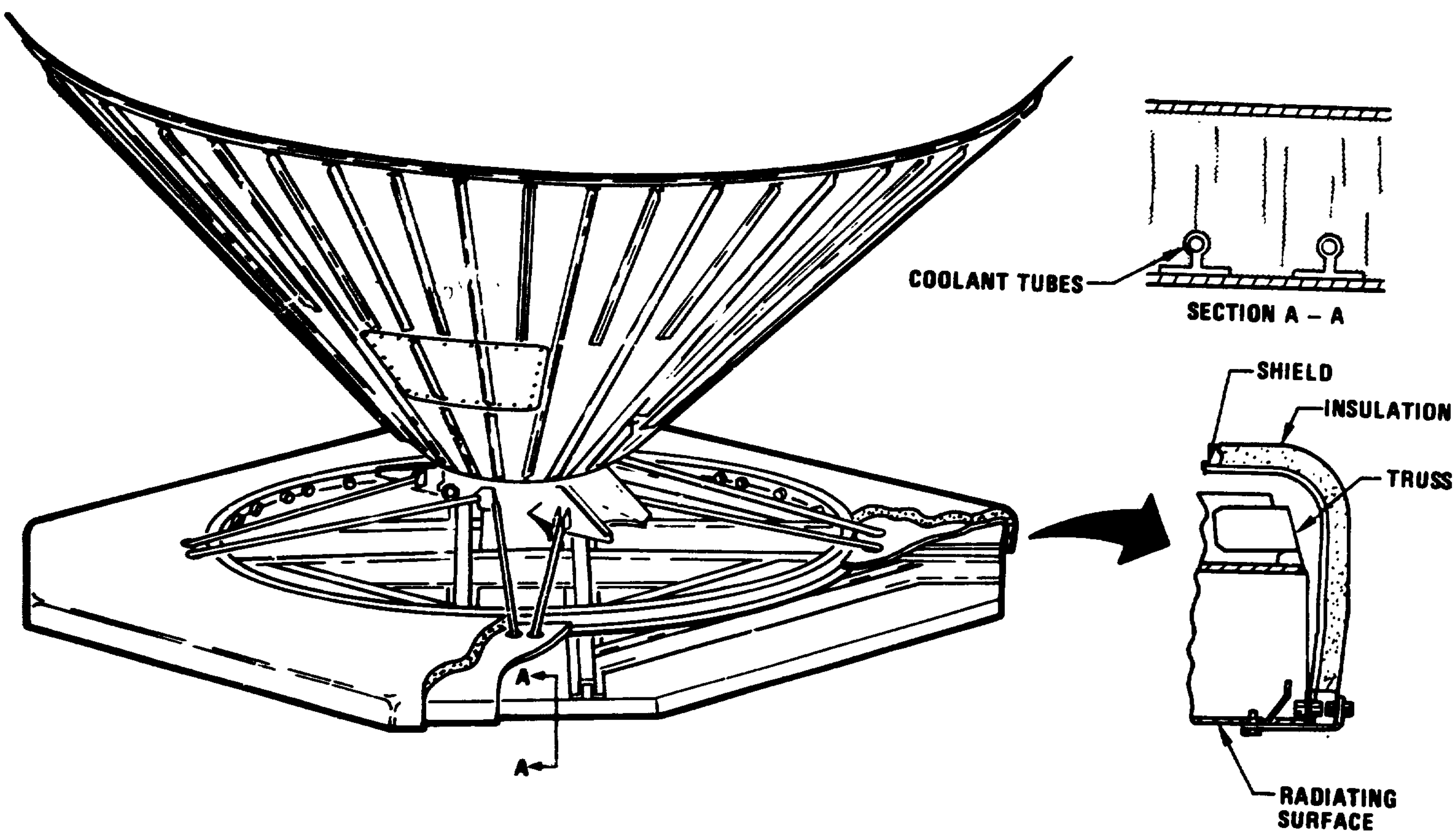


Figure 2.2.1.3-7

fitting from 7075-T7351 aluminum bar which was mounted to the thrust structure casting. (Figures 2.2.1.3-8, -9, -10, -11.) The radiator assembly bolted at four places at the center to the thrust structure casting through the fitting assembly. The fitting assembly bolted to the radiator support structure beams. This fitting assembly was made from A286 CRES which was embedded in glass fiber cloth, MIL-C-9084, type 8, class 2. A thermal control unit panel assembly supported the thermal capacitors, ground cooling heat exchangers, and other units used in conjunction with the radiator. The loads were transmitted to the thrust structure by 4 tubular struts and beams.

The panel, with all the radiator components mounted on it, and its installation was designed to withstand the acoustic and vibration environment. This package was located between the radiator and the sloping surface of the thrust structure inside the TACS bottle meteoroid shield. The panel, the same honeycomb design as the electrical equipment panels, had laminated fiberglass epoxy face sheets, 0.032 in. (.812 mm) thick, bonded to a 1.125 in. (2.85 cm) thick heat and cryogenic-resistant, phenolic-reinforced, glass fabric honeycomb core. Glass fiber cloth volan finish "C" channels 0.063 in. (1.6 mm) thick were installed around the periphery of the panel. Delron inserts were installed in the panel for mounting the radiator components. The panel was installed on the aft structure by four tabular 2024-T4 struts 7/8 in. (2.22 cm) O.D. and approximately 26 in. (66.04 cm) long. These struts bolted to the aft structure stringers and to an A286 CRES clevis (1B84673) which was bolted to four corners of the panel.

SKYLAB - ORBITAL WORKSHOP
RADIATOR SANDWICH CONSTRUCTION

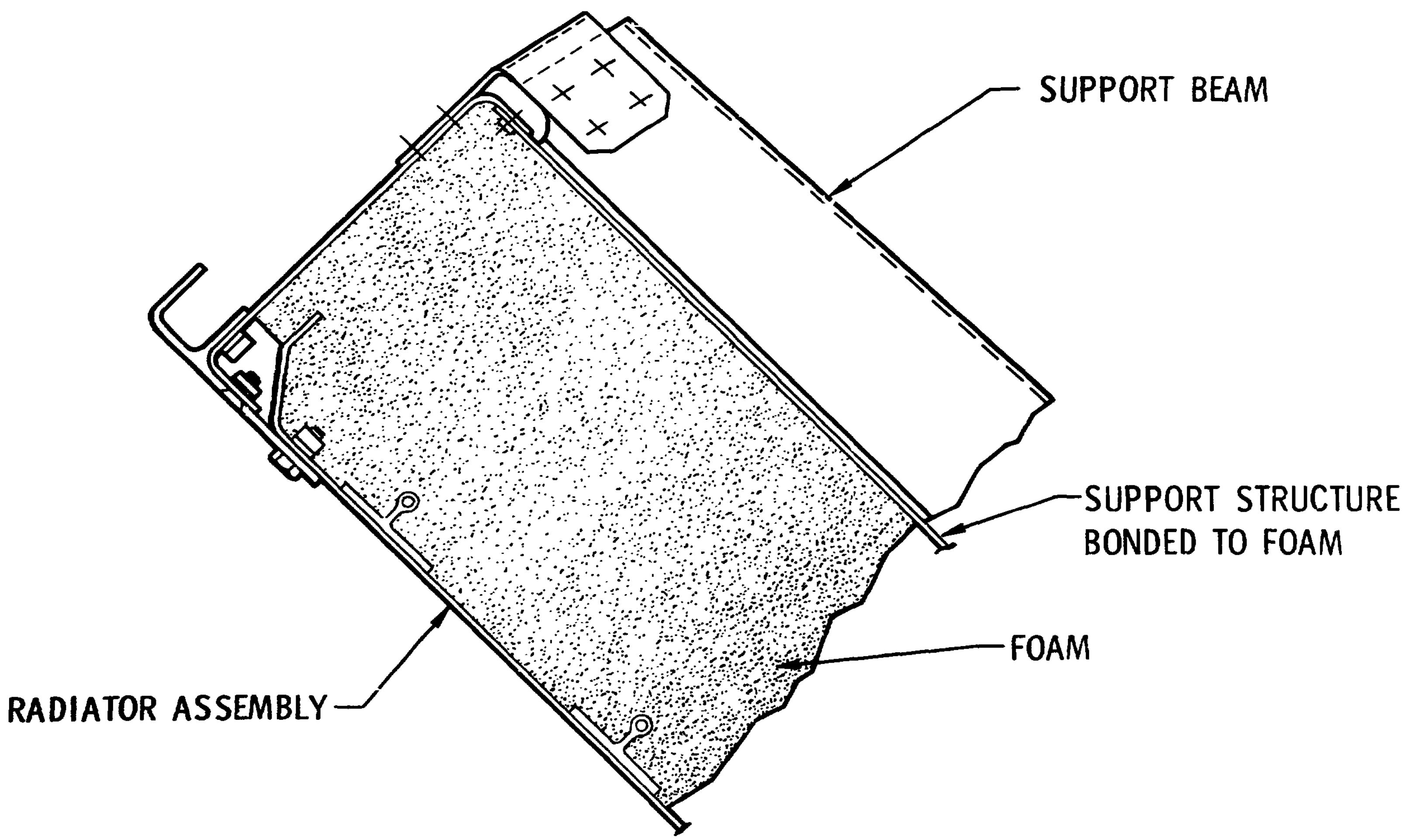


Figure 2.2.1.3-8

SKYLAB - ORBITAL WORKSHOP
RADIATOR - THRUST CASTING CONNECTION

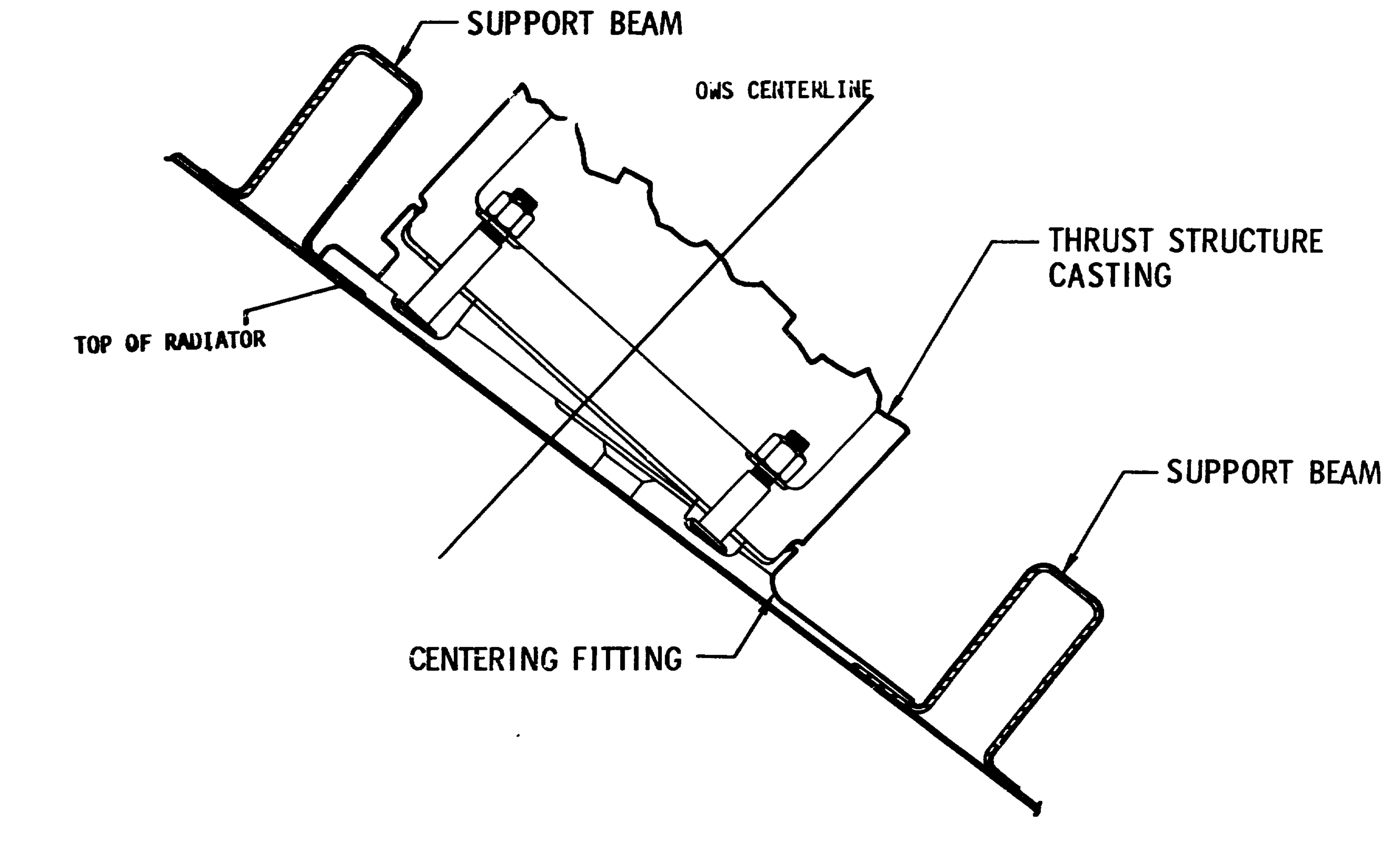
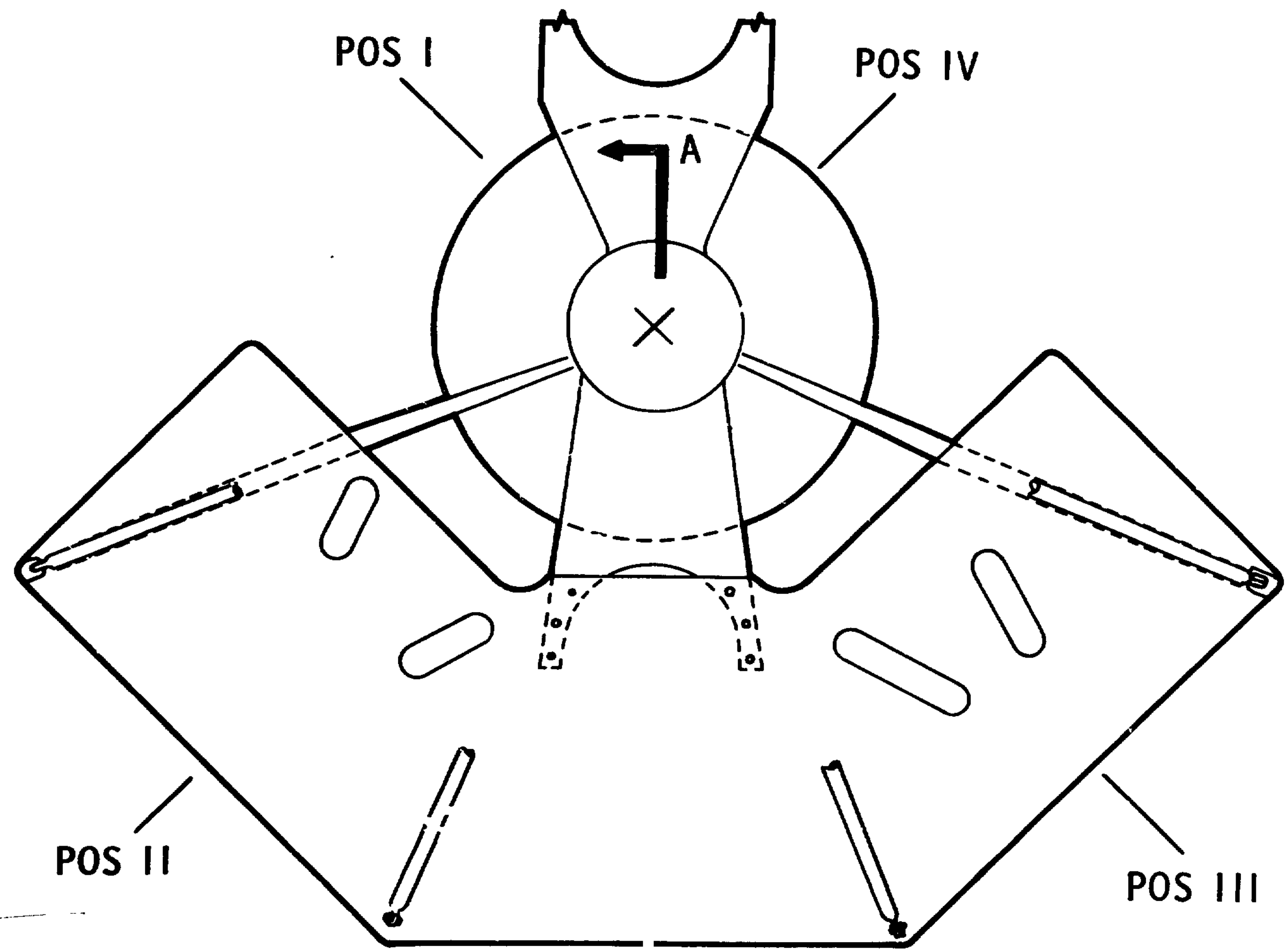


Figure 2.2.1.3-9

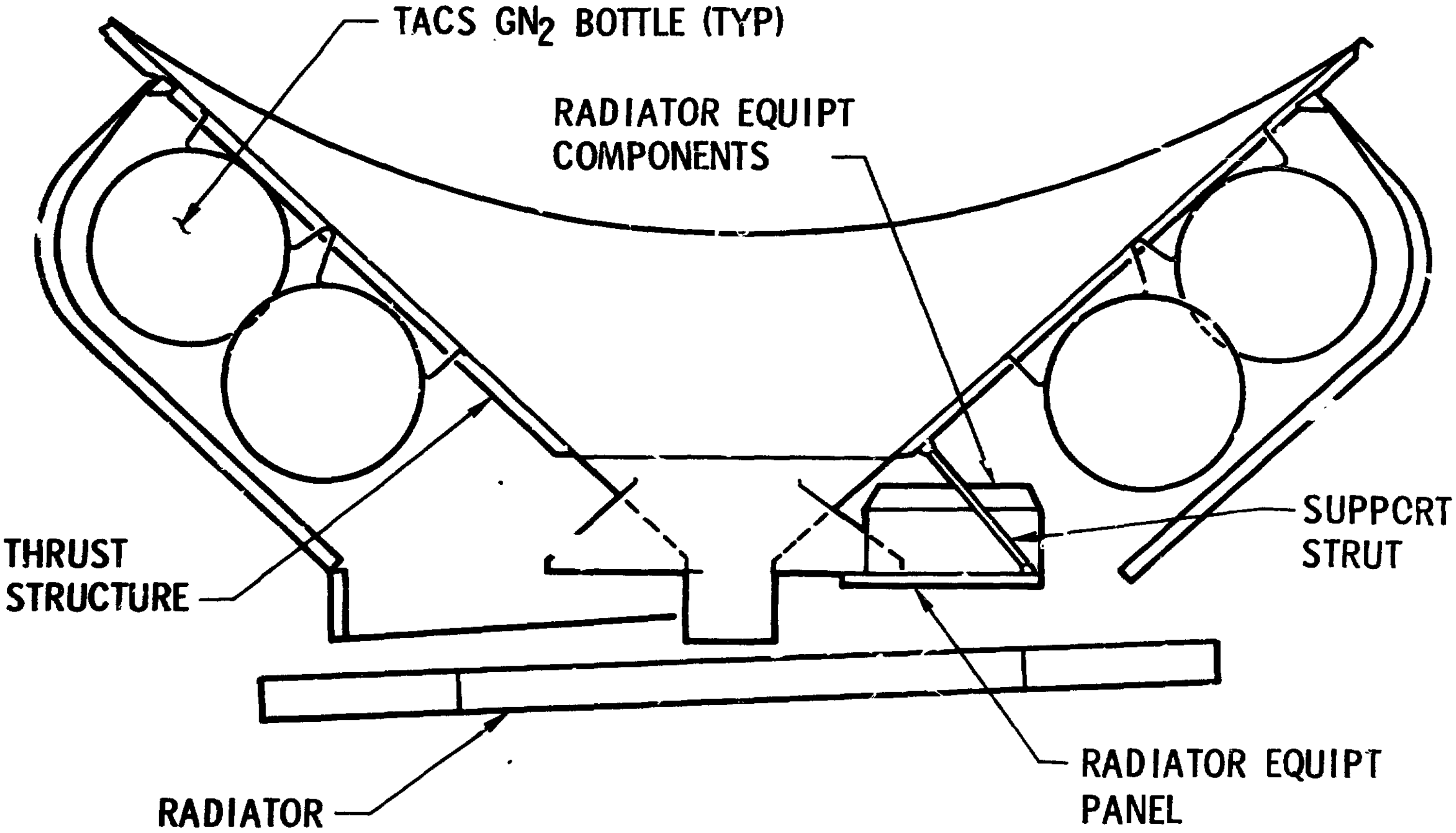
SKYLAB - ORBITAL WORKSHOP
THERMAL CONTROL UNIT INSTALLATION



See Figure 2.2.1.3-11

Figure 2.2.1.3-10

SKYLAB - ORBITAL WORKSHOP
THERMAL CONTROL UNIT INSTALLATION



SECTION A-A

See Figure 2.2.1.3-10

Figure 2.2.1.3-11

Two 2024-T42 sheet metal beams, located at the outermost ends of the panel, were bolted to the panel and the thrust structure casting. The inboard center of the panel was bolted to an extended leg of the aft structure casting.

A radiator impingement shield was attached to the radiating surface of the refrigeration system radiator. The shield provided protection for the radiator from retro-rocket plume impingement. Approximately 9 minutes after second-stage separation the shield was jettisoned by the IU automatic sequencer. A backup command was provided through the airlock module DCS. The radiator impingement shield was made from a lightweight 6061-T6 aluminum truss to which a sheet of insulation covered with Beta cloth was attached (Figure 2.2.1.3-12). The shield had a single support point at its center where it was attached to the jettison actuator by means of a ball release mechanism.

The 1B84989 support truss was welded and brazed aluminum structure composed of a tubular ring shaped to the peripheral contours of the radiator. Six tapered radial beams supported the ring.

The insulation was sandwiched between layers of Teflon-coated Beta cloth which were sewn to the outer ring of the truss.

The shield assembly was designed to be jettisoned by a single, symmetrical separation force at the center of gravity. To ensure a central center of gravity location which was 0.030 in. (.762 mm) within theoretical, the shield was balanced during assembly. Rotation of the installed assembly about the vehicle centerline was prevented by two anti-r   ion pins. The conical shape of these pins ensured instant disengagement during shield separation.

SKYLAB - ORBITAL WORKSHOP RADIATOR IMPINGEMENT SHIELD

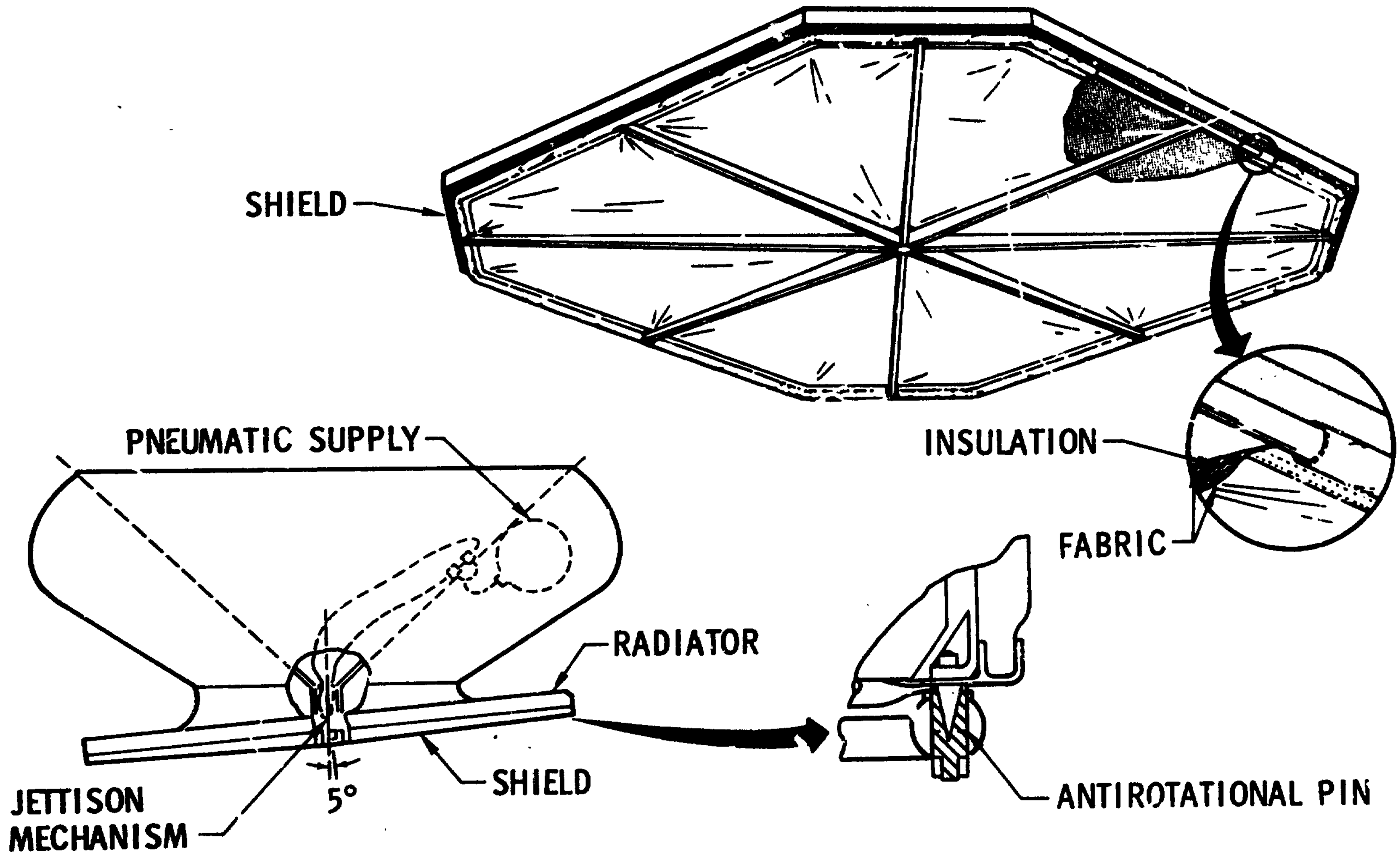


Figure 2.2.1.3-12

Shield separation was accomplished by energizing a dual-piston pneumatic actuator (Figure 2.2.1.3-13) that operates a ball release mechanism attached to the shield center. A separation spring mounted in the ball release mechanism provided enough force to eject the shield at a velocity sufficient to preclude stage recontact by the shield.

A plume impingement curtain located between the aft skirt and the aft structure provided protection from retrorocket plume impingement for the area between the aft skirt and the aft dome. The curtain was subjected to boost acceleration loads, vibrational loads, retrorocket gas impact, and thermal environments. This glass fabric curtain is attached to the aft frame of the aft skirt and the attach angle of the aft structure/aft dome joint. This construction did not constrain movement between the aft skirt and habitation area tank structure (Figure 2.2.1.3-14).

The curtain construction provided controlled ventilation of atmospheric pressure from the enclosed volume. Zippers in the curtain allowed access to equipment in the aft skirt area. To avoid loading the tubes and wire bundles which penetrated through the curtain, flexible boots were installed around the penetrations. The 1B65109-509 plume impingement curtain was made from aluminized, silicone-coated glass fabric. Eight curtain segments were jointed by leakproof zippers to form a truncated cone. At the skirt attach point, the curtain rested against the inner cap of the aft interstage forward frame. To prevent chafing of the curtain material, a silicone-rubber rub strip was bonded to the frame cap.

SKYLAB - ORBITAL WORKSHOP

RADIATOR SHIELD JETTISON MECHANISM

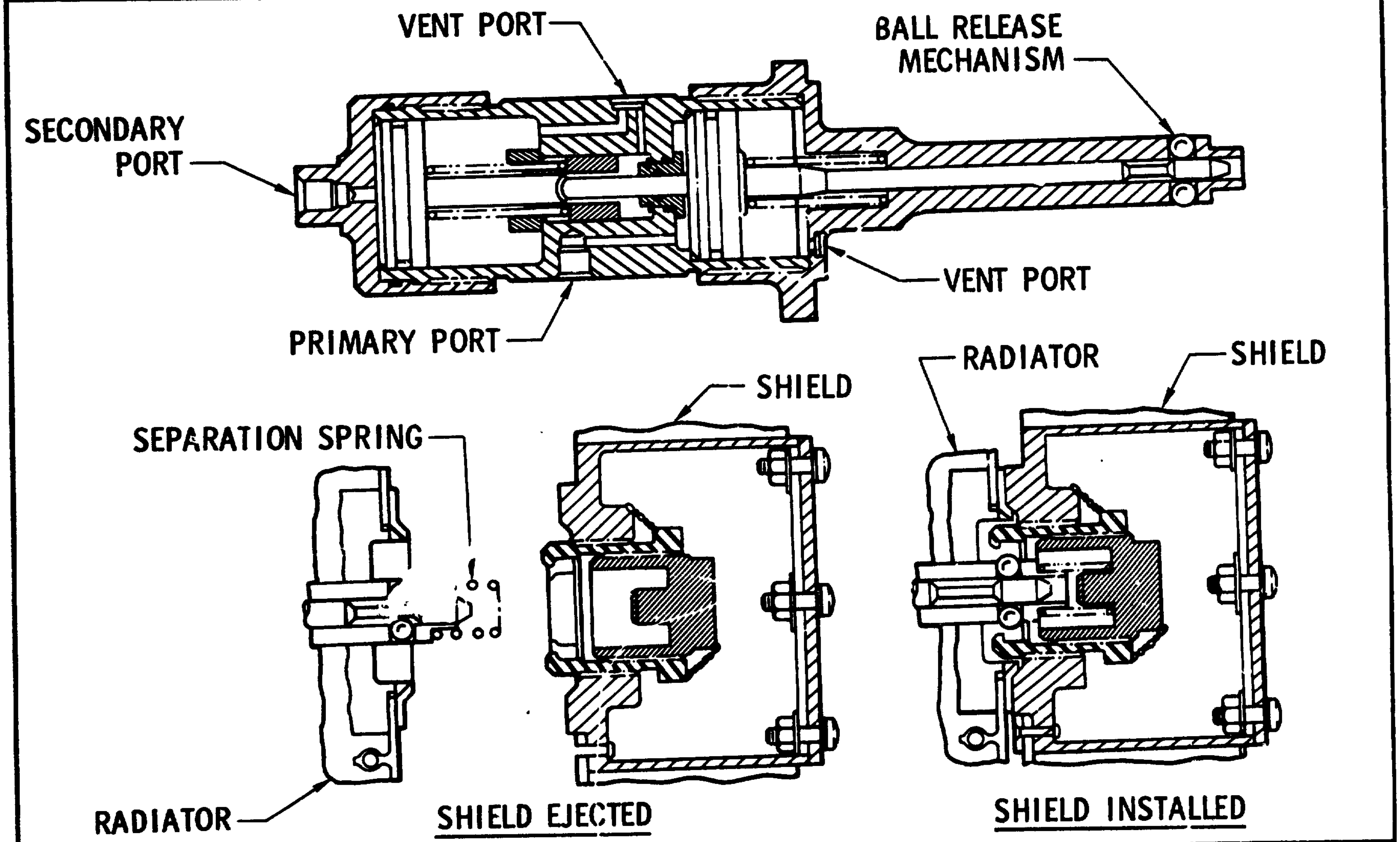


Figure 2.2.1.3-13

SKYLAB - ORBITAL WORKSHOP
PLUME IMPINGEMENT CURTAIN

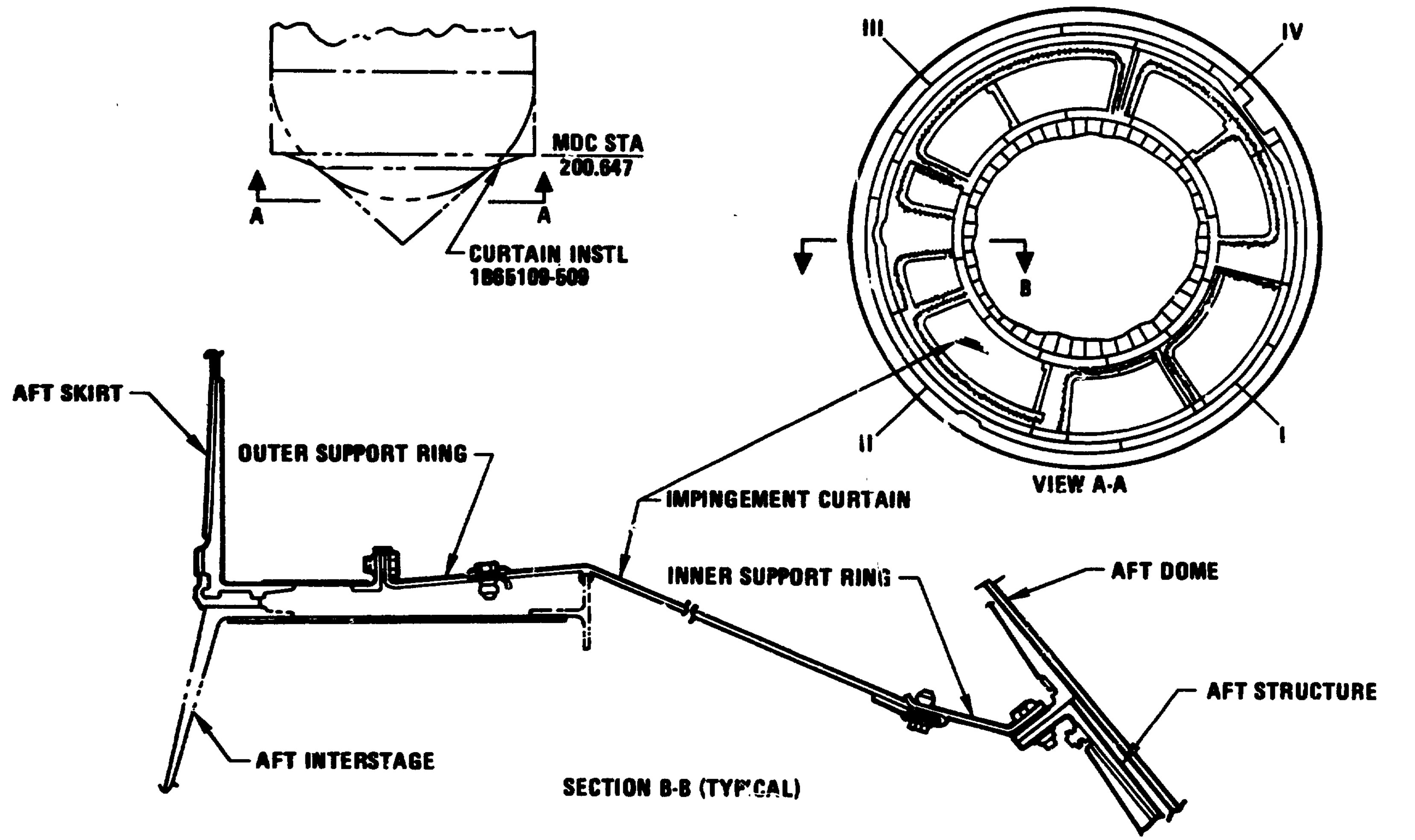


Figure 2.2.1.3.14

C. Testing - The DTA used flight-type aft structure and associated mounted mass simulated components during the acoustic and vibration tests performed on this vehicle. These tests demonstrated the structural integrity of the aft structure to withstand the loads induced by the launch acoustic and transient environment. Additionally, they confirmed the validity of the dynamic math models used to determine vehicle dynamic loads on the aft structure during launch and boost. Line Item TC-9 was required to qualify the TACS sphere installation because of the increased weight and vibrational loads with additional sphere and nitrogen gas when compared to the S-IVB/IB/V helium sphere installation. The qualification test consisted of a series of leak checks, pressure cycle tests, vibration tests, and a burst test to verify the structural integrity of the TACS sphere installation. Also, the test was used to qualify the following manifolding items.

1/ A section of the 3/4 in. (1.91 cm) O.D. manifold, consisting of 1B81127-1 tube, 1B81139-1 tube, braze fittings AE5171415A3A212 Tee, AE5170100A3A212 union, and 6238-1 Met-L-Flex mounts.

2/ Manifold to upper sphere connecting tube pigtail 1B81112-1.

3/ Manifold to lower sphere tube number 1B85324-1.

4/ 1B80406-1 and -501 bi-metal joints and 1B79580-503 temperature probe, both of which are part of the 1B80321 sphere assembly.

Table 2.2.1.3-1 lists all tests, test conditions and requirements, and test results. Each 1B80321 TACS sphere has been subjected to a

Table 2.2.1.3-1

TC-9 QUALIFICATION TEST: TACS SPHERE INSTALLATION

Test	Test Condition and Requirements	Results
Pretest proof test	Strap loads: 100^{+0}_{-100} lb ($45.36^{+0}_{-45.36}$ kg) Temp: 70° ± 25°F (21.1° ± 3.9°C) Pressure: $4{,}800^{+0}_{-100}$ psig ($330.9 \times 10^{5}{}^{+0}_{-6.89 \times 10^5}$ N/m²) Gas: GN_2	No failure
Leak test	Gas: GHe Pressure: $1{,}750^{+0}_{-100}$ psig ($120.6 \times 10^{5}{}^{+0}_{-6.89 \times 10^5}$ N/m²) Temp: 70° ± 25°F (21.1° ± 3.9°C) Allow leakage: 1×10^{-5} secs per joint using mass spectrometer in sniffing mode.	2.5×10^{-8} SCCS
Pressure cycle test	100 cycles Gas: GN_2 Cycle Definition: Pressurization-3 minutes from 100 ± 100 psig to 3,100 psig ($6.89 \times 10^5 \pm 6.89 \times 10^5$ to 213.7×10^5 N/m²) Hold-one minute at 3,100 ±100 psig ($213.7 \times 10^5 \pm 6.89 \times 10^5$ N/m²) Depress-3 minutes from 3,100+100 psig to 100 ±100 psig ($213.7 \times 10^5 + 6.89 \times 10^5$ to $6.89 \times 10^5 \pm 6.89 \times 10^5$ N/m²)	No failure
Leak test	Same as leak test above	1×10^{-9} SCCS

Table 2.2.1.3-1

TC-9 QUALIFICATION TEST: TACS SPHERE INSTALLATION (Cont'd)

	Pressure: 3,100 psig (213.7×10^5 N/m^2) Gas: GN_2 Strap loads:	
Leak test	Same as first leak test	9×10^{-10} SCCS
Sinusoidal	Flight axis: 3Hz to 60Hz	
Vehicle dynamic*	Lateral axis: 2 Hz to 20Hz	
Sinusoidal vibration evaluation criteria*	On three perpendicular axis 20 Hz to 2,000 Hz	No failure
Liftoff level random vibration criteria*	On three perpendicular axis 20 Hz to 2,000 Hz for one minute per axis	No failure
Boost level random vibration criteria*	On three perpendicular axis 20 Hz to 2,000 Hz for two minutes per axis	No failure
Leak test	Same as first leak test	3.5×10^{-8} SCCS
Dye penetrant inspection	All welds and joints	No anomalies
Burst pressure	Pressurize with water at 500 psig (34.5×10^5 N/m^2) per minute Temp 70° ± 25°F (21.1° ± 3.9°C) Design burst pressure 8,000 psig (551.52×10^5 N/m^2)	8,961 psig (617.8×10^5 N/m^2) for one minute. No failure. However, NAS1351C4-28 strap tie-down bolts yielded. As a result, bolt specification was changed to NAS1351-4-28.

production acceptance test. Test requirements and results are listed on Table 2.2.1.3-2.

The DTA included a flight version of the TACS sphere meteoroid shield. Structural adequacy of the shield skins was verified for fatigue loads from acoustic pressures by tests on the DTA and strength analysis. The shield was verified for critical low-frequency quasi-static dynamic loads by strength analysis only using an ultimate factor of safety of 3.00 and a yield factor of safety of 2.00. Therefore, data for determining statically equivalent dynamic load factors on the shield were not required from the DTA test results.

Under development test line item HS-31, vibration and acoustic tests per DAC-56620C, Zone 2-2, were successfully conducted on the radiator support structure. After modification of this test specimen to conform to the current flight configuration, the structure was successfully tested under test line item HS-77-2.

The difference between the development test (HS-31) and the qualification test (HS-77-2) was that in test HS-31 the support beams had been cut off approximately 9.00 in. (22.9 cm) from the end (4 places) and the upper skin had been cut away in these areas in order to repair and replace leaking manifolds (1T41060, Serial E.O. -001A). New sections of support beams and new sections of skin were spliced in and the test was completed. This same rework was accomplished on the qualification test specimen (1B79875 Serial E.O. -0002A) and test HS-77-2 was successfully completed.

Under development test line item HS-19-4 (run in conjunction with HS-31), vibration and acoustic tests per DAC 56620C, were run on the

Table 2.2.1.3-2

TACS NITROGEN GAS STORAGE SPHERES
PRODUCTION ACCEPTANCE TEST

Line Item	Test Title	Test Requirements	Results
1B80321-PATP1	Production Acceptance Test (PAT)	To verify that the TACS gas storage spheres meet design requirements for leakage and proof pressure as follows: A. Proof test (warm) Stabilize the sphere assembly at 150° $\pm$ 10°F (65.5° $\pm$ 12.2°C). Apply demineralized water to a pressure of 6000 $\pm$ 100 psig ($413.6 \times 10^5 \pm 6.89 \times 10^5$ N/m^2) at the inlet port. Maintain pressure for two minutes. B. Proof test (cold) Stabilize the sphere assembly at -60° $\pm$ 10°F (-51.1° $\pm$ 12.2°C). Apply nitrogen gas to a pressure of 6,000 $\pm$ 100 psig ($413.6 \times 10^5 \pm 6.89 \times 10^5$ N/m^2) at the inlet port. Maintain pressure for two minutes. C. Leak test Pressurize with nitrogen gas to 3100 $\pm$ 100 psig ($213.7 \times 10^5 \pm 6.89 \times 10^5$ N/m^2) at 70° $\pm$ 25°F (21.1° $\pm$ 3.9°C). Maintain pressure for two minutes. There shall be no external leakage as evidenced by bubbles with the sphere submerged in demineralized water. (Ref. 1B80321)	The tests were completed and all leakage and proof pressure requirements were met.

radiator control panel with all of the radiator components mounted on it. The test was completed satisfactorily.

The radiator equipment subsequently was redesigned with resultant heavier loads on the panel. The panel was also redesigned to accommodate the new loads imposed by the heavier equipment. The new panel (1B93135) and its installation (1B93134) with the new components mounted on it was then tested under line item HS-77-2.

Several development tests were conducted to determine the most suitable means to join the fabric on the plume impingement curtain. Joint configurations tested ranged from a single row of stitches, bonding only, to Velcro joints. The highest tension capability [170 pounds per inch ($11.72 \times 10^5 N/m^2$)] was obtained from a bonded joint of one inch overlap with a central single row of stitches. This joint was subsequently used in the manufacture of impingement curtains.

The plume impingement shield, in conjunction with the radiator, underwent development testing per line items HS-31 and HS-77. The purpose of these tests was to expose the specimen to simulated liftoff and boost vibration and acoustic environments, as specified in DAC-56620C, and note the effects thereof. Pre- and post-dynamic tests included proof, leakage, and functional tests. Development testing per HS-31 also included individual part testing, such as the separation spring and the actuator.

Test requirements, conditions, and results were documented in MDAC Technical Memoranda R-7060 and R-7068 for HS-31 and HS-77, respectively. No failures or deviations related to the plume impingement shield occurred during testing.

D. Mission Results

The aft structure with its mounting provisions for two of the most important system hardware components (TACS bottles and refrigeration radiator) on OWS for Skylab mission success, performed satisfactorily from launch through the entire mission. The TACS system functioned well after orbital insertion indicating that the bottles and manifolding attachments performed to their design expectation through the launch and boost phases where highest load conditions were experienced. The same holds true for the radiator with respect to function of the refrigeration system.

The plume impingement curtain functioned properly during separation as indicated by the consistent temperature curves of the components during and immediately after separation. The radiator impingement shield met all its objectives with shield separation occurring at the nominal time. Separation was initially verified by using radiator temperature profile predictions and final verification was by photographs taken by SL-2 crew.

E. Conclusions and Recommendations

A great deal of confidence, time, and cost saving was realized by using and modifying an existing S-IVB flight proven structural assembly for the aft structure on the OWS even though loading conditions were quite different. This minimized both analysis and testing. The volume available for equipment mounting on the aft structure was restricted by the separation clearance of the aft interstage, however adequate space was available for the OWS system component installations. On new space vehicle design, requirements for functional systems would undoubtedly be integrated into the structural system as the design configuration is developed. Some of the attachment methods such as that used for the TACS bottles would be considered worthy of study to obtain alternate (new) designs.

F. Development History

The aft support structure, a modified SIVB thrust structure, was developed for use on OWS after the change from wet to dry workshop. Wet workshop required this structure for mounting the J2 engine and to provide thrust load transfer to the SIVB shell.

Early dry workshop studies looked at methods of mounting OWS systems (TACS bottles and refrigeration radiator) external of the tank. Although the thrust structure was not the only structure considered for mounting systems, it appeared to be the most economical. It had been used to mount helium bottles for SIVB and the same size nitrogen bottle was required for TACS on OWS. The number of nitrogen bottles was greater and the increased weight and distribution required reinforcement of the structure and requalification for increased vibration loads. This location for the bottles required design and development of a meteoroid shield for a probability of no impact.

Different locations were also considered for the refrigeration radiator. With slight modification to the thrust structure casting, the radiator could be easily mounted with the radiator surface properly oriented with respect to the orbital attitude. The shape and size of the radiator was designed to account for separation clearance of the aft interstage.

Installation of other components of both the TACS and refrigeration systems were adapted for aft structure mounting.

2.2.1.4 Aft Interstage Structure

The OWS aft interstage was the S-IVB/Saturn V 514 aft interstage structure with a modified thermal insulation pattern and material.

A. Design Requirements

1/ Withstand vehicle body bending, shear, and axial loads and transfer these loads between the OWS aft skirt structure and the S-II stage.

2/ Static loads shall include a yield factor of safety of 1.10 and an ultimate factor of safety of 1.25. Random vibration loads shall include a yield factor of safety of 1.10 and an ultimate factor of safety of 1.40. Cyclic loads used in fatigue analysis shall include a factor of safety of 1.0 on 2 sigma dynamic loads.

3/ Withstand the vibration and acoustic environments.

4/ Withstand the aerodynamic pressures and heating.

5/ External coatings, paint and thermal insulation to satisfy the requirements of MSFC Specification 50M02442 for contamination due to outgassing.

6/ Provide structural support for electrical systems equipment, removable access platforms, the S-II retrorockets and the OWS aft umbilical GSE carrier.

7/ Access panel to be provided for entering the internal areas of the OWS aft skirt and the interstage.

8/ Provide structural support for inter-stage electrical systems quick disconnects at interface of OWS aft skirt and aft interstage.

9/ Provide vent openings to control differential pressure loads on the aft skirt, interstage and S-II forward skirt during ascent.

B. System Description

The aft interstage, as defined by Drawing 1B77176 (Figure 2.2.1.4-1), was a SIVB/V frustum-shaped semi-monocoque structure. The height of the structure was 227.5 in. (577.9 cm). The nominal diameter of the forward end was 260 in. (660.4 cm) and the aft end was 396.750 in. (1007.7 cm). The structure consisted of 144 externally mounted hat-section stringers, 040 in. (1.016 mm) thick sheet metal skin, 9 internally mounted ring frames, and intercostals in the forward and aft bays with a normal spacing of every third stringer. The major portion of the structure was fabricated from 7075-T6 aluminum alloy sheet metal and extrusions.

The basic cross section of the ring frames was two extruded "T" shaped (angle shaped on forward and aft frames) caps, a .025 in. (.635 mm) nominal thick sheet metal web and extruded angle stiffeners. The nominal depth of the seven intermediate frames was 16 in. (40.6 cm). The forward and aft frames were approximately 12 in. (30.5 cm) deep.

The intercostals in the forward 19 in. (48.3 cm) bay were .040 in. (1.016 mm) thick sheet metal cut to a "X" shaped planform, an extruded stringer attach angle and an extruded "T" shaped inboard cap. The aft 19 in. (48.3 cm) bay intercostals were hydroformed .032 in. (.813 mm) thick sheet metal, an extruded stringer attach angle and an extruded "T" shaped inboard cap (Figure 2.2.1.4-2).

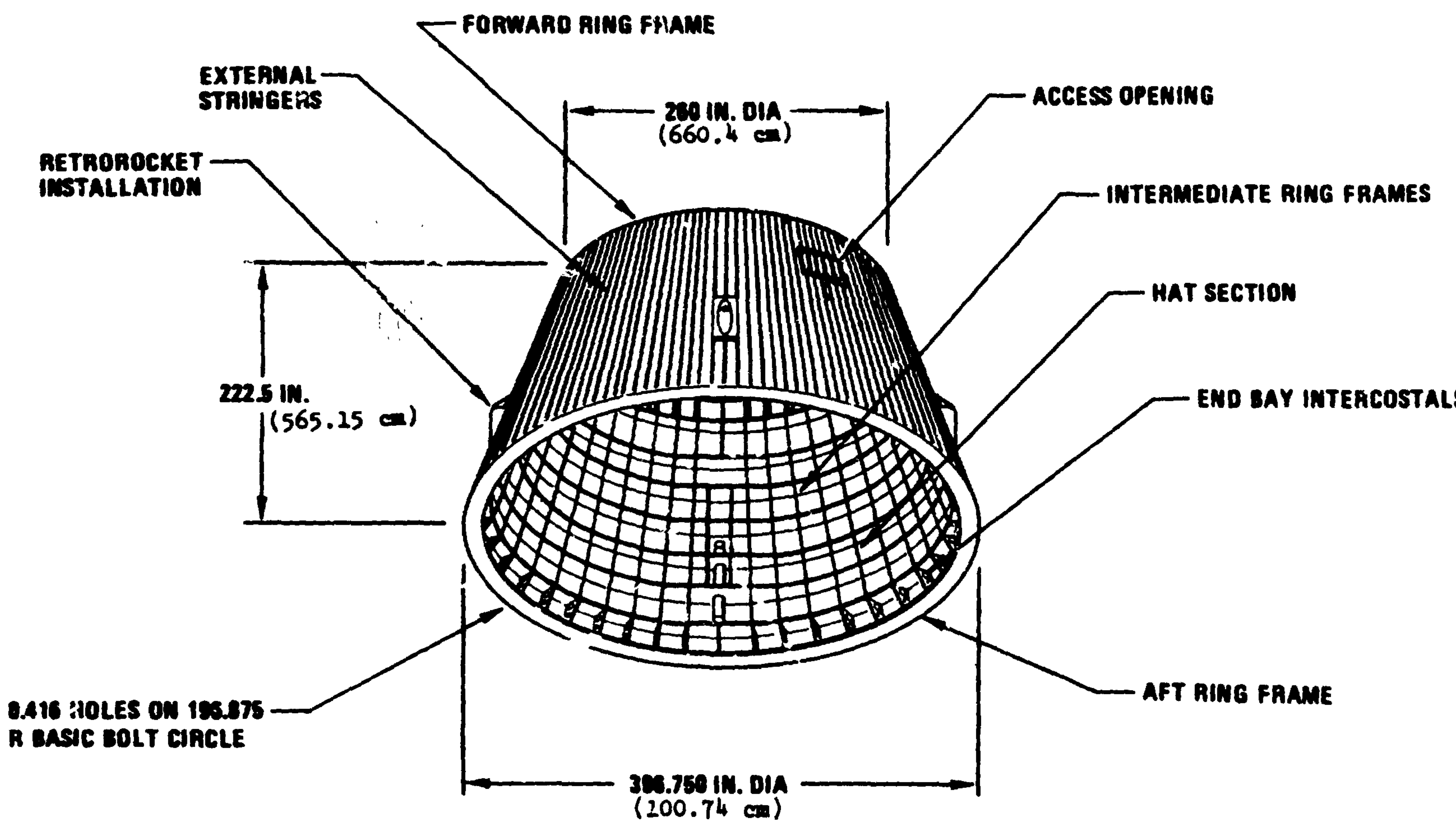


Figure 2.2.1.4-1. Aft Interstage – Basic Structure

Figure 2.2.1.4-1

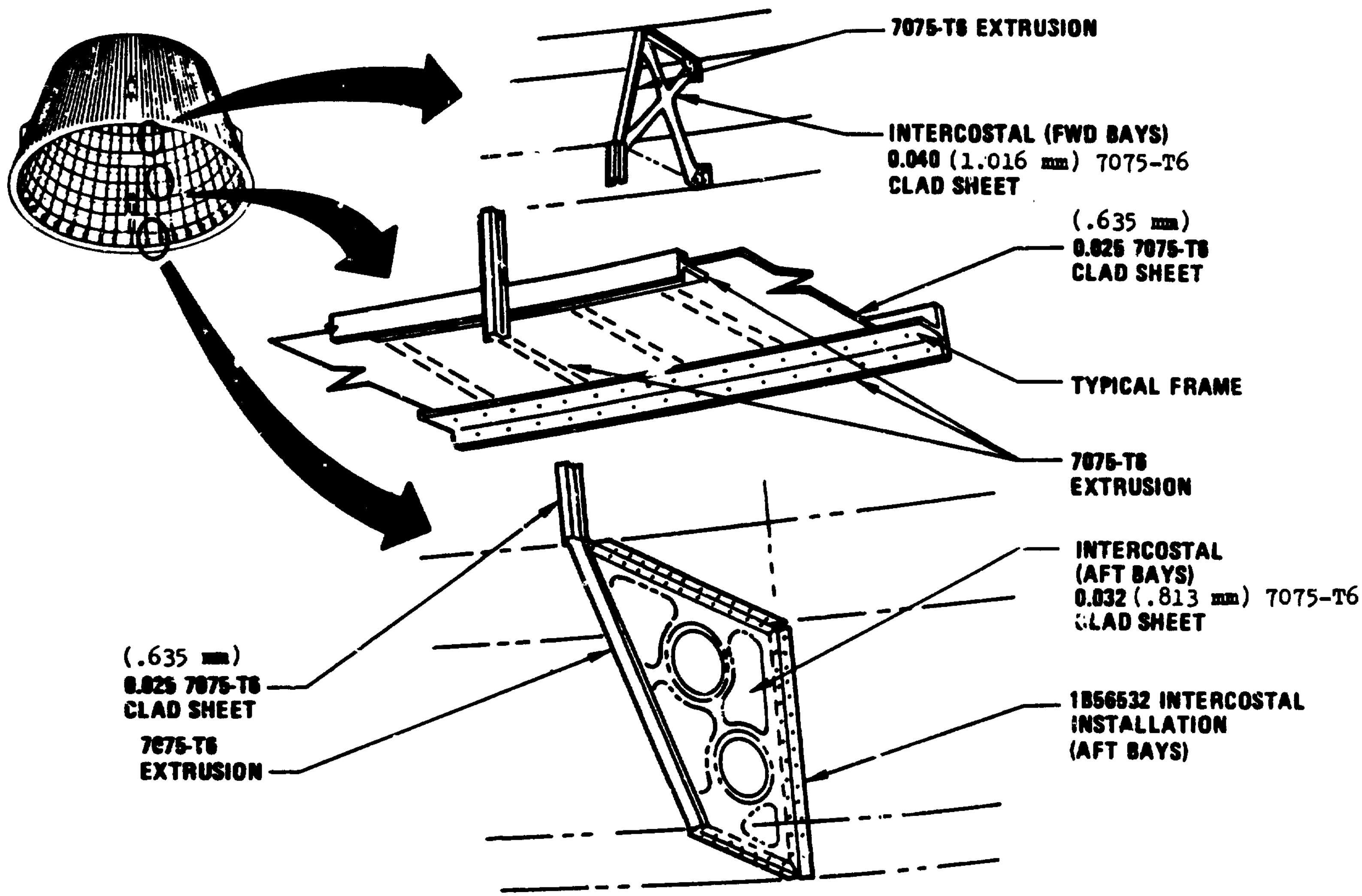


Figure 2.2.1.4-2. Ring Frames and Intercostals

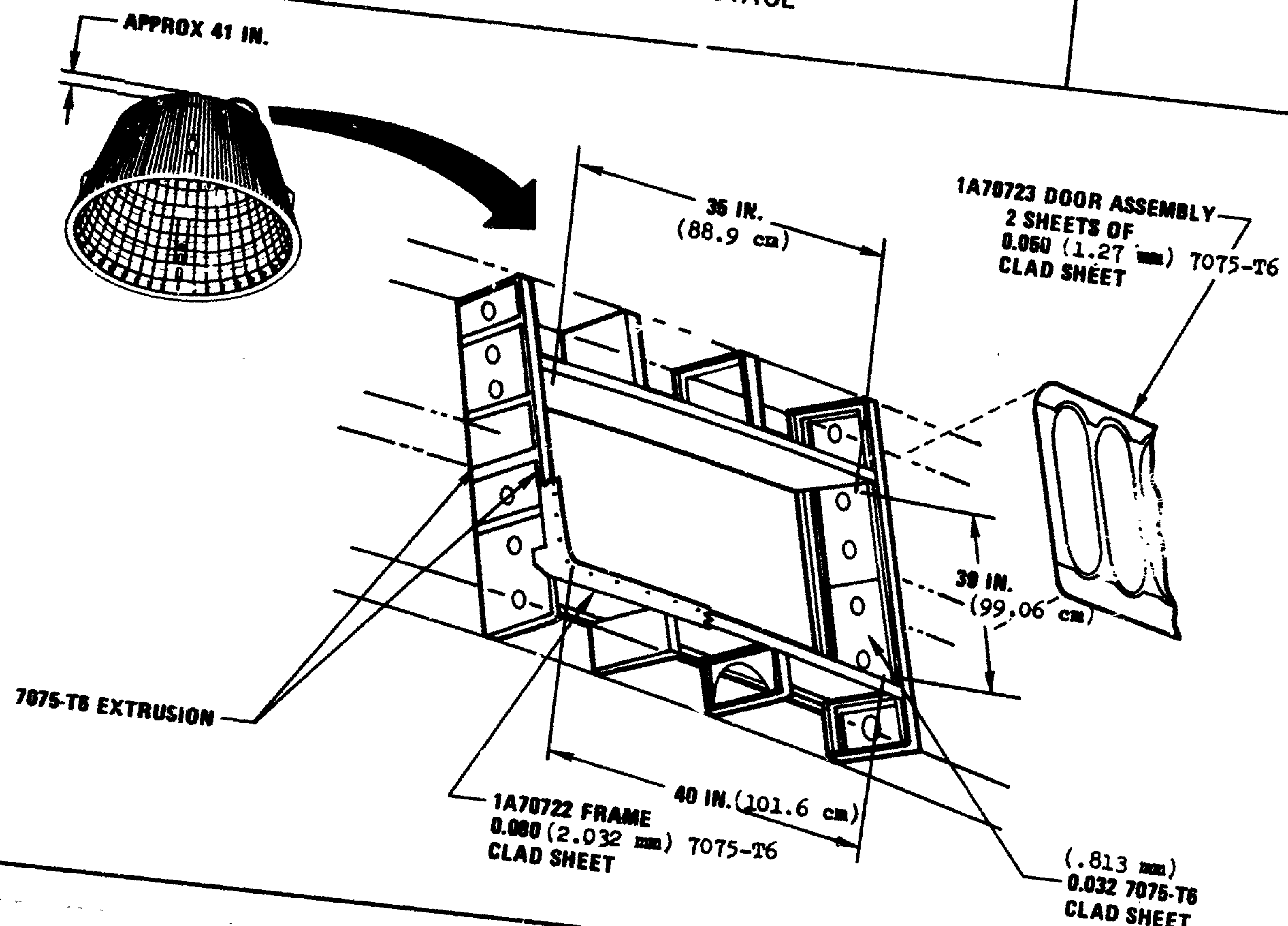
Figure 2.2.1.4-2

An access opening, 39 in. (99 cm) long, approximately 35 in. (90 cm) wide at the forward end and 40 in. (101.6 cm) wide at the aft end, located approximately 41 in. (104.14 cm) from the forward end of the aft interstage, provided access into the OWS aft skirt and interstage areas. A beaded panel, constructed of two sheets of .050 in. (1.27 mm) thick metal riveted back-to-back, was installed in the opening with Camloc high shear type fasteners prior to flight (Figure 2.2.1.4-3).

Vent openings, provided to control venting of the OWS aft skirt/aft interstage internal area to control the differential pressure loads on the aft skirt, interstage and S-II forward skirt during ascent and to provide an outlet for the purge gas during ground operations, were located approximately 84 in. (213.36 cm) from the aft end of the interstage. Two sets of 8 holes each were located (180°) apart. Each of the vent holes had a length-to-width ratio of 4:1 with the length parallel to a stringer. The area of each opening was 10 in.^{2} (64.5 cm^2). Rain shield assemblies were installed inside the aft interstage to prevent rain water from entering the interstage. A two-by-two .063 in. (1.6 mm) diameter wire mesh screen was installed on the rain shield assemblies to prevent birds and other solid objects from entering the interstage (Figure 2.2.1.4-4).

Support structure was provided for the four S-II retrorocket installations (Figure 2.2.1.4-5). Two intercostals, constructed of an .040 in. (1.016 mm) thick clad aluminum alloy sheet riveted to extruded caps and a formed clad aluminum alloy sheet angles with a

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
AFT INTERSTAGE



SKYLAB - OPBITAL WORKSHOP
BASIC SHELL STRUCTURE
AFT INTERSTAGE-VENT OPENINGS

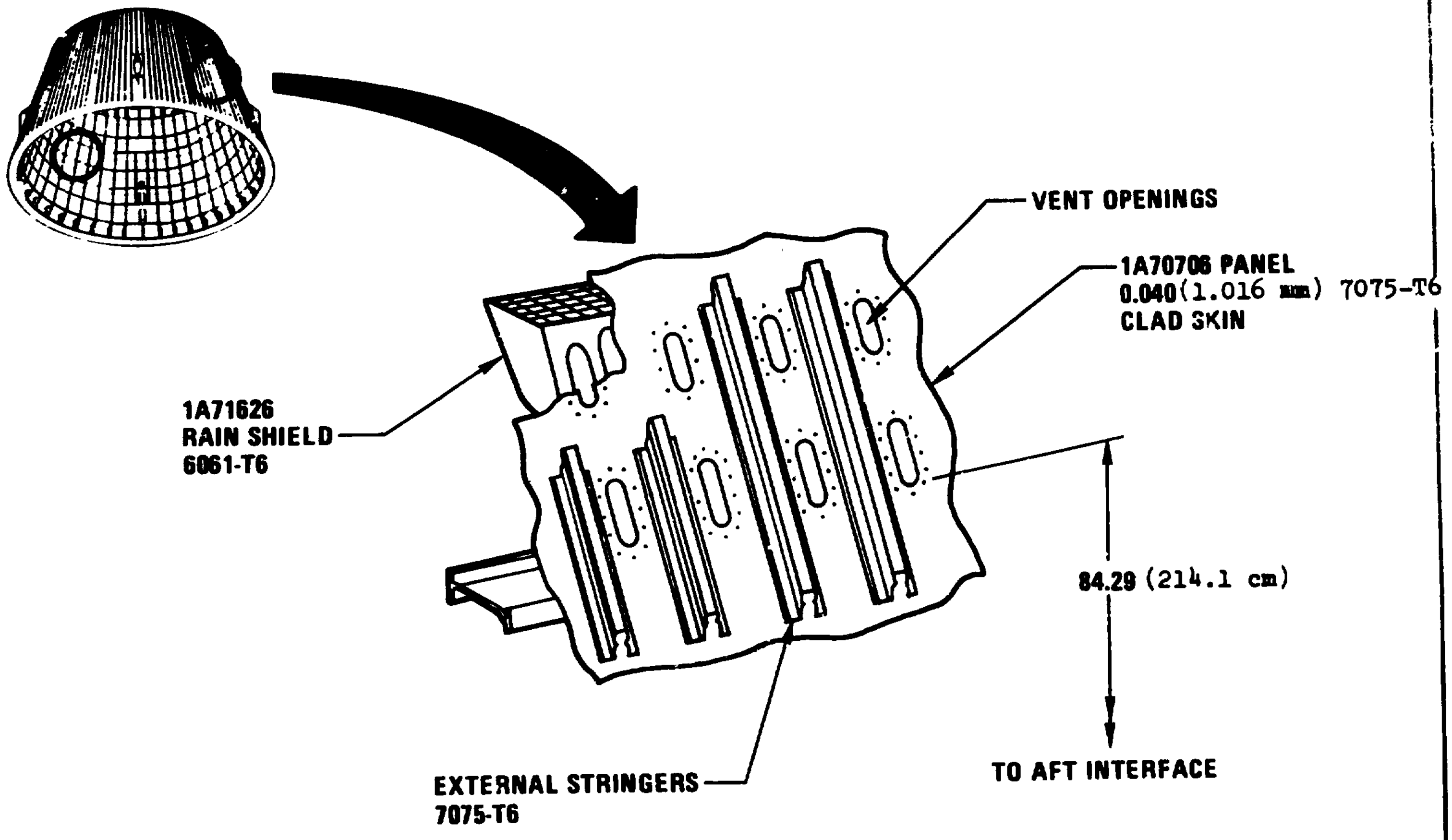


Figure 2.2.1.4-4

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
AFT INTERSTAGE

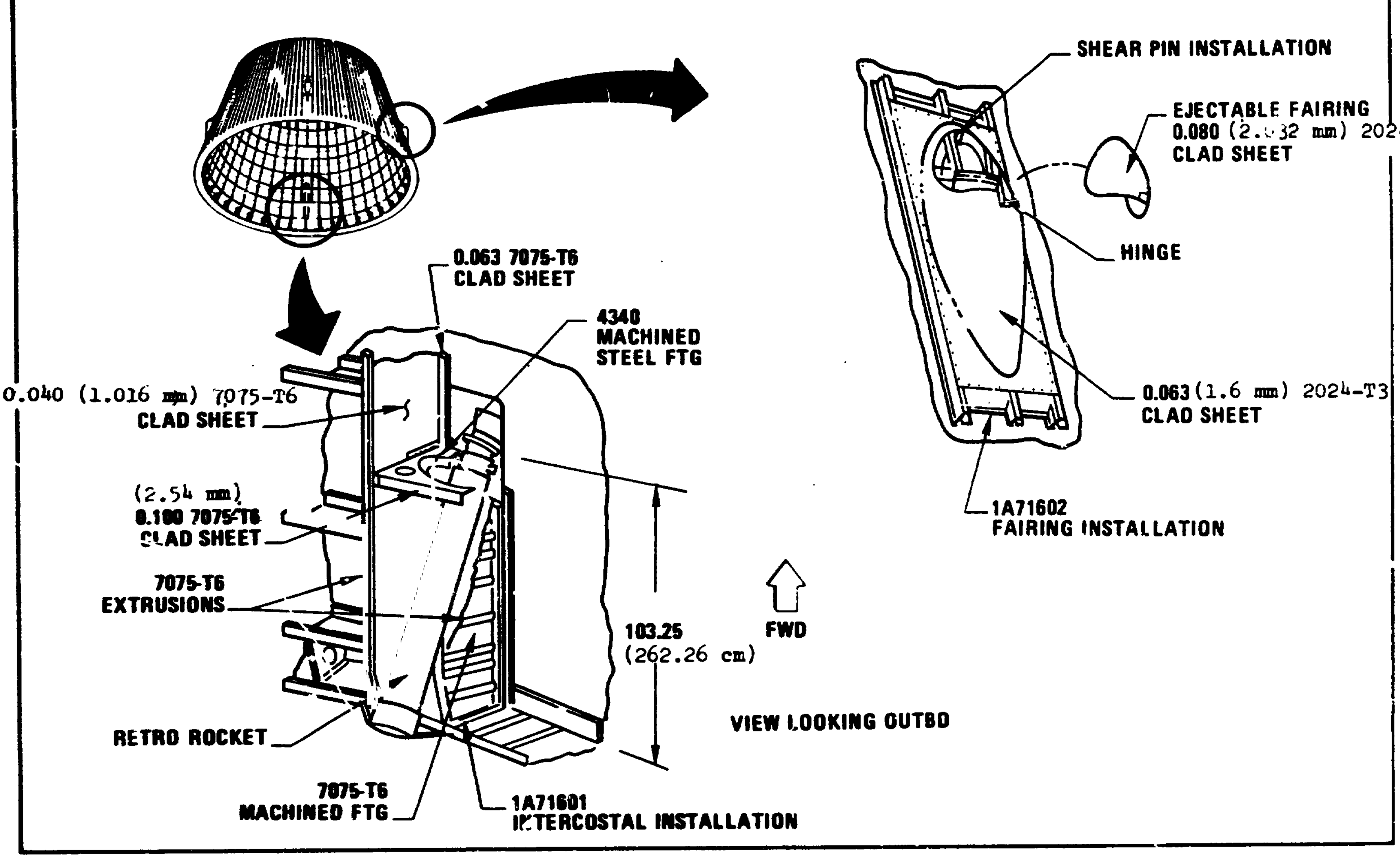


Figure 2.2.1.4-5

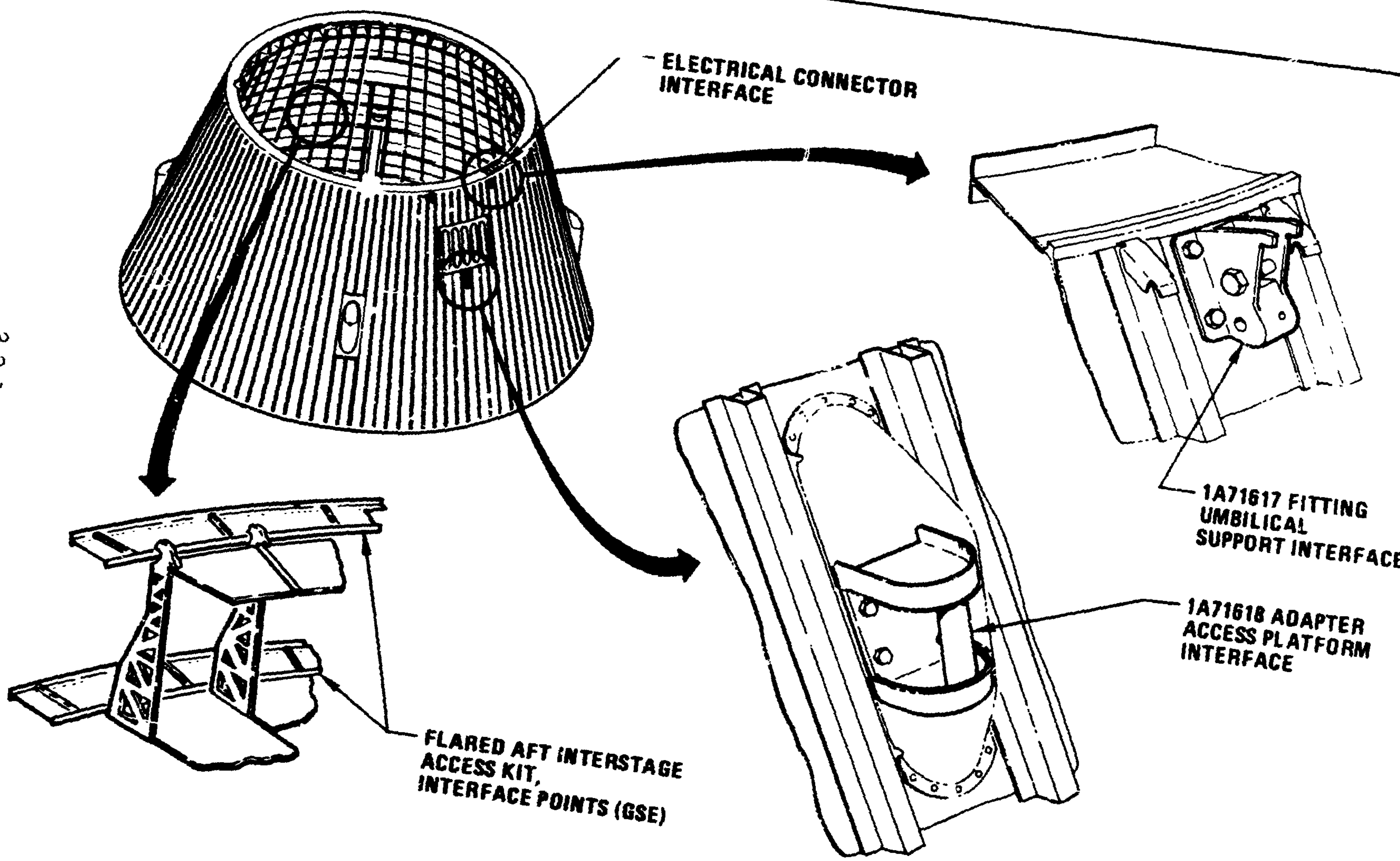
formed clad aluminum alloy sheet support between the intercostals, were installed at each retrorocket location. Each retrorocket was supported at the forward end with two trunion type 4340 steel fittings and the aft end with two tubular supports attached to the interstage aft frame inboard cap through two 2024-T351 aluminum alloy fittings.

A fairing with an ejectable nose section was installed at each retrorocket location. The fixed portion of the fairing, made of .063 in. (1.6 mm) thick 2024-T3 clad aluminum alloy sheet, was riveted to the exterior of the aft interstage. The ejectable nose section, made of .080 in. (2.03 mm) thick 2024-T42 clad aluminum alloy sheet, was attached to the fixed fairing with a hinge and shear pin (rivet) installation. Ejection was accomplished by firing the retrorocket causing pressure to build up under the fairing until a sufficient load was developed to rupture the shear pin. The release point of the hinge was positioned so that the path of the ejected fairing would not strike the OWS or the S-II Stage.

The ground support equipment (GSE) interfaces with the aft interstage were as shown in Figure 2.2.1.4-6.

The structural tie between the aft interstage and the OWS aft skirt was through 288 equally spaced 5/16 in. (.794 cm) diameter bolts. Flight separation at this joint was accomplished by fracturing a tension strap on the OWS aft skirt frame by means of a mild detonating fuse (See paragraph 2.2.1.1.B). Mating electrical connectors mounted to the OWS aft skirt aft frame and the aft interstage forward frame disconnect at separation. The structural

AFT INTERSTAGE/GSE INTERFACES



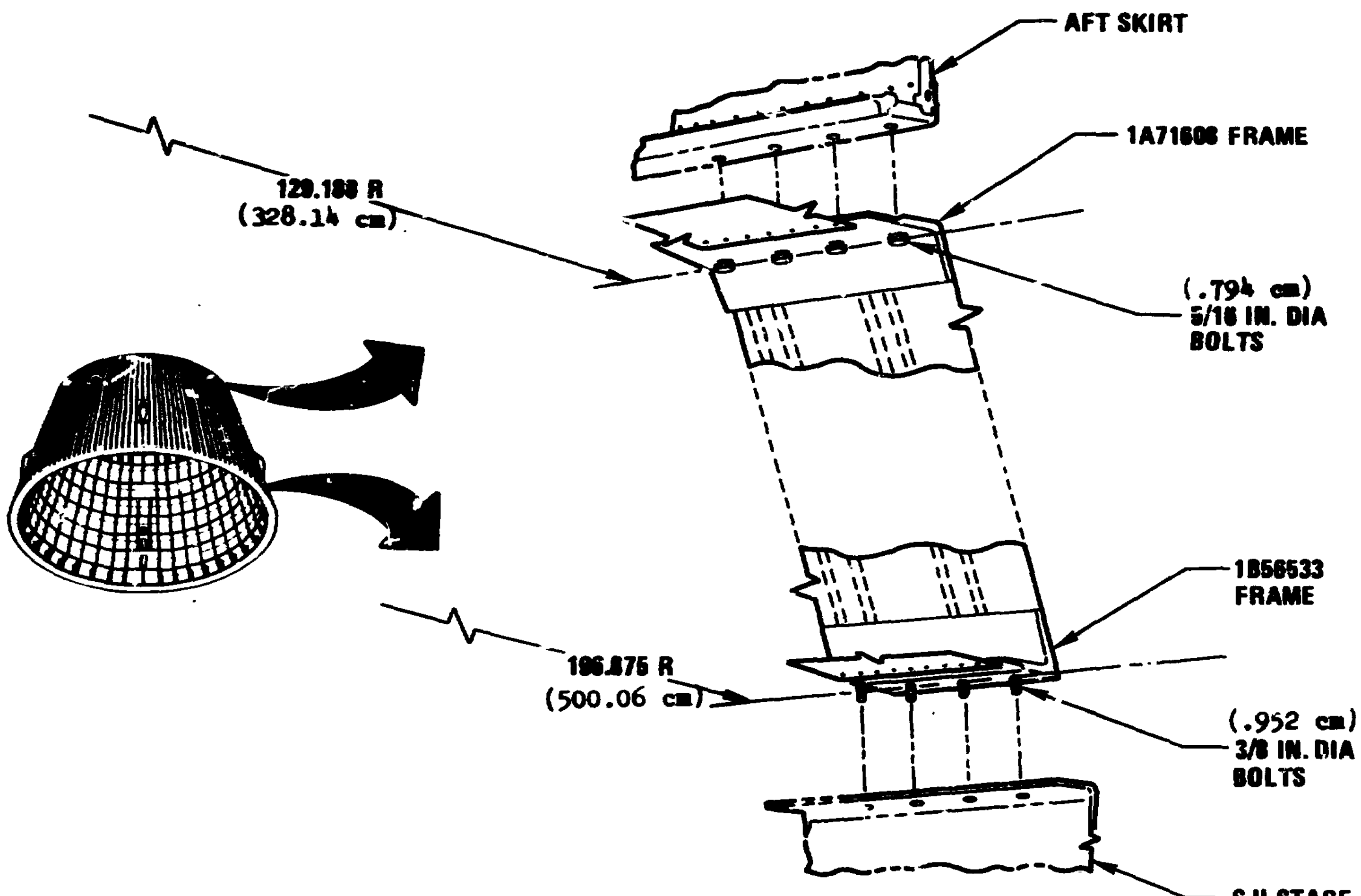
tie between the aft interstage and the S-II Stage forward skirt was through 288 equally spaced 3/8 in. (.952 cm) diameter bolts (Figure 2.2.1.4-7). Mating was facilitated by using three (3) equally spaced removable alignment fittings at each of the forward and aft interfaces.

The one difference in the OWS aft interstage and the S-IVB/V aft interstage was the material and pattern of the thermal insulation coating applied to external areas of the interstage. Non-ablative silicone insulation was used on the OWS aft interstage in lieu of the ablative Korotherm insulation used on S-IVB/V interstages to reduce contamination. The silicone insulation was applied only to highly stressed areas of the interstage that were subject to high heat fluxes due to the OWS aft skirt protuberances and the aft interstage retrorocket fairings.

C. Testing

The structure of the aft interstage was tested as a S-IVB/V structure for critical design load conditions (Reference MDAC Technical Memoranda A3-860-KBBC-TM-97, dated January, 1967). The design loads for the OWS aft interstage were less severe than for the S-IVB/V interstage. Therefore, the structural integrity of the OWS interstage was established primarily by comparing the OWS loads and environment to that of the S-IVB/V and the structural capability of the S-IVB/V aft interstage which was determined from the noted qualification test and engineering development tests. See Table 2.2.1.4-1 for a comparison of selected critical design load conditions for the aft interstage.

SKYLAB - ORBITAL WORKSHOP
BASIC SHELL STRUCTURE
AFT INTERSTAGE



AFT INTERSTAGE - DESIGN ULTIMATE LOADS

SA STA.	MAX. N_C LB/IN (kg/cm) S-IVB/V	MAX. N_C LB/IN (kg/cm) OWS	CECO (MAX. ACCEL) F_C LB/IN (kg/cm) S-IVB/V	CECO (MAX. ACCEL) F_C LB/IN (kg/cm) OWS	TENSION LOADS N_T LB/IN (kg/cm) S-IVB/V (DYNAMIC TENSION)	TENSION LOADS N_T LB/IN (kg/cm) OWS (MAX.)	TENSION LOADS N_T LB/IN (kg/cm) TENSION CAPABILITY
2746.5	(694.6) 3889	(436.0) 2441	(543.3) 3042	(263.3) 1474	(190.2) 1065	(202.9) 1136	(321.8) 1802
2519	(442.8) 2479	(269.5) 1509	(356.8) 1998	(179.4) 1005	(130.6) 731	(124.8) 693	(303.4) 1699

N_C = SHELL LOAD (COMPRESSION)

N_T = SHELL LOAD (TENSION)

(TABLE 2.2.1.4-1)

The aft interstage was subjected to higher temperatures on the OWS flight at S-IC center engine cutoff (CECO) than had been experienced on the S-IVB/V stages. However, the strength reduction due to the elevated temperatures was offset by the lower OWS flight loads to provide a higher margin of safety.

D. Mission Results

The aft interstage structure performed satisfactorily throughout the launch and ascent, and separated without incident from the OWS upon S-II staging command.

E. Conclusions and Recommendations

The aft interstage was a proven structural system with twelve (12) successful Apollo missions which was readily adaptable to the OWS mission application. The structure performed the same function on the OWS Program that it had on the S-IVB Program.

F. Development History

This hardware is SIVB Sat V structure which did not require any major changes or tests to develop it for OWS application.

2.2.1.5 **Mass Properties** - This section highlights the weight history of the Skylab Program from mid 1968 through the launch of SL-1 in May 1973.

In the following paragraphs, the term "Wet AAP" is used to identify the Apollo Applications Program which centered around the Wet Workshop; AAP-2 was the launch of the Wet Workshop; AAP-1, AAP-3a, and AAP-3 were CSM launches; and AAP-4 was the launch of the Lunar Ascent Module (LM-A) and Apollo Telescope Mount (ATM). A weight history is provided for the Wet AAP from mid 1968 through mid 1969.

The term "Dry SL" refers to the current Skylab Program. SL-1 was the Dry Workshop launch, while SL-2, SL-3 and SL-4 are CSM launches. A weight history is provided from mid 1969 on, after the program was changed to reflect a Dry Workshop launched by a two-stage Saturn V.

The final paragraph of this section details the weight history of the MDAC developed Orbital Workshop Module.

A. Wet AAP Weight History - The Wet AAP program was severely weight limited. Weight margins for the AAP-2 launch, based on data from MSFC and NASA/OMSF are shown below. The payload is defined as the injection weight of all systems which are added to the Propulsive Saturn IB, and consist of modifications to the S-IVB, the Instrument Unit (IU), the Airlock Module (AM), the Multiple Docking Adapter (MDA), and the non-jettisonable portion of the payload shroud (FAS):

Date	Wet AAP-2 Control Wt. at Injection Pounds (Kilograms)	Wet AAP-2 Payload Wt. at Injection Pounds (Kilograms)	Payload Margin Pounds (Kilograms)
6/15/68	30,300 (13,744)	30,491 (13,830)	-191 (-87)
	Potential Chgs.	+1,758 (+797)	-1,949 (-884)

2/69	33,925	31,740	+2,185
	(15,388)	(14,397)	(+991)
	Potential Chgs.	+1,220	+965
		(+553)	(+438)
4/15/69	33,925	31,524	+2,401
	(15,388)	(14,299)	(+1,089)
	Potential Chgs.	1,536	+865
		(697)	(+392)

The following constraints were placed on the Wet AAP program due to weight limitations:

1/ The cluster orbit, which was desired at 230 nautical miles (NM) (426 kilometers) was reduced to 210 NM (389 kilometers). Further, achieving this orbit required that the AAP-1 and AAP-2 be placed in eccentric orbit, with the entire cluster orbit to be circularized by the CSM.

2/ The ATM could not be placed in orbit by the AAP-2. The ATM was launched on AAP-4 and was only used for a portion of the third manned visit.

3/ The mode of launching the ATM required accomplishment of an unmanned rendezvous and the use of a modified lunar ascent module.

4/ Major modifications were required to the command and service module to provide extended electrical and water generation systems, and possibly a portion of the ECS CO_2 and GN_2.

5/ Development of a hypergolic Workshop Attitude Control System (WACS) was required. In addition to the usual problems of development, the WACS had the potential of creating a severe contamination problem.

6/ A nose cone had to be developed which could be jettisoned during powered flight. AAP-2 weight capability limitations precluded carrying the nose cone to orbit.

7/ Deployment of the cluster required extensive "drag-in" of equipment from the MDA, significantly complicating the activation of the cluster. Materials compatibility with liquid hydrogen severely limited the amount of preinstalled equipment.

8/ Quantities of experiments and life support systems were severely limited, not only by weight, but also by volume, as most of the Workshop equipment had to be stored in the MDA.

B. Dry Skylab Weight History - Initiation of the Dry Skylab Program provided immediate relief from the weight and volume constraints of the Wet AAP and permitted useful growth of life support and experiment systems.

The payload injection weights from MSFC, NASA/OMSF, and MDAC data for the Dry SL-1 are shown below. The payload is defined as the injection weight of everything forward of the S-II/S-IVB interstage and includes the Dry Orbital Workshop, IU, AM, MDA, ATM, and the complete payload shroud:

Date	Dry SL-1 Payload Capability at Injection Pounds (Kilograms)	Dry SL-1 Payload Wt. at Injection Pounds (Kilograms)	Payload Margin Pounds (Kilograms)
1/70	210,000 (95,254)	155,976 (70,750)	+54,024 (+24,505)
3/70	210,000 (95,254)	161,493 (73,252)	+48,507 (+22,002)
8/27/70	210,000 (95,254)	175,822 (79,752)	+34,175 (+15,502)
10/27/70	210,000 (95,254)	181,264 (82,220)	+28,736 (+13,034)
4/71	210,000 (95,254)	190,069 (86,214)	+19,931 (9,041)
2/73	210,000 (95,254)	195,771 (88,800)	+14,229 (+6,454)
5/73	210,000 (95,254)	197,015 (89,364)	+12,985 (+5,890)

Mission and hardware changes arising from the increased payload capability are summarized as follows:

1/ A total of four launches were required for a three visit mission, instead of the five launches required in the Wet AAP Program.

2/ The orbital assembly could be placed directly into the desired 234NM (433 Kilometer), 50° inclination orbit with no assist from a CSM.

3/ The ATM could be placed in orbit with the SL-1, thus eliminating one Saturn 1B launch, eliminating the Modified Lunar Module Ascent Stage, and eliminating the need for the unmanned rendezvous. Additionally, the ATM could then be utilized for three missions instead of only one.

4/ The need to make major modifications to the CSM was eliminated as water, electrical power, and ECS systems could be preinstalled in the SL-1 orbital assembly.

5/ The WACS was replaced by a cold gas system. This was possible because the ATM Control Moment Gyro's (CMG) became the prime method of control, and the extended passivation sequence was no longer required. This eliminated development of the WACS and the potential contamination problem.

6/ The nose fairing could be carried to orbit, thus eliminating the boost jettison requirement. In addition, due to the lack of weight constraints, the nose fairing could be made a low-cost, high safety factor assembly.

7/ "Drag-in" deployment was virtually eliminated, as most equipment could be pre-installed.

8/ Experiments were greatly expanded, and increased life support systems were provided, allowing extensive "open-ended" capability to the Skylab.

An unforseen use of a major portion of the final SL-1 payload margin of 12,985 pounds (5,890 kilograms) occurred during launch of SL-1. Separation of the Meteoroid Shield damaged the S-IC/S-II Interstage Separation System. The S-IC/S-II large interstage failed to separate during boost causing an additional 10,992 pounds (4,986 kilograms) to be carried to orbit. The unused SL-1 payload margin sufficiently covered this anomaly.

C. Orbital Workshop Module Weight History - The following paragraphs highlight the weight history of the OWS Module of the Skylab Program.

1/ Final OWS-1 Weight Status - The final reported weight of the OWS Module was 544 pounds (247 kilograms) below specification weight, as follows:

Item	Weight Pounds (Kilograms)
Basic CEI Specification Weight (Per Supplemental Agreement 94)	50,665 (22,981)
Authorized changes to the CEI Specification Weight	28,362 (12,865)
Final Specification Weight	79,027 (35,846)
Final Reported Weight	78,483 (35,599)
Underweight	544 (247)

Included in the OWS-1 weight is 32,655 pounds (14,812 kilograms) of Government Furnished Parts (GFP). This weight is included and identical in value in both the CEI specification weight and final reported weight. The underweight status, therefore, reflects MDAC performance related to Contractor Furnished Equipment (CFE).

2/ OWS-1 Weight Control - Although the two-stage Saturn V booster for SL-1 launch provided an extremely large initial payload capability margin, it was necessary to implement weight control procedures for the Skylab. This was necessary for the following reasons:

a. The committed booster capability was weight limited by booster guidance system networks. These circuits required a predicted launch weight in their design and a prediction tolerance. If the prediction was exceeded, a circuit redesign was required.

b. The Control Moment Gyro Attitude Control System was sized for the predicted inertia of the cluster assuming one CMG inoperative. The inertia was controlled by assigning maximum weight limits to each module of the cluster.

c. The availability of the dry OWS volume and the desirability of maximizing experiment and crew systems assured that a large weight growth would occur.

The basis of OWS module weight control was the establishment of a module specification weight. The OWS specification weight imposed upon MDAC was less than the maximum module weight limit but greater than the estimated weight at that time. Initial negotiated OWS specification weight margins were adequate to insure that no costly and unnecessary weight reduction or weight optimization efforts were undertaken.

The joint efforts of NASA/MSFC and MDAC-W established a highly effective specification weight management procedure. Major features of the procedure were that both MDAC-W and NASA/MSFC program management were fully cognizant of weight impact prior to approval of design changes, and the rapid update of the CEI Specification Document.

Authorized modifications to the OWS-1 specification weights are substantiated in the specification weight model by 400 line items of Engineering Change Proposal (ECP) data.

3/ OWS-1 Weight Growth - The weight of the OWS-1 increased from 47,570 pounds (21,577 kilograms) (MDC G0139, dated November 1969) to 78,483 pounds (35,599 kilograms) (MDC G2645M, dated March 1973) during the design and development phase. Total weight growth was 30,913 pounds (14,022 kilograms).

Of the 30,913 pounds (14,022 kilograms) growth 28,362 pounds (12,865 kilograms) was out-of-scope (authorized changes and GFP) and 2,551 pounds (1,157 kilograms) was in-scope weight growth. These changes were explained by 1,710 formally published reasons for change reported in the Skylab A Orbital Workshop Mass Properties Status Reports.

Table 2.2.1.5-1 provides a breakdown of the OWS-1 weight growth, organized in the functional grouping used in the OWS Status Reports. Note that increases in the experiments, crew systems, and cabin atmosphere system groupings account for over 60-percent (18,582 pounds or 8,429 kilograms) of the total OWS-1 weight growth.

The final Skylab design, at 78,483 pounds (35,599 kilograms) provided a highly versatile spacecraft which maximized usable experiments and provided for significant open-ended life support capability.

TABLE 2.2.1.5-1

SL-1 ORBITAL WORKSHOP MODULE

WEIGHT GROWTH

Description	Nov. 1969 MDC G0139 Weight Pounds (Kilograms)	March 1973 MDC G2645M Weight Pounds (Kilograms)	Weight Change Pounds (Kilograms)
Experiments	1,962 (890)	7,207 (3,269)	+5,245 (+2,379)
Exterior Structure	13,243 (6,007)	13,994 (6,348)	+751 (+341)
Interior Structure	3,780 (1,715)	8,491 (3,851)	+4,711 (+2,136)
Propulsion System	4,251 (1,928)	6,651 (3,017)	+2,400 (+1,089)
Electrical Power Supply System	4,936 (2,239)	8,051 (3,652)	+3,115 (+1,413)
Cabin Atmosphere System	510 (231)	2,101 (953)	+1,591 (+722)
Crew Systems	14,886 (6,752)	26,632 (12,080)	+11,746 (+5,328)
Electronics	1,400 (635)	2,131 (967)	+731 (+332)
Launch Control Subsystems	2,603 (1,181)	2,449 (1,111)	-154 (-70)
Mission Spares and Stowage	0 (0)	990 (449)	+990 (+449)
Actual Weight Adjustment	0 (0)	-214 (-97)	-214 (-97)
Total SL-1 OWS Module at Launch	47,570 (21,578)	78,483 (35,600)	30,913 (14,022)

D. Conclusions and Recommendations

1/ Mass properties limits (primarily moments of inertia) on the orbital cluster were controlled by assigning weight limits on the individual modules. This proved to be an effective method of overall mass properties control, and avoided the added complexity of tracking center of gravity and moment of inertia changes against individual ECP's.

2/ Mass properties data on Government Furnished Parts (GFP) were required by MDAC to fulfill the following needs:

a. Determine the mass properties of the overall Orbital Workshop Module.

b. Provide handling and transportation weights.

c. Design of major structures (such as floors and walls). This requires detailed local mass distributions to determine loads.

d. Design of mounting brackets and local stiffening. This requires individual "Package" weights plus C.G. and MOI data for both static and dynamic loads analyses.

e. Design of containers and cabinets.

It is recommended that methods of obtaining current, accurate, complete, consistant data be established for future programs. This could be accomplished by having one coordinator assigned this responsibility.

3/ An excellent degree of weight visibility was provided during the OWS program. This visibility resulted from the support of both NASA and MDAC Management in establishing an effective Weight Management Program, which included a negotitated specification weight, early visibility of design change weight impacts, and visibility of remaining module weight margin. It is recommended that specification weights and associated control procedures be implemented as early as possible on future programs of this type.

2.2.2 Meteoroid Shield System

2.2.2.1 Launch/Flight Configuration

A. Design Requirements for this configuration contained the following constraints and parameters:

1/ The shield shall meet the extension requirements of 65ICD9001 during prelaunch operations.

2/ The system shall be free of flutter or other divergent instabilities during flight.

3/ The system shall be designed to the environments produced by the trajectory defined in S&E-AERO-MMM-16-69.

4/ The system shall be designed to withstand the dynamic environment produced by launch and ascent.

5/ The system shall be designed for compatibility with the flight pressure in the habitation area of 22 psia to 26 psia (1.51×10^5 to 1.79×10^5 N/m^2).

6/ Static loads shall include a yield factor of safety of 1.10 and an ultimate factor of safety of 1.25. Random vibration loads shall include a yield factor of safety of 1.10 and an ultimate factor of safety of 1.40. Cyclic loads used in fatigue analysis shall include a factor of safety of 1.0 on 2 sigma dynamic loads.

7/ The system shall be designed for the worst case combinations of items 4, 5, and 6 superposed upon rig loads. Appropriate factors of safety defined in (6) shall be used.

8/ The shield system shall provide protection for the external thermal optical coatings on the OWS cylinder from detrimental ground environments.

9/ A fairing, designated the auxiliary tunnel, shall be an integral part of the system to provide ground and flight protection for electrical cables, tubing, and related equipment.

10/ Two openings shall be provided in the shield to allow on-orbit operation of the solar and anti-solar scientific airlocks. Another opening shall be provided in the shield to allow on-orbit use of the OWS viewing window located in the wardroom. This opening shall be covered during pre-launch, launch and ascent to provide protection for the window from ground and ascent environments. These openings shall meet the size and tolerances of the following ICD's.

13M12381 S019

13M12391 S020

13M12401 S063

13M12351 S149

13M13521 S183

13M13540 S190B

13M12301 T025

11/ Provisions shall be made for emergency ground entry into the OWS through the side hatch without de-rigging the shield.

12/ The system shall include provisions for ground checkout of the Metabolic Analyzer (M171) and the Lower Body Negative Pressure (M092) vacuum outlets and provide sealing for these outlets during ground and ascent conditions.

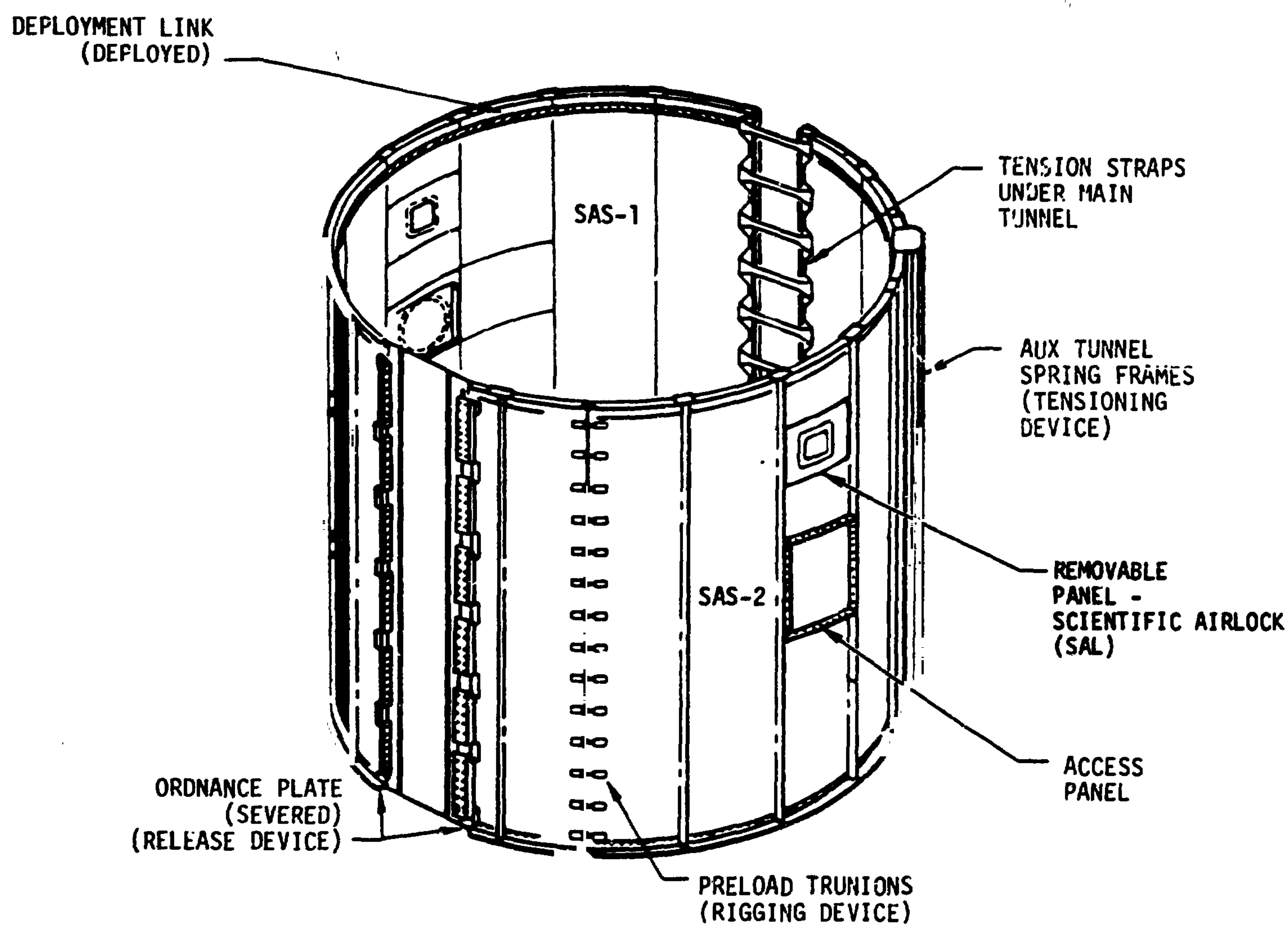
13/ The shield shall provide an interface with the Solar Array System which will allow adequate sealing for ground environmental protection (purging) of the Solar Array System as well as sealing to provide proper distribution and levels of delta pressure during ascent.

14/ Materials shall be compatible with MSFC-Specification-101A and also shall not "out-gas" or "off-gas" to prevent detrimental contamination.

B. System Description

1/ Hardware Configuration (Figure 2.2.2.1-1) - The habitation area meteoroid shield was designed to be wrapped tightly around the habitation area during launch and ascent of the OWS. On-orbit, the shield was to be released by one of two longitudinal contained expandable ordnance tubes, which fractures a tension tie assembly permitting the shield to be deployed approximately 5.00 in. (12.7 cm) radially from the 260 in. (660.4 cm) diameter tank.

The deployed configuration was to provide meteoroid protection for the habitation area. Deployment was to be achieved by energy stored in 16 torsion bars acting on link arms mounted through the tank-skirt flanges. The linkage mechanism was a series of four-bar linkages in which the shield was a flexible interconnecting bar and the tank flange was a rigid bar. The change in circumference required for deployment was to be accomplished by hinged fold-over panels located under the ordnance separation joint. The frames of the auxiliary tunnel were designed to act as hoop tension springs to assure intimate contact between the tank and meteoroid shield and also to accommodate growth of the tank due to changes of differential pressure and temperature.

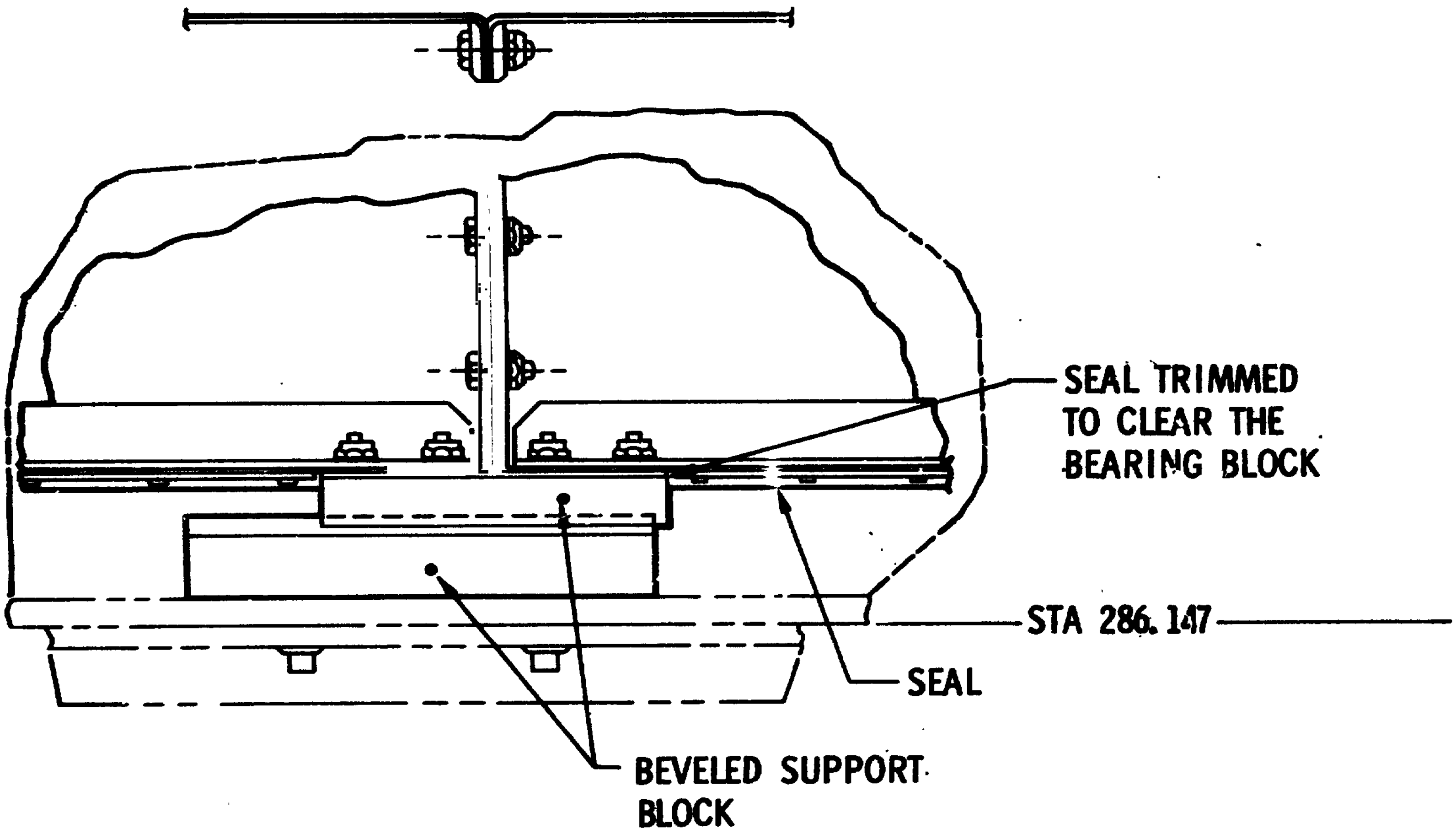


Meteoroid Shield

Figure 2.2.2.1-1

2/ Subsystem Configuration - The deployable meteoroid shield assembly was a series of 16 curved 135 in. (342 cm) radius, 0.025 in. (.635 cm) thick 2014-T6 aluminum panels bolted together through brake formed flanges at each edge to form a complete cover around the cylindrical section of the habitation area. The weight of the shield at launch was transmitted to the aft tank flange by beveled 6061-T651 aluminum support blocks located at the panel joints (Figure 2.2.2.1-2). The forward and aft ends of each panel had a 7075-T6 closing angle. Attached to the closing angles was a silicone rubber bulb seal which was adjusted after the shield was rigged to provide ground weather protection for the thermal coating on the exterior habitation area surface. Also, attached to the closing angles were 7075-T651 link fittings for attaching the deployment links to the panels. At both ends of the individual panels were a series of slotted 0.005 in. (.127 mm) thick 301 CRES boots that were pre-formed to form an annular closure at both ends of the shield when deployed (Figure 2.2.2.1-3). This closure was required for meteoroid protection and thermal control on-orbit. When the shield was closed, these boots were pressed between the shield and the tank. The interior surface of the panels and boots were Teflon coated to minimize the coefficient of friction with the tank surface which had a gold coating. Several of the panels had removable sub-panels to permit access through the rigged meteoroid shield to the viewing window, the side access panel, and the two SAL's. The panel containing the auxiliary tunnel had twenty-eight

SKYLAB - ORBITAL WORKSHOP
METEOROID SHIELD PANEL JOINT



SKYLAB - ORBITAL WORKSHOP
METEOROID SHIELD BOOT

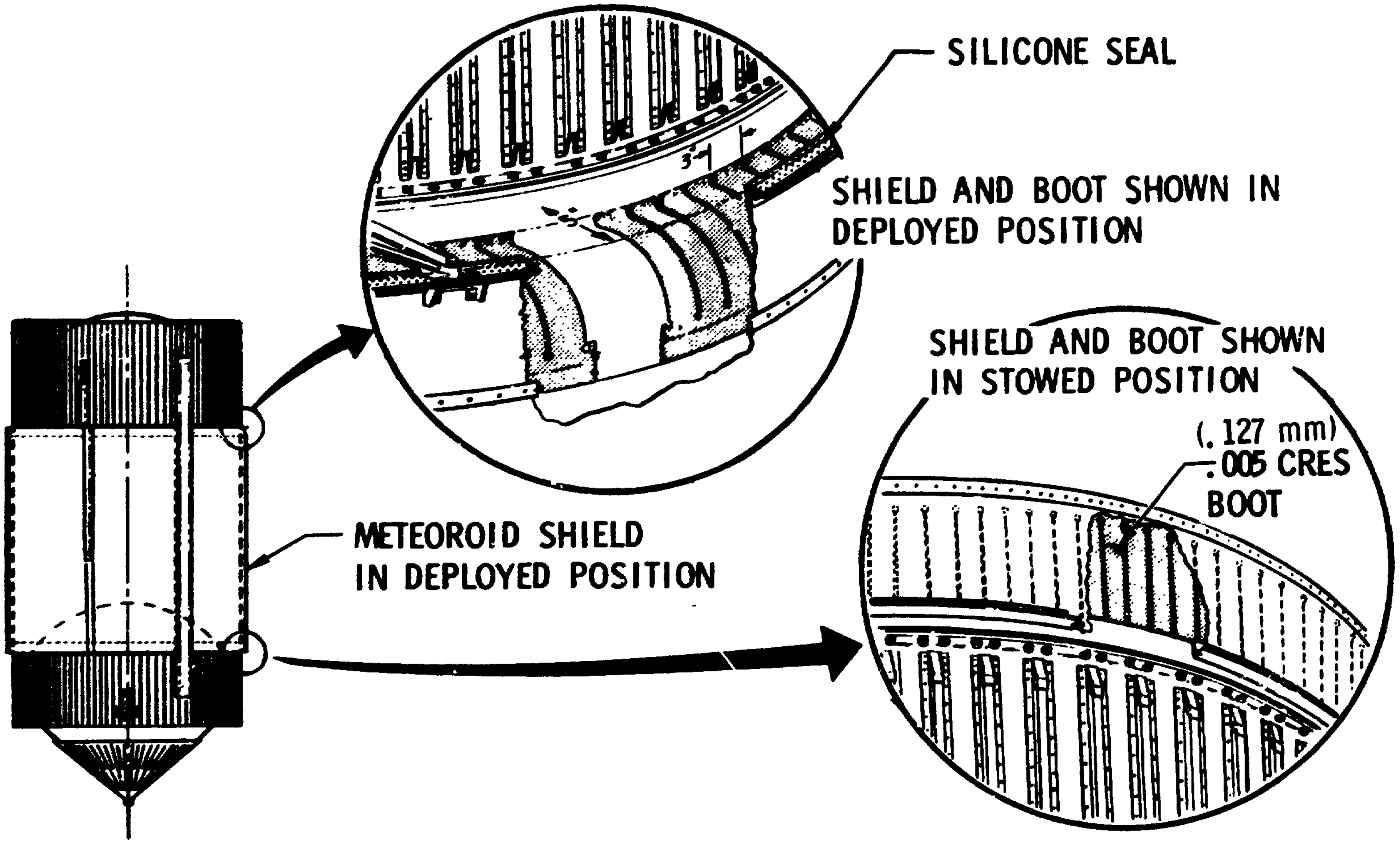


Figure 2.2.2.1-3

6AL-4V titanium alloy frames which acted as springs to maintain a hoop tension load in the shield. (Figure 2.2.2.1-4). They were deflected circumferentially 1.25 in. (3.175 cm) (nominal) when rigged and were capable of additional deflection to accommodate tank growth due to differential pressure of the habitation area tank. The frames were attached to a 6061-T651 aluminum base on the panel by dry-lubed cadmium plated hinge pins.

The auxiliary tunnel enclosed auxiliary equipment running the entire length of the tank and was covered by a three-piece, 0.025 in. (.635 mm), 2014-T6 aluminum beaded skin which interfaced with the fixed tunnel extensions on both the forward and aft skirts.

The folded panel assembly contained the expandable ordnance device and five folded 0.025 in. (.635 mm) thick 2014-T6 aluminum panels connected at the edges by continuous 2024-T6 hinges with teflon coated spacers located between hinge lobes. After deployment, the foldover panels were held away from the tank by four 0.010 in. (.254 mm) thick, 320 CRES scroll springs which were held flat between the panel fold when rigged. (Figure 2.2.2.1-5).

The shield assembly was attached to the vehicle on both sides of the main tunnel and by sixteen A-286 CRES link arms. A 718 INCONEL stud attached each link arm to a panel. A keyed 718 INCONEL shoulder sleeve was pressed into the other end of the link arm and passes through the tank to the skirt flange. The dry-lubed sleeve rotated in two A-286 CRES bearing blocks mounted on the tank-skirt flange. The sleeve was retained by

SKYLAB - ORBITAL WORKSHOP
METEOROID SHIELD AUXILIARY TUNNEL FRAME
AND BEADED PANEL COVER

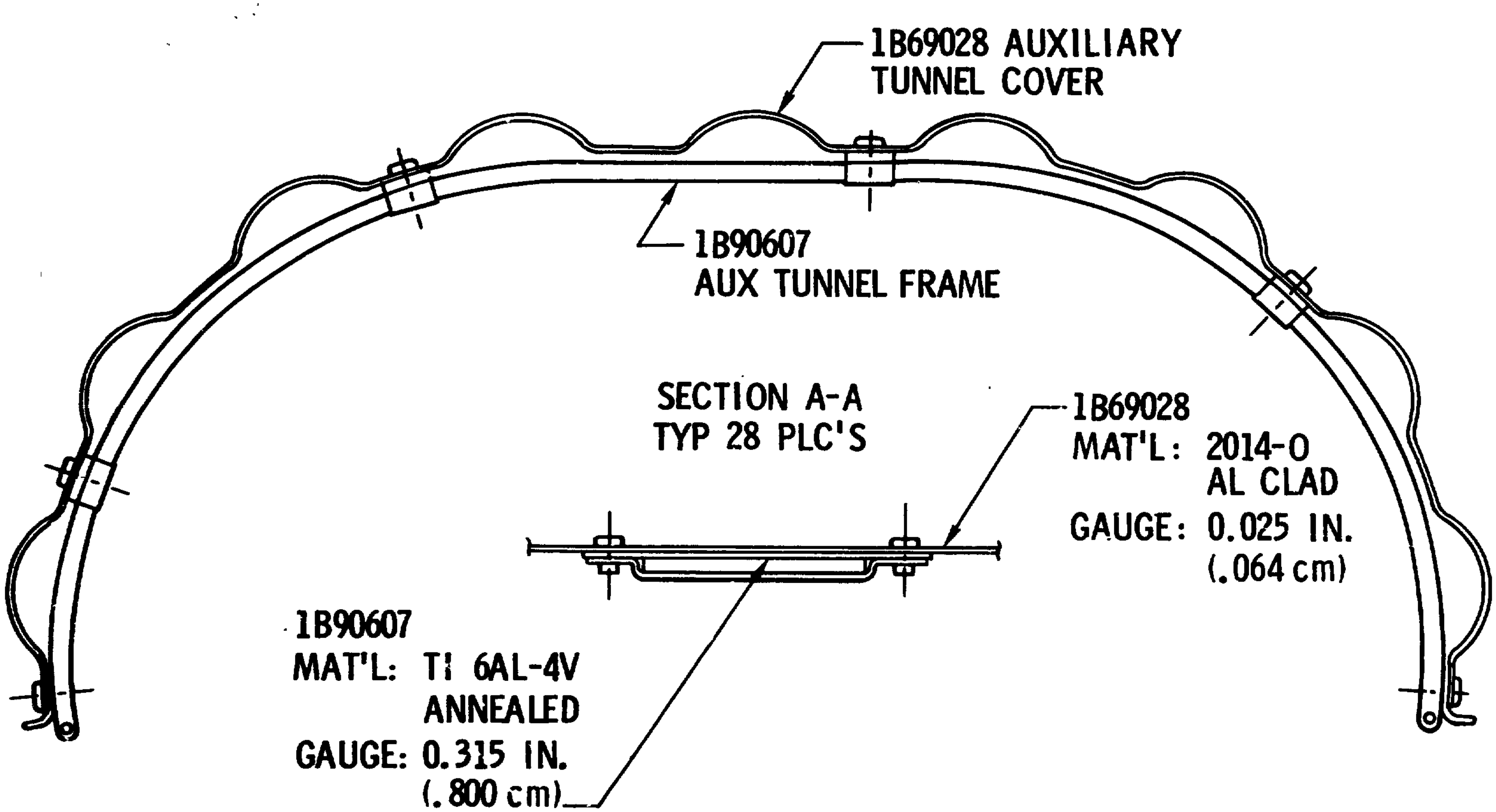


Figure 2.2.2.1-4

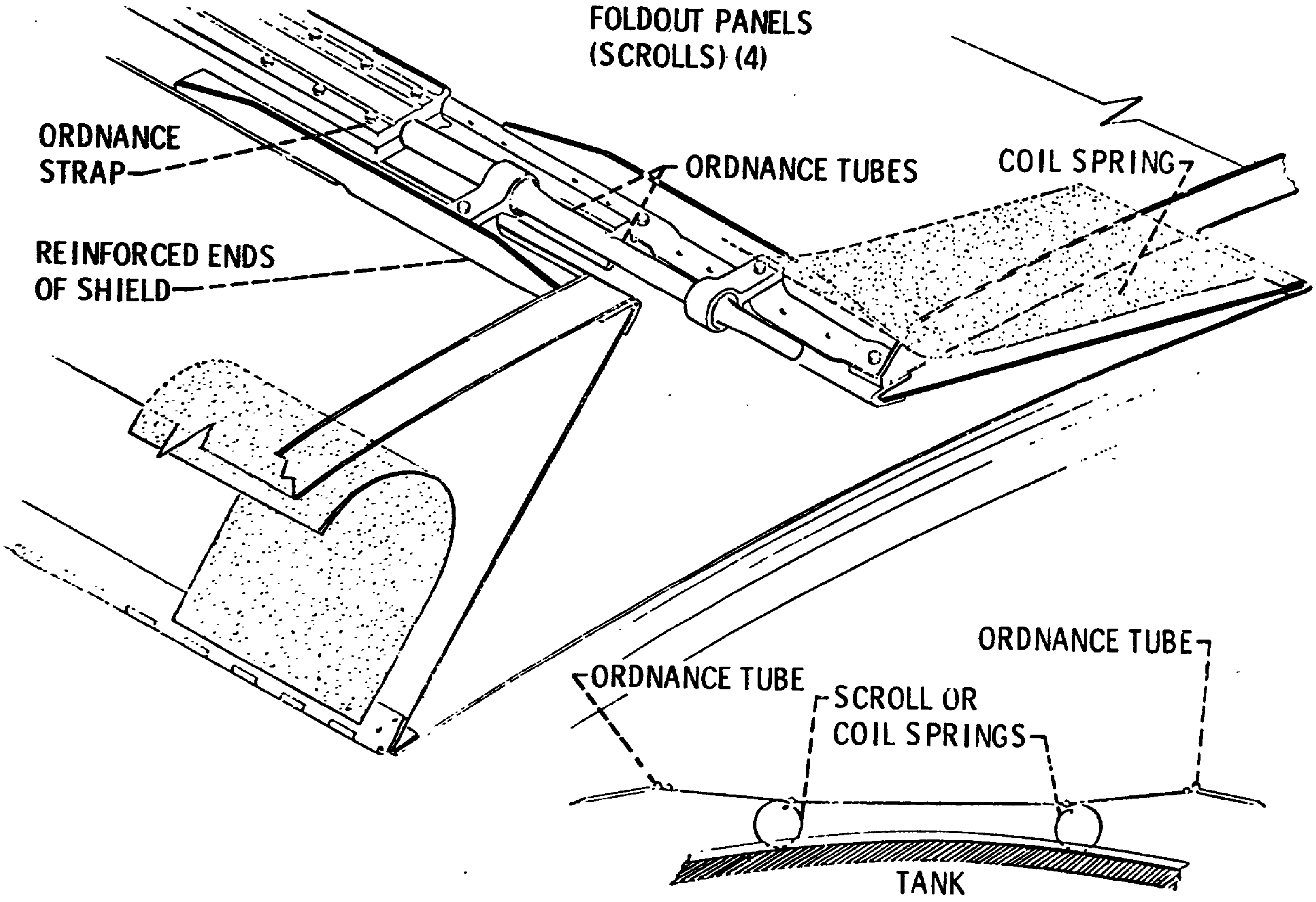


Figure 2.2.2.1-5 - Meteoroid shield deployment ordnance and foldout panels.

two A-286 CRES jam nuts. A 6Al-4V titanium alloy torsion bar was passed through each sleeve mounting the rotatable links and connected at one end to the sleeve by an indexed spline. (Figure 2.2.2.1-6). The torsion bars were preloaded manually and supplied the energy required to deploy the shield. The anchor end of the torsion bar had a A-286 CRES keeper which slid on the hexagon of the torsion bar to pre-load it. The keeper was retained by a 7075-T651 aluminum anchor fitting located between stringers of the forward and aft skirt.

On each side of the main tunnel was a 2014-T651 aluminum full-length hinge attached by dry-lubed cadmium plated steel hinge pins to twelve 2014-T651 aluminum tension straps which were bonded to the tank with a polyurethane adhesive. The mating panel hinges were 6061-T651 aluminum and were also joined with dry-lubed cadmium plated steel hinge pins. The thrust loads were reacted by aluminum Teflon coated spacers located between hinge lobes. The tension straps spanned the main tunnel by extending under the wiring and piping which ran between the forward and aft skirts. (Figure 2.2.2.1-7).

The meteoroid shield was pre-loaded in hoop tension by two overlapping panels, each having fourteen 0.040 in. (1.016 mm) thick 7075-T6 aluminum loop straps attached. The straps were joined by A-286 CRES trunnions and 303 CRES take-up bolts (Figure 2.2.2.1-8). The viewing window was protected at launch by a two-piece 0.010 in. (.254 mm) thick 301 CRES Teflon-coated cover. Each piece of the cover was pre-formed into a scroll. When the shield was rigged, the shade segments were flattened

SKYLAB - ORBITAL WORKSHOP
METEOROID SHIELD

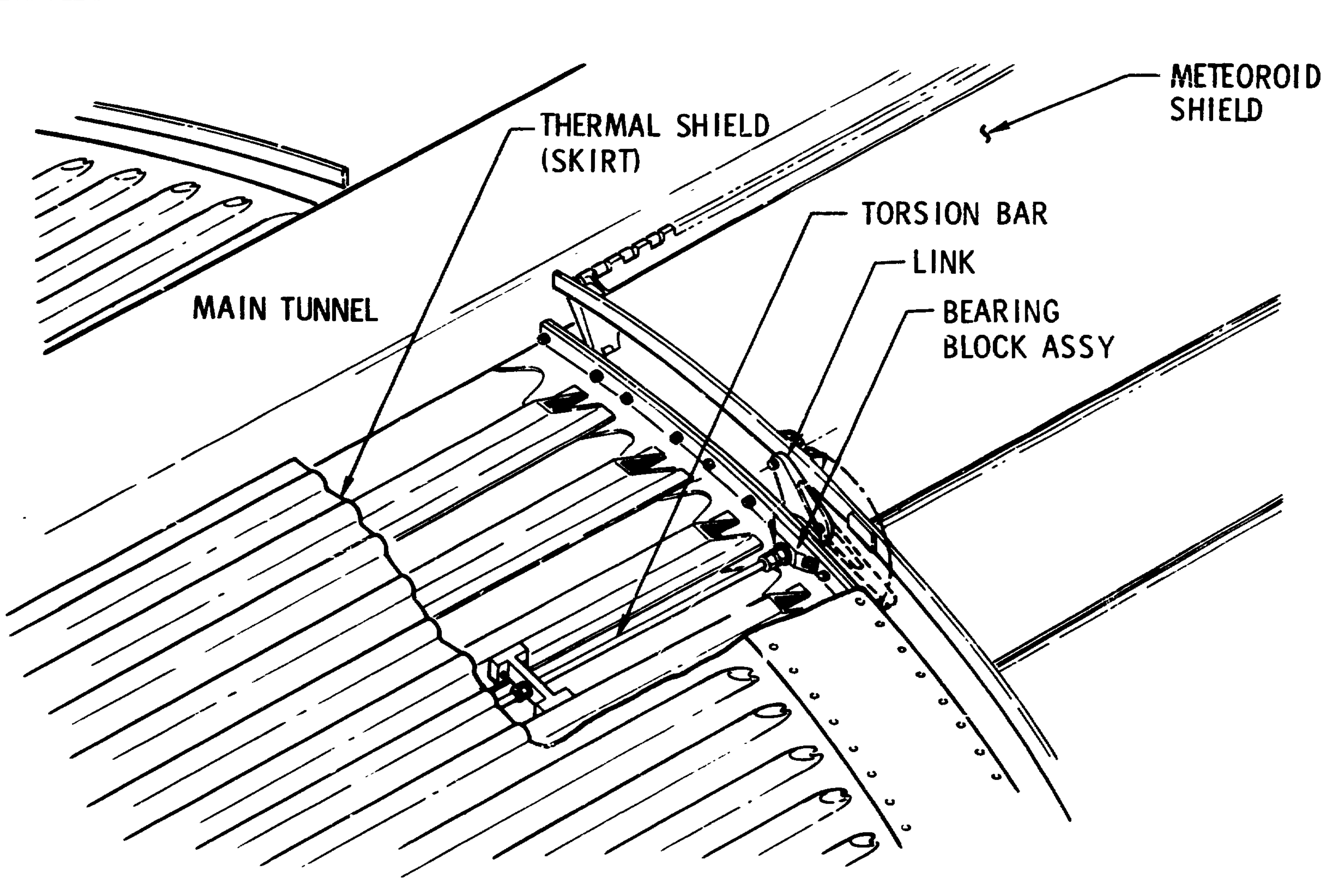


Figure 2.2.2.1-6

SKYLAB - ORBITAL WORKSHOP
DEPLOYABLE METEOROID SHIELD

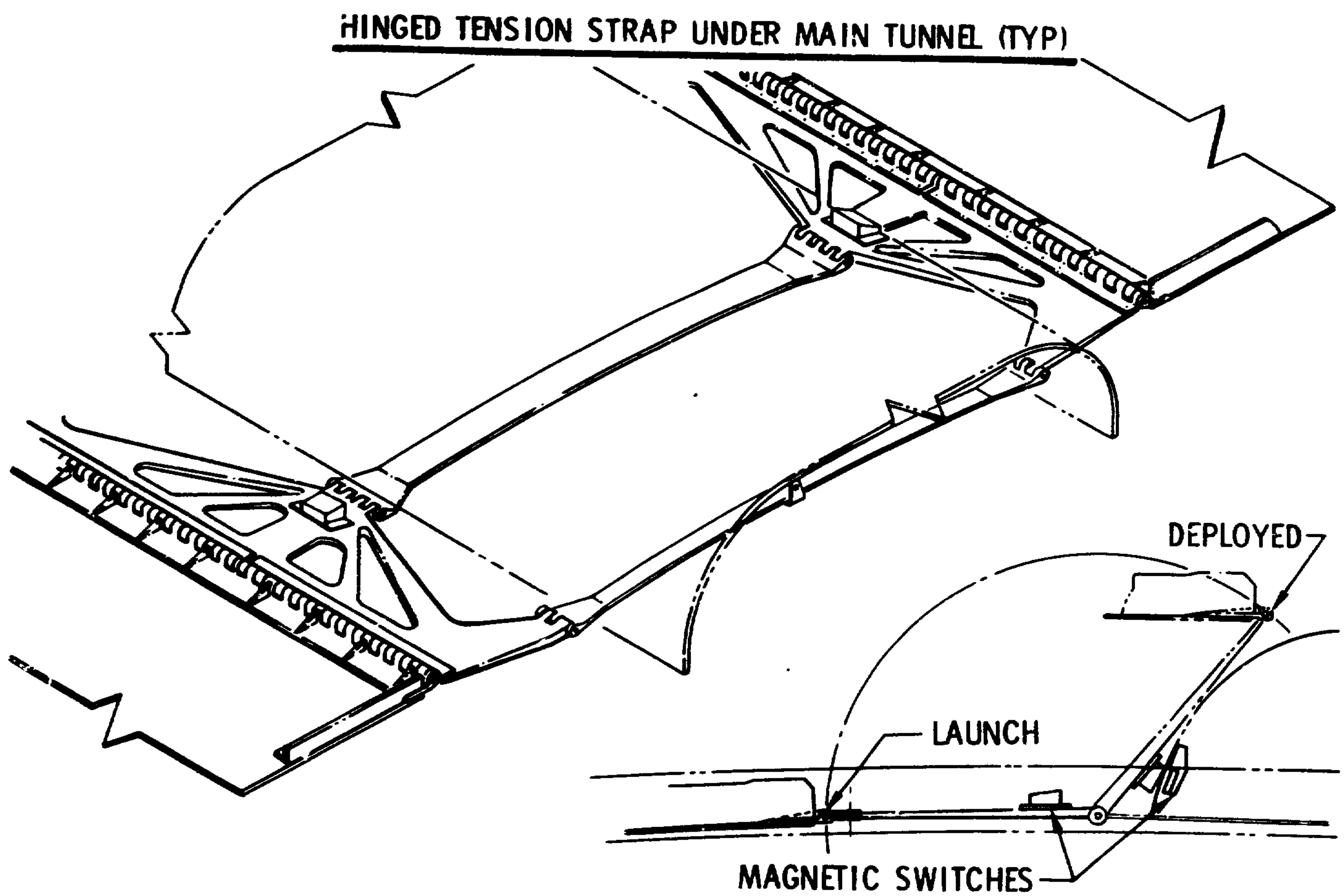


Figure 2.2.2.1-7

SKYLAB - ORBITAL WORKSHOP

METEOROID SHIELD RIGGING DEVICE

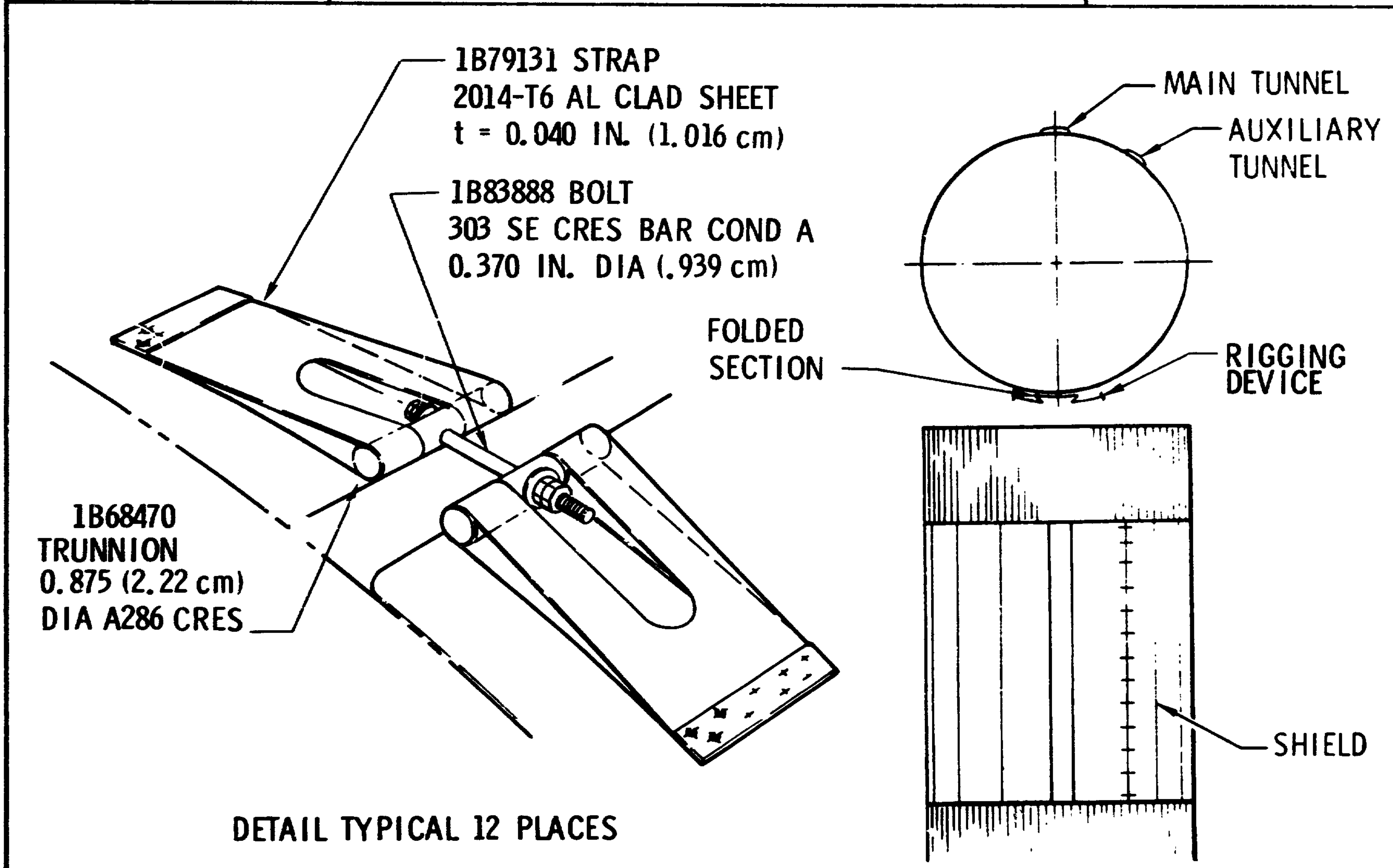


Figure 2.2.2.1-8

between the shield and the tank. At shield deployment, each section returned to its pre-formed shape, thus exposing the window. A 17-4 PH CRES hook was attached to each of the four link arms adjacent to the main equipment tunnel. This hook engaged a 17-4 PH CRES latch which was mounted on a 17-4 PH CRES latch base. The latch base was attached to the tank flange. The latching mechanism stopped rebound of the shield during deployment. The latching force was achieved by a 302 or 304 CRES compression spring.

3/ Mechanical Interfaces (Internal and External) - When the shield was rigged for launch, the shield interfaced with the SAS seals and the auxiliary tunnel on both the forward and aft skirts.

The SAS seals interfaced with the panel surfaces in the constant section of the tank. Adjacent to the tank flanges, a ramp was attached to the panel, raising the sealing surface to the height of the sealing surfaces on the forward and aft skirts. The forward end of the meteoroid shield had a teflon seal interfacing with a flat bulkhead on the aft end of the auxiliary tunnel on the forward skirt. The aft end of the meteoroid shield auxiliary tunnel was sealed by a silicone rubber seal held in place by a slotted frame that formed around the aft skirt fairing stub. The frame was attached to the panel with rollers to permit shield-to-tank movement.

The shield provided a method of entry into the habitation area through the side access panel. This was accomplished by removal of a section of the segmented panel after the shield was rigged. After removal of this section, the side access panel could be removed. After deployment of the shield, the two cutouts for the SAL's in the meteoroid shield were approximately centered over the SAL openings in the tank to permit deployment of the experiments. Also, the cutout for the window was approximately centered over the viewing window.

4/ Ordnance System Description - Release of the meteoroid shield to permit deployment was accomplished by means of an ordnance system. The meteoroid shield was held against the OWS by means of a tension strap assembly that extended along the full length of the meteoroid shield. The tension strap assembly contained six tension straps and two explosively actuated expandable tube assemblies. (Figures 2.2.2.1-9 through through 2.2.2.1-11). The tension strap assembly was installed during manufacture of the meteoroid shield folded panel assembly. The folded panel assembly contained five hinged panels that were folded on themselves to permit proper retraction of the shield to the smaller OWS diameter.

The meteoroid shield release system contained redundant EBW detonators and electronic firing units located on a panel in the forward skirt. A redundant fuse train, consisting of the confined detonating fuse, extended between the detonator block and the expandable tubes. The expandable tubes consisted of two flattened 1/2 in. (1.27 cm) diameter steel tubes extending the full length of the shield. Each tube contained a single 14-grain/foot RDX mild detonating fuse. The tubes were encased in the tension strap which was severed along its

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METEOROID SHIELD RELEASE SYSTEM

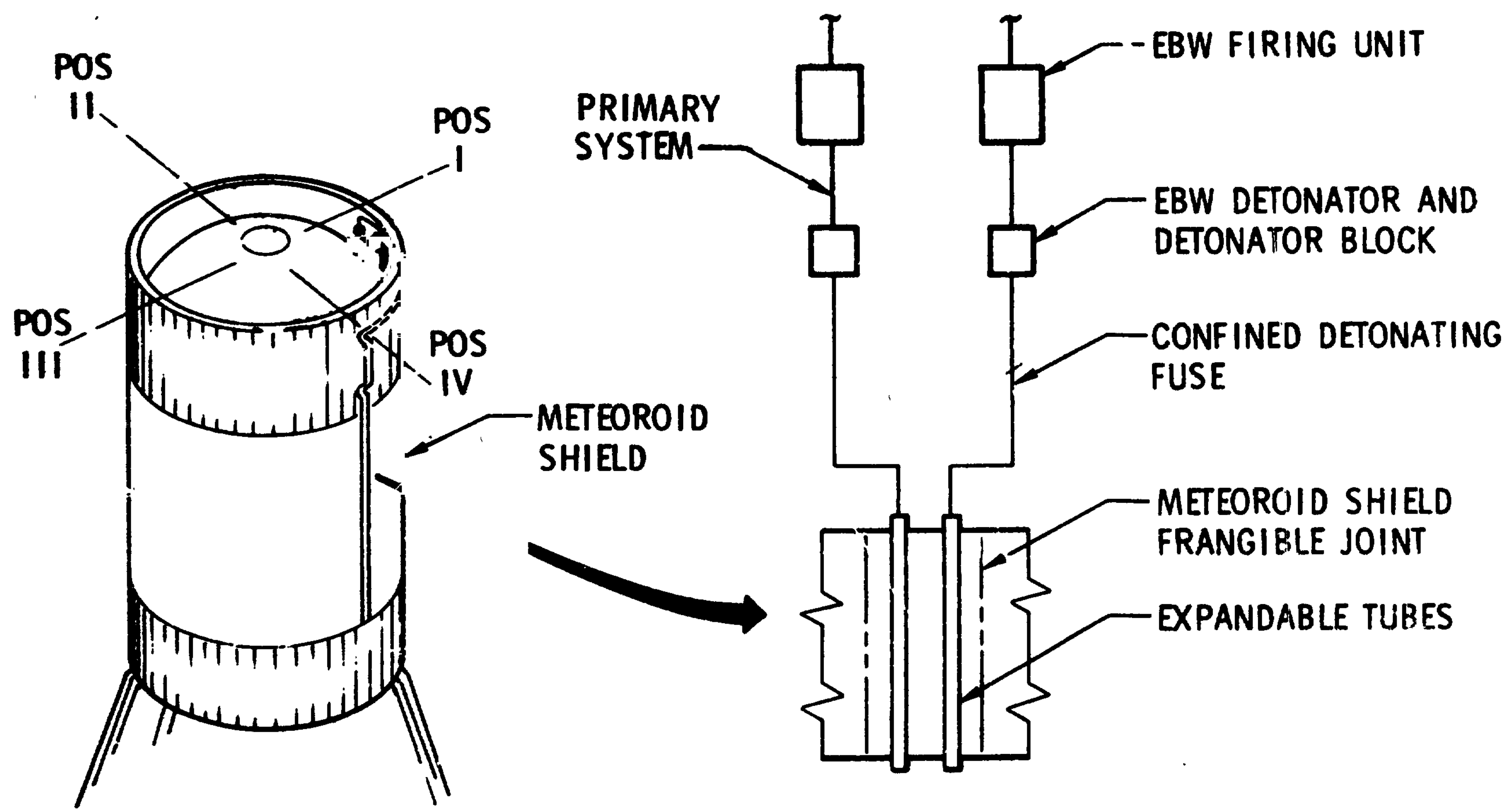

Figure 2.2.2.1-9

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SHIELD RELEASE DEVICE

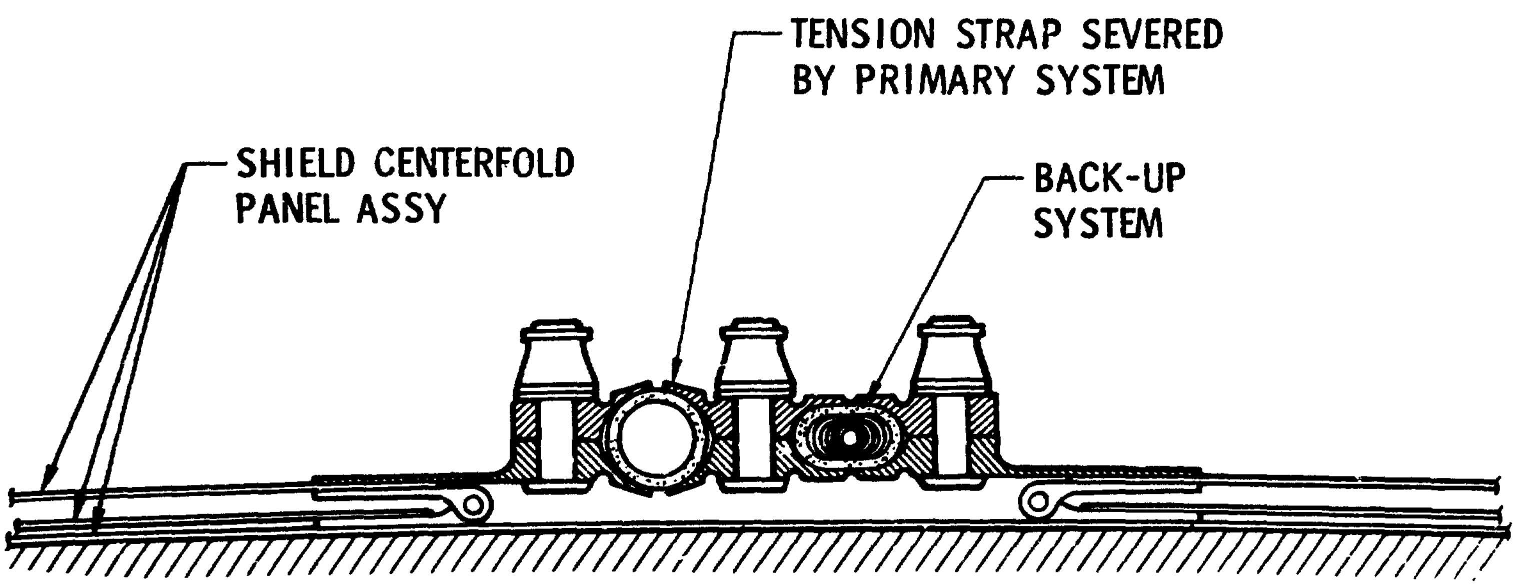

Figure 2.2.2.1-10

SKYLAB - ORBITAL WORKSHOP
METEOROID SHIELD RELEASE MECHANISM

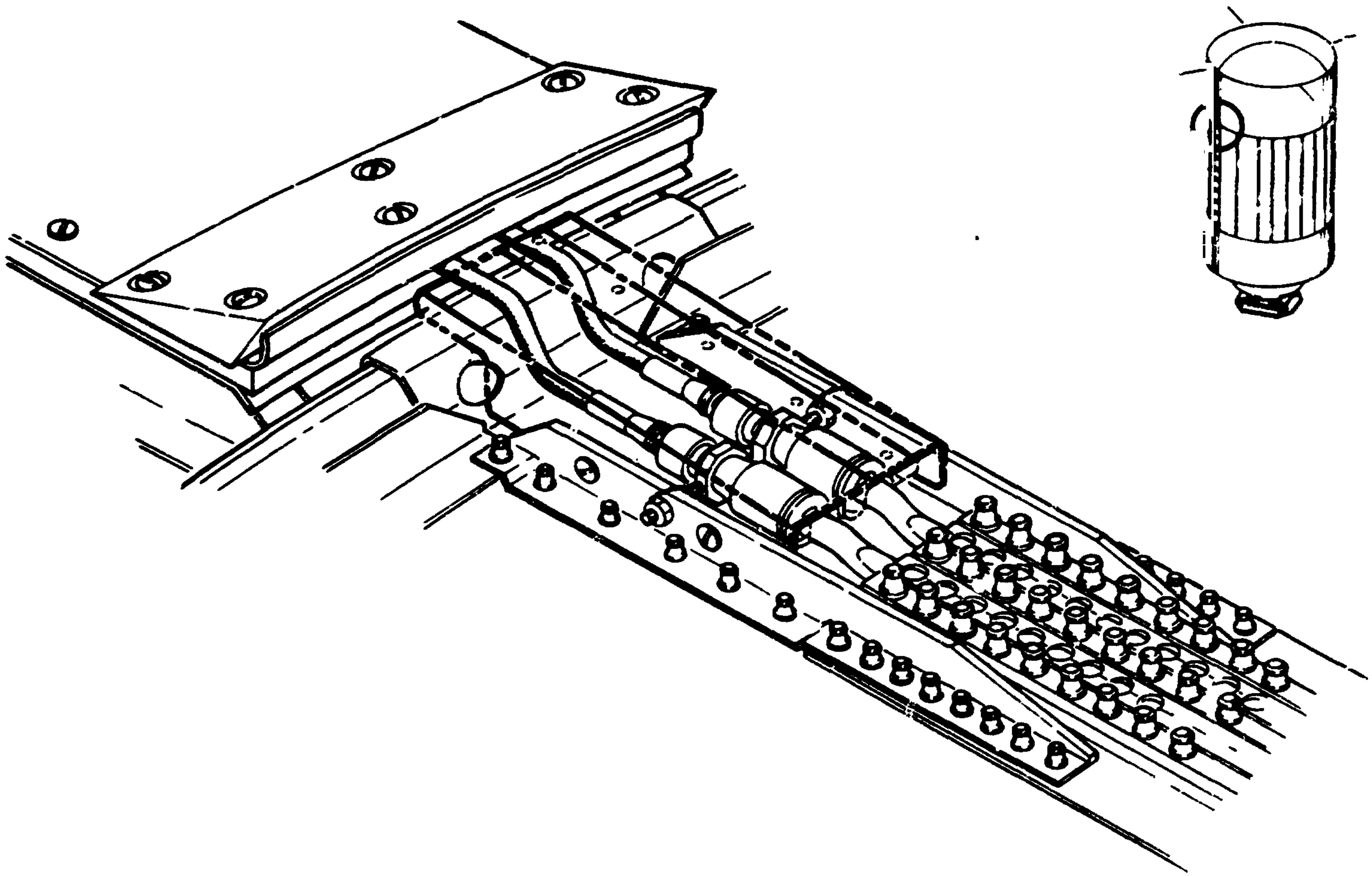

Figure 2.2.2.1-11

entire length by the expansion of either tube of the redundant system. The folded panels were extended to the deployed position by pre-loaded torsion bars and springs after the restraining tension strap assembly was severed.

Each of the two explosive trains was initiated using an EBW system containing an electronic firing unit and a detonator. The plan was to initiate one system; the other system remaining available as a backup.

The electrical design provided for charge and firing commands from the IU with backup signals from the AM. This system utilized redundant ordnance trains which were designed so that a single failure did not inhibit shield deployment.

5/ Firing Circuit - This subsystem utilized both AM Bus 1 and AM Bus 2 power. During the first 7-1/2 hours of flight, the system utilized the IU. There were two methods to release the shield: a primary, and a secondary. The primary method utilized the IU automatic flight control computer to issue proper control commands to the OWS switch selector. Through relay contacts controlled by the switch selector, AM Bus 2 power was utilized to charge the firing unit and fire the ordnance, which released the shield. The relays utilized by this circuit were 10-amp mag-latch type and 10-amp general purpose. The fire command circuit was inhibited by a normally open relay contact controlled by the OWS solar array deployment command. This sequence of deployment was required to prevent possible damage to the OWS solar arrays during array deployment.

The backup method of deploying the shield utilized ground digital command system to control 10-amp general-purpose relays via the AM. This circuit utilized AM Bus 1 power to charge the firing unit and fire the ordnance to permit deployment of the shield.

C. Development and Qualification Tests

1/ The first phase of test was performed primarily to verify that the shield thickness selected could withstand the specified meteoroid environment.

During May, 1971, a deployment test was performed on OWS-1. This test was conducted to verify the contained ordnance system, the torsion bar linkage mechanism, ordnance break wire instrumentation, deployed sensor switches, and fit and function of all other components. The test was performed with one "G" environment and in the atmosphere. During the test, the confined ordnance system ruptured and the meteoroid shield did not fully deploy. All other systems and fit and function of the other components conformed to design expectations except for difficulty in rigging and problems with the seal at the forward end of the auxiliary tunnel which required modification.

Because full deployment was not achieved, the deployment sensing switches were not activated. A backup system, consisting of redundant strain gage installations on each torsion bar, was deemed necessary to indicate position of deployment obtained. This system was tested in Line Item CA-32 and W-ST-E-14 tests. These tests were performed by subjecting a torsion bar with sixteen full pairs of strain-gage bridges to humidity, temperature, and life cycles. The position indication system was also incorporated on the ST-14 test.

Failure of the meteoroid shield to fully deploy on OWS-1 was investigated. Included in this investigation was a test conducted to see if the 6AL-4V titanium alloy torsion bar was susceptible to creep. The test, W-ST-S-12, was performed by placing a torsion bar for several days at various known angular deflections. No discernible creep was observed within the range of operating torque loads.

An analytical investigation was conducted to determine the effect of friction imposed on a typical linkage mechanism due to the weight of the shield. This investigation showed that the primary cause of failure of the shield to fully deploy from friction. A counterbalance kit, Model DSV7-371, to simulate zero gravity was created for use in future tests.

Line item CA-34 test was performed on the latch mechanism by varying the angular velocity and impact momentum of the link arm to ensure that the latch would capture the link arm.

A meteoroid shield was installed on the Static Test Article (STA) at Huntsville. Deployment tests (ST-14) were performed in February and April, 1972, using the DSV7-371 counterbalance kit. A manual release panel was fabricated from the expended ordnance folded panel assembly used on the OWS-1 test. A manual release deployment was performed successfully in February, 1972. An ordnance deployment test in February, 1972, was successful with exception of the following discrepancies: a) the latches on the links on the window side did not latch, b) the distance between the tank and the fold-over panel was considered insufficient for meteoroid protection, and c) the forward and aft bulb seals did not provide adequate sealing around the entire periphery.

The latches were revised to provide greater latching freedoms and a method of adjustment was incorporated into the design. The forward and aft bulb seals were made adjustable and stand-off scroll springs were added to the fold-over panel assembly. Also, the manual release panel assembly was revised to carry hoop tension loads due to differential pressure of the habitation area tank. After these changes were incorporated into the hardware, three manual release deployments (one with differential pressure on the habitation tank) were successfully performed, followed by two successful ordnance release deployments with differential pressure on the habitation tank. Subsequent to the deployment tests, the STA habitation tank was taken to the design ultimate differential pressure. The hinge lobes on the window side of the main tunnel at three tension strap locations and the mating hinge lobes attached to the shield failed in bearing. This failure also caused a bending failure in the hinge pins in these joints.

A butterfly hinge test (ST-28) was conducted to substantiate the failure conditions and to verify the adequacy of the proposed rework. The rework was an extension with additional lobes bonded to the main tunnel strap and a mating part mechanically attached to the butterfly hinge. The first specimen (without rework) failed, in a manner similar to the STA failure, at 5,671 lbs (2572 kg). The redesigned specimen failed in the main tunnel strap at the test fixture connection at 11,600 lbs (5262 kg), thus demonstrating a capability of at least 181 percent of the design requirement. The results substantiated both the structural and functional adequacy of the redesigned meteoroid shield hinge. Test verification is summarized in Table 2.2.2.1-1.

A subsequent STA pressure test was conducted at MSFC with the meteoroid shield and reworked butterfly hinges installed. A bonding failure occurred between the add-on hinge lobe and tunnel strap at three locations. Failure was due to poor bonding techniques. OWS-1 and OWS-2 were inspected to determine bond adequacy and paper reviewed to determine that proper procedures and materials were used. Use of proper procedures and materials were verified. Inspection of OWS-1 at KSC indicated a partial debonding of the reworked doubler of the main tunnel strap hinge/butterfly hinge. This led to a strap doubler test (ST-38), conducted to evaluate partial bond capability of the 1B94294-1 fitting to the 1B68056-1 strap. Three test specimens were involved: (1) reworked design with the strap doubler unbonded, (2) reworked design with strap doubler partially unbonded to simulate OWS-1 Strap 5, and (3) reworked design with strap doubler unbonded and with a series of bonded glass cloth layers used to join the 1B94291-1 fitting to the simulated 1B68056-1 strap.

Table 2.2.2.1-1

METEOROID SHIELD
TEST VERIFICATION SUMMARY

Line Item	Test Title	Test Requirements	Results
CA-28	Expandable Tube and Strap Assy.	Phase I To verify redesign of tube and strap and prevent rupture	Tube did not rupture, but loose pieces broke off from tension strap
		Phase II Parameters changed to prevent loose pieces	Test completed successfully in all respects, 4 September 1971 Documented in Report No. MDC G4022.
CA-30	Expandable Tube and Strap Assembly - Acoustic Test	To qualify a meteoroid shield release system containing a redesigned expandable tube and tension strap assembly [8 ft (2.44 m) long]. Tension strap to be severed by ordnance device following simulated flight environment (Ref. 1T41957).	Test completed successfully March, 1972. Documented in Report No. MDC R7042A.
CA-31	Expandable Tube and Strap Assembly	To qualify a full scale [22 ft (6.7 m)] installation of redesigned expandable tube and strap assemblies on backup structure.	Test completed successfully April, 1972. Documented in Report No. MDC R7043.
CA-32	Meteoroid Shield Position Instrumentation Development Test	To determine if strain gages can maintain stability and accuracy in a torqued position for an extended period of time.	Test completed successfully. Documented in Report No. MDC G3373.
CA-34	Meteoroid Shield Deployment Latch, Development Test	To verify proper latch operation by repeated latching runs over an expected range of meteoroid shield mass and velocity (Ref. 1T43177).	Test completed successfully March, 1972. Documented in Report No. MDC G3374.

Table 2.2.2.1-1

METEOROID SHIELD TEST VERIFICATION SUMMARY (Continued)

Line Item	Test Title	Test Requirements	Results
ST-14	Meteoroid Shield Deployment Test	Series 1 (Ref. 1T42765 Rev. A) Manual Deployment Demonstration of meteoroid shield deployment. Proper deployed position of meteoroid shield. Demonstration that counterbalance preloads do not cause linkage to gall, bind, or prevent proper MS translation and rotation. Ordnance Deployment Primary test objectives Proper deployed position of MS Proper proximity switch activation. Proper position indicator readout. Proper link latch actuation Demonstrate proper debris containment and no evidence of contaminants by the ordnance expandable tube Secondary Test Objectives Proper tension strap breakwire signal Demonstrate no significant effects from pressurization Proper zero g deployment time.	Test completed successfully in all respects, except latches on window side did not latch properly because of improper latch position. Difficulty was corrected for series 2 tests. Series 1 test completed March, 1972. Documented in Report No. MDC G3369.

Table 2.2.2.1-1

METEOROID SHIELD TEST VERIFICATION SUMMARY (Continued)

Line Item	Test Title	Test Requirements	Results
ST-14 (CONT)	Meteoroid Shield Deployment Test - STA	Series 2 (Ref. 1T42765 Rev. C) Mechanical deployment - no tank pressure Demonstrate that ordnance deployment test objectives A1 thru A3, B2, and B3 (below), are met. Mechanical deployment - with tank pressure Demonstrate that ordnance deployment test objectives A1 through A3, B1 through B3 (below) are met. Ordnance Deployment A. Primary test objectives 1. Proper deployed position of MS 2. Proper proximity switch actuation 3. Proper link latch actuation 4. Demonstrate no evidence of contaminants by ordnance expandable. 5. Proper tension strap break-wire signal.	Test completed successfully in all respects. Test completed April, 1972; Documented in Report No. MDC G3369.

Table 2.2.2.1-1

METEOROID SHIELD TEST VERIFICATION SUMMARY (Continued)

Line Item	Test Title	Test Requirements	Results
ST-14 (CONT)		B. Secondary test objectives 1. Demonstrate no significant effects from pressurization 2. Zero g deployment time 3. Proper continuous position indicator readout.	

Table 2.2.2.1-1

METEOROID SHIELD TEST VERIFICATION SUMMARY (Continued)

Line Item	Test Title	Test Requirements	Results
ST-28	Hinge Test	Hinge to function at 6,540 lbs (2967 kg) with a rotational torque of 39 in-lbs (4.4 N·m) or less.	The original design was functioned at 5500 lbs (2495 kg) and the lobes and pin structurally failed at 5671 lbs (2572 kg). The reworked design was functioned at 10,000 lbs (4536 kg) and the test strap failed at 11,600 lbs (5262 kg). NOTE: Rotational torques were less than 39-inch pounds (4.4 N.m)
ST-38	Hinge Test	Hinge to function at 6,540 lbs (2967 kg) with a rotational torque of 39 in-lbs (4.4 N·m) or less.	The reworked design with strap doubler unbonded but restrained from rotation functioned at 6750 lbs (3062 kg) and failed at the lobes and pin at 6840 lbs (3103 kg). This test demonstrated unbonded rework increased the structural capability to exceed the design requirement. The reworked design with strap doubler partially unbonded to simulate OWS No. 1 Strap No. 5 functioned at 8250 lbs (3742 kg) and failed at the lobes and pin at 9750 lbs (4423 kg). The reworked design with strap doubler totally unbonded, but with a 3 ply 181 fiberglass band-aid fix applied to the external surface of the doubler and strap, functioned at 8250 lbs (3742 kg) and failed at the lobes and pin at 9900 lbs (4491 kg).

All three specimens in this test demonstrated functional capability after sustaining loads greater than the 6,540 lb (2967 kg) requirement. It was concluded that the reworked butterfly hinge would be adequate even in the complete absence of bonding.

2/ Development and Qualification Test Problems

During the testing of OWS-1, it became apparent that the meteoroid shield was not capable of deployment at full gravity loading on the linkage mechanisms. Since deployment was to occur in space in a weightless condition, a counterbalance kit to simulate zero gravity was fabricated for future tests. During the ST-14 test at Huntsville utilizing the zero "G" Kit, the distance between the meteoroid shield fold-over panels and the tank did not provide the meteoroid protection required. This problem was solved by the incorporation of four (4) standoff scroll springs to the hinges of the fold-over panels. On the first series of tests, the latches did not engage on the window side of the meteoroid shield during the ordnance deployment. This problem was solved by redesign of the latches and changes in their adjustment. These changes were proven successful in the Series 2 tests of ST-14. The hinge lobes on the right side of the main tunnel at three tension straps and the mating hinge lobes attached to the shield failed during the ultimate pressure test of the STA at MSFC. Both sides of the hinge were reinforced and proven to be structurally and functionally adequate by the ST-28 tests. After a bonding failure during a pressure test at MSFC, inspection of OWS-1 at KSC and OWS-2 at Huntington Beach indicated a partial debonding of the reinforced main tunnel strap hinge on both vehicles. After completion of a test program (ST-38), it was concluded that the rework would be adequate even in the complete absence of bonding. (See Table 2.2.2.1-2).

3/ KSC Testing

The first mechanical deployment test was conducted per TCP KO-3018 on October 14, 1972. The two latches on the right hand (window) side of the meteoroid shield failed to latch. A troubleshooting team was organized and a series of meetings were held to determine the causes of the non-latching. Many Interim Discrepancy Reports, most of which were subsequently upgraded to Discrepancy Reports, were generated during the trouble shooting operations. Extensive rework was made on the seals in the link areas to avoid pinching and binding. A proximity switch mounting was changed to eliminate interference. These two items were considered the main contributors to the non-latching condition.

On October 16, 1972, the No. 5 aft torsion rod was found to be yielded (twisted) in the hex portion of the rod, resulting in a Discrepancy Report with a disposition to remove and replace. Instrumentation on two other rods indicated low residual torque readings. A mechanical test for residual torque was run. Based on these results, one rod required replacement. Replacement was completed on October 20, 1972, except for final glass cloth and RTV coatings over the strain gages on

Table 2.2.2.1-2

PROBLEM SUMMARY STRUCTURES SUBSYSTEM

HABITATION AREA TANK - METEOROID SHIELD

Item	Problem	Solution
1.	Meteoroid shield failed to achieve full deployment on OWS-1 with full gravity loads on links. Ordnance tube ruptured.	Counterbalance kit used to simulate actual deployment conditions - redesigned ordnance tube/strap.
2.	Distance between tank and fold-over panel insufficient to provide meteoroid protection	Added scroll spring to hold fold-over panels away from tank.
3.	Rebound latches did not engage during deployment.	Revised design to increase clearance and provided adjustment.
4.	Hinge failed in bearing during deployment.	Hinge reinforced on both the strap and butterfly hinge sides.
5.	Hinge reinforcement had debond after ultimate pressure test.	Tests showed debond acceptable.

the new rod. Torsion Rod No. 8 aft was found to have a tapped hole in the spline end (permitted from an earlier configuration) that was deeper than expected. The results of a stress analysis determined it to be acceptable.

A multi-page deviation was written to rerun the mechanical deployment test. The second deployment was performed on October 22, 1972. The shield deployed fully with the exception that the upper right latch failed to engage. A waiver (MDAC-OWS-WR-02) was written and NASA concurrence received to allow any three of the four latches to engage for an acceptable deployment. More Discrepancy Reports were generated which concerned bulb seals, gold foil, thermal boot fingers, and paint, after this deployment. These discrepancies were fixed. The mechanical panel was removed and the ordnance panel was installed in its place. Flight rigging was accomplished. Subsequently, it was determined that the shield was gapping locally in two places at the aft end. An engineering disposition to the Discrepancy Report was written to correct this situation and when implemented, the meteoroid shield was satisfactorily rigged for flight per KO-3018.

D. Mission Results

Skylab 1 was launched from Pad 39A at the Kennedy Space Center on 14 May 1973. Just prior to 63 seconds, the Orbital Workshop film vault accelerometer indicated an onset of vibration that is attributed to the beginning of the meteoroid shield failure. Shortly after, the shield temperature measurements were lost, ordnance strap breakwires separated and SAS Wing No. 2 tie-downs were broken, indicating a meteoroid shield structural failure. The Skylab Mission continued with the approximate 1 month visit of SL-2, related EVA, (see Solar Array System, 2.2.5) and the SL-3 and SL-4 visits of 59 and 84 days respectively. Total storage time between inhabitations was approximately 3 months. No meteoroid hits resulting in an OWS habitation area pressure loss occurred during the full mission duration of over 9 months.

The passive thermal control system was significantly influenced by the meteoroid shield anomaly.

E. Conclusions and Recommendations

Three prominent areas of discussion resulted from design trade-offs, analysis, ground test, and the flight of the OWS (SL-1).

1/ The choice of a deployable meteoroid shield for the OWS was the logical configuration to meet the design and program requirements. Emphasis in testing was placed upon the deployment aspects of the system with repetitive test success attained after significant development and hardware changes. Proof of structural integrity for launch and ascent conditions was accomplished by component tests and conventional analysis techniques. If a total trade-off study leads to a light weight, thin sheet, deployable shield, it is recommended that a design of this type must have and maintain a greater intimacy with the Spacecraft in all modes from launch to deployment. This ability must be shown by test, or by analysis using an unusually large factor of safety on all possible failure modes.

2/ A complete re-examination of system design requirements and trade-offs, made subsequent to SL-1 launch, showed that it was possible to obtain an OWS configuration which did not require a meteoroid shield. This was accomplished by changing external optical coatings, reallocation of power, and higher risk of meteoroid penetration. If these parameters were considered acceptable, a much less complex configuration results. It is recommended that trade-offs between systems be made continuously as a program matures and new "State-of-the-Art" technical data is obtained to be certain that current design requirements are valid.

3/ It is recommended that in future design requirements, the risks involved by the use of EVA as a primary or backup mode for necessary deployment or fixes be traded against possible weight reduction, simplified system design, and cost effectiveness.

F. Development History

The meteoroid shield for OWS was originally developed for the wet workshop. A thin shield wrapped tightly around the cylinder which could deploy about 5 in. (12.7 cm) from the cylinder on orbit provided many advantages especially with liquid hydrogen in the tank. By using a gold coating on the tank surface in conjunction with the shield paint pattern and the internal insulation, a simple passive thermal control system could be achieved. No cryo-pumping or purging was required for the shield. Although there was concern of ice forming to prevent deployment, early testing and analysis proved the shield would deploy. Early testing was also performed to determine standoff distance for meteoroid protection.

When the change was made to dry workshop, a fixed shield was considered; however, other systems such as the SAS had already been designed to the deployable shield configuration and would be greatly affected by a change. Also, a weight penalty plus several other problems such as flutter, condensation, etc. would be encountered with a fixed shield. The deployable shield development continued with the addition of other OWS systems. The wardroom window required a shield cover. The auxiliary tunnel was designed as a spring to maintain tension in the shield. Scroll springs were developed to insure proper deployment standoff distance at the fold over panels. Interface of the SAS with the shield was developed to provide sealing of the SAS beam fairing. All of these changes were supported by development, qualification, and acceptance testing.

2.2.2.2 Deployment and On-Orbit Configuration

A. Design Requirements for this configuration contained the following constraints and parameters:

1/ The OWS shall be designed to withstand meteoroid impact without loss of pressure in the habitation compartment and to be protected against loss of functional capability when subjected to the meteoroid environment specified in TMX-53957. The probability of no pressure loss, in the habitation compartment, shall meet or exceed an overall requirement of 0.995 during a minimum period of eight consecutive months. Maximum inherent protection shall be provided against functional loss.

2/ The meteoroid shield deployment shall be initiated during orbit by a signal from the IU via the switch selector. A backup deployment command from the AM shall be available at the OWS/AM interface. The system shall be designed so that a single failure will not inhibit deployment.

3/ On-orbit deployment shall be initiated by redundant ordnance devices which shall result in no contamination when activated. The system shall be capable of operation in a temperature range of -140°F to 120°F (-95.5°C to 48.8°C).

4/ A closure shall be installed at each end of the meteoroid shield covering the annulus formed by the shield in the deployed condition to provide thermal control and meteoroid protection.

5/ Latching mechanisms shall be provided to maintain the shield in the deployed position after initial deployment.

6/ Instrumentation and telemetry provisions shall be made to allow determination of the position of the deployed shield on-orbit.

7/ Particular coatings shall be used on the external surface of the meteoroid shield to contribute to the passive Thermal Control System.

B. System Description

1/ Meteoroid Protection - The cylindrical habitation area meteoroid shield acted as an outer protective wall (bumper), thus forming a two-barrier protection system with the existing structure and augmenting the pressure vessel penetration resistance. The subsystem of interest here was the entire OWS. The various structural subassemblies which formed the pressure vessel, constituted a unique meteoroid protection system, and are delineated in Figure 2.2.2.2-1. Also, shown in the figure are functional subsystems which, although not falling under the no-pressure loss criterion, were still checked for protection integrity.

a. Major Components - From basic probability theory, it was known that the probability of no failure (pressure loss) for the entire OWS was the product of no-failure probabilities to each contributing subassembly. Thus, each unique subassembly shielding configuration could be analyzed separately to obtain the overall resultant no-failure probability. A description of these component shielding systems follows.

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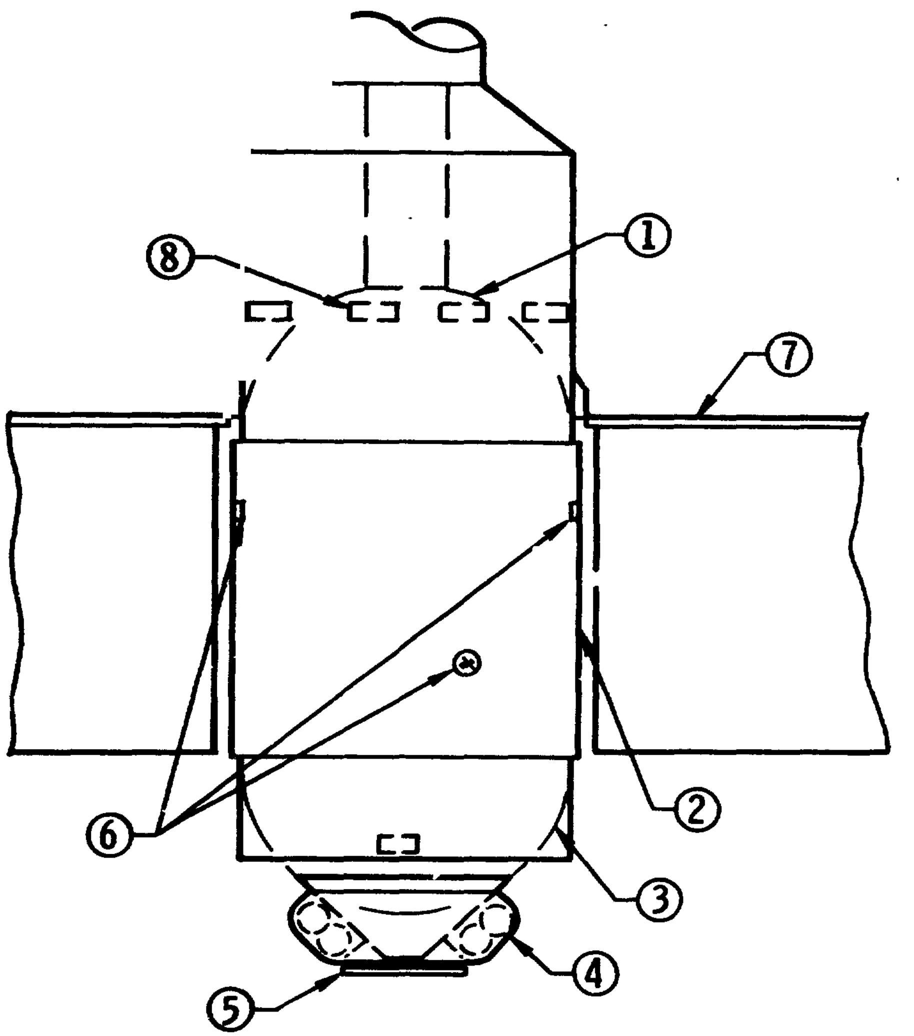
WORKSHOP FLIGHT SYSTEMS



(1) FORWARD DOME
(2) TANK CYLINDER
(3) AFT DOME
(4) TACS SHIELD
(5) HSS RADIATOR
(6) VIEW PORT AND AIRLOCKS
(7) SAS
(8) EXTERNAL CONTROL PANEL

Figure 2.2.2.2-1

The forward dome functioned as the forward pressure bulkhead for the crew compartment. It was completely enclosed by the OWS forward skirt and the aft airlock module structures, as shown in Figure 2.2.2.2-2. Although meteoroids could impact any of the five different structures, penetrations of the dome from impacts upon the structure forward of the skirt were assumed negligible because of large distances involved.

The habitation area tank wall was the cylindrical portion of the pressurized compartment. The tank wall was protected over most of its area by the meteoroid shield. This outer cylinder had three cutouts: Two scientific airlock cutouts and one viewing window cutout. These are shown in Figure 2.2.2.2-3. There was also a ring of unprotected tank at both ends of the bumper, as shown in Figure 2.2.2.2-4. The habitation area aft dome and the common bulkhead functioned as the aft pressure wall of the crew compartment. The various shielding barriers are shown in Figure 2.2.2.2-5.

b. Mechanical Interface (Internal and External) - The OWS interfaced with other Skylab components shown in Figure 2.2.2.2-6. The airlock module provided complete shielding of the forward dome from meteoroid debris created by impact forward of the forward dome. The SAS provided partial shielding to the pressurized crew compartment.

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FORWARD DOME STRUCTURAL CONFIGURATIONS

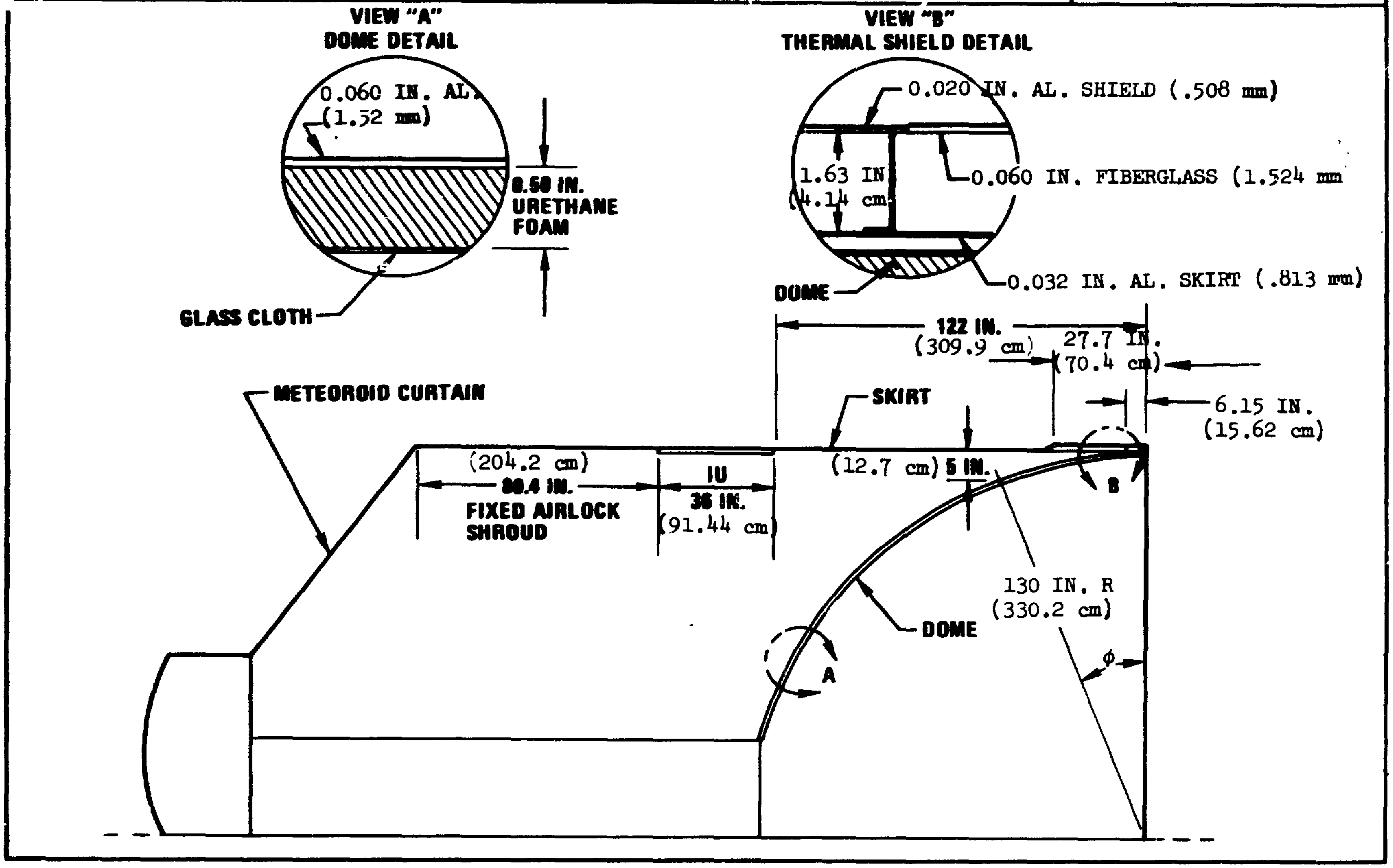

Figure 2.2.2.2-2

SKYLAB - ORBITAL WORKSHOP CUTOUT IMPACT ANGLES

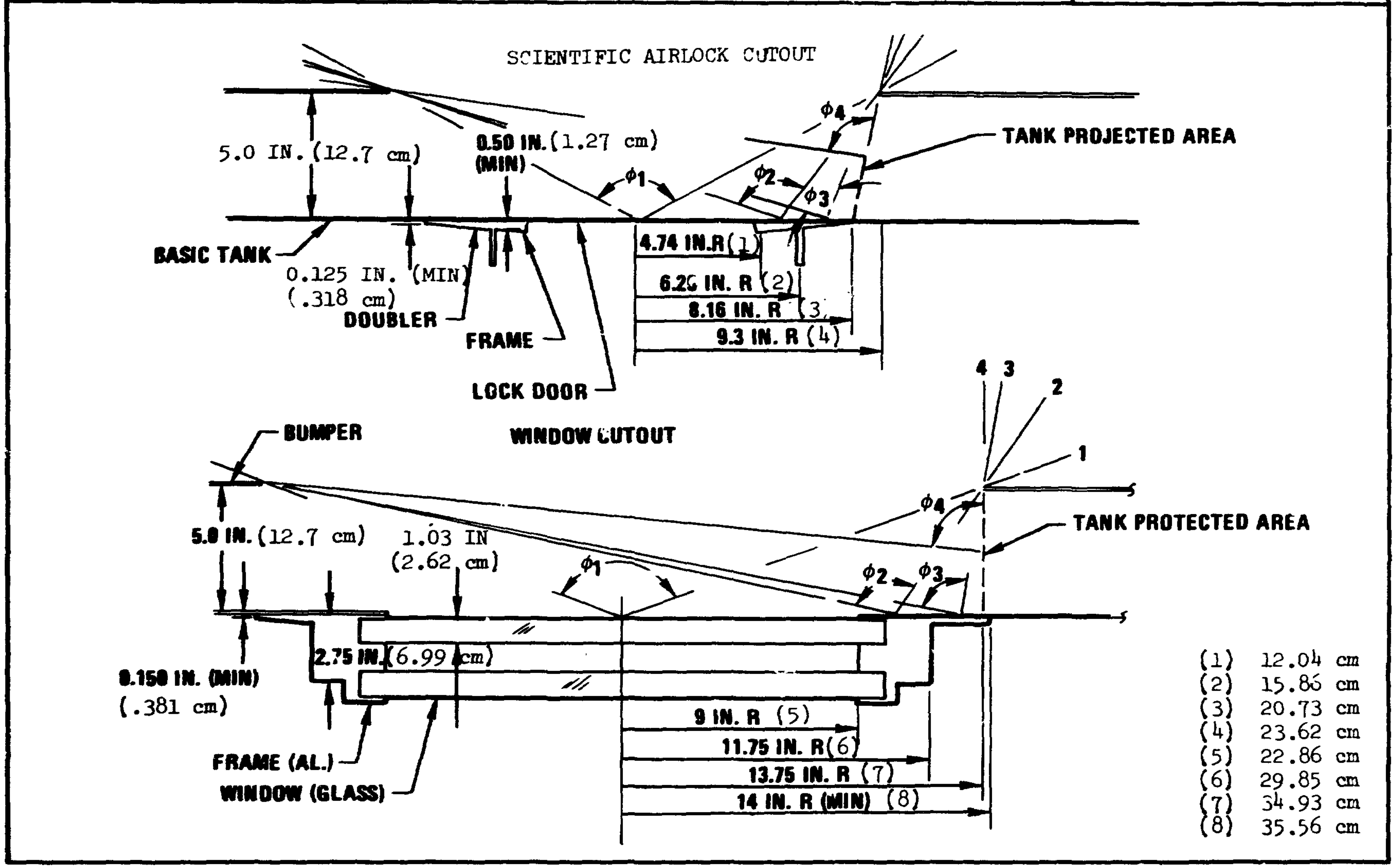

Figure 2.2.2.2-3

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END CLOSURE CONFIGURATION

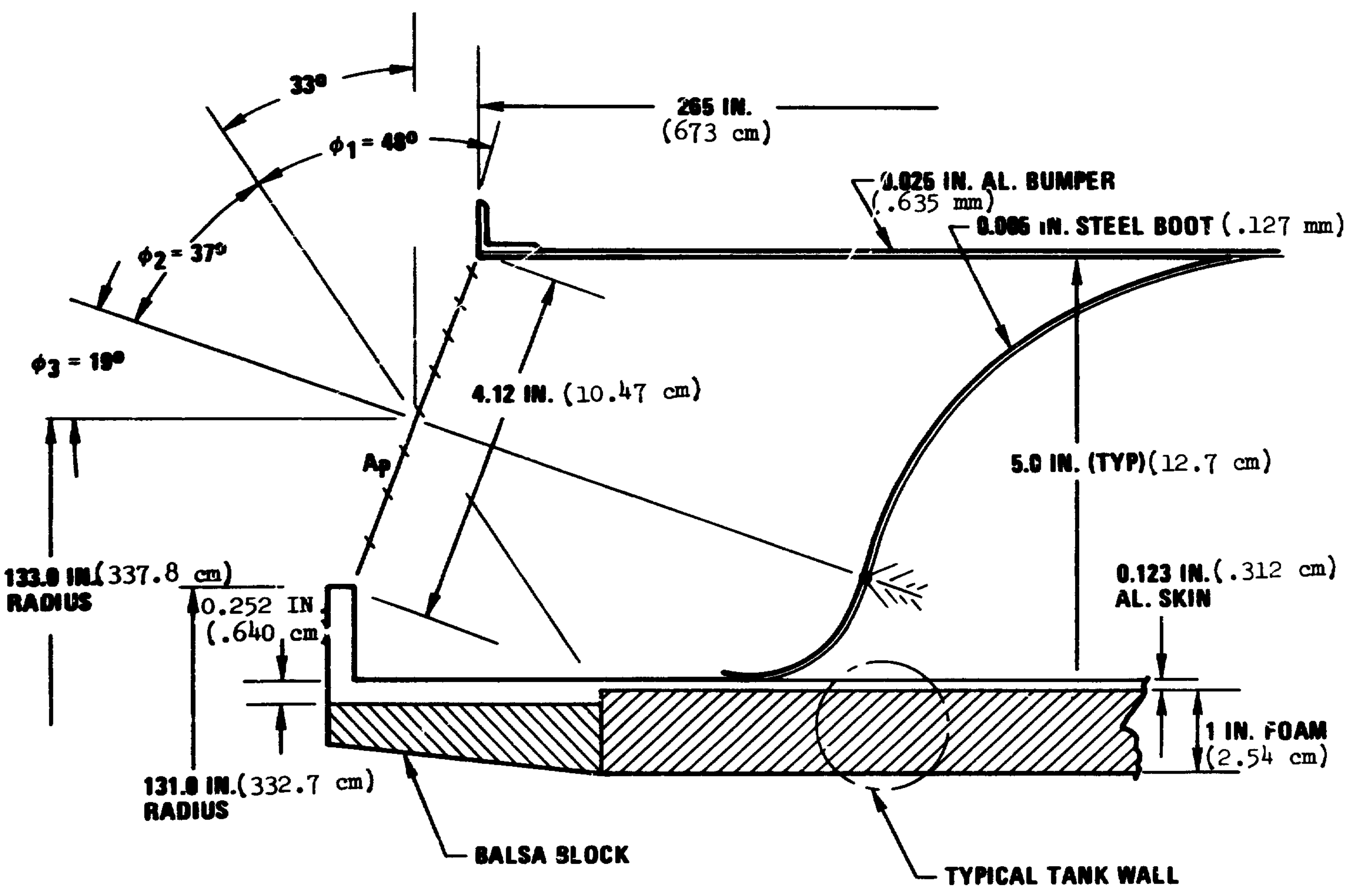
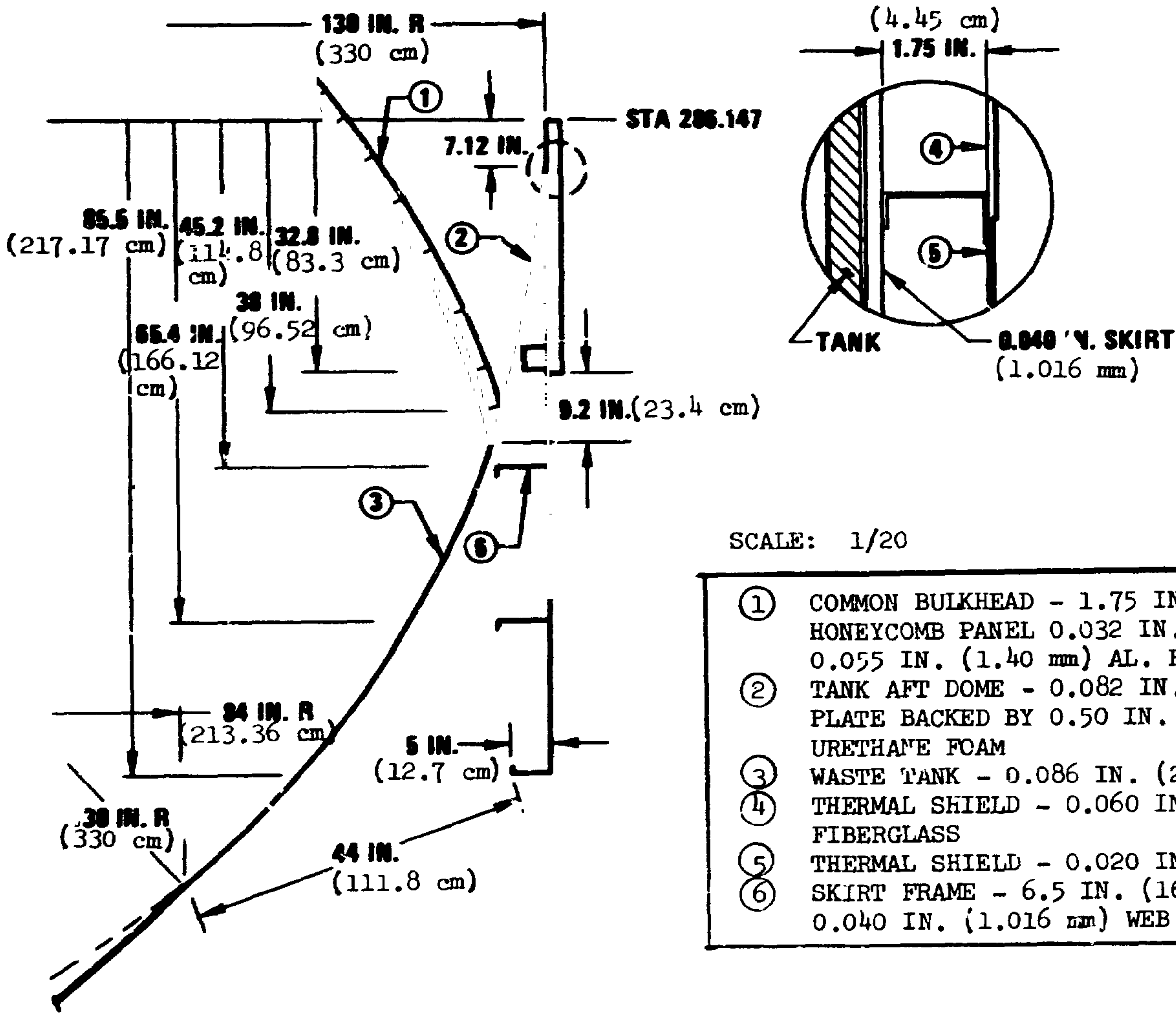


Figure 2.2.2.2-4

SA-4026



1. COMMON BULKHEAD - 1.75 IN. (4.445 cm) THICK HONEYCOMB PANEL 0.032 IN. (.812 mm) AND 0.055 IN. (1.40 mm) AL. FACESHEETS
2. TANK AFT DOME - 0.082 IN. (2.08 mm) AL. PLATE BACKED BY 0.50 IN. (1.27 cm) URETHANE FOAM
3. WASTE TANK - 0.086 IN. (2.18 mm) AL. PLATE
4. THERMAL SHIELD - 0.060 IN. (1.52 mm) FIBERGLASS
5. THERMAL SHIELD - 0.020 IN. (.508 mm) AL.
6. SKIRT FRAME - 6.5 IN. (16.5 cm) DEEP, 0.040 IN. (1.016 mm) WEB

Figure 2.2.2.2-5. Aft Dome ... rt Configurations

Figure 2.2.2.2-5

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SKYLAB CONFIGURATION

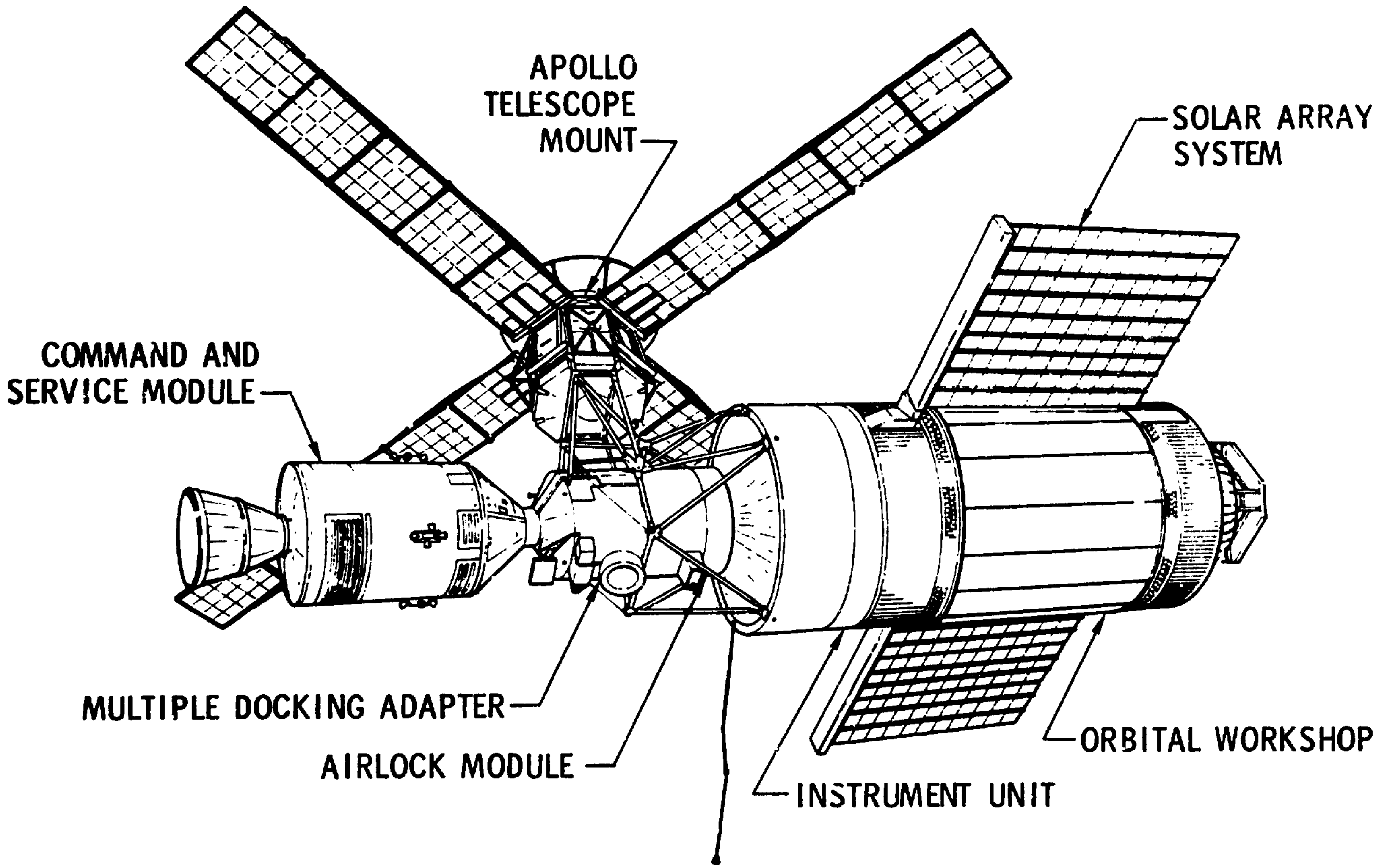


Figure 2.2.2.2-6

c. Application of OWS Test Program Results - Light-gas gun ballistic tests were conducted to evaluate meteoroid shielding capabilities of various OWS subsystem configurations. The environmental model in NASA TMX-53957 specifies that the average meteoroid be of 0.5 gm/cc density and have a velocity of 20 kilometers/second. The available light-gas gun was not capable of launching particles with this velocity, so the test mass was increased by an amount governed by a momentum equivalence equation defined in NASA TMX-53798. In the cases of the habitation area tank meteoroid shield system and viewing window, the actual flight configuration was subjected to simulated meteoroid impacts. In other cases, configurations were used that would give conservative results when test data was applied to actual configurations. The results of this testing are summarized in Table 2.2.2.2-1.

The equivalent meteoroid masses that the test configurations successfully defeated (no penetration of pressure vessel wall) were used in the analysis for computing probabilities of defeating all meteoroid masses defined in the environment model.

d. Analytical Results - The probability of no penetration of the pressurized crew compartment was a function of three parameters: (1) the exposure duration, (2) the exposed area, and (3) the flux of meteoroids with mass greater than that which would cause incipient penetration in the configuration.

Table 2.2.2.2-1

METEOROID PROTECTION TEST VERIFICATION SUMMARY

Line Item	Test Title	Test Requirements	Results
CA-18	Viewing Window	Evaluate window glazing when subjected to hypervelocity impact with a particle representing a micro-meteoroid whose mass was scaled to test velocity.	Glazing defeated particle with no detached spallation off the back surface, thus meeting design requirement.
	Hypervelocity impact testing on targets simulating various areas of habitation tank enclosure	Obtain empirical data to support assessment of OWS meteoroid protection capability by evaluating ability of simulated areas of habitation tank enclosure to defeat a test mass representing a properly scaled meteoroid mass.	Data was satisfactory for use in evaluating OWS.

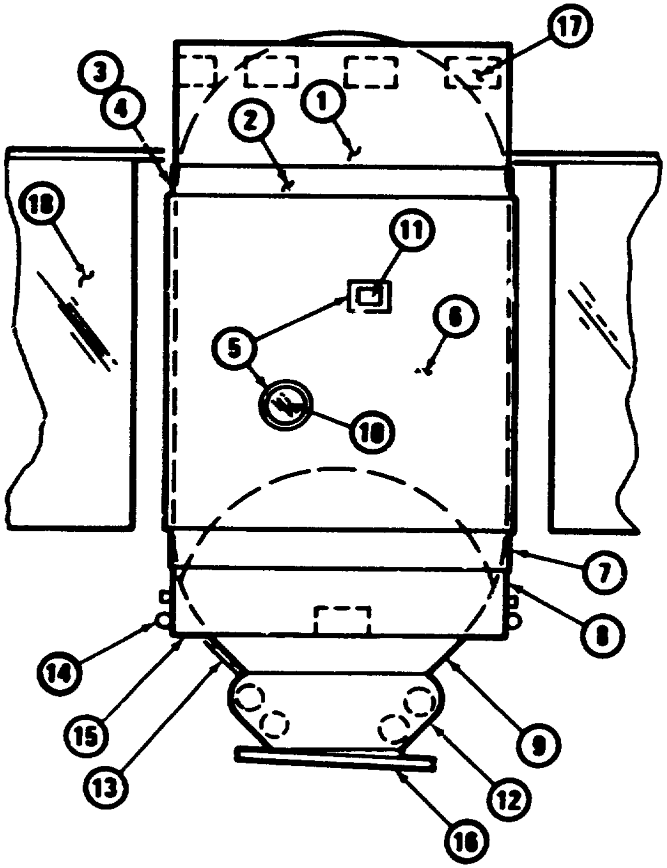
The exposure duration was a known constant of 240 days, representing the worst eight consecutive months of May through December.

Each unique component configuration was analyzed separately, with the overall no-pressure loss probability being the product of component no-failure probability. Thus, the areas used in the analysis were the exposed areas for each unique shielding configuration (defined in Figures 2.2.2.2-1 through 2.2.2.2-5).

The mass that will initiate penetration of the pressure wall was determined from either test data or the Environment and Penetration Criterion by V. C. Frost (Aerospace Corporation Report. No. TOR-269 (4560-40-2)). The number of these masses that will impact an area in a given time (termed the flux) was then specified by the environmental model of NASA TMX-53957.

With the above three quantities known, probabilities were readily computed. The resulting no-penetration probabilities for the structural components which constitute the pressurized crew compartment are summarized in Figure 2.2.2.2-7. The probabilities were based on an exposure duration of 230 days, originally taken as the mission timeline. The No-Pressure Loss (Table 2.2.2.2-2) is revised to account for SAS shielding, total mission duration of 240 days, and shield irregularities of auxiliary tunnel and folded panel.

- 8 MONTH MISSION (230 DAYS)
- 235 NAUTICAL MILE ORBIT
- TMX-53957 ENVIRONMENT



	SUBSYSTEM OR STRUCTURAL SUBASSEMBLY	TEST DATA	P_0
1.	FORWARD DOME (SKIRT IMPACT)	▲	.999344*
2.	FORWARD DOME (THERMAL SHIELD IMPACT)		.999185*
3.	TANK WALL (DIRECT IMPACT)		.998491*
4.	TANK WALL (THERMAL BOOT IMPACT)		.999921*
5.	TANK WALL (DIRECT IMPACT)		.999196*
6.	TANK WALL (BUMPER IMPACT)	▲	.999334*
7.	AFT DOME (THERMAL SHIELD IMPACT)		.999674*
8.	AFT DOME (SKIRT IMPACT)	▲	.999957*
9.	COMMON BULKHEAD (DOME IMPACT)	▲	.999988*
10.	VIEWING WINDOW	▲	.999964*
11.	SCIENTIFIC AIRLOCKS		.999999*
12.	TACS BOTTLE SHIELD		.999264
13.	TACS THRUSTER FEEDLINES		.999711
14.	TACS THRUSTER MODULES (SKIRT IMPACT)		.999924
15.	TACS THRUSTER MODULES (DIRECT IMPACT)		.999685
16.	HSS RADIATOR		.999994
17.	ELECTRICAL PANELS—FWD & AFT (SKIRT IMPACT)		.999734
18.	SAS (90% POWER CAPABILITY)		.999577
19.	TACS CABLES IN AUXILIARY TUNNEL (240 DAYS)		.998207
	OVERALL PRESSURE VESSEL NO—PRESSURE LOSS PROBABILITY*		.99506
	OVERALL OWS NO FUNCTIONAL LOSS PROBABILITY (EXCLUDING WIRE BUNDLES)		.99297

Figure 2.2.2.2-7. Meteoroid Damage Probabilities

Figure 2.2.2.2-7

SKYLAB - ORBITAL WORKSHOP METEOROID PROTECTION STRUCTURAL EVALUATION SUMMARY

(No Pressure Loss)

	BASELINE ANALYSIS	BASELINE ANALYSIS + SAS SHIELDING	BASELINE ANALYSIS + SAS SHIELDING + SHIELD IRREGULARITIES	BASELINE ANALYSIS + SAS SHIELDING + SHIELD IRREGULARITIES + 240-DAY MISSION	REMARKS
BASELINE	0.99506				OWS STRESS ANAL.RPT VOL III 10.5.11-2
SAS SHIELDING	0.99539	0.99539			OWS STRESS ANAL.RPT VOL III 10.5.9-4
SHIELD IRREGULARITIES	0.99497	0.99530	0.99530		OWS STRESS ANAL.RPT VOL III 10.5.10-23
240-DAY MISSION	0.99485	0.99521	0.99509	0.99509	OWS STRESS ANAL.RPT VOL I 5.4-4

Table 2.2.2.2-2

It is seen that the final probability of no-pressure loss exceeds the requirement of 0.995.

2/ Ordnance Deployment Subsystem

See 2.2.2.1 B4/ and 5/.

3/ End Closures and Latching Mechanisms

See 2.2.2.1B2/.

4/ Instrumentation

The instrumentation of the meteoroid shield consisted of three breakwires across the ordnance strap assembly, four deployment magnetic switches at the main tunnel (See Figure 2.2.2.1-7), and 16 torsion bar strain gages.

The deployment magnetic switches were proximity switches located on either side of the OWS main tunnel with two of the switches on the forward end and two on the aft end. The switches closed independently of each other when the meteoroid shield deployed.

The strain gages were located on each of the torsion bars which deployed the meteoroid shield. There were 16 strain gages, one for each torsion bar. A measurement of zero degrees rotation corresponds to the meteoroid shield secured position.

5/ External Coatings

The meteoroid shield was painted with Lusterless Black Paint per STM 0537-01, except local areas of S-13G white paint. These areas are shown in Figure 2.2.2.2-8.

C. Testing - See 2.2.2.1C, 2.2.2.2B 1/c. and Table 2.2.2.2-1.

D. Mission Results - See 2.2.2.1D.

E. Conclusions and Recommendations - See 2.2.2.1E.

F. Development History - See 2.2.2.1F.

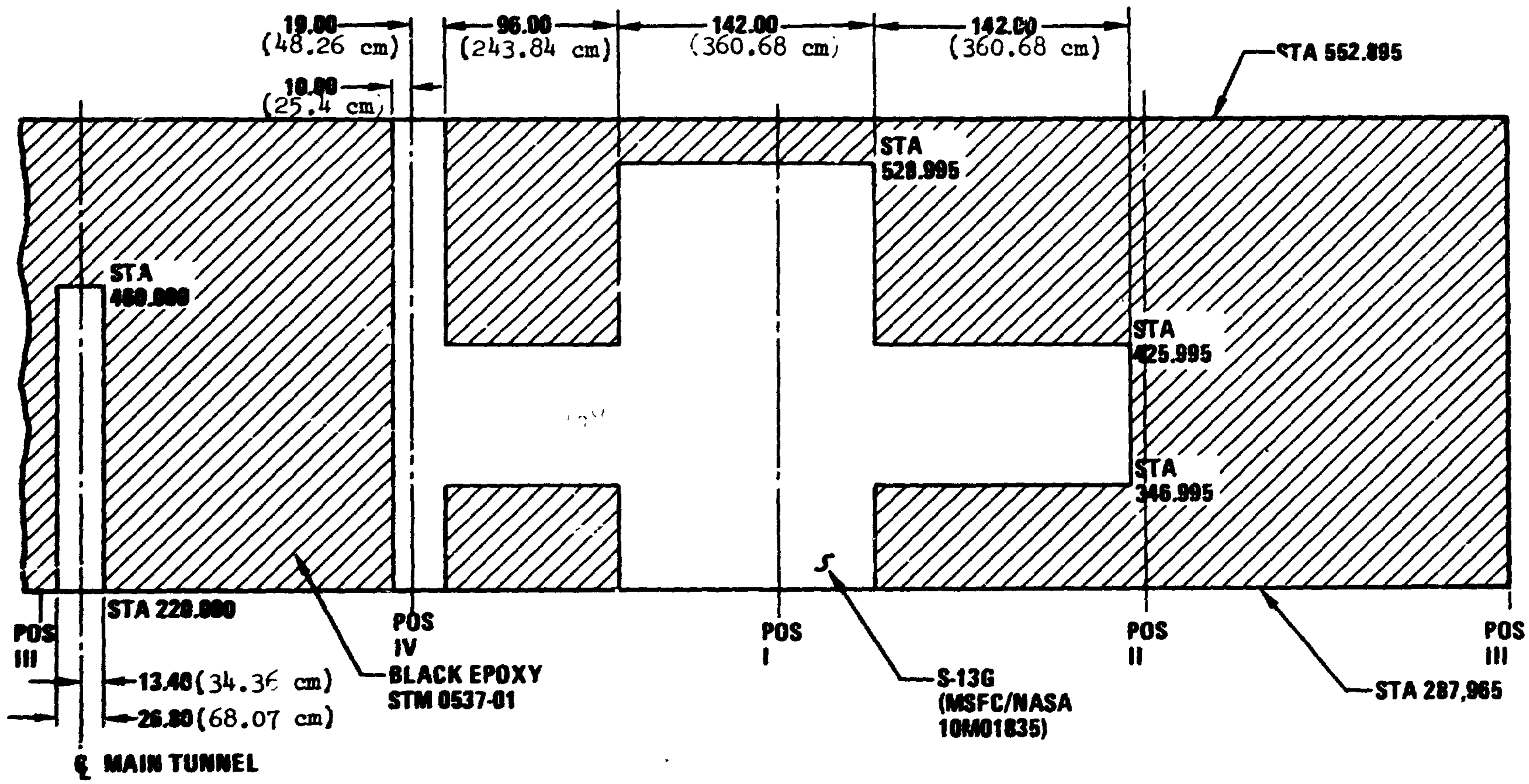


Figure 2.2.2.2-8. Meteoroid Shield Paint Pattern

Figure 2.2.2.2-8

2.2.3 Environmental/Thermal Control Subsystem (E/TCS)

The E/TCS design was based upon passive thermal control of the OWS environment with augmentation by convective heating and cooling of the atmosphere during manned phases and radiative heating of the internal structure during unmanned phases. The E/TCS was thus made up of two basic subsystems: i.e., an active thermal control subsystem including ventilation and a passive thermal control subsystem. The atmospheric gas was a mixture of 70 percent oxygen and 30 percent nitrogen at 5 psia (3.45 N/cm^2) during habitation.

The active thermal control subsystem provided continuous control of the OWS internal environment during periods of astronaut habitation. The cabin gas temperature was to be controlled by cabin gas heat exchangers in the Airlock Module (AM) and by the convective heaters in the three VCS ducts. Reconditioned and purified gas from the AM was to be mixed with recirculated gas in the OWS. Prior to habitation, radiant heaters were to be used to maintain temperature above the minimum levels that satisfied food and film storage requirements.

The passive thermal control subsystem design relied on optical property control of the OWS interior and exterior surfaces, high performance insulation (HPI) on the forward dome, polyurethane insulation lining the inside of the OWS pressure shell and heat pipes attached to structural penetrations of the interior insulation. The exterior surface finishes and the HPI blanket were to control the net energy balance between the OWS and the external space environment. The heat transfer rates from the habitation area to the meteoroid shield, and from the forward and aft dome areas, were to be regulated by finish control of the respective surfaces. Also, the interior habitation area wall temperature were to be made more uniform with optical property control of these surfaces and with the heat pipes.

With the loss of the meteoroid shield, the passive thermal control system could not function as designed since the shield surface was a primary control for the direct solar inputs. The erection of the parasol sunshade in-orbit on DOY 147 and later the sail sunshade on DOY 218 provided passive control approaching the original design levels, but biased toward higher internal temperatures as a result of higher external heat inputs. Because of the warmer habitation area environments, the convective heaters in the VCS ducts were never utilized in orbit. The radiant heaters were only activated once, which was shortly after insertion on DOY 134 before it was realized that meteoroid shield had been lost.

2.2.3.1 Design Requirements

A. Active Environmental/Thermal Control System (E/TCS)

The E/TCS design requirements were defined as follows:

1/ Comfort Conditions

During the habitation period following activation, the OWS atmosphere and mean radiant wall temperature shall be controlled to produce the conditions (on a 24-hour average basis) within the zone shown in Figure 2.2.3-1, while maintaining the performance noted in Table 2.2.3-1. The following periods are exempt from the thermal control requirements:

a. EVA/IVA including three hours of post-EVA/IVA

b. During and after mole sieve bakeout (Ref. ICD 13M02519)

c. During and 7.5 hours after the Z-LV (E) Earth resources maneuvers

The comfort conditions shall be met considering the following parameters:

a. Minimum OWS electrical power dissipated as heat internal to environment	525 W continuous
b. Maximum sensible metabolic rate	1000 Btu/hr (293 W)
c. CLO value variation	0.35 to 1.0 CLO*
d. Minimum Humidity	0.018 specific

* 1 CLO = 0.88°F-ft^2 - hr/Btu (0.155 K-m^2/W)

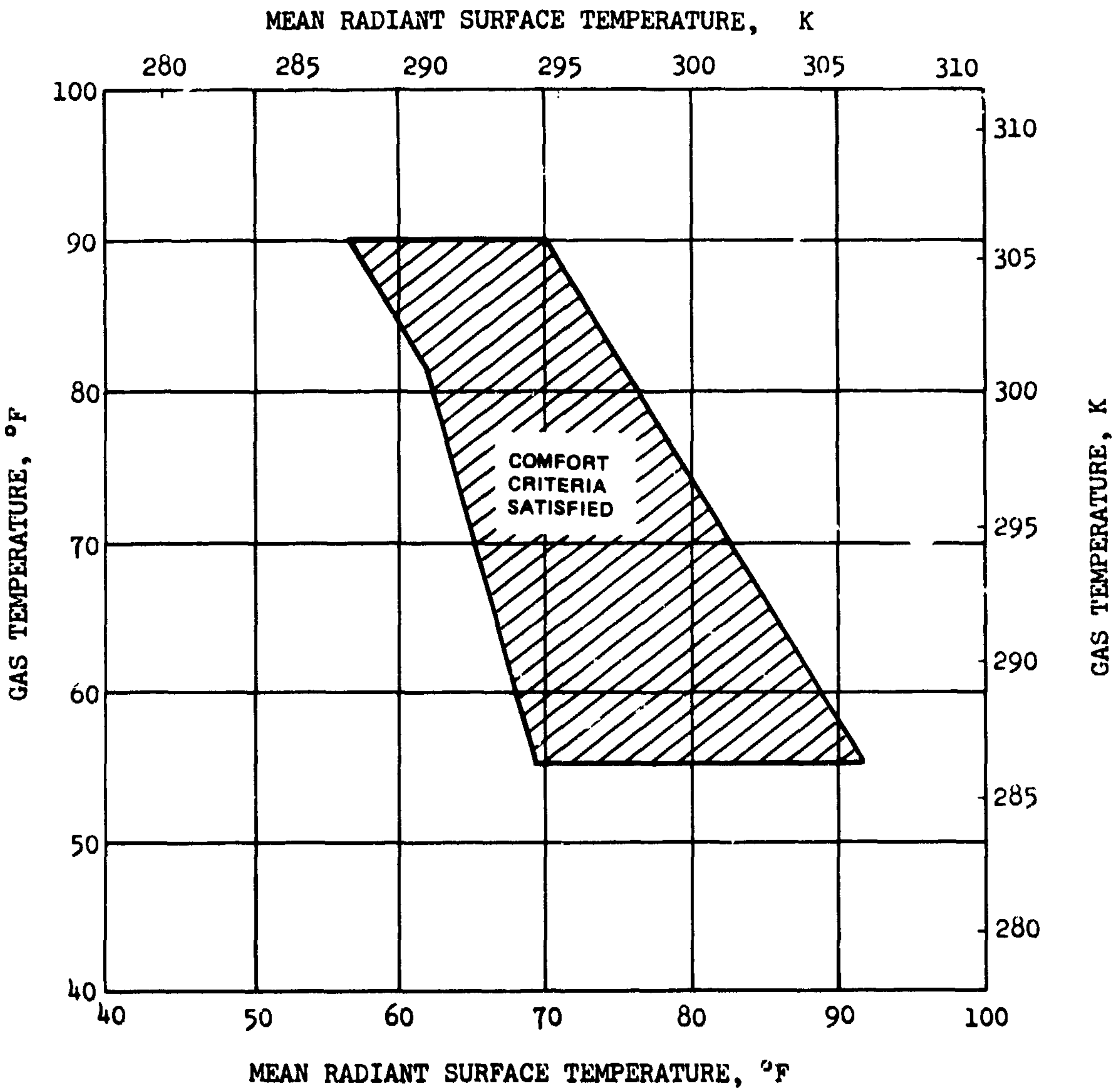


FIGURE 2.2.3-1 CREW COMFORT CRITERIA

Table 2.2.3-1

THERMAL CONTROL SYSTEM PERFORMANCE SUMMARY

Condition	Beta Angle Range	Maximum Average Duct Heater Power for Cold Case*, Watts	Minimum Electrical Heat Removal for Hot Case*, Watts
Nominal	$\|\beta\| < 60$ Deg	825	1,300
Nominal	$\|\beta\| > 60$ deg	825	850
2-Sigma	$\|\beta\| < 60$ deg	1,170	1,050
2-Sigma	$\|\beta\| > 60$ deg	1,170	550

*24-Hour average capability

e. Maximum humidity — 95 percent relative

f. Internal surface temperature — 55 to 105°F (286 to 314 K)

g. Atmospheric velocity requirements — 15 to 100 ft/min (4.6 to 30.5 m/min)

h. Angle variation between orbit plane and sun line — -73.5 to 73.5 deg

i. System Performance Summary — (Table 2.2.3-1)

The internal surface temperature deviations to the requirement of 55 to 105°F (286 to 314 K) noted previously will be as follows:

Equipment	Calculated Maximum Temperature °F	Calculated Maximum Temperature K
Forward Compartment Instrumentation Panel (Front Surface)	120	322
Waste Management Compartment Control and Display Panel (Front Surfa..)	120	322
Wardroom Control and Display Panel (Front Surface)	115	320

Equipment	Calculated Maximum Temperature °F	K
Power and Display Console, Circuit Breaker Panel No. 2 (Area around duct heaters circuit breakers)	110	317
Power and Display Console, Control and Display Panel No. 2		
a. Area around duct heaters switches	120	322
b. Area around tape recorder lights	115	320
Power and Display Console Structure (Lower right hand side panel, approximately one foot above floor)	110	317
Pressure Transducers mounted on water tanks support structure, near Position Plane II (Transducers Surface)	125	325
Digital Display Unit (GFP)(Mounting bracket)	120	322
Viewing Window (Pane inboard surface)	127	326
Fire Detection System Control Panel (GFP) (Mounting interface)	115	320

Equipment	Calculated Maximum Temperature °F	K
Portable Light Housing (Assembly exterior surface)	118	321
Food Reconstitution Water Heater Quick Disconnect Cap (Quick Disconnect cap external surface)	135	330
Hygiene Water Heater Quick Disconnect Cap (Quick disconnect cap external surface)	115	320
Interior Light Housing (Assembly grid enclosing light)	123	324
Food Table (Area adjacent to table interface with food trays)	130	328
Waste Processor Control and Display Panel Advisory Lights	130	328

2/ Noise Control

For orbital operation, the OWS systems identified below shall be designed to limit the sound pressure levels (volume average) in the OWS to the values listed below and 74 db overall when the summation of these sources is considered at any given time. These values do not apply within 3 feet of any sound source.

Frequency Range (Hz)	Sound Pressure Levels
37.5 to 75	70 db
75 to 150	Decreasing 10 db/octave

Frequency Range (Hz)	Sound Pressure Levels
150 to 300	60 db
300 to 600	Decreasing 5 db/octave
600 to 1,200	55 db
1,200 to 2,400	55 db
2,400 to 4,800	55 db
4,800 to 9,600	Increasing 5 db/octave
9,600 to 19,200	60 db

The OWS systems to which the above requirements collectively apply are:

Nomenclature	Quantity
Fan Clusters	3
Odor Fan	1
Fecal Collector	1
Refrigeration System	1
Portable Fans	3

Where possible, preference shall be given to reducing interference with voice communications (600 to 4,800 Hz) provided that suppression methods shall not cause any of the other above limits to be exceeded.

3/ Gas Distribution Components

A duct system which extends from a mixing chamber to the mixing volume aft of the ceiling shall be provided. The mixing chamber shall connect to the AM ECS supply duct. The gas transmission capability of the system shall be sufficient to ensure internal OWS gas velocities which are consistent with CO_2 removal and comfort condition requirements. Diffusers and outlets shall be provided to supply gas velocity and distributions which are consistent with CO_2 removal and crew comfort requirements. All circular diffusers shall be located on the upper (forward) surface of the crew quarters floor.

Manually adjustable flow dampers shall be installed in each of the circular diffuser outlet standpipes in the OWS to facilitate adjustment of the gas flow. Filtering to keep hair, lint, etc., from entering the duct system shall be provided.

4/ Duct Fan Clusters

Mounting provision shall be provided to permit installation of Apollo post-landing ventilation (PLV) fans. The mounting provisions shall be designed to limit the dynamic input to each fan/shroud assembly to the profile as shown in Figure 2.2.3-2. The fans (GFP) shall provide the force required for sufficient air movement throughout the OWS. Sound suppression shall be provided as required to meet the limits specified under A.2/ Noise Control.

5/ Portable Fans

The portable PLV fan shrouds shall be modified to permit installation of dovetails (GFP) for the placement of the portable PLV fans (GFP) using universal mounts (GFP). Sound suppression devices shall be added to the fans in order to limit the noise level to that specified under A.2/ Noise

SKYLAB - ORBITAL WORKSHOP
MAXIMUM DYNAMIC INPUT TO PLV FAN/SHROUD
ASSEMBLY (IN ALL THREE AXES)

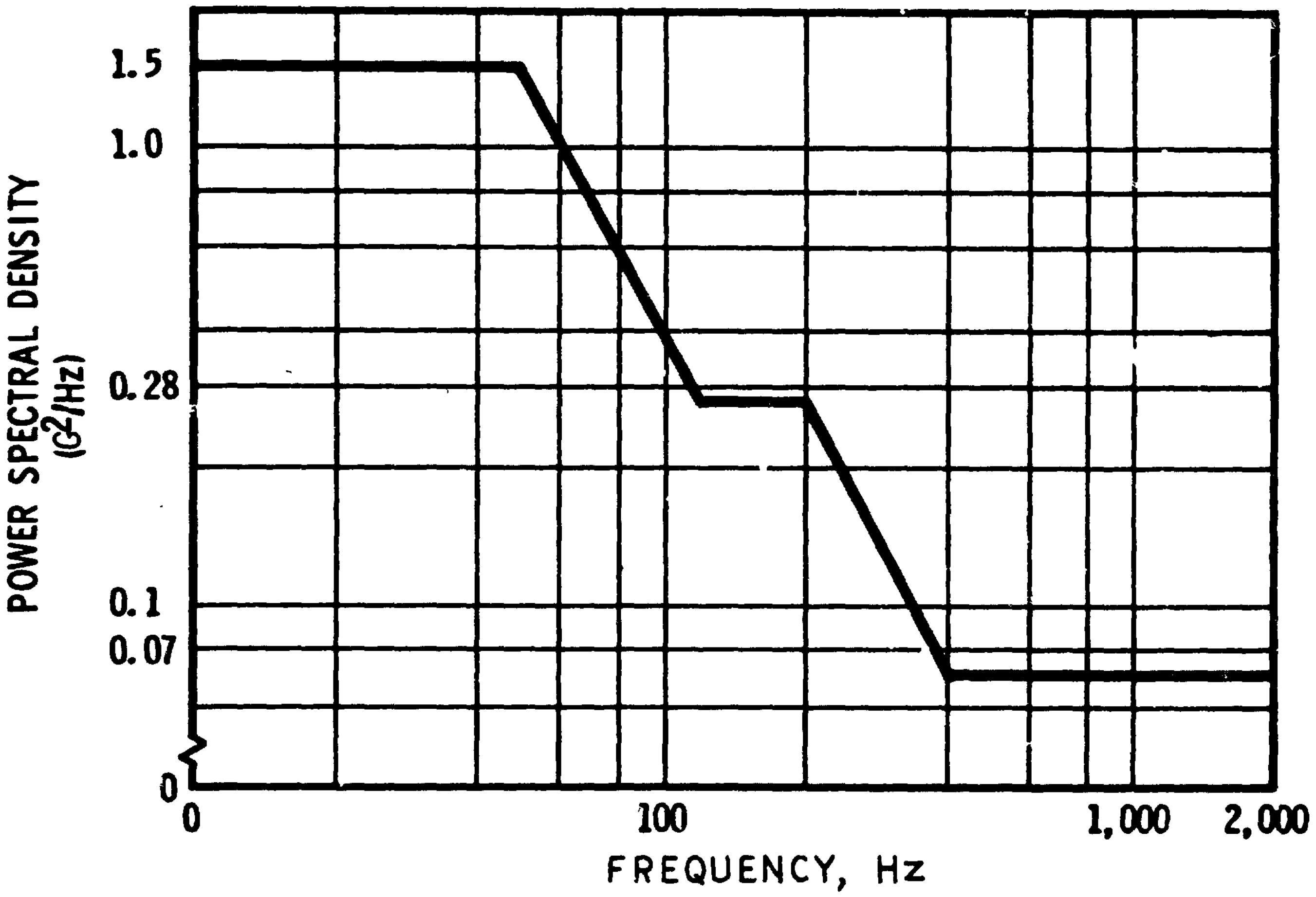


Figure 2.2.3-2

Control. The portable fans shall employ an adjustable diffuser. Protection devices for the fan inlets and outlets shall be provided.

6/ Duct Heaters

Duct Heaters shall provide a total heating capacity of 1,500 watts. Heater power of 1,000 watts shall switch in increments of 250 watts in the automatic mode. The other 500 watts shall be controlled manually in increments of 250 watts. High-limit thermostats shall be installed to provide heating element protection.

7/ Radiant Heaters

Radiant heaters shall be provided for warm-up of the habitation spaces prior to occupancy. Additionally, the radiant heaters shall be capable of being energized or de-energized as required to maintain the temperature of the food and film between 40°F (278 K) and 85°F (303 K) during uninhabited orbital periods. The heaters shall have an output of 1,000 watts.

8/ Temperature Control

A temperature control subsystem shall be provided which is capable of automatic and manual operation. In either mode of operation the subsystem shall meet the following requirements:

a. The subsystem shall provide crew quarters atmosphere temperature monitoring for display and telemetry.

b. Visual displays of heater operation shall be provided.

c. The subsystem shall be capable of sensing and deactivating a faulty heater. Heater failure detection shall be based upon a suitable sensing device in the electrical network.

The temperature control subsystem shall have an automatic mode of operation which shall meet the following additional requirements:

a. The subsystem shall be capable of monitoring the crew quarters atmosphere temperature and selectively operating heaters or heat exchangers to regulate the OWS atmosphere temperature.

b. A temperature selection control shall be provided which will allow temperature selection over a range of 60°F to 90°F (289 K to 306 K). This control shall be provided by either 1°F (0.56 K) increments or by means of an analog-type device.

A manual override control shall be provided which shall enable the crewman to select individual on-off control of the heaters and heat exchangers.

9/ Wardroom Window Heater

A wardroom window heater shall be furnished to satisfy the requirement for a window defogger.

10/ Airlock Module/OWS Interface

MSFC ICD 13M02519, Revision B, dated 27 January 1971, Zone 21, defines the centerpoint of the "Interface OWS to ECS Duct" as 13.1 inches (33.0 cm) below the nominal interface plane (V.S. 3230.311/AM-11.454) at a radius of 31.1 inches (79.0 cm) from the centerline of the reusable hatch for the duct from the AM whose centerline crosses the nominal interface plane perpendicular to it, and then makes a 65$\pm$ 1 degree bend away from the hatch centerline in the plane formed by it and the centerpoint of the duct interface. In all operating modes the OWS shall transmit through the dome hatch a total net gas flow rate equal to that supplied by the AM including OWS leakage.

The requirements imposed on the OWS E/TCS are shown in Table 2.2.3-2. In addition, conditions imposed on the AM portion of the TCS by the OWS portion are shown for reference.

11/ Experiment and Operational Film to Film Vault Interface

The requirements imposed on the E/TCS by ICD 13M13519 are that the film vault in the OWS must be thermally maintained between 40°F (278 K) and 80°F (300 K) for $|\beta| < 60$ degrees and between 40°F (278 K) and 85°F (303 K) for $|\beta| > 60$ degrees.

12/ Skylab Food to OWS Interface

The requirements imposed on the E/TCS by ICD 13M20926 are that the food storage containers must be thermally maintained between 40°F (278 K) and 85°F (303 K).

Table 2.2.3-2

A/M COOLING TO OWS

Operating Mode	Flow Rate (1) lb/hr (kg/hr)	OWS Return Gas Dry Bulb Temperature °F (K)	AM Sensible Heat Removal from OWS (2) (3) Atmosphere No Crewmen In OWS (4) Btu/hr (W)	All Crewmen In OWS (5) Btu/hr (W)	Remarks
Interchange duct only; no aft Hx fans	203 (92)	60 (6) (289) 72 (296) 83 (7) (302)	-28 (-8.2) -28 (-8.2) -28 (-8.2)	161 (47.2) 372 (109) 566 (166)	Cooling table does not apply during AM molecular sieve bakeout/cooldown operation when gas supply temperature to OWS may be as high as 82°F (301 K) immediately following bakeout, and OWS cooling capability is reduced for a 22-hr period.
Interchange duct plus four aft Hx fans	432 (196)	60 (6) (289) 72 (296) 83 (7) (302)	300 (87.9) 816 (239) 1287 (377)	500 (146.5) 1257 (368) 1951 (572)	The AM aft heat exchanger fans shall be deactivated during EVA/IVA. Cooling table does not apply during AM molecular sieve bakeout/cooldown operation when gas supply temperature to OWS may be as high as 70°F (294 K) immediately following bakeout and OWS cooling capability is reduced for a 22-hr period. See Note (8) for associated reduction in maximum internal OWS heat generation rates.

See notes on following page

Table 2.2.-3-2 (Cont'd)

A/M COOLING TO OWS

NOTES: Gas flow is from AM through ECS ductwork

1. AM gas flow rates based upon 26 volts input to fans and compressors.
2. AM sensible heat removal based upon a cluster latent heat load of 750 Btu/hr (220 W) and gas temperatures equal in the AM and OWS.
3. AM sensible heat removal based upon the solar inertial vehicle attitude.
4. AM condensing heat exchanger gas flow directed to MDA.
5. AM condensing heat exchanger gas flow directed to OWS.
6. OWS return gas low temperature limit, based upon a minimum internal OWS heat generation rate of 1300 W, will be maintained when the gas temperature is 60°F (289 K) and $|\beta| \leq 40$ deg.
7. OWS return gas high temperature limit based upon maximum internal OWS heat generation rates at an OWS gas temperature of 83°F (302 K) not exceeding 1300 W for $40 \leq |\beta| \leq 60$ deg or 800 W for $|\beta| > 60$ deg.
8. OWS return gas high temperature limit based upon maximum internal OWS heat generation rates at an OWS gas temperature of 83°F not exceeding 1250 W for $40 \leq |\beta| \leq 60$ deg or 750 W for $|\beta| > 60$ deg.

B. Passive Thermal Control Subsystem

1/ Insulation

Insulation shall be installed on the forward dome exterior, forward of the debris shield. The insulation shall provide a thermal conductance of 0.02 Btu/hr-ft^2-°F (0.11 W/m^2•K) or less. A dry nitrogen gas purge shall be provided to condition the insulation during prelaunch operations. Within the habitation area, the Saturn S-IVB configuration of internally installed polyurethane insulation shall be utilized. In addicion to this insulation, polyurethane foam shall be added over the plenum surfaces below the crew quarters floor.

2/ Thermal Shielding

Thermal shadowing of the forward and aft joint areas shall be provided. The shield providing the shadowing shall shadow approximately the aft 31 inches (78.7 cm) of the forward skirt and the forward 33 inches (83.8 cm) of the aft skirt. The thermal shields include the skirt thermal shield and the meteoroid shield closures.

3/ Thermal Control Surface Coatings

The thermal control systems shall be capable of controlling to specification limits with a thermal surface absorptivity change of 10 percent.

2.2.3.2 System Description

The E/TCS design was developed from the parameters associated with the ventilation and active and passive thermal control subsystem elements and those shown in Table 2.2.3-3. The extreme values for the solar constant, albedo, and Earth radiation were combined statistically along with the thermal control coating property variation to establish the design conditions.

A. Active Environmental/Thermal Control Subsystem

1/ Ventilation Control

The Ventilation Control Subsystem (VCS) consisted of a gas supply duct, a mixing chamber, distribution ducts containing fan clusters, a plenum, floor diffusers and portable fans. The VCS flow schematic is shown in Figure 2.2.3-3. Conditioned gas was brought from the AM to the OWS by the supply duct, which attached to the mixing chamber located in the forward compartment near the OWS dome. The three distribution ducts were routed from the mixing chamber to the plenum below the crew quarters floor. The OWS bulk gas flow was maintained by a four-fan cluster in each duct. The crew quarters floor was equipped with adjustable diffusers which allowed the air to circulate through the crew quarters and back to the forward compartment, a portion of which then was returned to the AM for conditioning.

Three portable fan units were included in the OWS to provide a localized increase in flow velocity as required by the crew. The performance of the VCS was dependent upon the excitation voltage for the fans, the number of duct and

TABLE 2.2.3-3

Environmental/Thermal Control Subsystem Design Parameters

Parameter	Design Value Minimum	Design Value Maximum
Electrical/Electronic component Heat load (Table 2.2.3-1)	525 W	550 to 1300 W
Sensible metabolic rate (Nominally 650 Btu/hr or 190.6 W)	0	1000 Btu/hr (293 W)
CLO Value	0.35 CLO (0.0543 K·m^2/W)	1.0 CLO (0.155 K·m^2/W)
Humidity	0.018 lb H_2O/ lb dry gas	95% Relative
Beta (Angle between sun line and orbit plane)	0	± 73.5 deg
Heater power requirement	0	1170 W
Atmospheric cooling (Table 2.2-3-2)	0	1950 Btu/hr (571 W)
Solar constant (Nominally 420 Btu/hr-ft^2 or 1352 W/m^2)	410 Btu/hr-ft^2 (1295 W/m^2)	443 Btu/hr-ft^2 (1413 W/m^2)
Albedo (Nominally 0.3)	0.18	0.42
Earthshine (Nominally 75 Btu/hr-ft^2 or 236 W/m^2)	62 Btu/hr-ft^2 (19.6 W/m^2)	88 Btu/hr-ft^2 (278 W/m^2)

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VCS SCHEMATIC

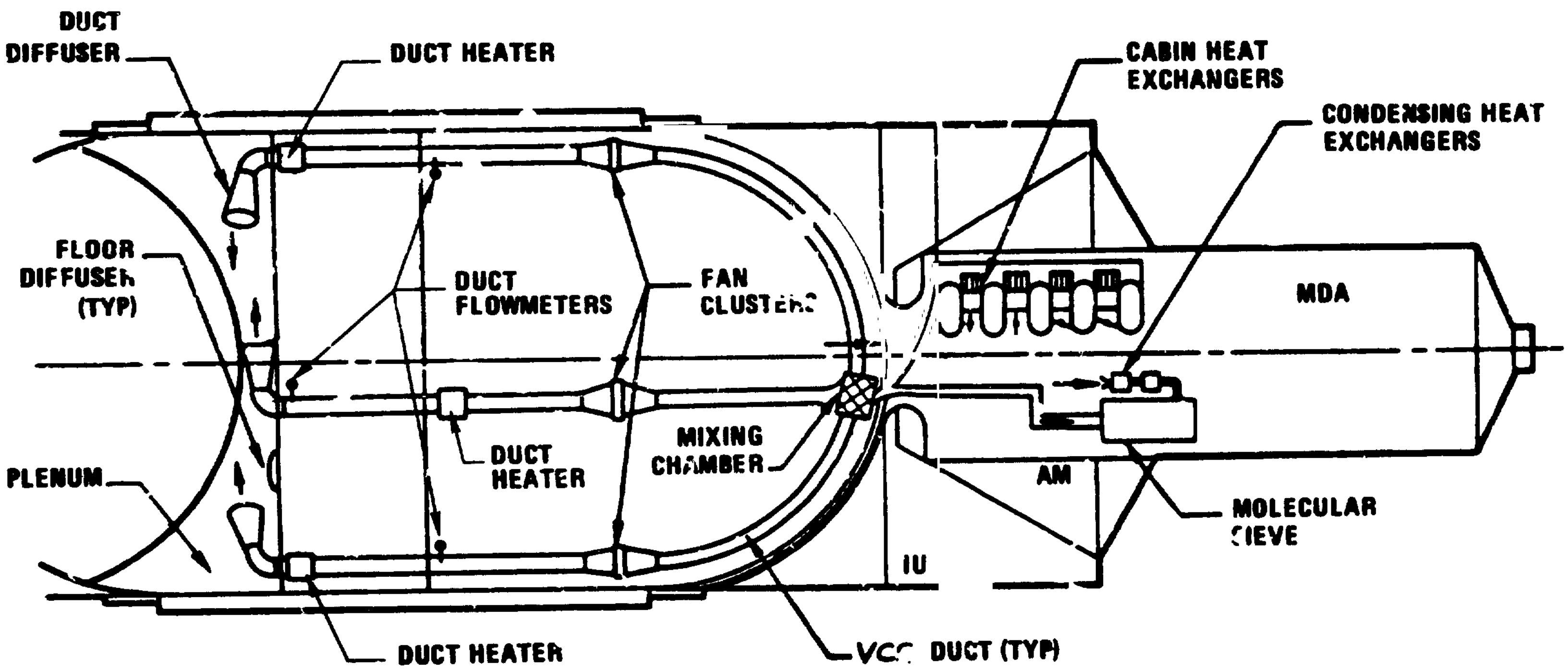


Figure 2.2.3-3

portable fans operating, and the adjustments of the floor diffusers and chambers. The system design operating voltage range was from 24 to 30 Vdc. For normal operation during habitation all 12 duct fans were operating continuously. The three portable fans were operated as designed by the astronauts. Adjustment of the diffusers and dampers was also made by the astronauts as desired. The range of flow anticipated was from 450 to 700 cfm (12.7 to 19.8 m^3/min) per duct.

a. Supply Duct

The supply duct carried conditioned gas from the AM/Environment Control System (ECS) to the VCS mixing chamber (Figure 2.2.3-4). The duct was 8 inches (20.3 cm) in diameter and was designed for a maximum flow of 2.6 lb/min (1.18 kg/min) at a maximum pressure differential across the duct of 0.05 inch (0.127 cm) of water. The conditioned gas from the AM and the recirculated gas from the OWS were combined in the mixing chamber (Figure 2.2.3-4) and fed into the distribution ducts. The recirculated OWS gas entered the mixing chamber through the screens covering five sides of the chamber.

b. Distribution Ducts

The three VCS distribution ducts (Figure 2.2.3-5) supplied gas to the plenum below the crew quarters floor. The fan cluster inlet portion (Figure 2.2.3-6) in each duct was a wire-reinforced two-ply Armalon cloth construction.

SKYLAB - ORBITAL WORKSHOP
AIRLOCK TO WORKSHOP INTERFACE AND
MIXING CHAMBER

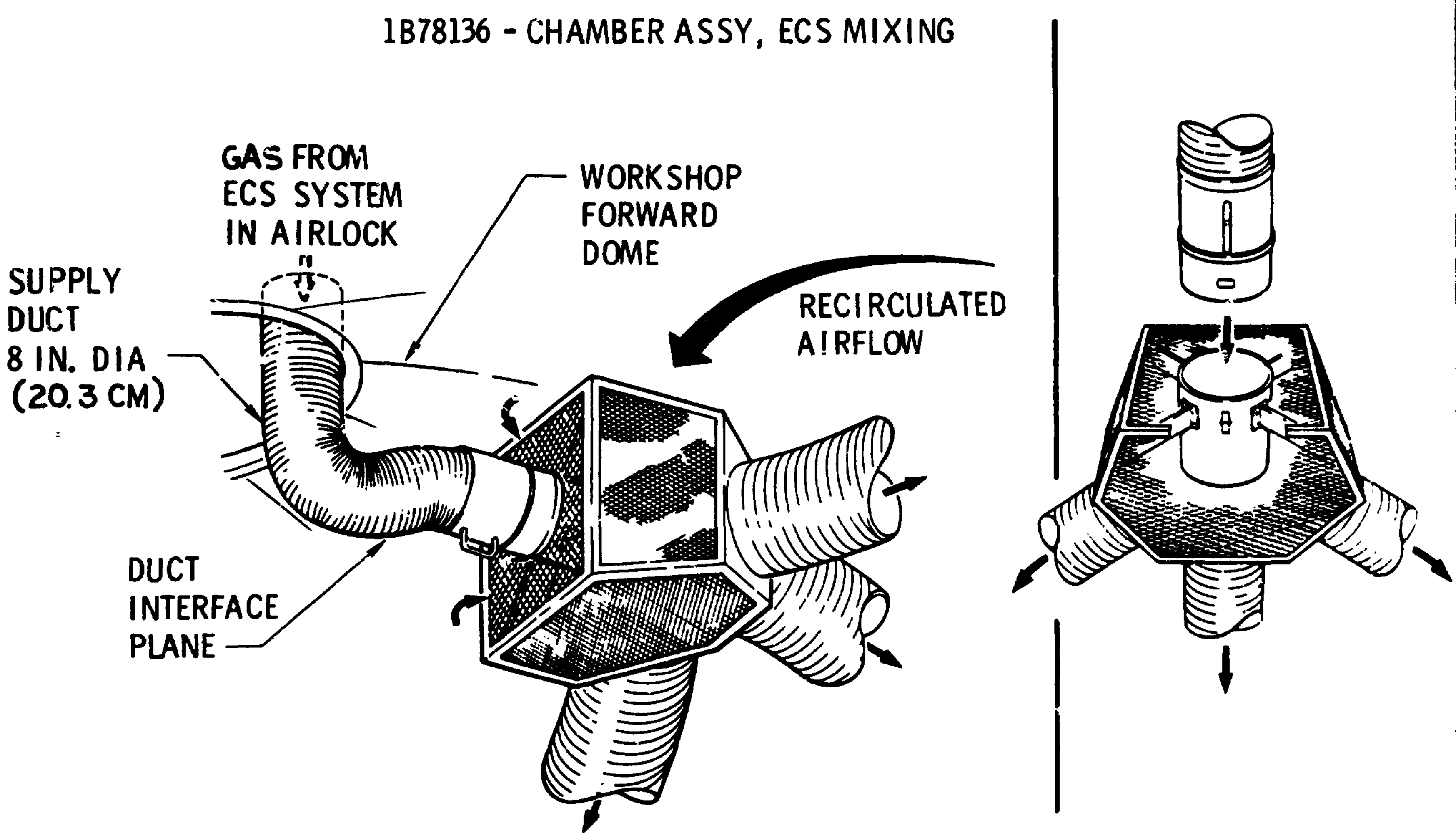


Figure 2.2.3-4

SKYLAB - ORBITAL WORKSHOP
VENTILATION

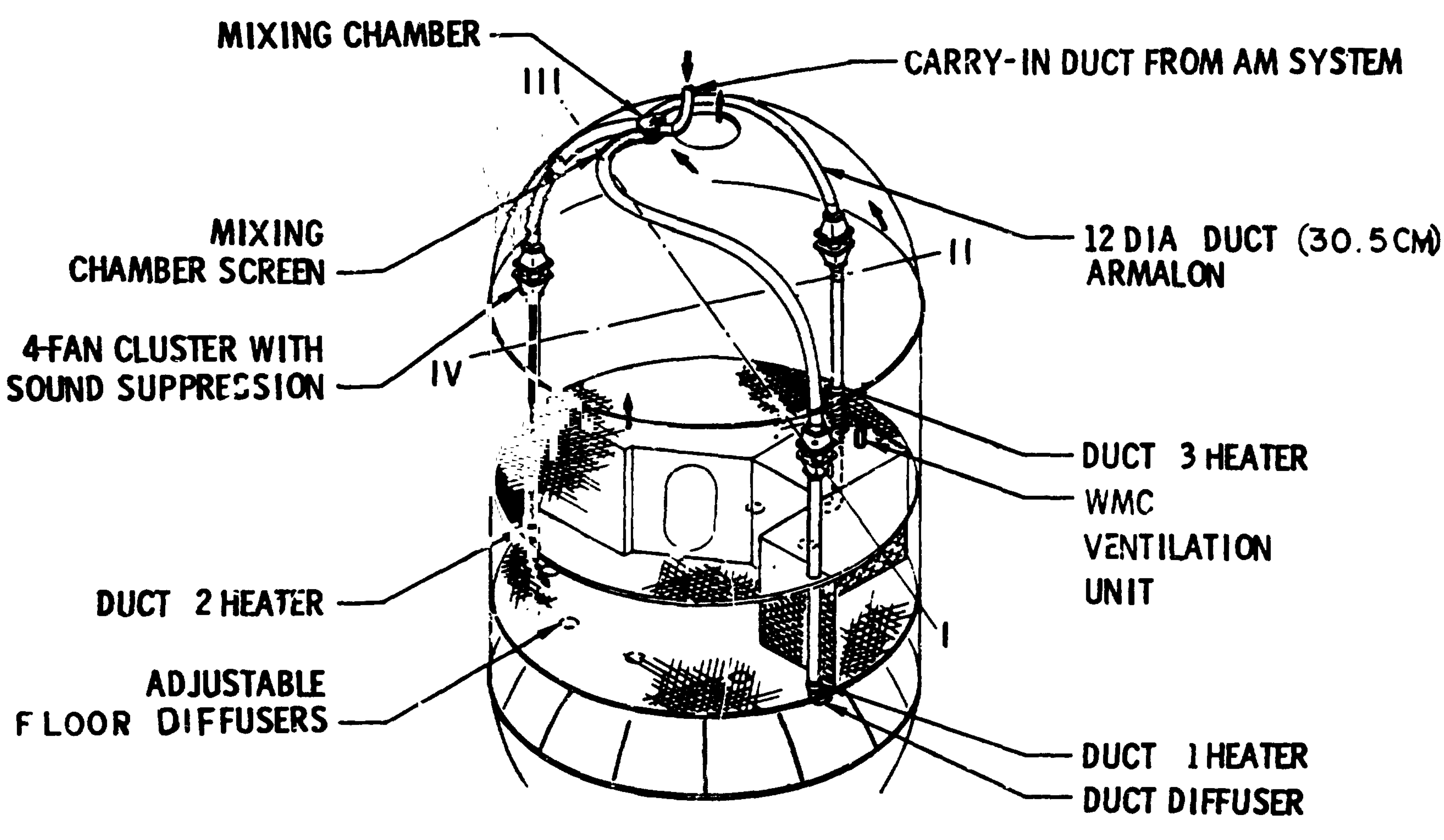


Figure 2.2.3-5

SKYLAB - ORBITAL WORKSHOP
FAN INLET DUCT

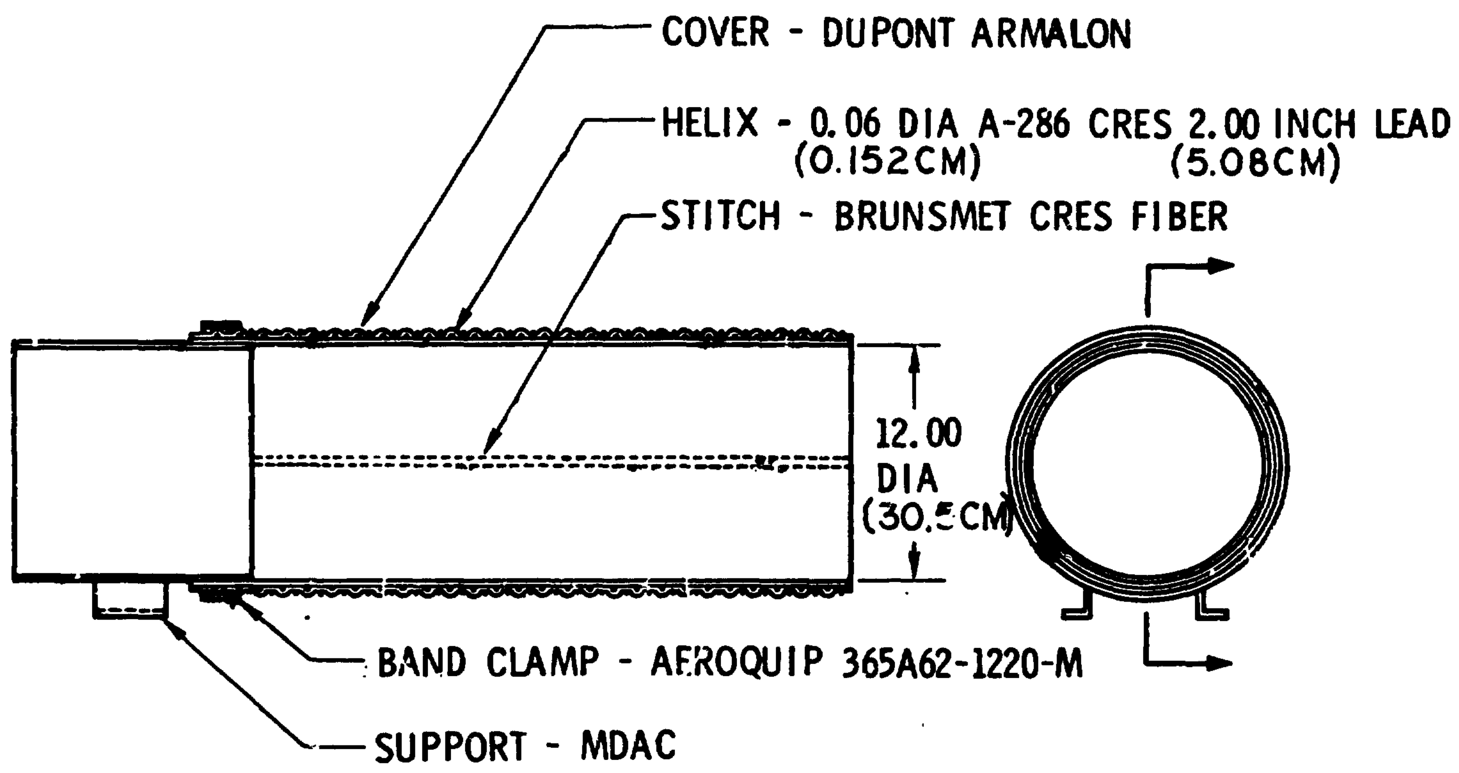


Figure 2.2.3-6

The discharge duct (downstream of the fan cluster) was a single-ply Armalon construction.

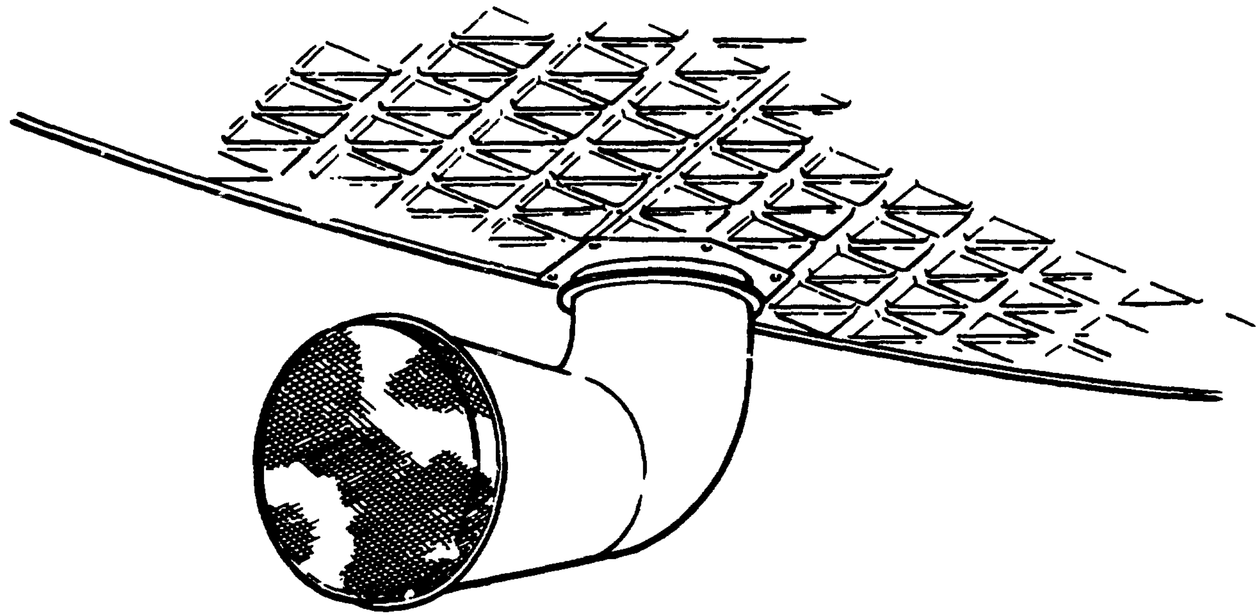
A diffuser (Figure 2.2.3-7) was placed on the end of each duct to direct the flow in the plenum and provide a uniformly mixed gas supply to the floor diffusers.

c. Duct Fan Clusters

The fan cluster (Figures 2.2.3-8 and 2.2.3-9) in each duct consisted of a baffled resonant chamber, an inlet and outlet muffler; and four PLV fans to deliver the conditioned and recirculated gas to the plenum. The fan cluster design provided access to each fan through the four fan access doors for fan replacement.

The cluster fan shown in Figure 2.2.3-10 was designed for easy replacement in the cluster. The fan could be removed by opening the fan access door (Figure 2.2.3-8) and squeezing the handle which retracted the fan retainer pins and released the fan.

SKYLAB - ORBITAL WORKSHOP
DUCT DIFFUSER



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FAN CLUSTER AND MUFFLER ASSEMBLY

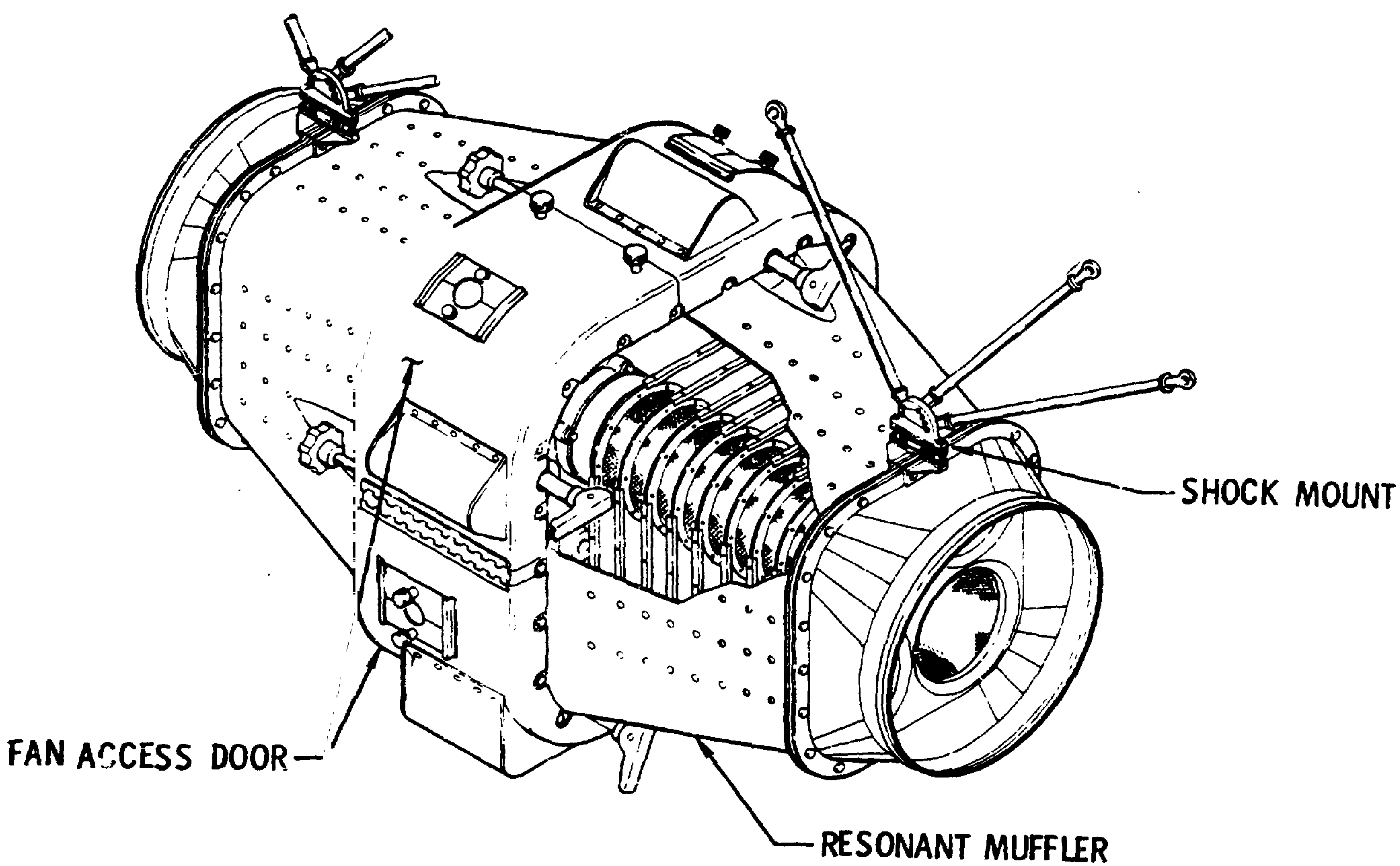
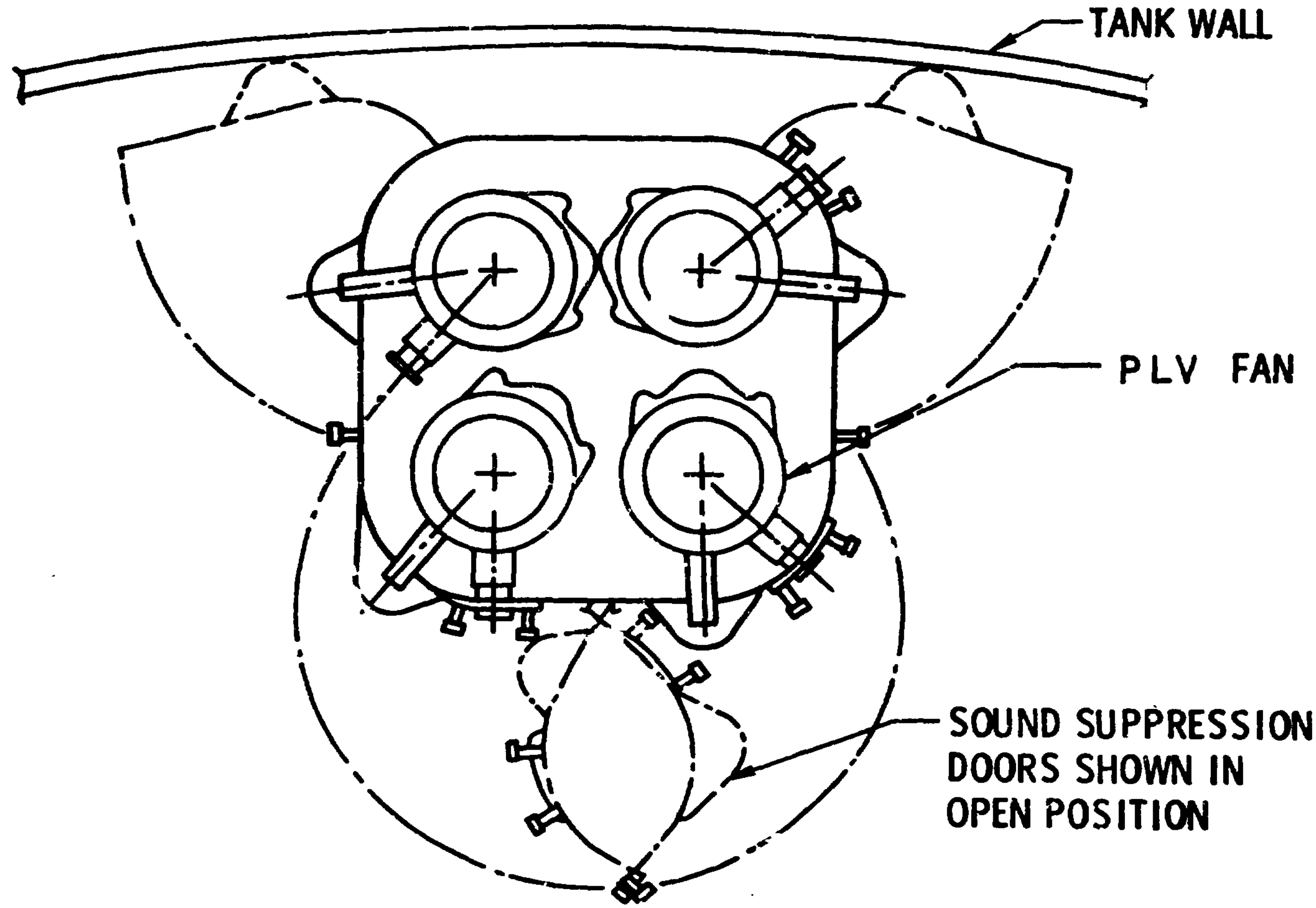


Figure 2.2.3-8

SKYLAB - ORBITAL WORKSHOP
OWS VENTILATION SYSTEM
FAN CLUSTER ASSEMBLY



DUCT FAN

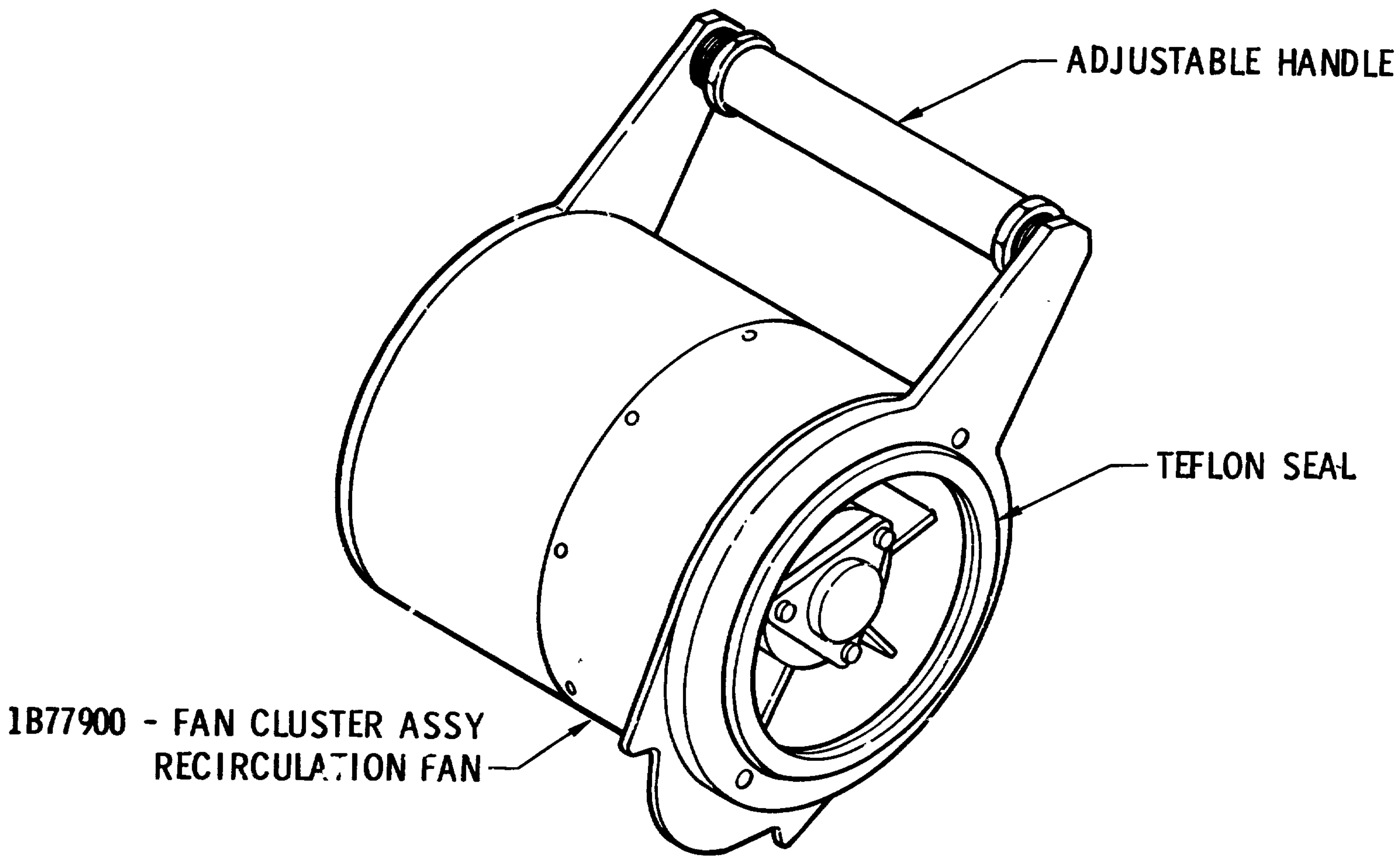


Figure 2.2.3-10

d. Floor Diffusers

The two types of adjustable velocity-profile diffusers are shown in Figures 2.2.3-11 and 2.2.3-12. The circular diffuser is shown in Figure 2.2.3-11, and the locations in the OWS are shown in Figure 2.2.3-13. As shown in Figure 2.2.3-11, the flow through the diffuser was controlled by the position of the damper and of the diffuser vanes. The position of the diffuser vanes was controlled by turning the diffuser vane control to either the narrow or wide flow pattern. The volumetric flow was controlled primarily with the damper control, although a small change in flow could occur when adjusting from narrow to wide pattern.

Figure 2.2.3-12 shows the sleep compartment rectangular diffuser configuration. Flow volume and direction were controlled by manually adjusting the movable diffuser vanes. The locations of the diffusers in the sleep compartment are shown in Figure 2.2.3-13.

e. Portable Fans

The portable fan (Figure 2.2.3-14) utilized the fan in a portable support that could be located on any of the OWS structural grids, the fireman's pole or the handrails, and could be connected to any three ampere utility outlet. This portable support incorporated sound suppression and inlet and outlet screens to protect the fan from debris. The fan operated from a three-way switch giving a high, low, and off position.

SKYLAB - ORBITAL WORKSHOP
FLOOR/AIR DIFFUSER ARRANGEMENT

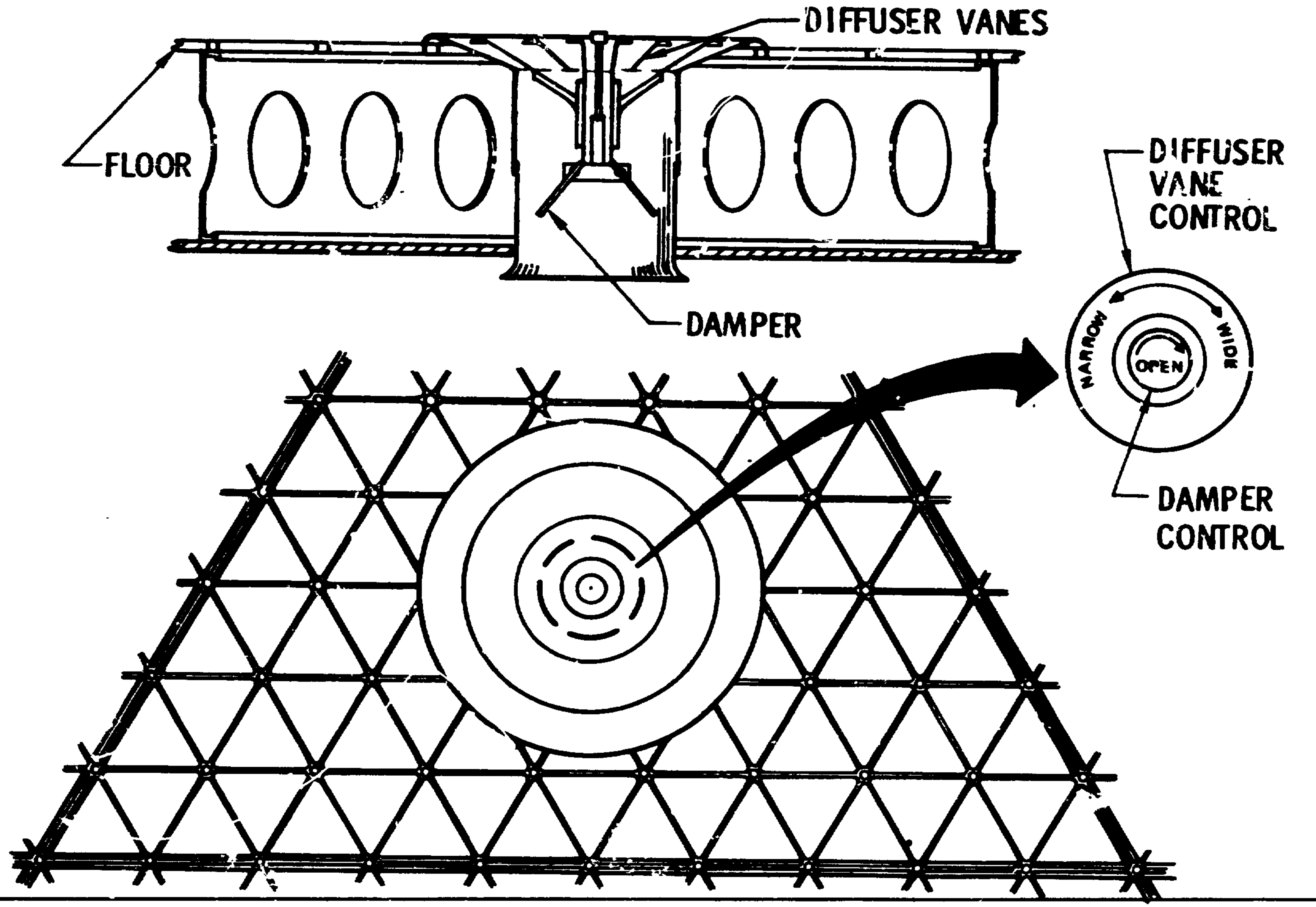


Figure 2.2.3-11

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SLEEP AREA AIR OUTLET

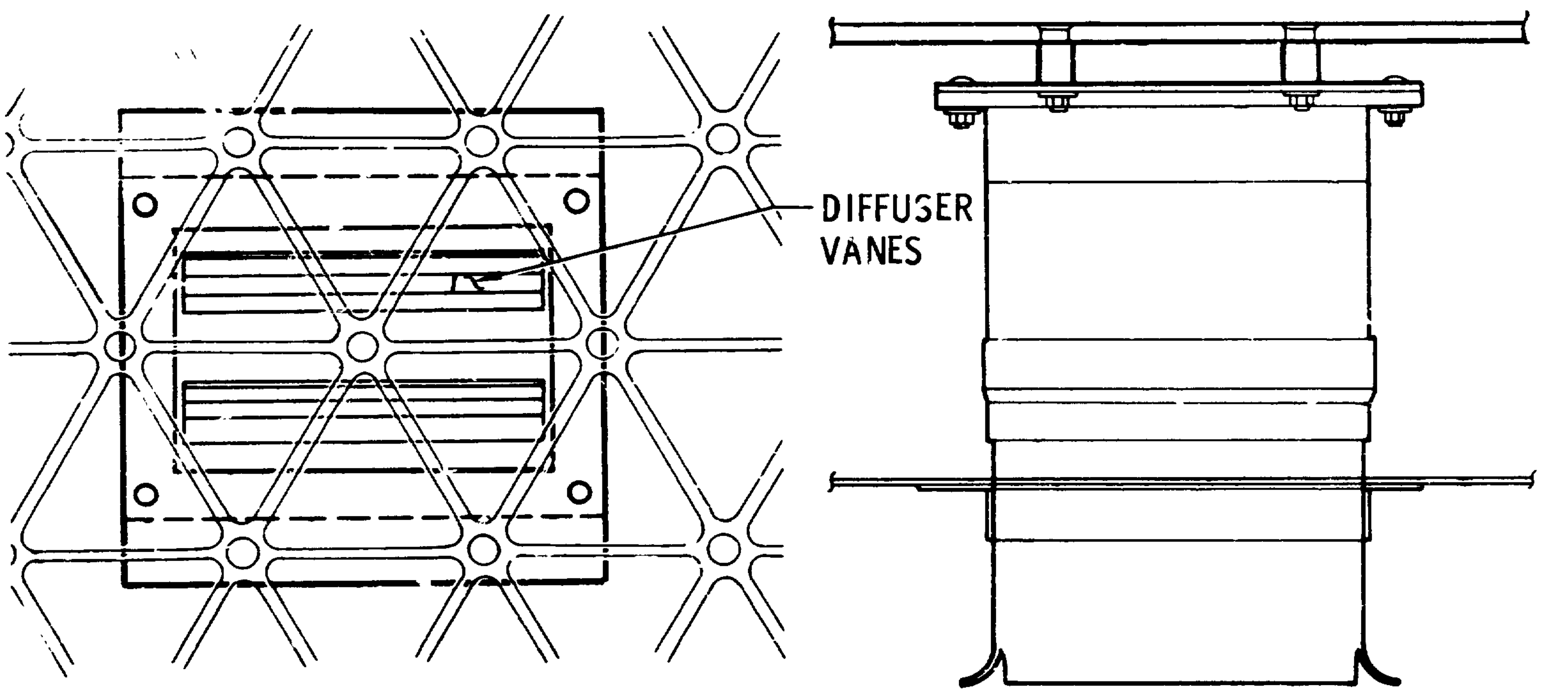

Figure 2.2.3-12

ORBITAL WORKSHOP VENTILATION CONTROL SYSTEM DIFFUSER LOCATIONS

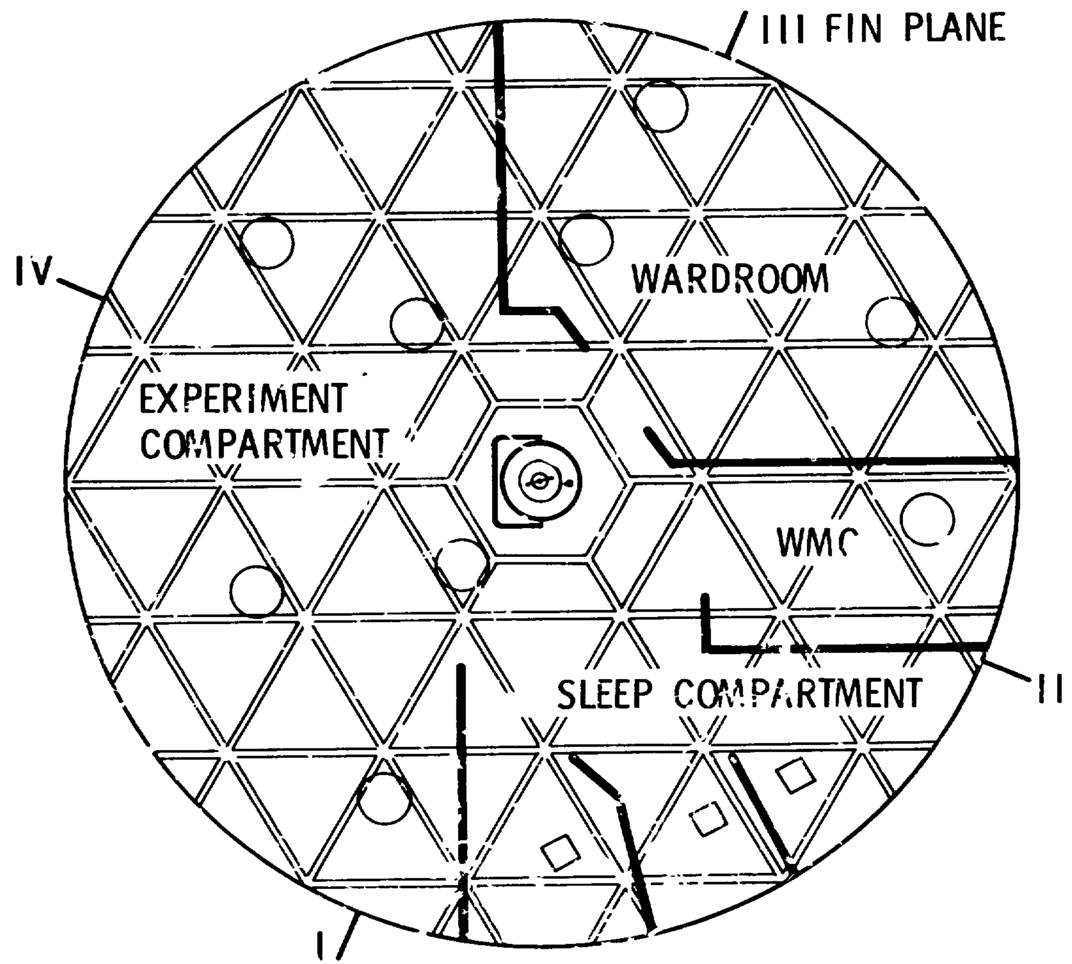

Figure 2.2.3-13

C-5

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PORTABLE FAN WITH SOUND SUPPRESSION

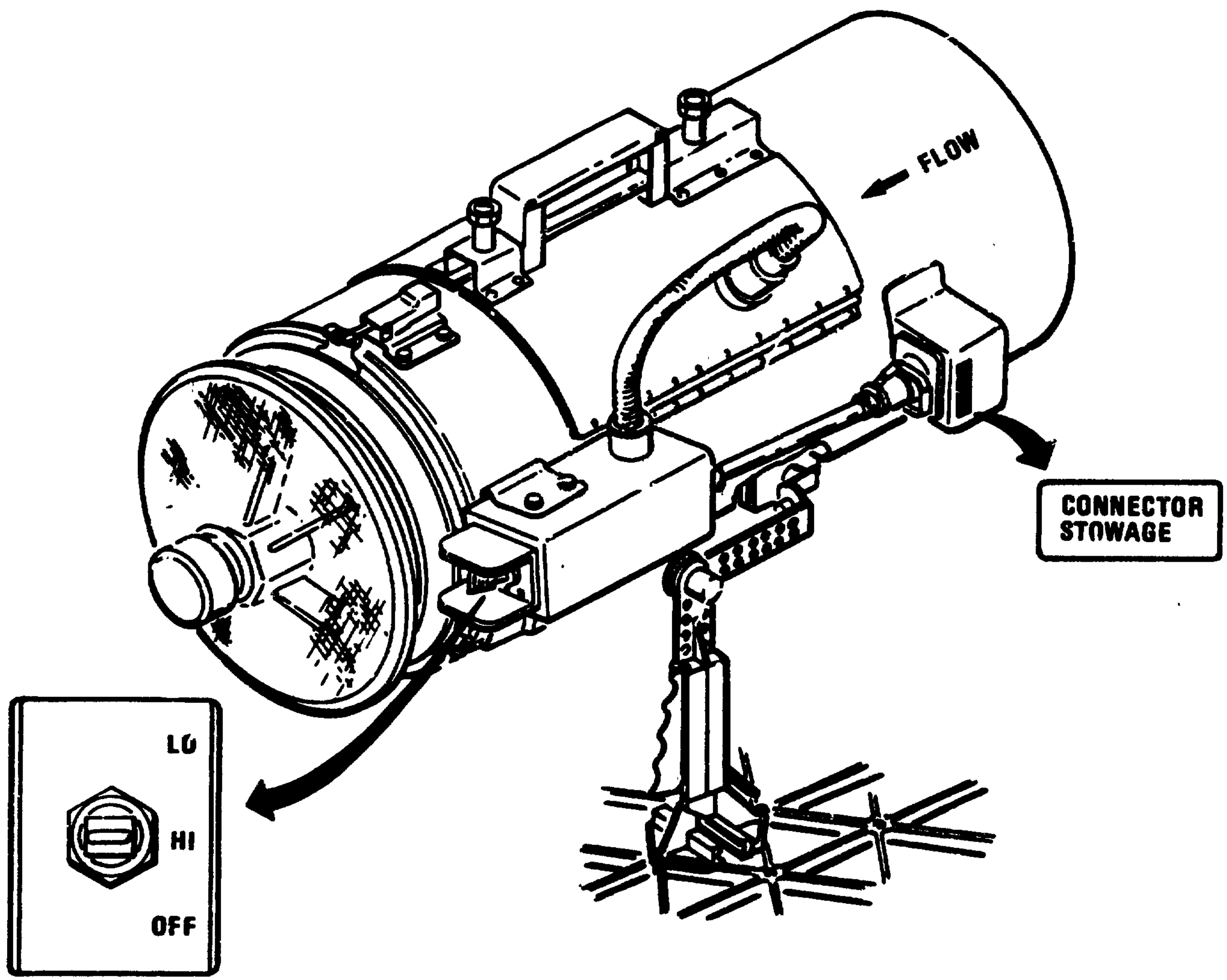


Figure 2.2.3-14

f. VCS Control Electronics

The VCS duct fans could be controlled by circuit breakers located on Circuit Breaker Panel 614 (Figure 2.2.3-15). The fans in VCS Ducts 1 and 2 were powered from OWS Power Buses 1 and 2, respectively, as shown in Figure 2.2.3-16. The fans in Duct 3 could be powered from either bus or split between Bus 1 and Bus 2 as shown in Figure 2.2.3-17.

2/ Active Thermal Control

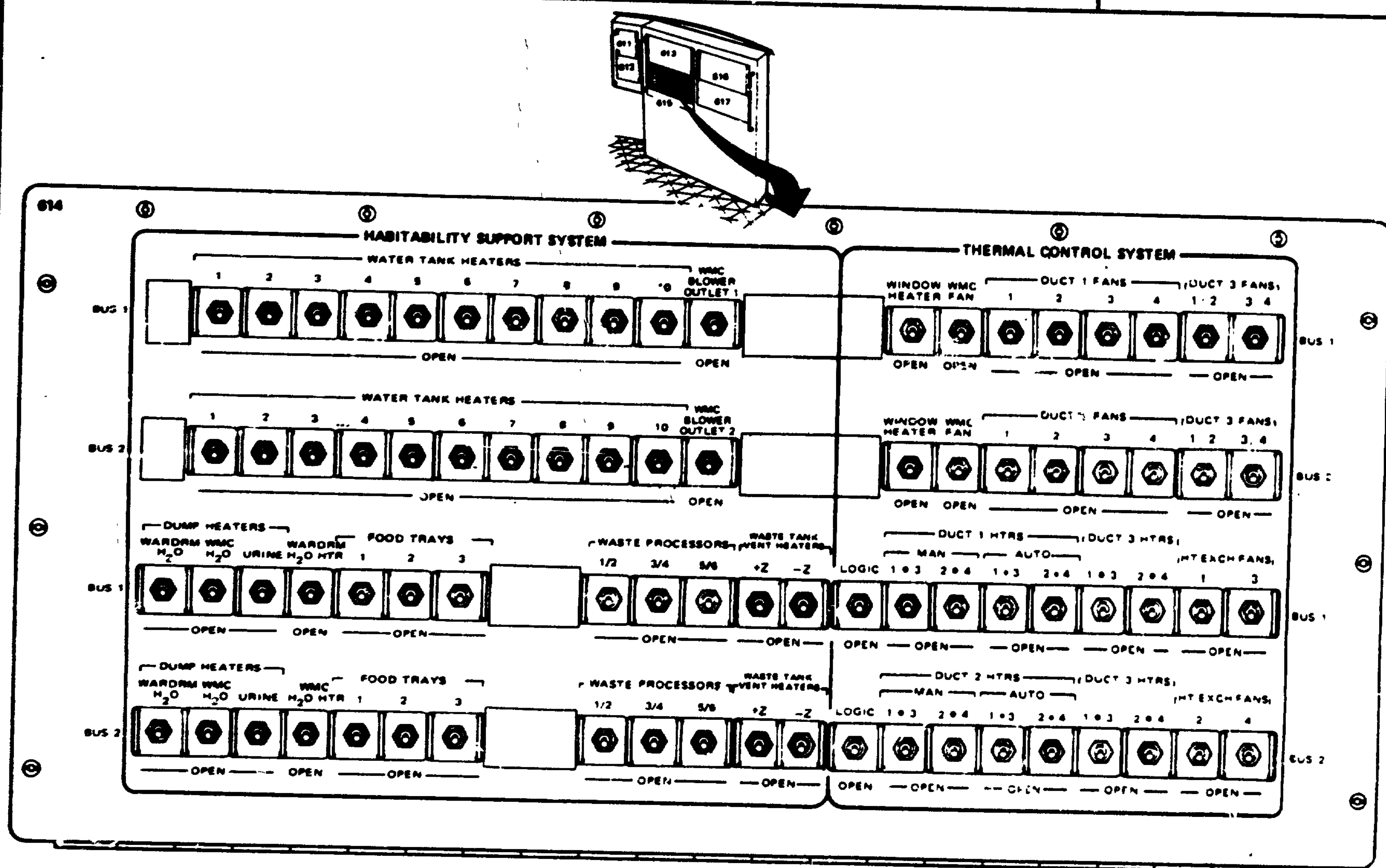
The active thermal control subsystem provided continuous control of the OWS internal environment during periods of astronaut habitation. The cabin gas temperature was controlled by cabin gas heat exchangers in the airlock module (AM) and by convective heaters in the three VCS ducts. Conditioned gas from the AM was mixed with recirculated air in the OWS. Radiant heaters were designed to maintain temperatures above the minimum levels that satisfied food and film storage requirements prior to habitation.

a. Convective Heaters

The convective heater design parameters are shown in Table 2.2.3-4. As shown, the heater power requirement was 500 watts dissipation at 24 Vdc. The heater was capable of operating over an applied voltage range of 24 to 32 Vdc.

SKYLAB - ORBITAL WORKSHOP

CIRCUIT BREAKER PANEL 614



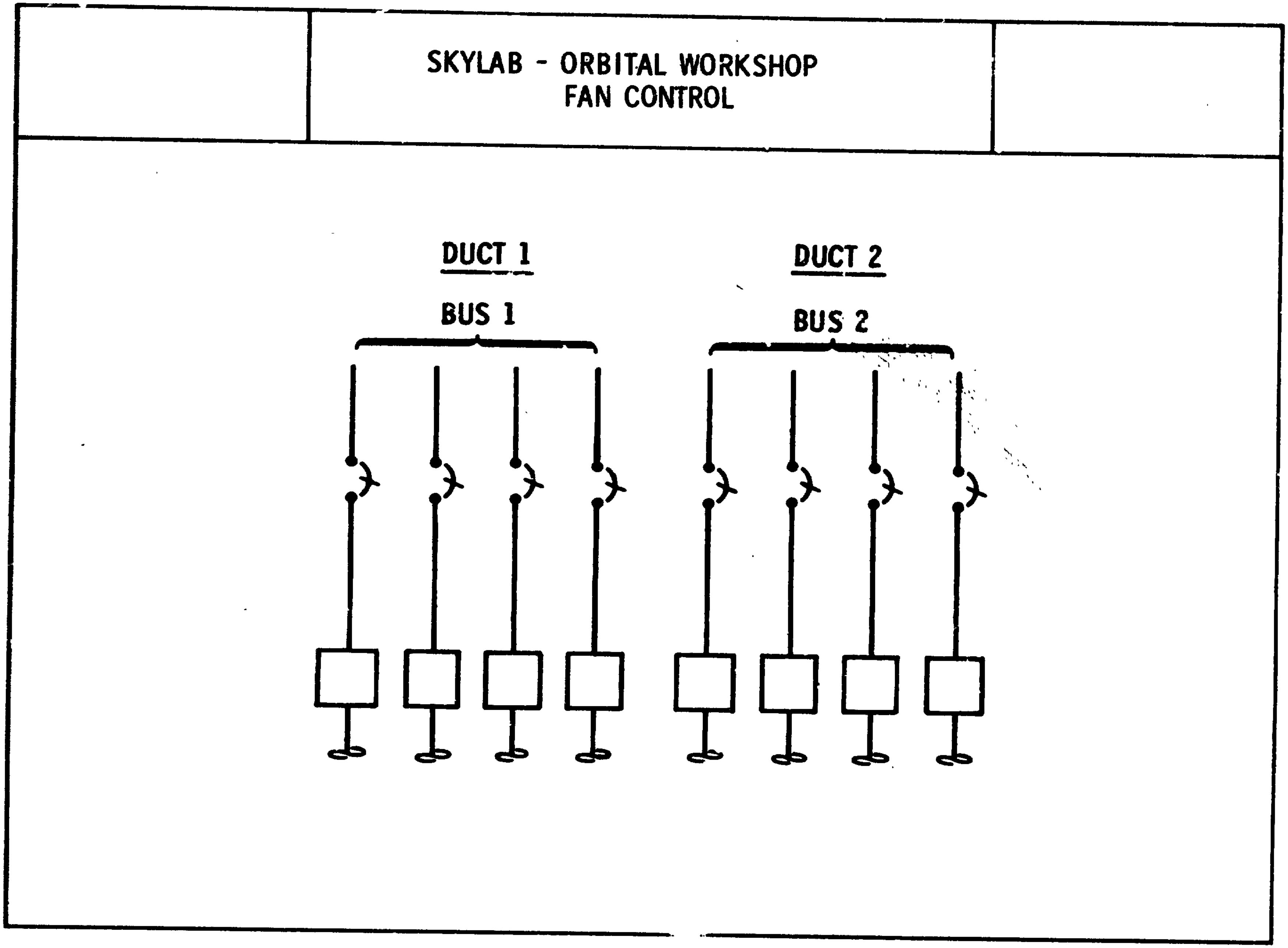


Figure 2.2.3-16

ORBITAL WORKSHOP
FAN BUS SELECTION

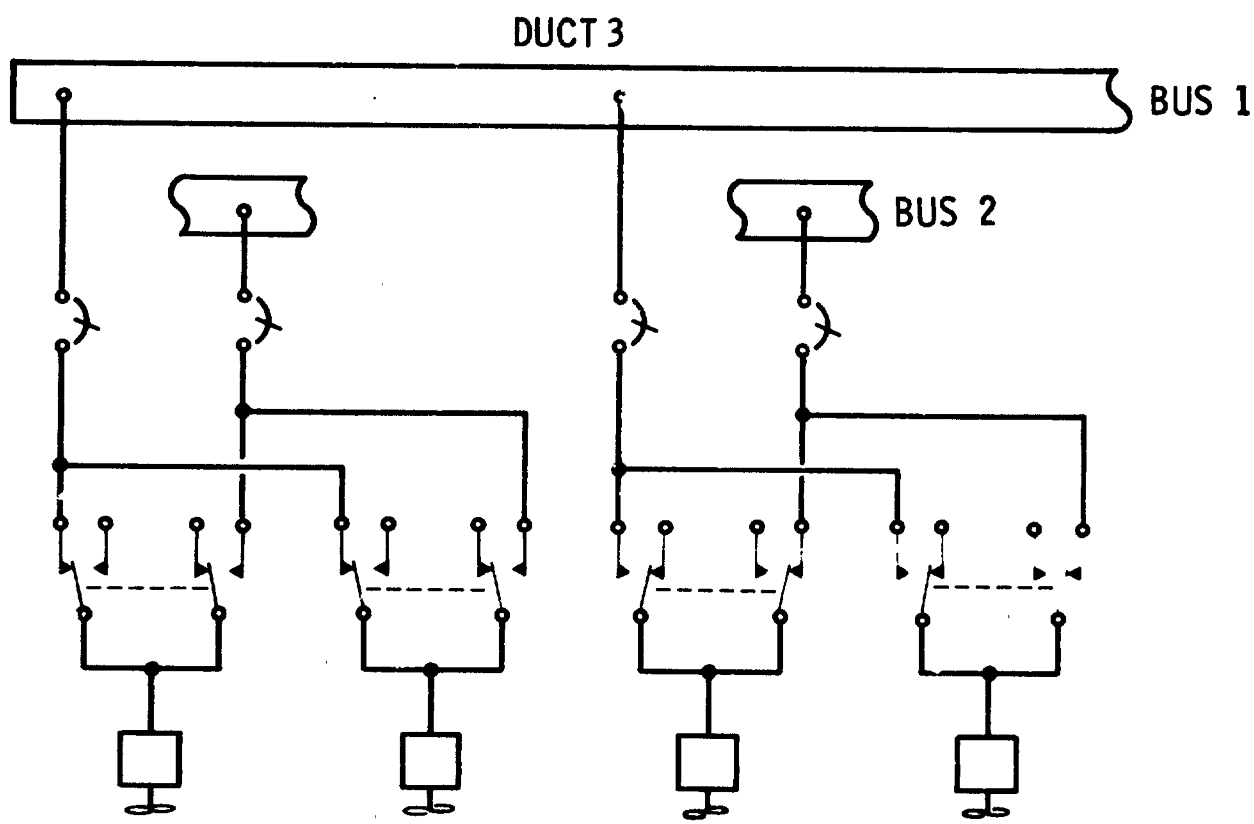


Figure 2.2.3-17

Table 2.2.3-4

CONVECTIVE HEATER DESIGN PARAMETERS

Parameter	Minimum	Criteria Nominal	Maximum
Heater power, watts	500	N/A	N/A
Applied voltage, Vdc	24	N/A	32
Element resistance, ohms	4.15	N/A	4.61

One convective heater assembly was located in each of the three OWS VCS ducts (Figure 2.2.3-5). In Ducts 1 and 2 the heaters were positioned just above the lower floor. In Duct 3 the heater was mounted just above the ceiling. Each unit contained four heating elements, designed to dissipate 125 watts each at 24 Vdc. Each element contained two thermostats for high limit temperature control with redundancy. The thermostats were designed to open at 260 ±10°F (400 ± 5.6 K) and close at 240 ± 10°F (389 ± 5.6 K).

The heater assembly is shown in Figure 2.2.3-18. V-band couplings were provided for heater assembly replacement on-orbit. A cover over the coupling mechanism was used to provide astronaut protection. The four heater elements were stacked in a 6-inch (15.24 cm) long by 13.9-inch (35.3 cm) diameter, 0.050-inch (0.127 cm) wall aluminum housing. The four elements were brazed to either side of two spoke assemblies (8 spokes per assembly) attached to the housing with screws.

ORBITAL WORKSHOP
CONVECTIVE HEATER

1B77902 - HEATER ASSY, THERMAL CONTROL

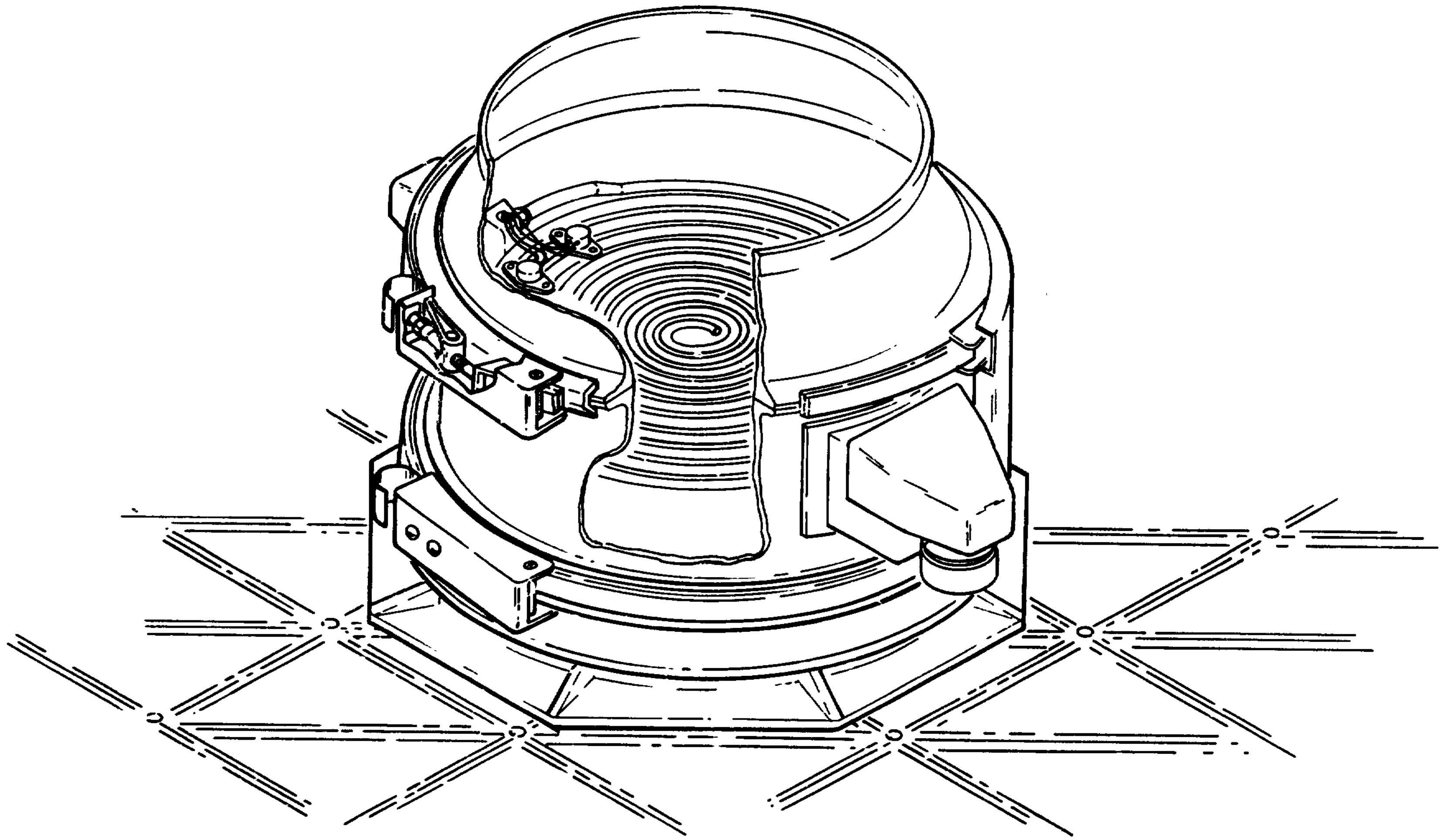


Figure 2.2.3-18

Each element was a spiral design with a total length of approximately 24 feet (7.3 m). The outer wall of the 0.125-inch (0.318 cm) diameter element was stainless steel of 0.025-inch (0.0635 cm) thickness. The heater wire within the element, 0.025-inch (0.0635 cm) diameter nickel, doubled back on itself at the center of the spiral such that both ends terminated at the large radius end of the element. Magnesium oxide was the insulating material used between the nickel wire and the stainless steel sheath. The wiring from an element and its thermostats terminated at one of two junction boxes located on either side of the assembly.

The convective heaters, located in the VCS ducts, were designed to be used in conjunction with the AM cooling heat exchangers to provide thermal control of the OWS atmosphere during habitation. The Duct 1 and Duct 2 heaters were to be controlled automatically by the Thermal Control Assembly (TCA), but could be controlled manually by switches located on Control Panel 617 (Figure 2.2.3-19). The TCA and control panel switches were arranged such that heaters could be turned on and off in increments of 250 watts. The 250 watt increments of the Duct 3 heater could be controlled only manually.

SKYLAB - ORBITAL WORKSHOP
CONTROL AND DISPLAY PANEL -617
THERMAL CONTROL SYSTEM

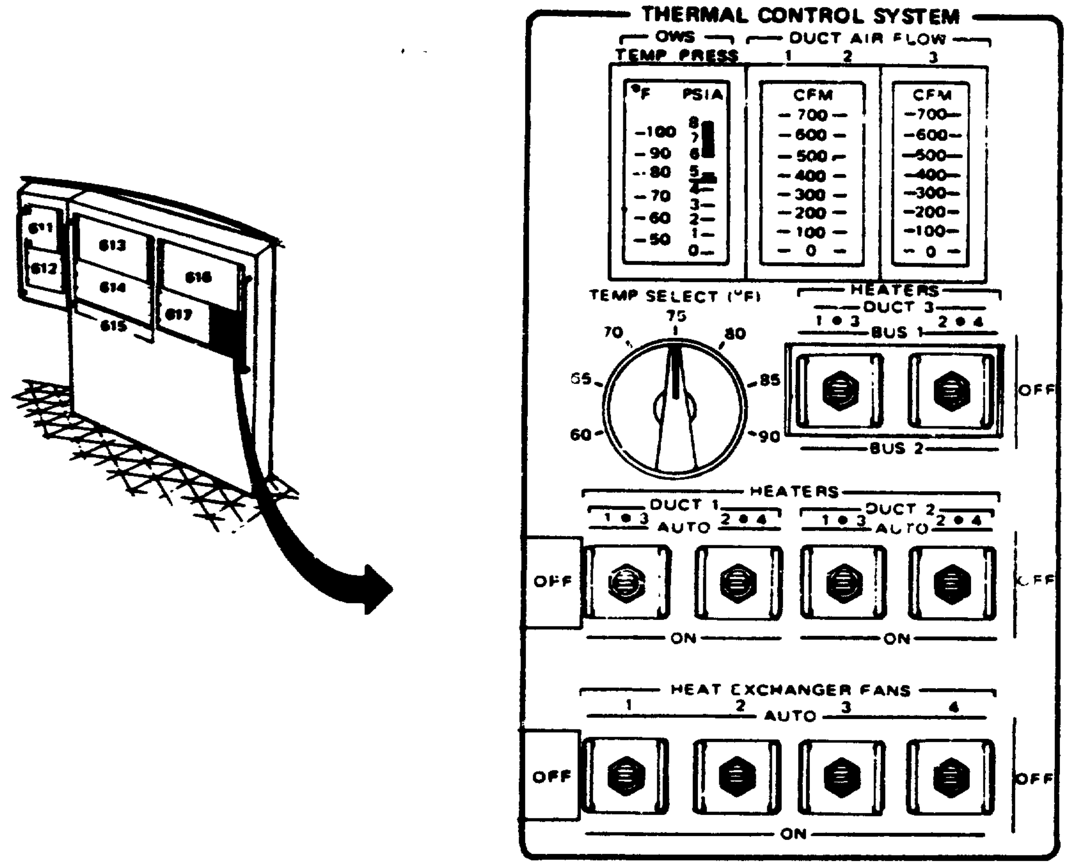


Figure 2.2.3-19

The convective heater system electrical schematics are shown in Figures 2.2.3-20 and 2.2.3-21. As shown, the heaters in Ducts 1 and 2 were controllable by both the TCA automatic (AUTO) and manual switches (OFF/ON). The heaters in Duct 3 were controllable by the manual switches only. In addition, the heater in Duct 3 could be powered from either bus, whereas the Duct 1 and 2 heaters could only be powered from Buses 1 and 2, respectively.

b. Radiant Heaters

The radiant heater design parameters are shown in Table 2.2.3-5. As shown, the power requirement for each of eight heaters was 125 watts minimum at 24 Vdc. The heaters were to be capable of withstanding an applied voltage of 32 Vdc.

TABLE 2.2.3-5

RADIANT HEATER DESIGN PARAMETERS

Parameter	Minimum	Criteria Nominal	Maximum
Heater power, watts	125	N/A	N/A
Applied voltage, Vdc	24	N/A	32
Surface temperature, °F (K)	N/A	N/A	190 (361)

SKYLAB - ORBITAL WORKSHOP
HEATER CONTROL

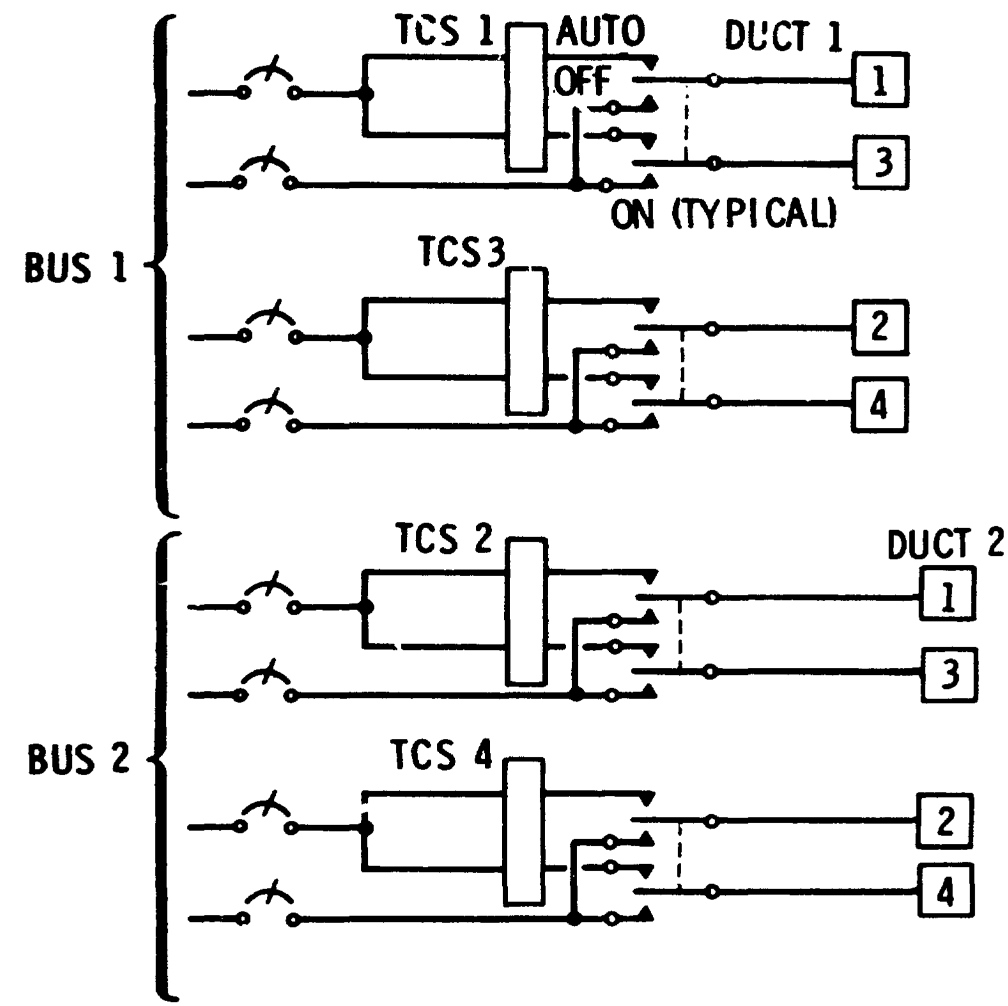


Figure 2.2.3-20

SKYLAB - ORBITAL WORKSHOP
HEATER CONTROL DUCT 3

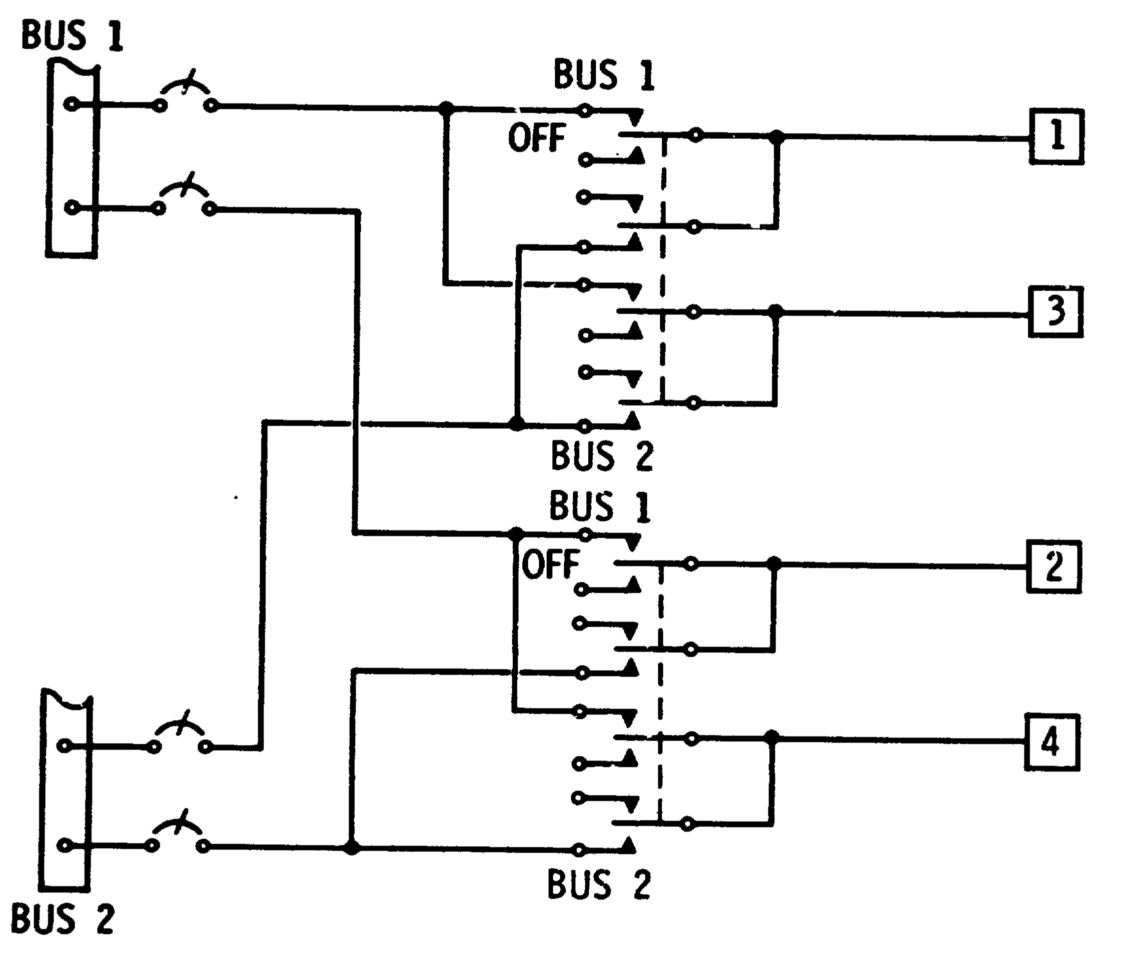


Figure 2.2.3-21

The function of the radiant heaters was to heat the OWS after boost and prior to habitation, and during orbital storage periods. The heaters were designed to dissipate a total of 1000 watts (125 watts each) at 24 Vdc. Three heaters were located in the forward compartment and the remaining five in the crew quarters as shown in Figure 2.2.3-22.

The heaters, which had a semi-circular cross section were 31 inches (78.7 cm) long and 10.8 inches (27.4 cm) in diameter (Figures 2.2.3-23 and 2.2.3-24). The heating element, located beneath the 0.025-inch (0.0635 cm) aluminum shell was 0.001-inch (0.00254 cm) Inconel. The inner and outer surfaces of the element were insulated with silicone rubber impregnated cloth. A flame barrier of 0.012-inch (0.0305 cm) fluorocarbon impregnated fiber glass cloth on either side of the insulation was provided. One-inch (2.54 cm) thick polyurethane foam was used to insulate the inner surface of the heater. The 0.020-inch (0.05 cm) aluminum inner shell and 0.003-inch (0.0076 cm) aluminum foil outer covering provided fire protection by enclosing the foam. Other structural members of the heater included a 0.188-inch (0.477 cm) base plate and three 0.03[illegible] inch (0.0813 cm) bulkheads.

SKYLAB - ORBITAL WORKSHOP

RADIANT HEATERS

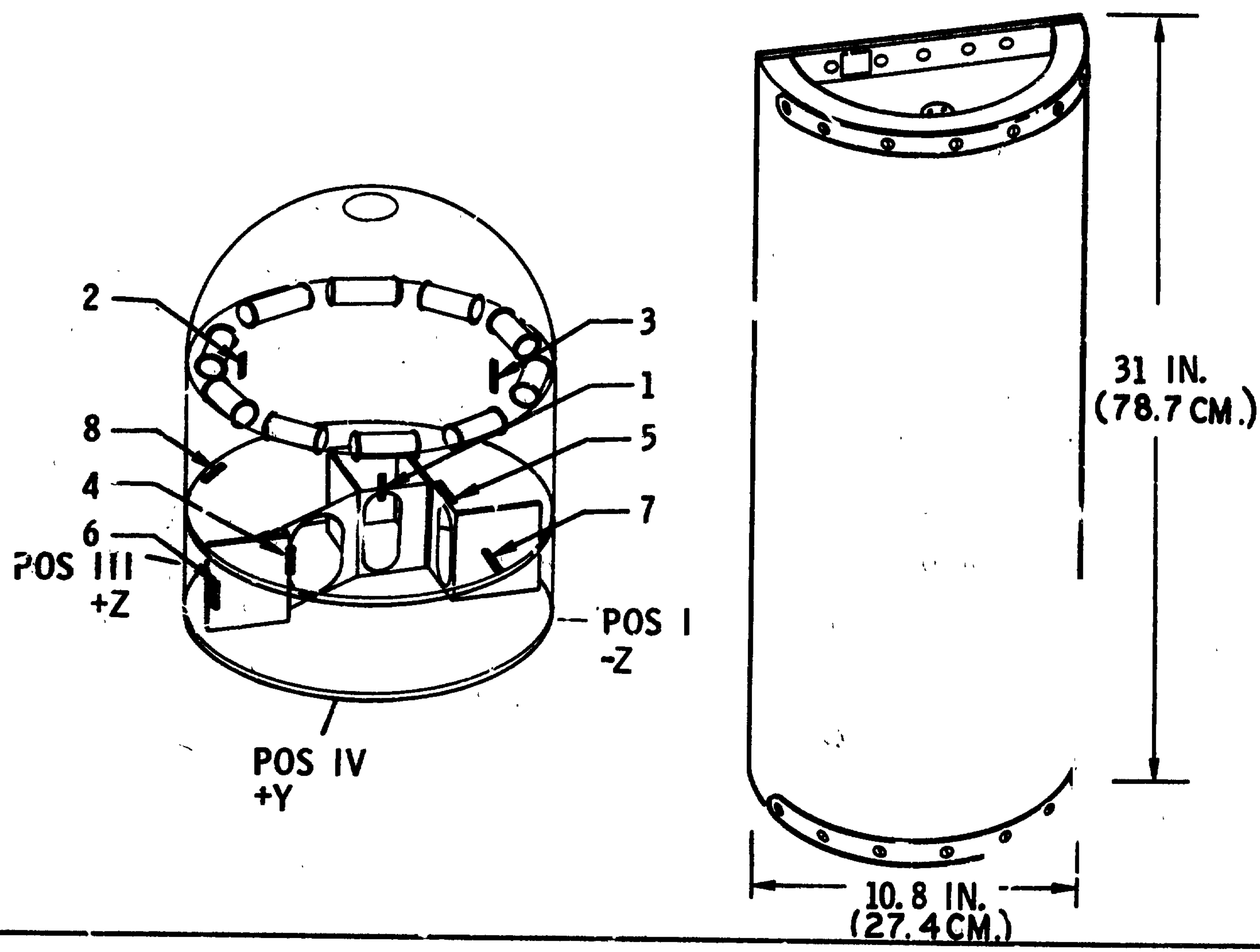


Figure 2.2.3-22

SKYLAB - ORBITAL WORKSHOP
RADIANT HEATER - 1B81046

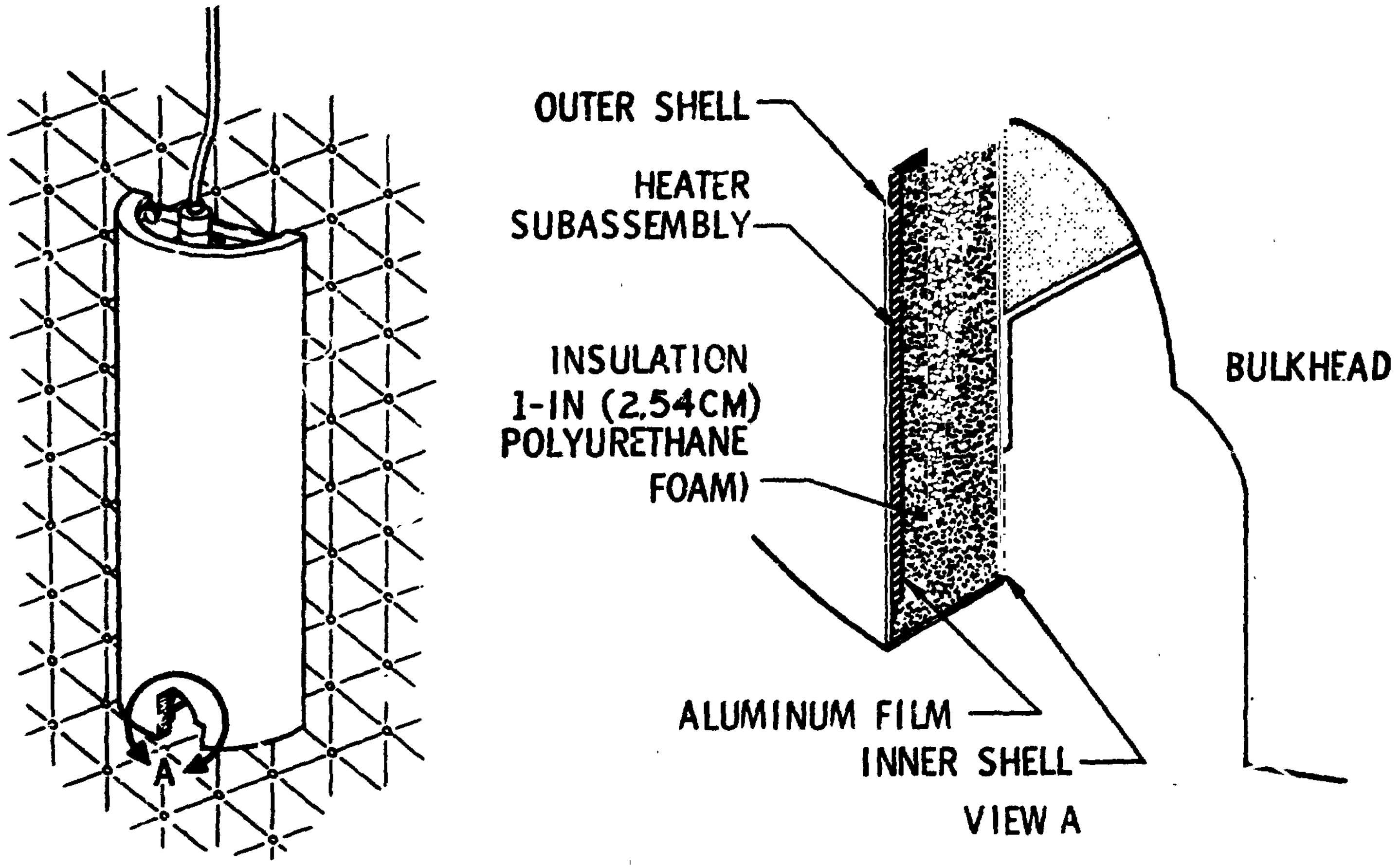


Figure 2.2.3-23

SKYLAB - ORBITAL WORKSHOP
RADIANT HEATER COMPONENTS

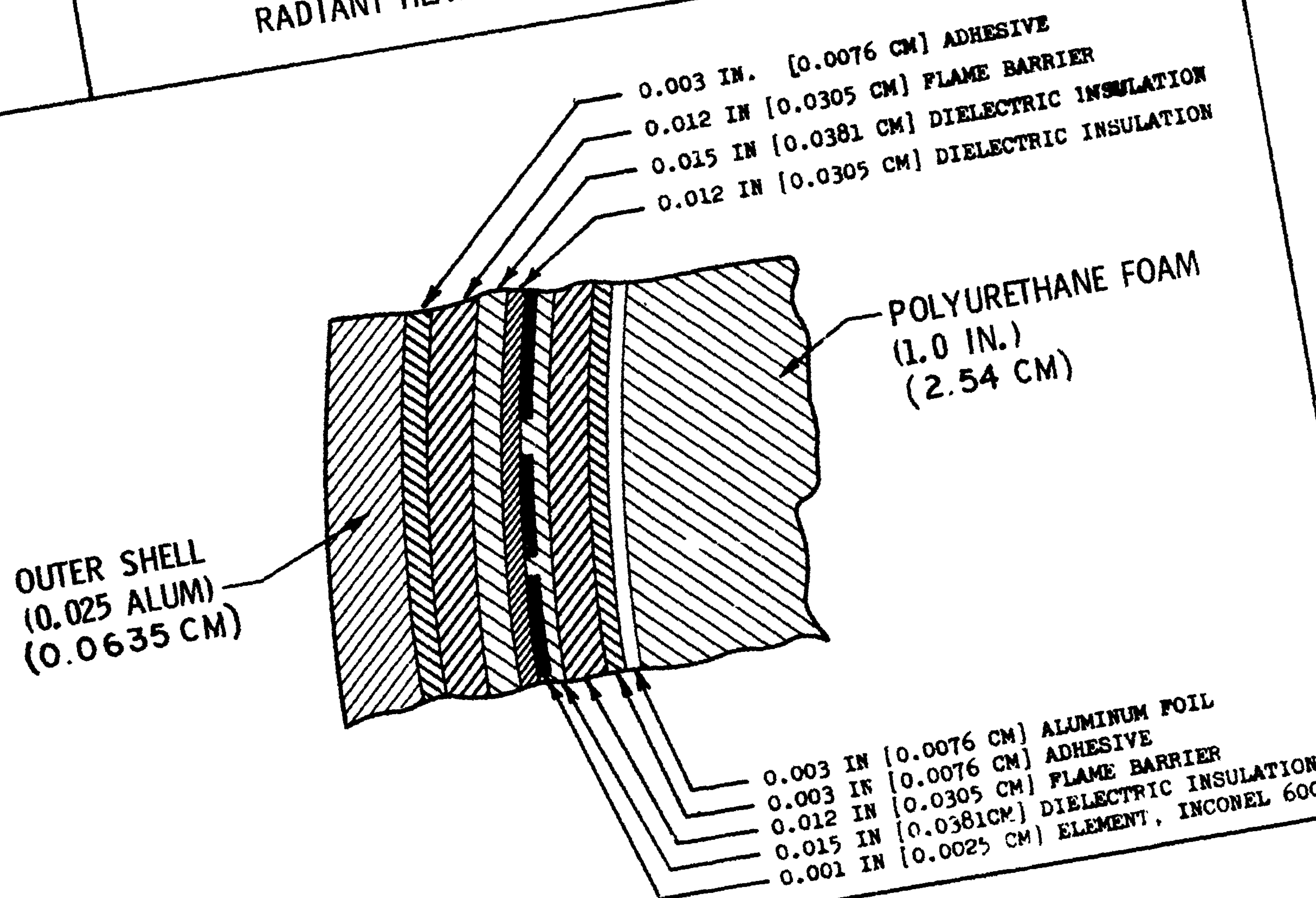


Figure 2.2.3-24

The eight radiant heaters were provided to maintain the temperatures of food and film above 40°F (278 K) during unmanned periods after orbital insertion and to warm up the habitation area prior to OWS activation. The heaters were designed to be remotely controlled in groups of four, both through the Digital Command System (DCS) and manually through switches in the AM (Panel 203, Figure 2.2.3-25). The radiant heater system electrical schematic is shown in Figure 2.2.3-26.

c. Wardroom Window Heater

The Wardroom window heater design parameters are shown in Figure 2.2.3-27. Any combination of gold coating emissivity and electrical resistance falling within the envelope shown met the design criteria for defogging and maintaining a minimum window temperature of 55°F. In addition, the window heater was capable of operating over an applied voltage range of 24 to 32 Vdc.

The Wardroom window heater was an electrically conductive (EC) gold film 3 to 5 Angstroms thick, vapor deposited on the outboard surface of the inboard glazing of the window (Figures 2.2.3-28 and 2.2.3-29). The window had two bus bars bonded to the EC coating as shown in Figure 2.2.3-29. When voltage was applied to the bus bars, current flowed in the gold film generating E^2/R heat.

SKYLAB - ORBITAL WORKSHOP
ECS CONTROL PANEL 203

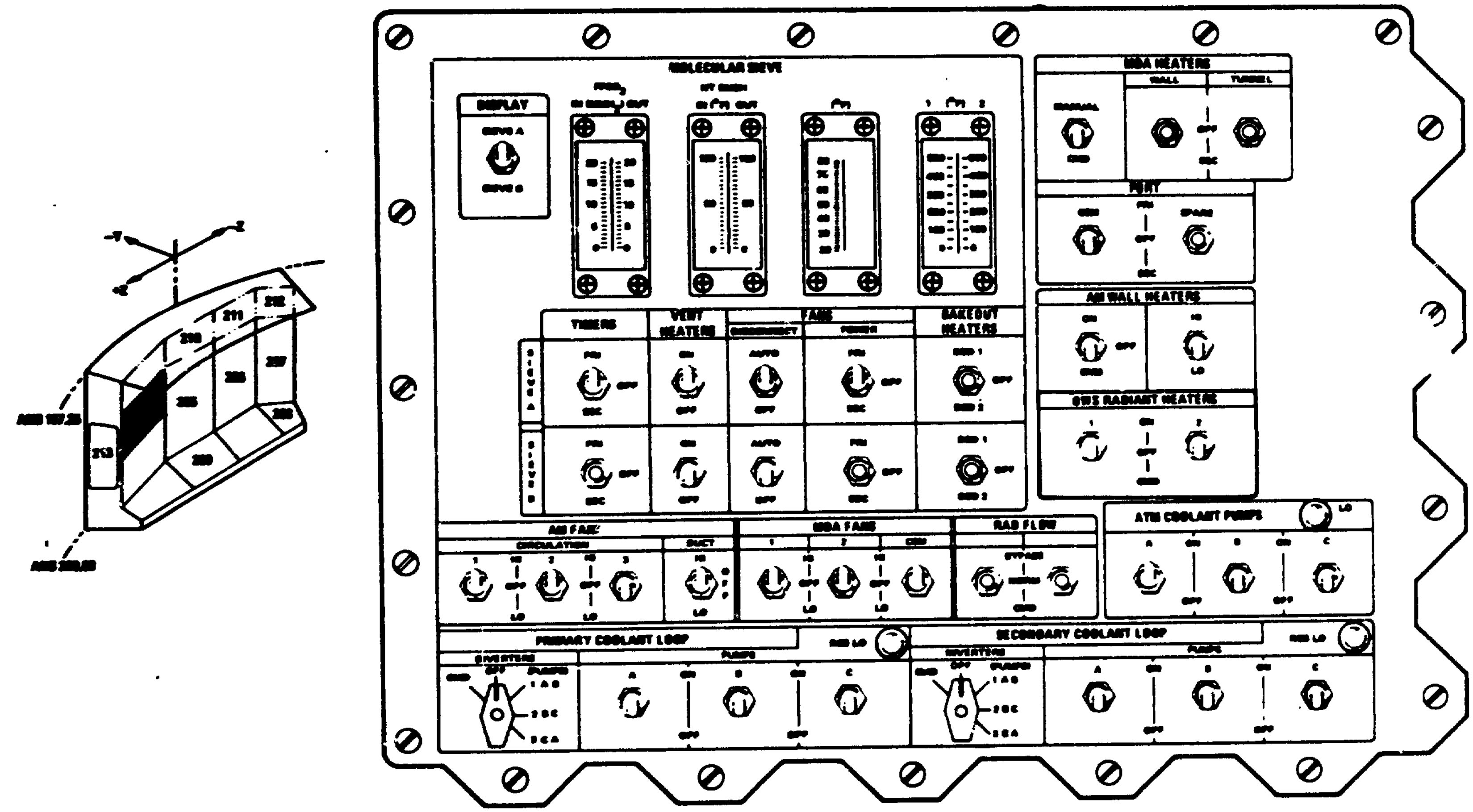


Figure 2.2.3-25

SKYLAB - ORBITAL WORKSHOP
RADIANT HEATER CONTROL

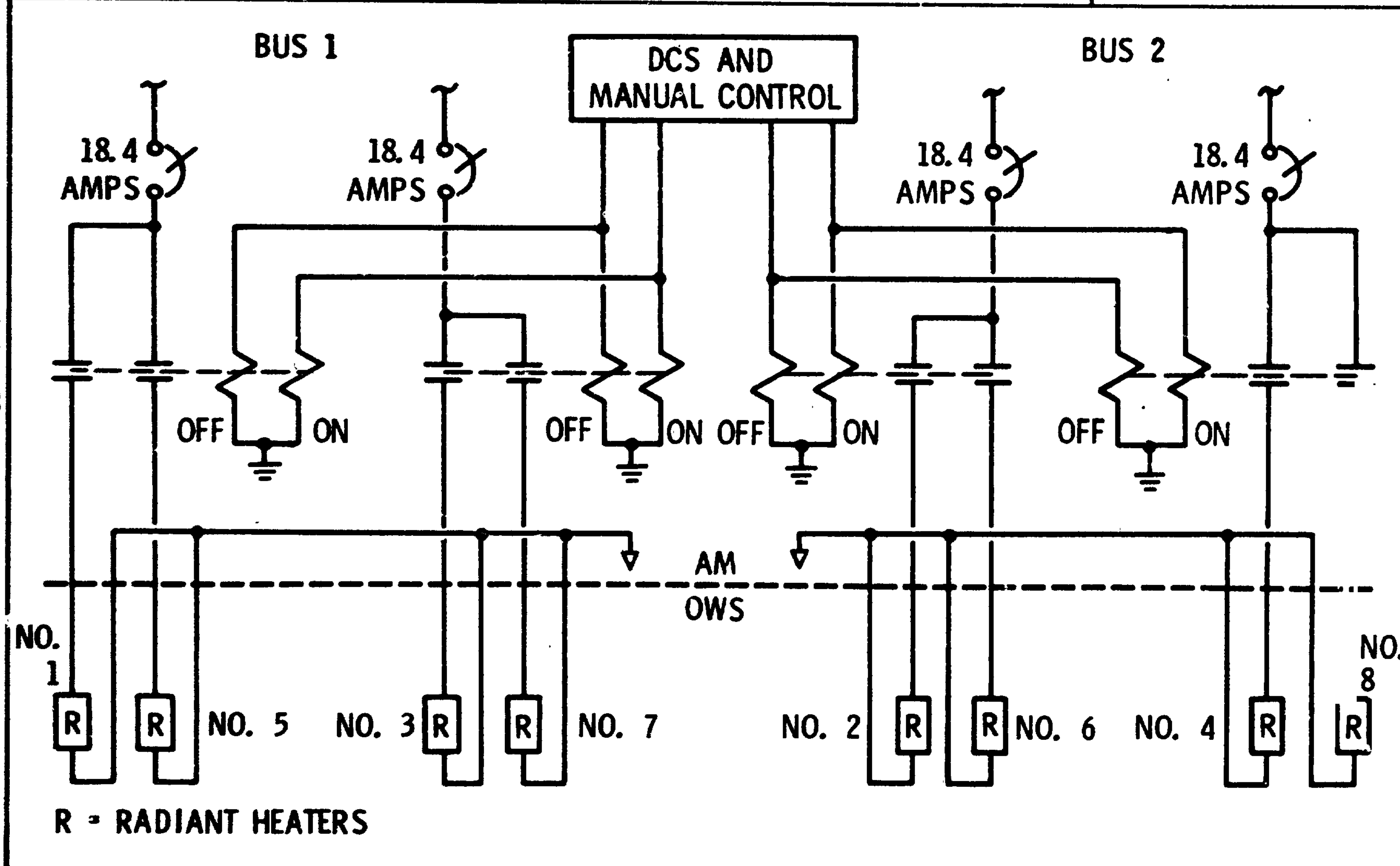
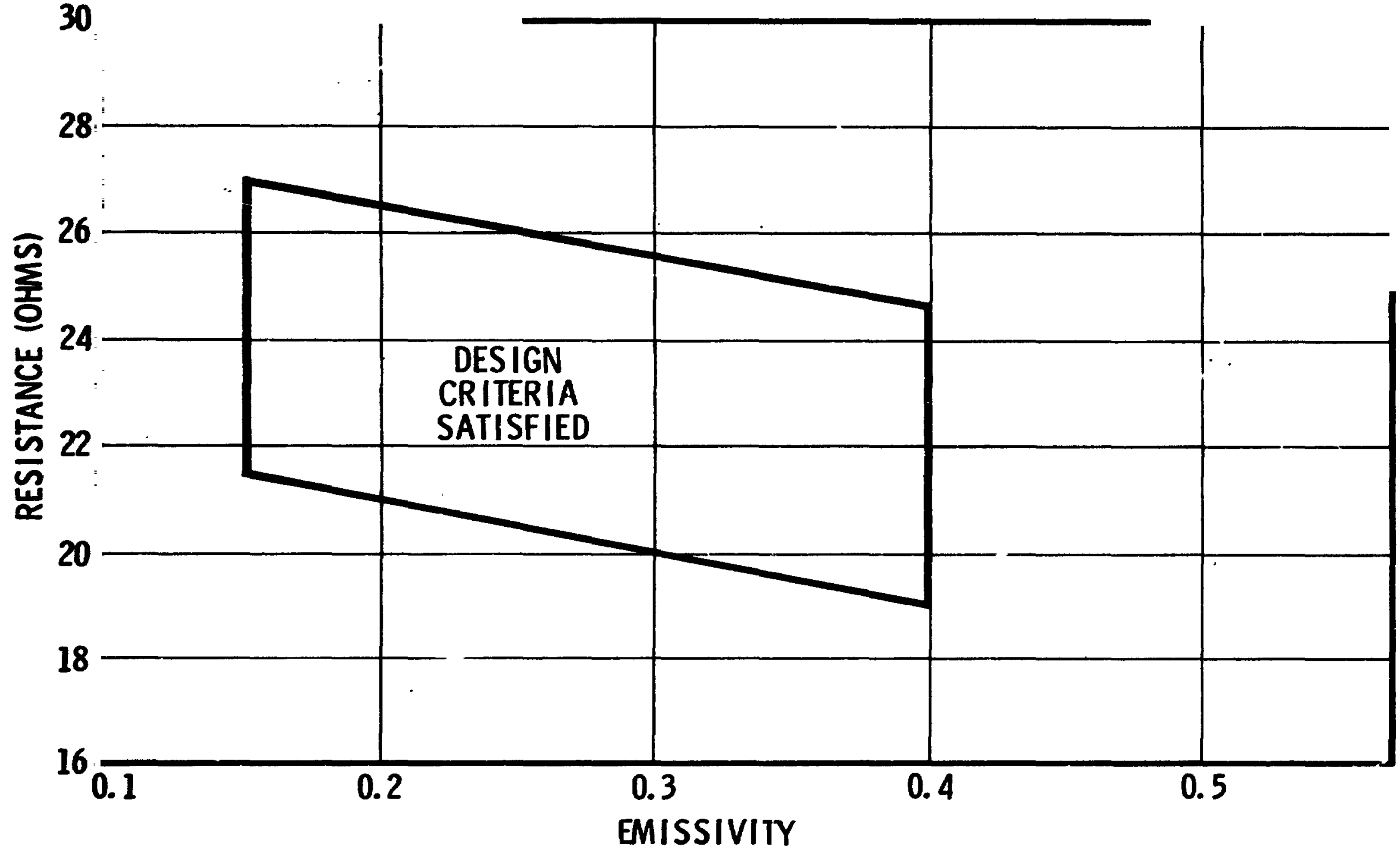


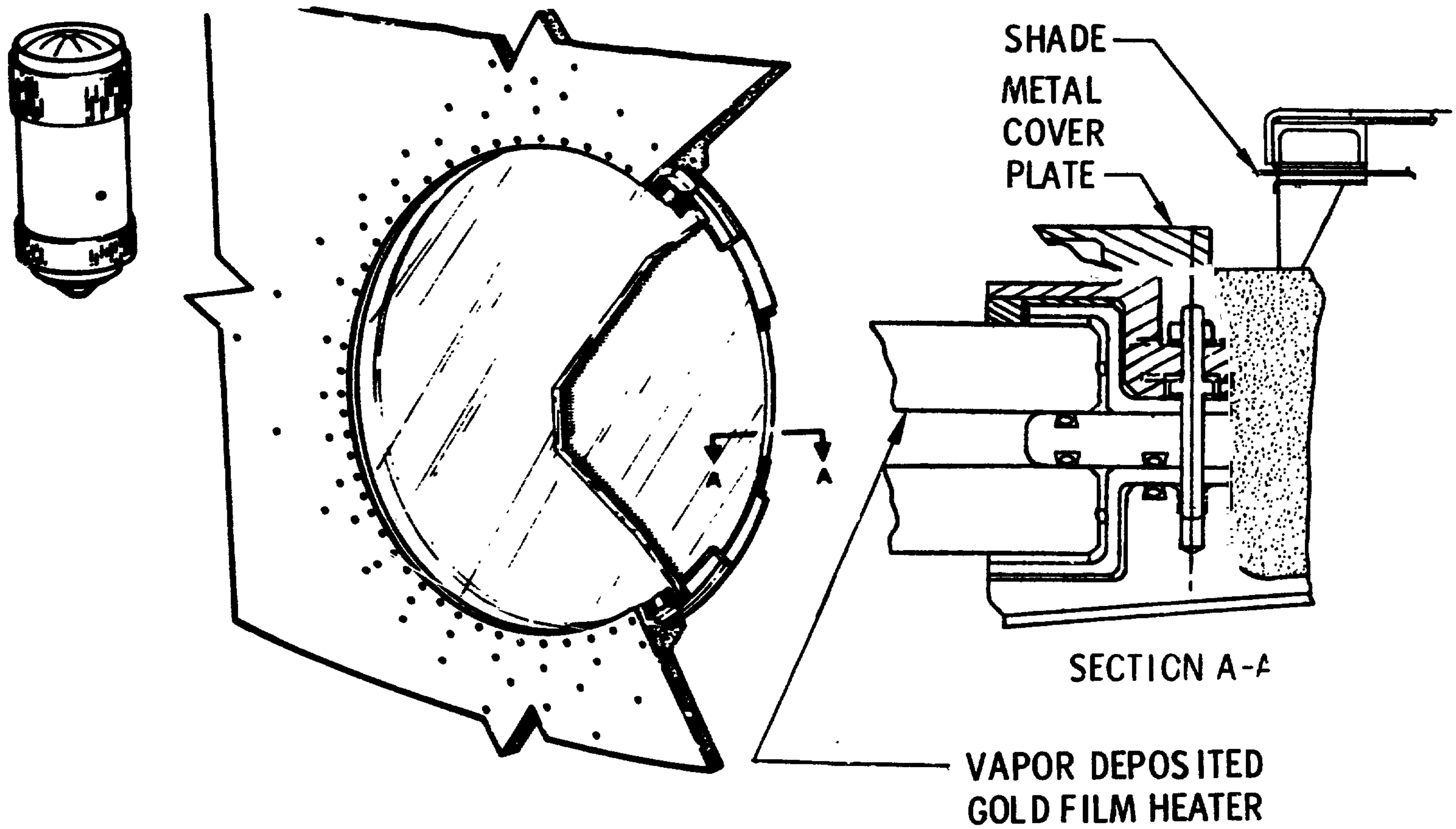
Figure 2.2.3-26



OWS VIEWING WINDOW
DESIGN CRITERIA

Figure 2.2.3-27

ORBITAL WORKSHOP WARDROOM WINDOW



SKYLAB - ORBITAL WORKSHOP WARDROOM WINDOW

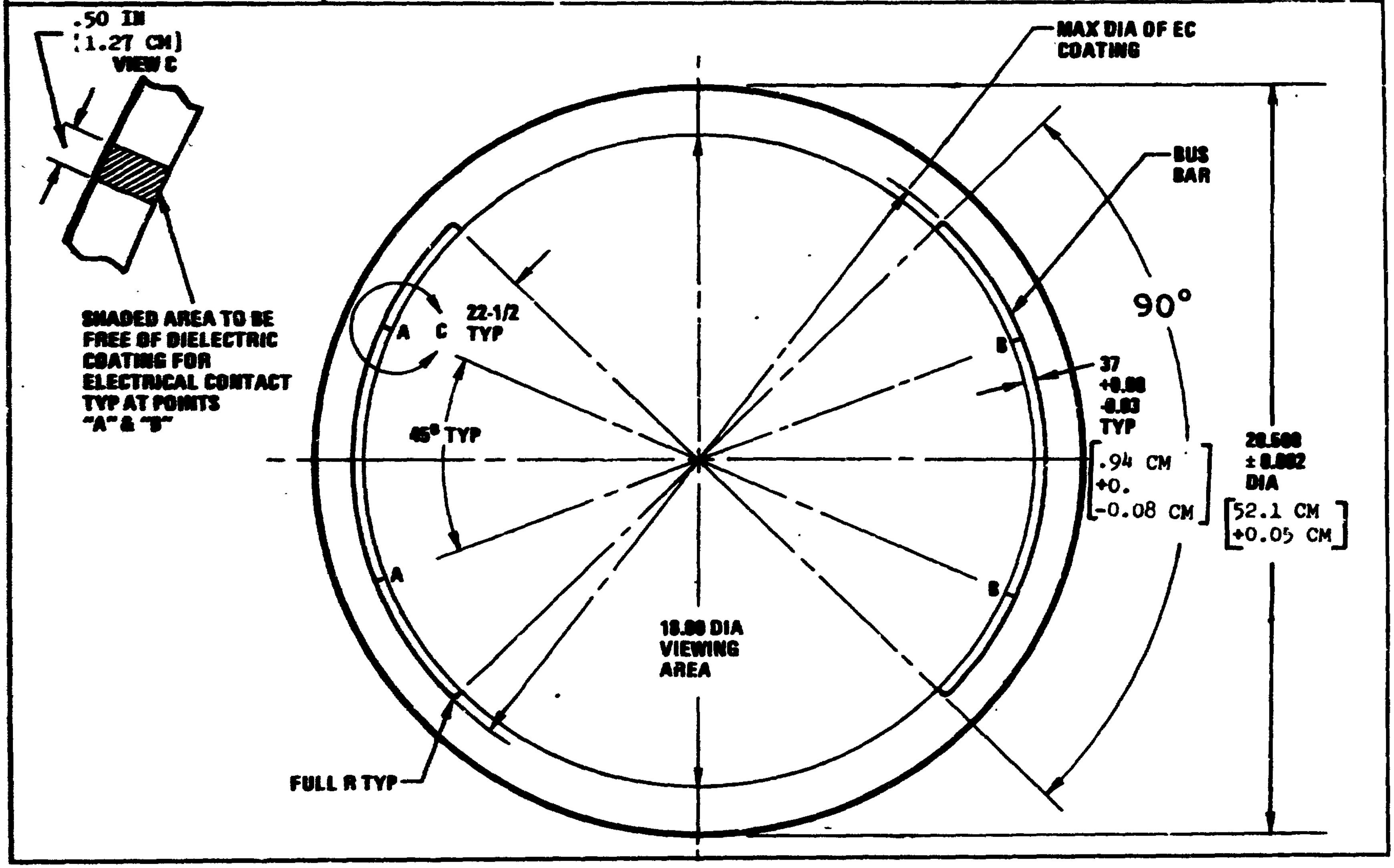


Figure 2.2.3-29

The Wardroom window heater was controlled by circuit breakers on Panel 614 (Figure 2.2.3-15) and a manual switch on Panel 700 in the Wardroom (Figure 2.2.3-30). The window heater was normally on during the manned phase of the mission and off during the unmanned phases.

The window heater electrical schematic is shown in Figure 2.2.3-31. As shown, the window heater was switchable between OWS Bus 1 and Bus 2. In addition, the heater circuit was protected by 6 ampere circuit breakers.

d. Controls and Displays

The Thermal Control Assembly (TCA) was the electronic assembly that was designed to provide the control functions for the TCS convective heaters and AM heat exchangers when the control switches (Panel 617) were in the AUTO position. The TCA was located behind a non-removable panel of the main control and display console.

The TCS controls and displays were located on Panel 617 (Figure 2.2.3-19). The switches were designed to control the mode of operations of the convective heaters and AM heat exchangers (AUTO/OFF/ON). The TEMP SELECT switch set the temperature to which the TCA controlled the OWS atmosphere and the OWS TEMP display showed the OWS atmospheric temperature.

SKYLAB - ORBITAL WORKSHOP
WARDROOM CONTROL AND DISPLAY PANEL - 700
(1B84376-1)

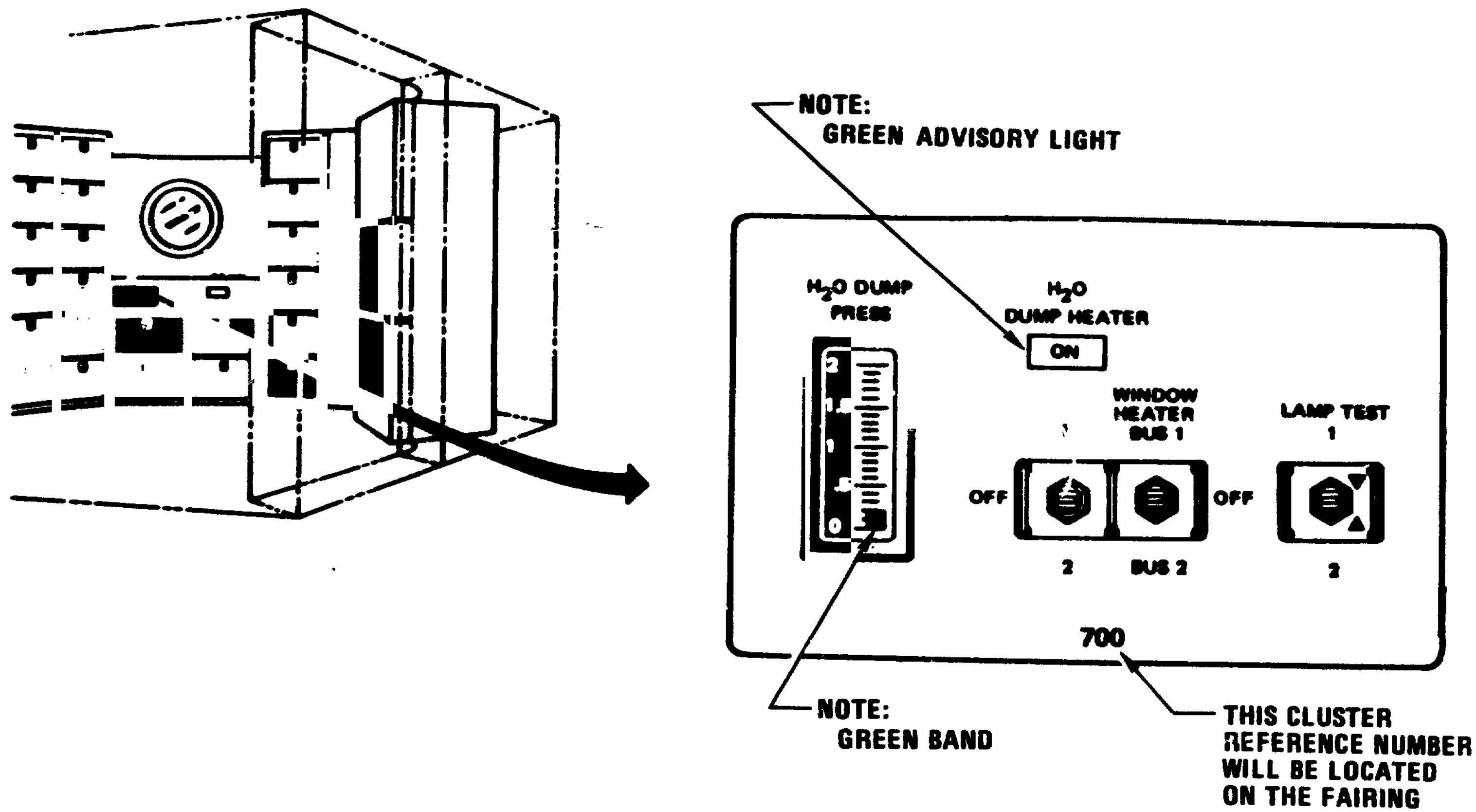


Figure 2.2.3-30

SKYLAB - ORBITAL WORKSHOP
WARDROOM WINDOW HEATER
ELECTRICAL SCHEMATIC

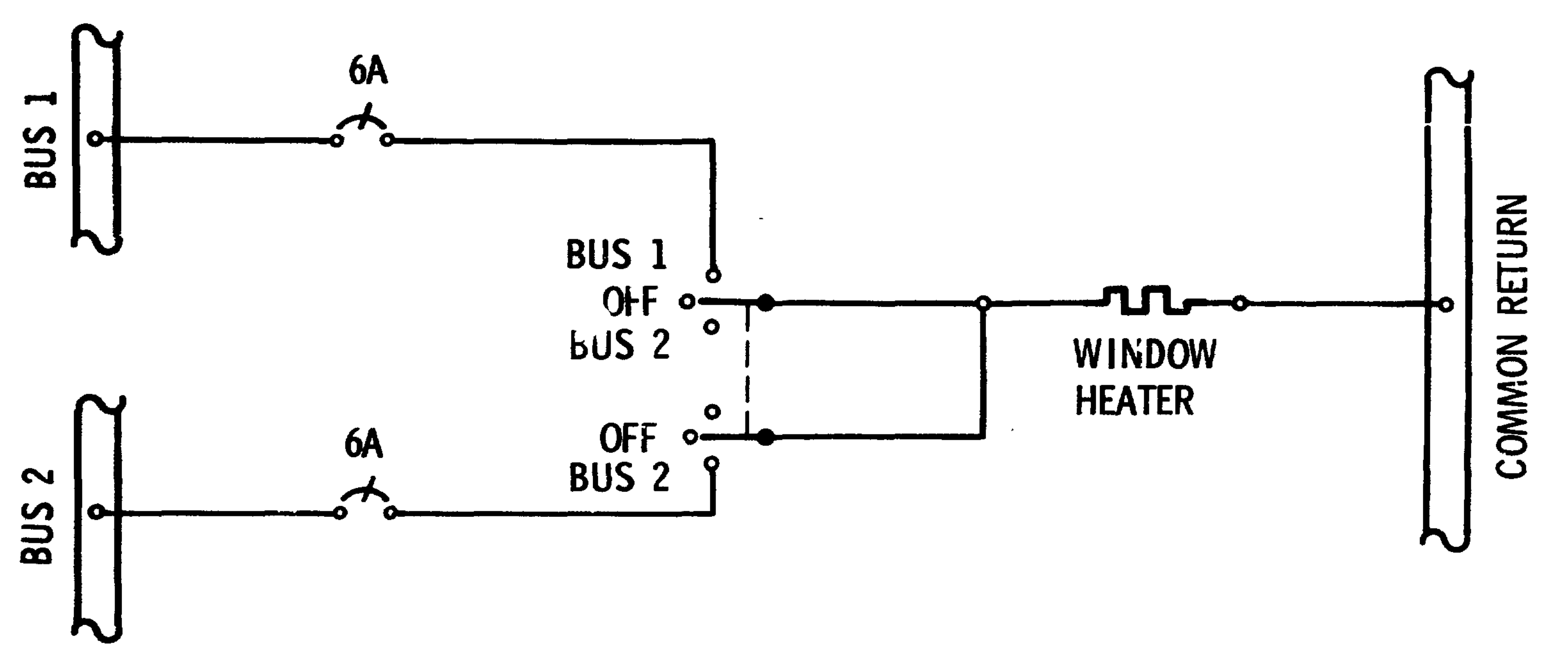
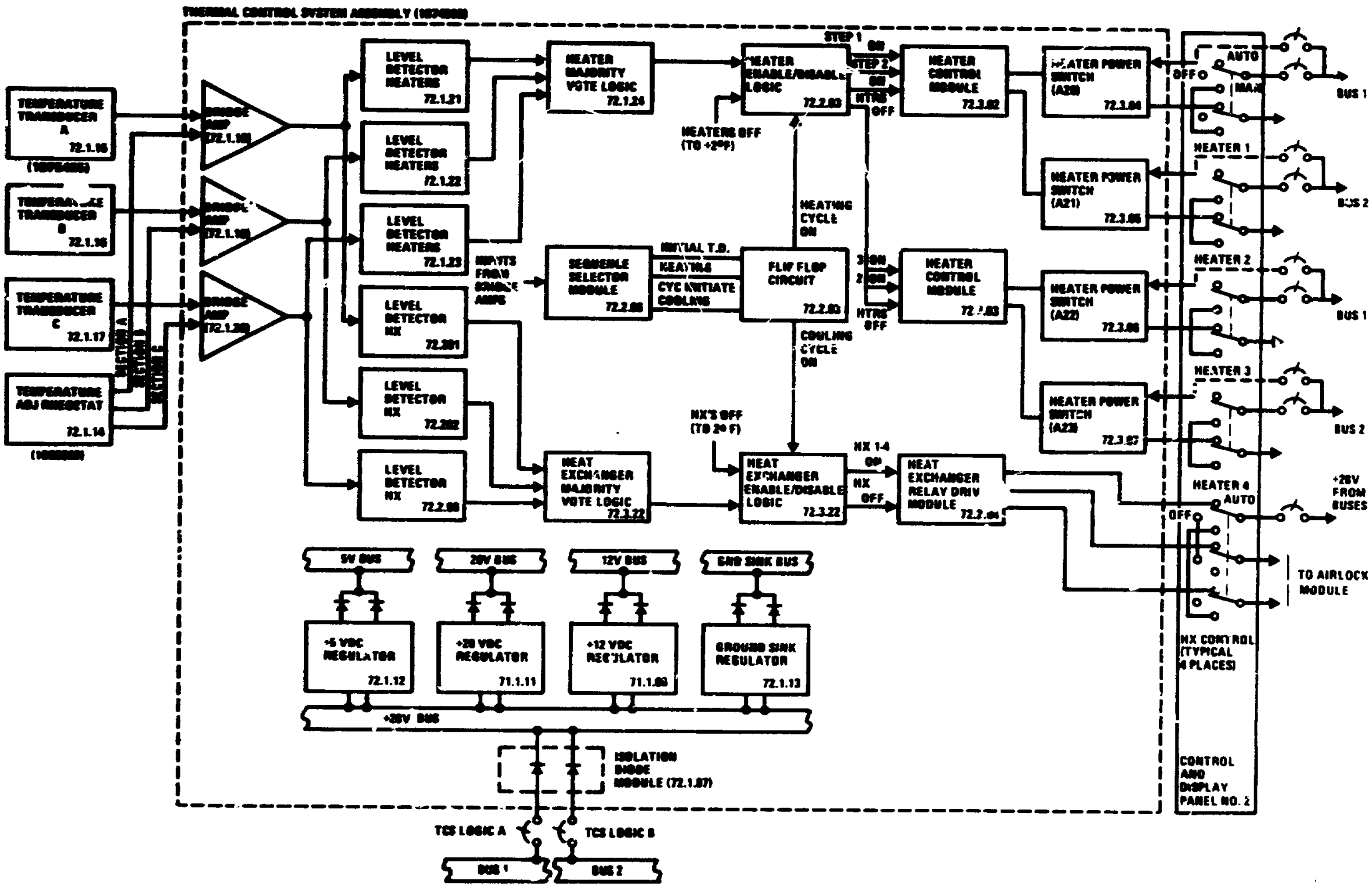


Figure 2.2.3-31

The TCA bridge amplifiers (Figure 2.2.3-32) were designed to take inputs from the three temperature transducers, compare them with the input from the temperature adjust rheostat and send an input to the sequence selector module. The sequence selector module could trigger the flip-flop circuit to enable either the heating cycle circuit or the cooling cycle circuit. The heater or heat exchanger majority voting logic then controlled the number of heaters or heat exchangers based on inputs from the respective level detector circuits. The TCA could activate the heaters or heat exchangers as required to stabilize the cabin gas temperature at the selected temperature. The design allowable temperature deviation was ±4°F (2.2 K) from the temperature setpoint. If the gas temperature increased by 4°F (2.2 K), the thermal control system was designed to enter a cooling cycle and all four heat exchangers would be turned on. When the gas temperature decreased 2°F (1.1 K) below the temperature setpoint, all four heat exchangers would be turned off. Then, if the gas temperature increased, one heat exchanger would be turned on for each 1°F (0.56 K) sensed above the setpoint

SKYLAB - ORBITAL WORKSHOP
THERMAL CONTROL SYSTEM SCHEMATIC



to a maximum of four heat exchangers. When the temperature dropped below the setpoint by 2°F (1.1 K) the heat exchanger(s) would be turned off and the thermal control system would remain in the cooling cycle until the cabin gas temperature dropped 4°F (2.2 K) below the temperature setpoint. Then the system would enter the heating cycle and the converse of the cooling cycle would be performed.

B. Passive Thermal Control Subsystem

As designed the passive subsystem design parameters were established by the overall heat balance requirements of the OWS Thermal Control System (TCS), the OWS CEI Specifications and launch vehicle imposed conditions.

1/ High Performance Insulation (HPI)

The HPI subsystem design parameters are shown in Table 2.2.3-6. As shown, the design capability was consistent with the requirements. The primary requirement for the HPI performance was the effective thermal conductance of system.

The High Performance Insulation on the outboard surface of the OWS forward dome consisted of two preinstalled sections and 18 panels which were installed at KSC.

The preinstalled HPI was installed prior to the attachment of the forward skirt and installation of the wiring and plumbing lines. It consisted of two sections: one in the cavity area between the forward skirt and the forward dome, the other around the forward hatch opening (Figures 2.2.3-33 and 2.2.3-34). Both sections consisted of an outer cover comprised of two epoxy impregnated Dacron cloth sheets, each 0.012-inch (0.0305 cm) thick, and 10 layers of single goldized 0.0005-inch (0.00127 cm) Kapton with nine 0.007-inch (0.018 cm) Dacron net separators. The assembly was held together with nylon fasteners. The cavity HPI had tabs bonded to the forward dome; the forward hatch section was held in place by the wire support grid standoffs.

Table 2.2.3-6

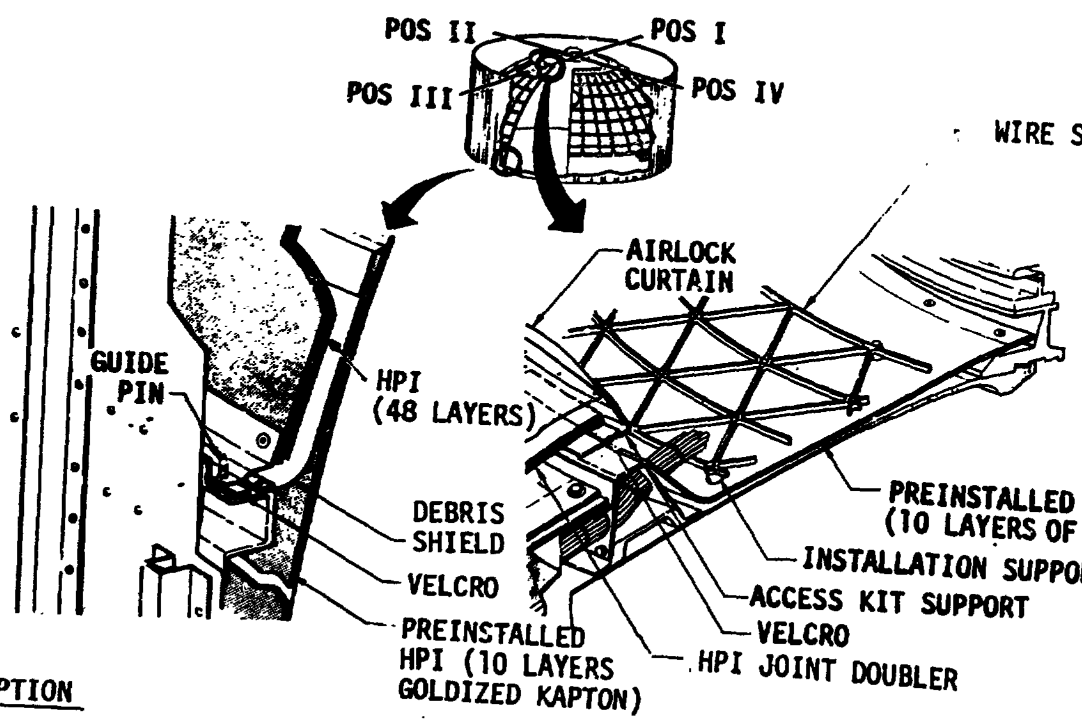
HPI DESIGN PARAMETERS

Parameter	Criteria		
	Minimum	Nominal	Maximum
Effective Thermal Conductance* Btu/hr-ft^2-°F (W/m^2·K)	0.001 (0.0057)	0.005 (0.0285)	0.015(0.0855)
Sheet Emissivity	N/A	0.04	0.10

*CEI Specification calls for conductance $\leq$ 0.02 Btu/hr-ft^2-°F (0.114 W/m^2·K)

ORBITAL WORKSHOP

PREINSTALLED HIGH PERFORMANCE INSULATION



COVER SHEETS - .012 DACRON - EPOXY

REFLECTORS - .0005 GOLDIZED KAPTON

SEPARATORS - .007 DACRON NET

FASTENERS - NYLON BUTTONS

DESCRIPTION

THE PREINSTALLED INSULATION IS INSTALLED PRIOR TO THE ATTACHMENT OF THE FORWARD SKIRT AND INSTALLATION OF THE WIRING AND PLUMBING LINES.

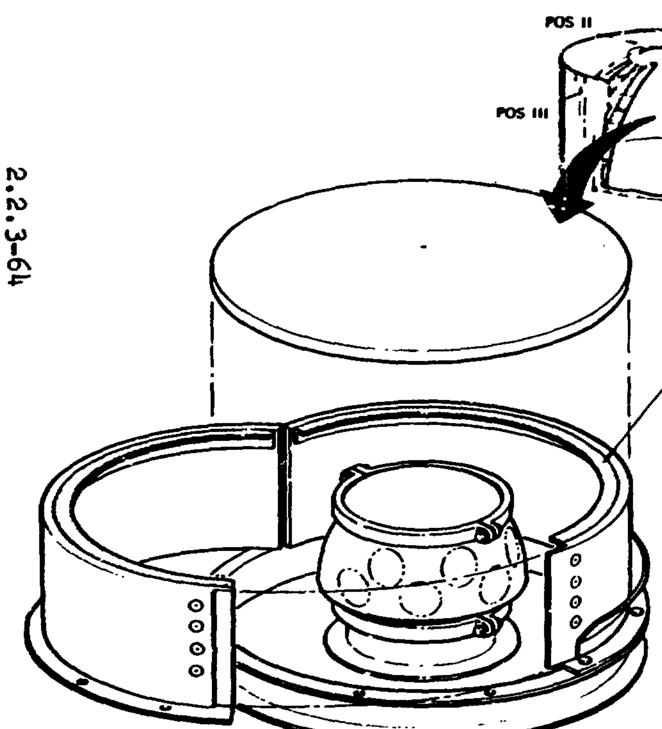
THERE ARE TWO SECTIONS INSTALLED, ONE IN THE CAVITY AREA BETWEEN THE SKIRT AND THE FORWARD DOME, THE OTHER AROUND THE FORWARD HATCH OPENING. THE HPI IS MADE OF TWO COVER SHEETS OF IMPREGNATED DACRON CLOTH, TEN LAYERS OF GOLDIZED KAPTON REFLECTIVE SHIELDS, AND NINE PLIES OF DACRON NET SEPARATORS. THEY ARE HELD TOGETHER WITH NYLON FASTENERS.

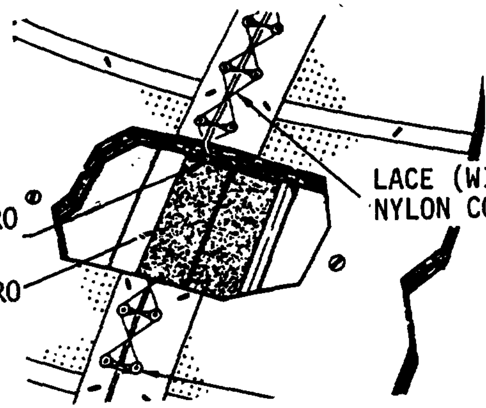
THE LOWER SECTION IS BONDED TO THE TANK; THE FORWARD SECTION IS HELD IN PLACE BY THE WIRE SUPPORT GRID STANDOFFS AND VELCRO FASTENERS.

Figure 2.2.3-33

ORBITAL WORKSHOP

FORWARD DOME HIGH PERFORMANCE INSULATION





MATERIAL

PROTECTIVE COVER - NOMEX COATED NYLON

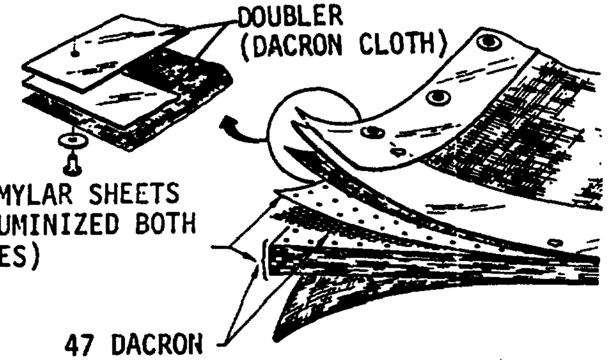
HPI COVER - DACRON EPOXY

REFLECTIVE LAYERS - ALUMINIZED MYLAR

SEPARATOR SHEETS - DACRON NET

ATTACHMENTS - NYLON TEES

INSTALLATION ATTACHMENT - VELCRO & NYLON LACING



DESCRIPTION ·

THE FORWARD DOME HPI CONSISTS OF 18 LARGE SEGMENTS COVERING THE DOME FROM THE PREINSTALLED HPI IN THE CAVITY AREA TO THE PREINSTALLED HPI AT THE HATCH. THE CONSTRUCTION OF THE HPI CONSISTS OF TWO COVER SHEETS OF EPOXY-IMPREGNATED DACRON CLOTH, 48 SHEETS OF .00015 INCH ALUMINIZED MYLAR, AND 47 SEPARATOR SHEETS OF DACRON NET. THE COMPOSITE IS HELD TOGETHER BY NYLON TEE FASTENERS. THE THREE ELECTRICAL FEEDTHRU FITTINGS ARE INSULATED WITH HPI HAT BOXES OF SIMILAR CONSTRUCTION.

THE HPI IS ATTACHED TO THE STRUCTURE BY INSULATED BOLTS AT THE FORWARD END AND AT RADIAL SPLICES BY VELCRO AND NYLON LACING. A COVER MADE OF NOMEX-IMPREGNATED NYLON IS USED TO ENCOMPASS THE HPI PANELS, SERVES TO PROTECT THE HPI, AND ACTS AS A SEAL TO RETAIN THE GAS DURING PURGING. THE COVER IS MADE OF SIX SEGMENTS ATTACHED TOGETHER WITH PLASTIC ZIPPERS.

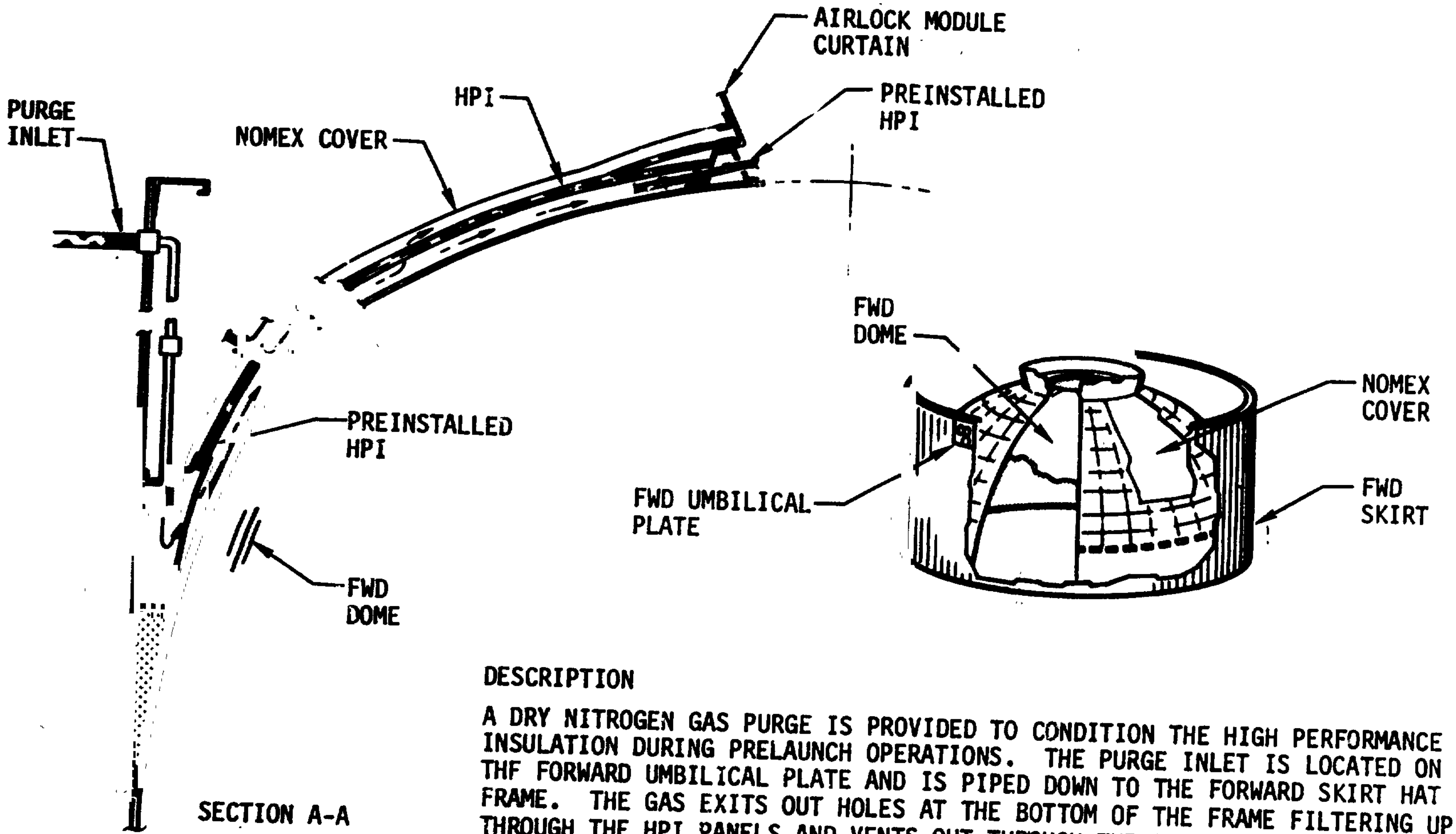
Figure 2.2.3-34

Each of the 18 panels which was placed on the forward dome between the two preinstalled sections (Figure 2.2.3-33) consisted of two cover sheets of epoxy impregnated 0.008-inch (0.020 cm) Dacron cloth, 48 sheets of 0.00015-inch (0.00038 cm) double aluminized Mylar, and 47 separator sheets of 0.008-inch (0.020 cm) Dacron net. The layers were held together by intermittently spaced nylon fasteners (buttons) which penetrated the multilayer insulation and held the assembly to a uniform 0.5-inch (1.27 cm) thickness. The forward end of each panel was bolted to the support ring. The aft end was attached to a hat-shaped ring frame on the forward skirt by Velcro fasteners.

The 18 panels were fastened together with nylon lacing and Velcro fasteners (Figure 2.2.3-34). The three electrical feedthrough areas were insulated with HPI hat boxes of similar construction (Figure 2.2.3-34).

In order to prevent trapped moisture from degrading the aluminized Mylar surface emittance, a dry nitrogen purge was provided following installation of the 18 HPI panels on the OWS. The purge system is shown in Figure 2.2.3-35.

ORBITAL WORKSHOP
HIGH PERFORMANCE INSULATION PURGE SYSTEM



DESCRIPTION

A DRY NITROGEN GAS PURGE IS PROVIDED TO CONDITION THE HIGH PERFORMANCE INSULATION DURING PRELAUNCH OPERATIONS. THE PURGE INLET IS LOCATED ON THF FORWARD UMBILICAL PLATE AND IS PIPED DOWN TO THE FORWARD SKIRT HAT FRAME. THE GAS EXITS OUT HOLES AT THE BOTTOM OF THE FRAME FILTERING UP THROUGH THE HPI PANELS AND VENTS OUT THROUGH THE AIRLOCK MODULE CURTAIN.

Figure 2.2.3-35

The function of the HPI was to reduce the heat loss and gain from cold and hot sides, respectively, of the OWS forward dome region. The multiple layers of low emissivity insulation provided a high thermal resistance between the exterior surface of the forward dome and its environment. The low emissivity layers were separated by Dacron net and perforated to allow venting after launch. Since the layers were separated and vented, gas convection and conduction were virtually eliminated. Overlapping joints were used in assembling the HPI to minimize the heat leaks at the joints.

2/ Thermal Control Coatings and Shielding

a. Design Configuration

The OWS thermal control coating design parameters are shown in Table 2.2.3-7. The selected coatings met the outgassing requirements of MSFC Drawing 50M02442. The thermal control coatings were used to regulate the radiant thermal energy exchange between the OWS and its environment. As designed the exterior of the OWS was black except for the white painted areas noted in Figures 2.2.3-36 and 2.2.3-37. The white and black paint pattern on the hot side of the meteoroid shield was designed to absorb sufficient solar energy such that the TCS would meet the astronaut temperature comfort criteria using the specified convective heater power.

Table 2.2.3-7
ORBITAL WORKSHOP OPTICAL COATINGS

Locations	Optical Properties	Coating
A. Forward tank exterior		
1. Between meteoroid shield and tank wall	$\varepsilon \leq 0.04$ ≥ 0.8	Goldized Kapton on tank Green Teflon on M/S
2. Between forward dome (aft of debris shield and skirt)	$\varepsilon \leq 0.05$	Goldized Kapton (both surfaces)
3. Between aft dome and skirt	$\varepsilon \leq 0.05$	Goldized Kapton (both surfaces)
B. Meteoroid shield		
1. Exterior (except locally)	$\alpha = \varepsilon \geq 0.85$	Cat-A-Lac flat black enamel
2. α locally on sun side	$\alpha \geq 0.22$	S-13G white paint
C. Skirt thermal shields		
1. Between shield and skirt	$\varepsilon \leq 0.05$	Goldized Kapton on shield
2. Exterior	$\alpha = \varepsilon \geq 0.85$	Cat-A-Lac flat black enamel
D. Meteoroid shield closures		
1. Interior	$\varepsilon \geq 0.75$	Green Teflon
2. Exterior	$\alpha = \varepsilon \geq 0.85$	Cat-A-Lac flat black enamel
E. Forward skirt	$\alpha = \varepsilon \geq 0.85$	Cat-A-Lac flat black enamel
1. Exterior		
F. Aft skirt exterior		
1. Aft 36 in. (0.914 m)	$0.19 < \alpha/\varepsilon$ ≤ 0.36	S-13G white paint
2. Sta. 236.6 to 253.1	$\alpha = \varepsilon \geq 0.85$	Cat-A-Lac flat black enamel
G. Thrust structure exterior	$\alpha = \varepsilon \geq 0.85$	Cat-A-Lac flat black enamel

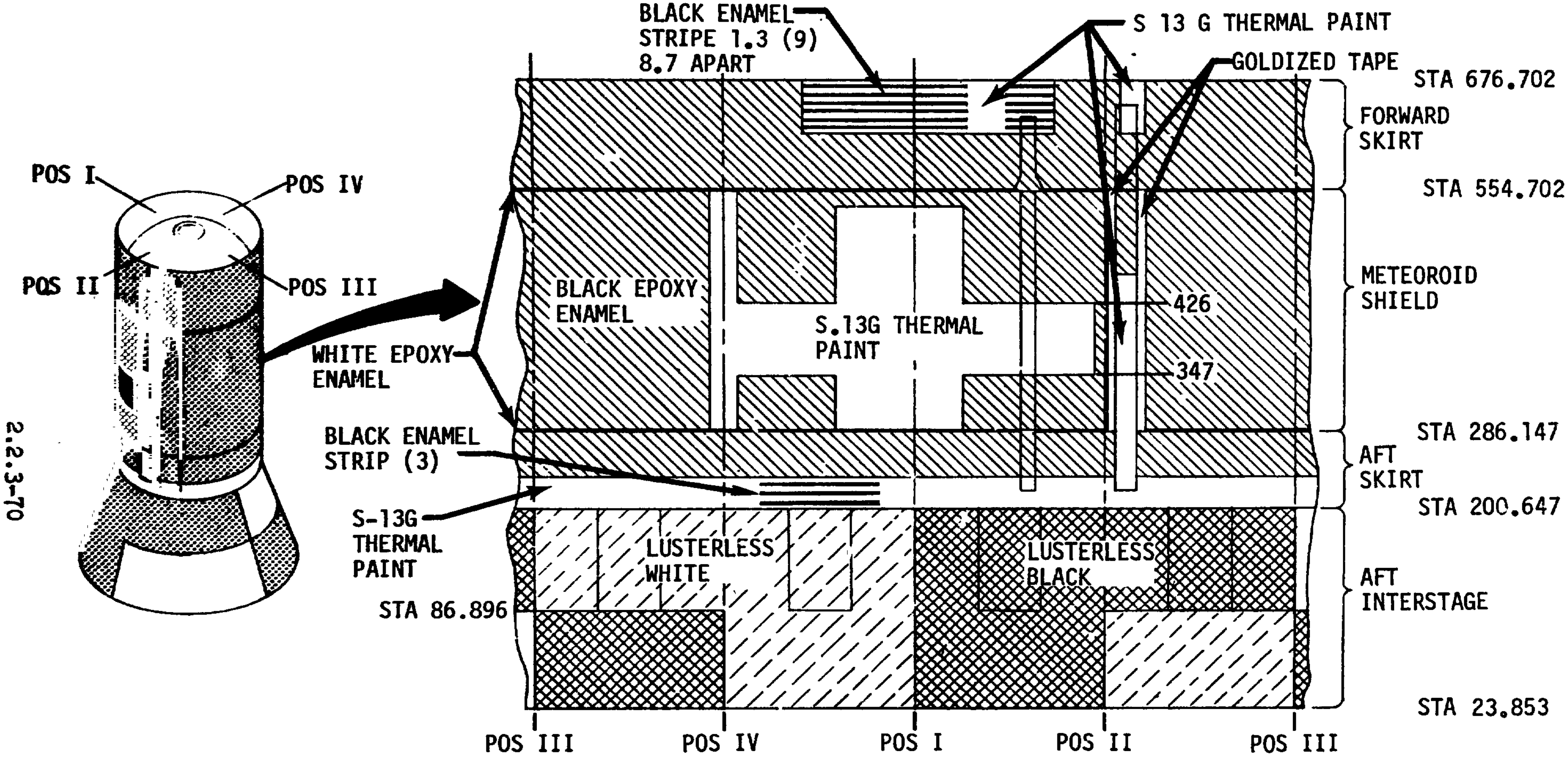
Table 2.2.3-7 (continued)

ORBITAL WORKSHOP OPTICAL COATINGS

Locations	Optical Properties	Coating
H. Crew quarters vertical walls	$\varepsilon \geq 0.75$	Colored anodize
I. Fire retardant liner	$0.5 \leq \varepsilon \leq 0.9$	Colored Teflon
J. Crew quarters floor	$\varepsilon \geq 0.75$	Yellow anodize
K. Crew quarters ceiling	$\varepsilon \geq 0.75$	Yellow anodize

ORBITAL WORKSHOP

EXTERNAL PAINT PATTERN



PAINT PATTERN - SHOWN FLAT (UNDEPLOYED METEOROID SHIELD)

DESCRIPTION

THE EXTERNAL SURFACE OF THE VEHICLE INCLUDING THE METEOROID SHIELD IS PAINTED BLACK AND WHITE.

THE AFT INTERSTAGE IS COATED WITH WHITE AND BLACK LUSTERLESS PAINT. AN AREA BETWEEN EACH POSITION PLANE 30° WIDE BETWEEN STA 86.896 AND 200.647 IS COATED WITH A 0.10 INCH [0.254 CM] THICK SILICONE INSULATION.

THE FORWARD SKIRT, AFT SKIRT, MAIN TUNNEL, AND METEOROID SHIELD IS COATED WITH A WHITE (S-13G) THERMAL CONTROL PAINT AND BLACK EPOXY ENAMEL.

THE CYLINDER IS COATED-BOTH FORWARD AND AFT BETWEEN THE GOLD TAPE AND SKIRT ATTACH ANGLES WITH WHITE EPOXY ENAMEL. GOLD TAPE IS VISIBLE ALONG EACH SIDE OF THE MAIN TUNNEL.

SKYLAB - ORBITAL WORKSHOP
EXTERNAL WHITE PAINT PATTERN

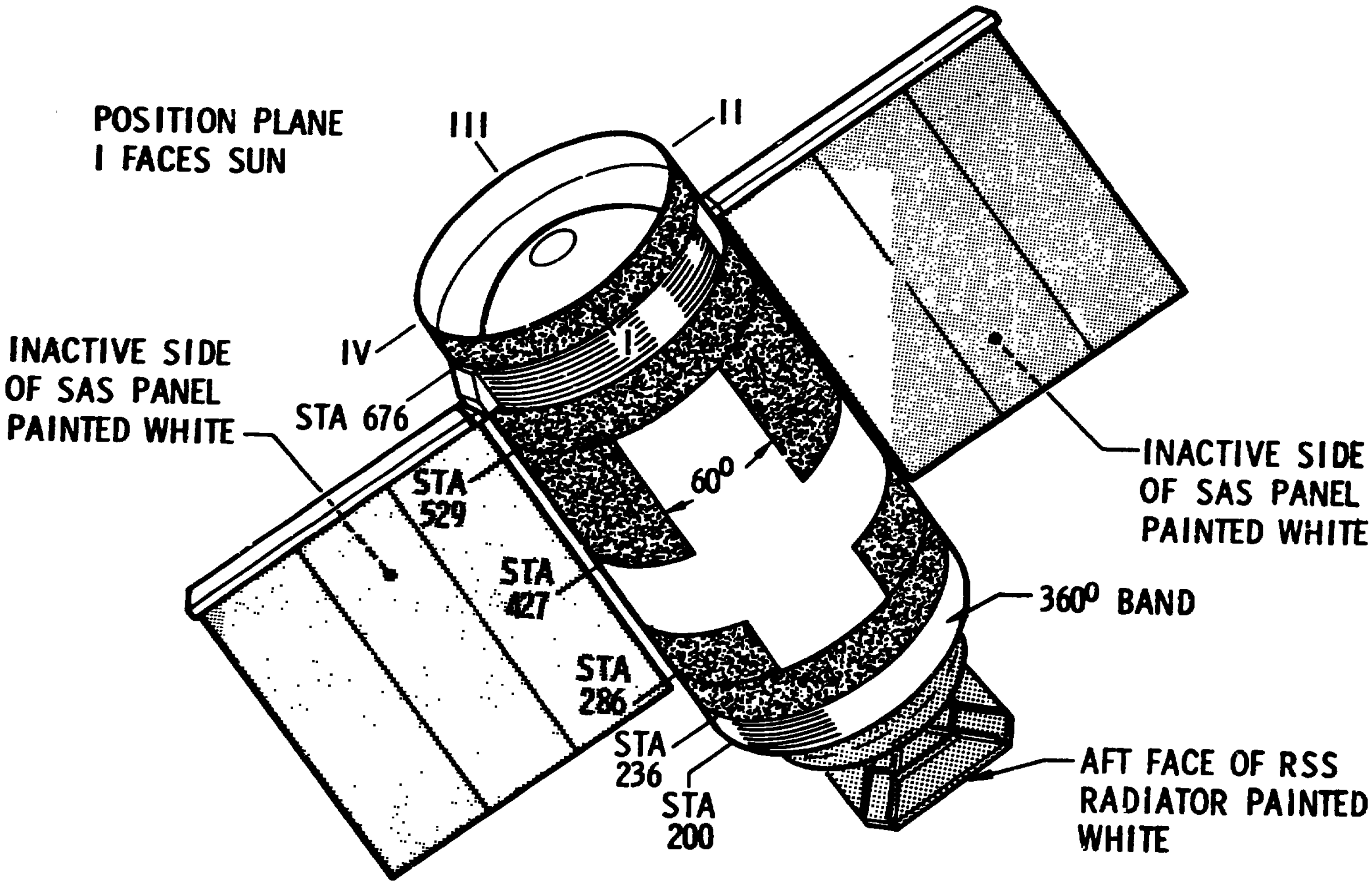
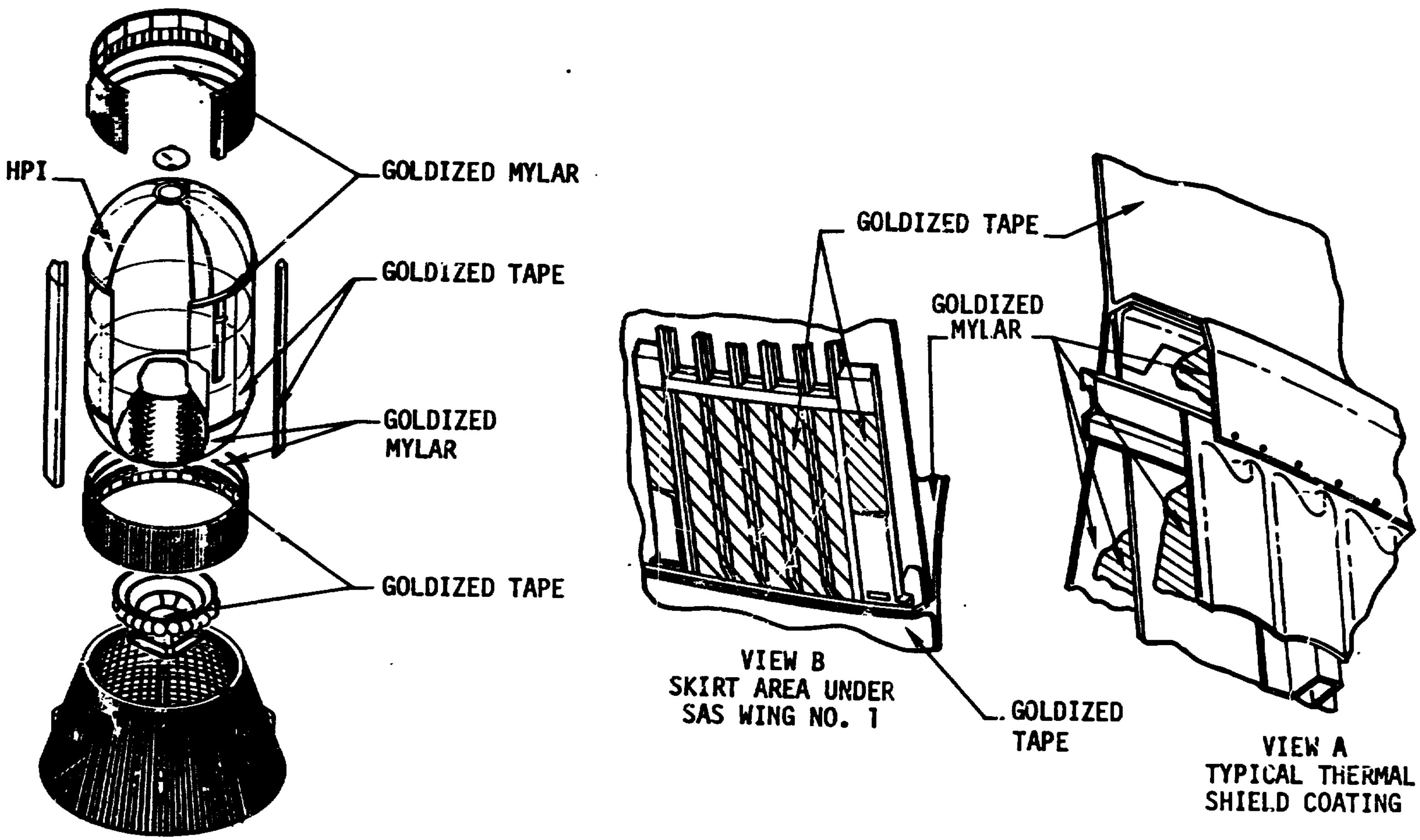


Figure 2.2.3-37

The exterior OWS wall beneath the meteoroid shield was covered with a low emissivity coating (goldized Kapton tape) to minimize the heat transfer between the wall and shield (Figure 2.2.3-38). This coating was intended to dampen the amplitude of the internal wall temperatures caused by the day/night orbital cycles. Meteoroid shield boots were designed to reduce the heat loss from the annulus formed by the meteoroid shield and the OWS tank wall (Figure 2.2.3-39). The low emissivity coatings used on the skirts and the domes adjacent to the skirt-tank joints minimized the heat transfer through the joints. The skirts and domes would have acted as radiative fins for heat conducted across the joints if the low emissivity coatings were not provided. Thermal shields covered portions of the OWS skirts adjacent to the skirt-tank joints (Figure 2.2.3-40 and 2.2.3-41) and protected the coatings in this region from degradation both prior to and during flight.

The OWS interior surfaces in the habitation area had high emissivities to provide thermal radiation interchange between all walls and partitions.

ORBITAL WORKSHOP
THERMAL RADIATION COATING



MATERIAL

GOLDIZED TAPE - POLYIMIDE FILM, PRESSURE SENSITIVE ADHESIVE TAPE STM0570-01

GOLDIZED MYLAR - METALIZED POLYIMIDE FILM STM0524-01

Figure 2.2.3-38

SKYLAB - ORBITAL WORKSHOP
METEOROID SHIELD BOOT

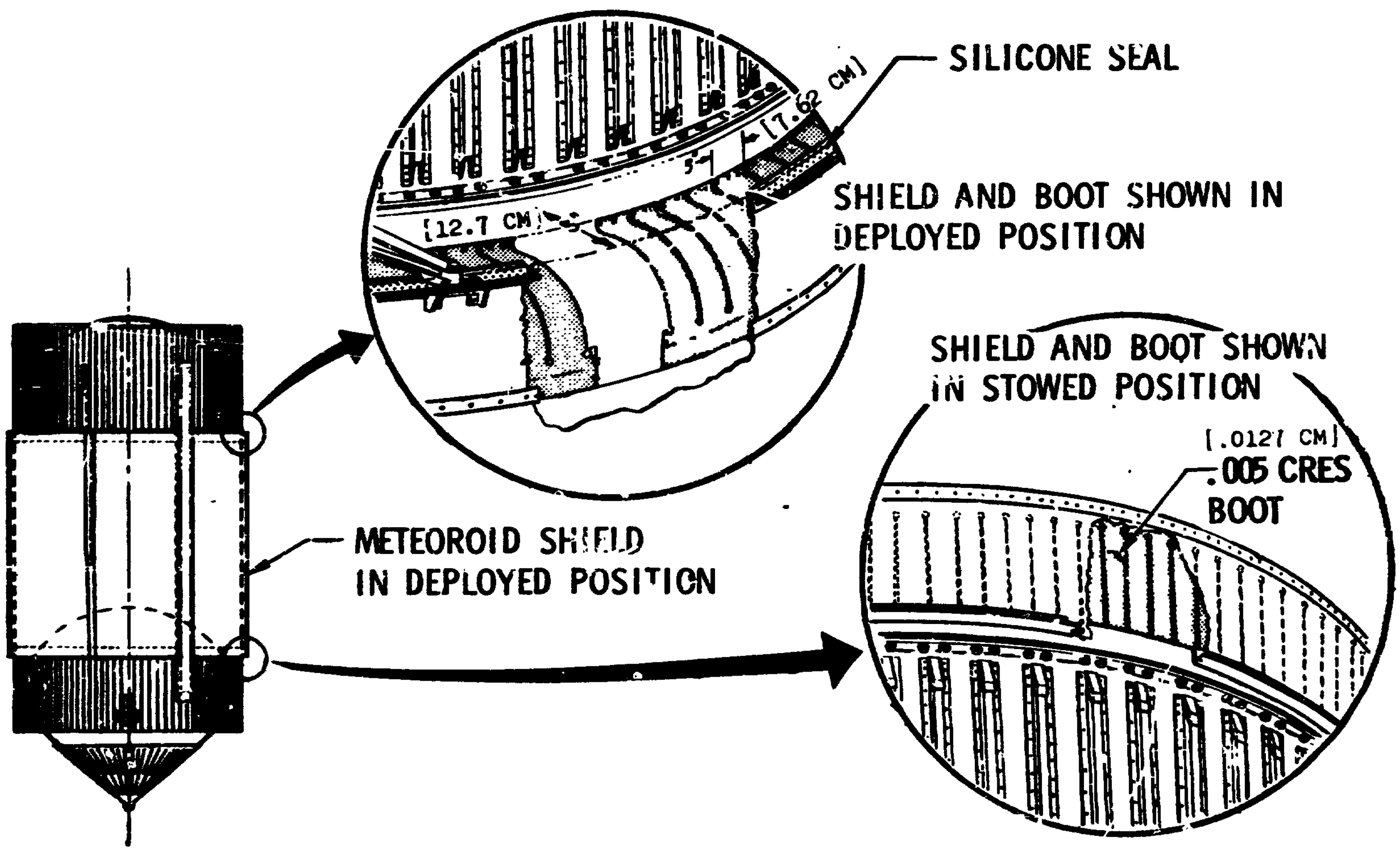


Figure 2.2.3-39

SKYLAB - ORBITAL WORKSHOP FORWARD THERMAL SHIELD

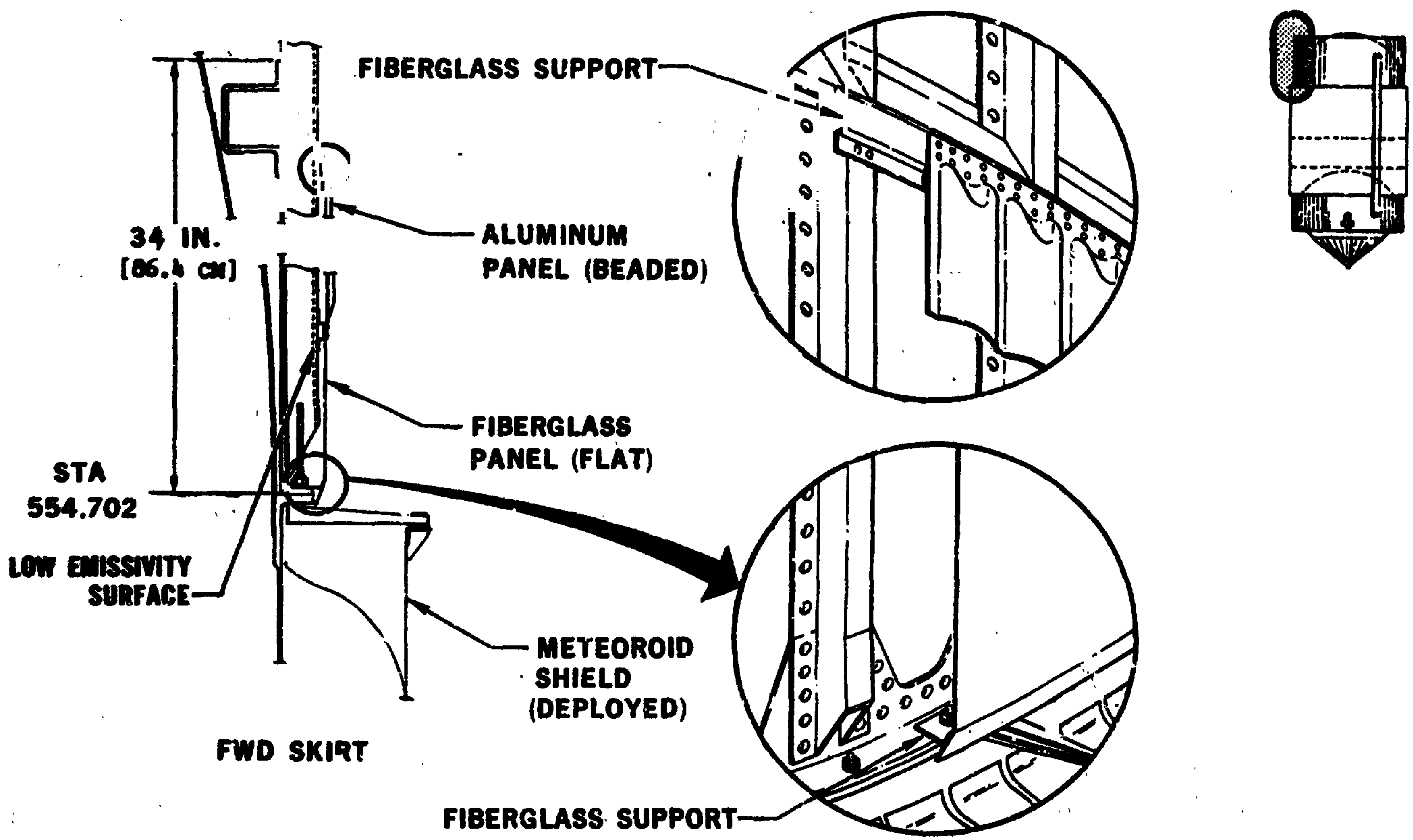


Figure 2.2.3-40

ORBITAL WORKSHOP
AFT THERMAL SHIELD

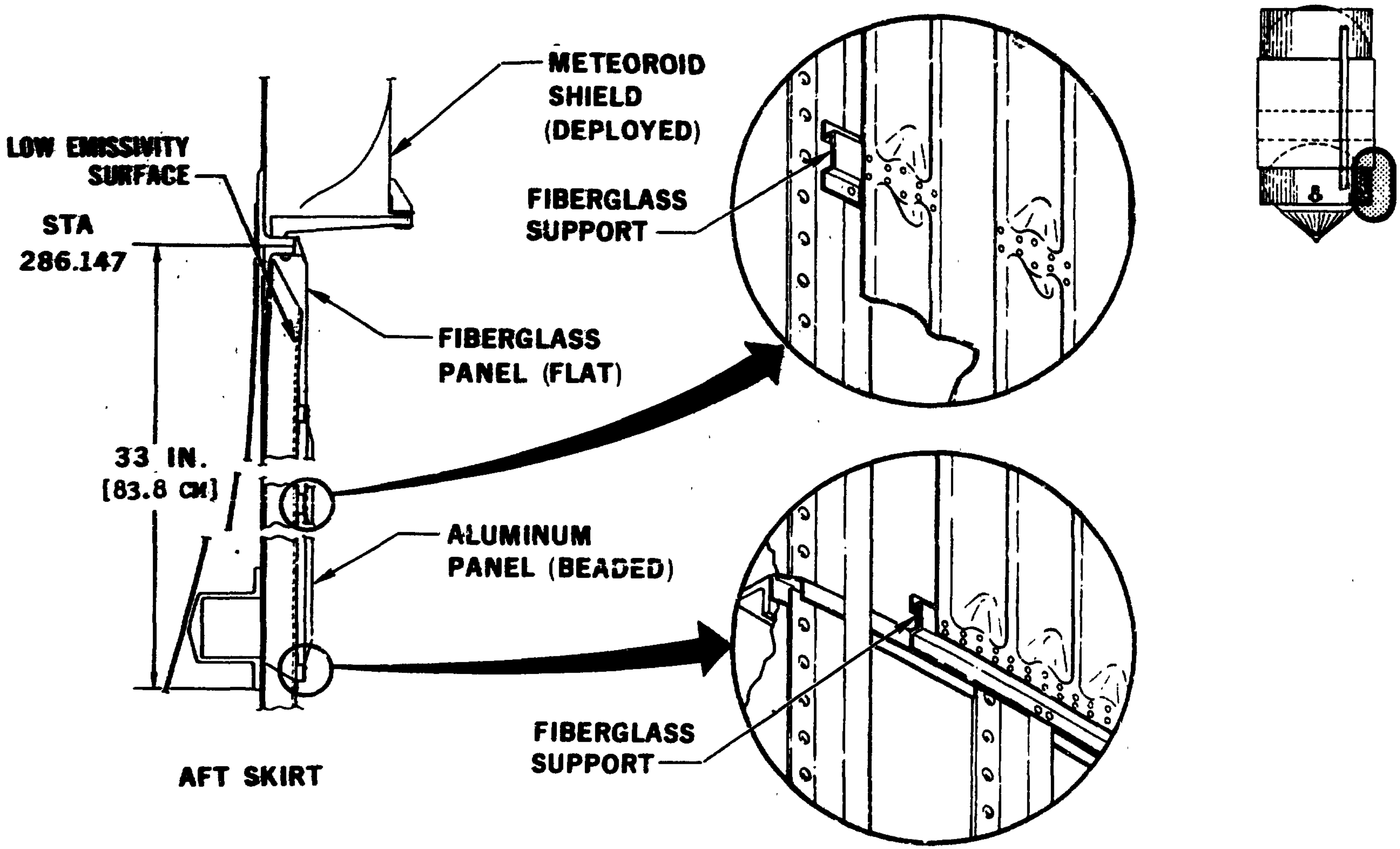


Figure 2.2.3.41

t. Orbital Configuration

Because of the loss of the meteoroid shield which was to passively control the external environment heat inputs to the OWS Habitation Area cylindrical wall, it was necessary to revise the vehicle's orbital configuration during flight to achieve an acceptable internal temperature range. Thermal analyses showed that thermal control of the OWS could be achieved by deploying a sunshade over the goldized kapton tape on the sun side of the vehicle. It was determined that the low emittance (ϵ) surface of the gold tape in combination with a low ϵ coating on the sunshade facing the OWS would allow greater flexibility in the selection of an acceptable material for the shade, since for this condition, the shade side facing the sun could have a relatively high solar absorptance (α_s) as long as $\alpha_s/\epsilon \leq 1$.

Several shade designs employing immediately available materials were developed for both EVA and IVA deployment. The first deployed was an IVA type through the solar side Scientific Airlock on DOY 147. Known as the JSC Parasol, this sunshade had a segmented central pole with four telescoping spring loaded rods that deployed a rectangular fabric canopy. The shade reduced the temperature inside the OWS to a habitable range despite the fact that it did not deploy to its full rectangular shape. It was estimated from fly-around photographs that 135 ft^2 (12.5 m^2) of goldized Kapton projected area was exposed to direct solar impingement. Figure 2.2.3-42 shows the configuration of the Skylab with the parasol fully deployed. The canopy was made of orange ripstop nylon laminated with aluminized mylar. The nylon side, α_s/ϵ = 0.37/0.86, faced the sun and the aluminized side, α_s/ϵ = 0.08/0.02, faced the vehicle. A more complete description may be found in the Skylab Program Operational Data Book, Appendix B.1.

The second shade, known as the MSFC Sail, was deployed by the SL-3 crew during an EVA on DOY 218. It was deployed over the parasol to provide additional coverage. The sail material was similar to the parasol except the aluminized side of the mylar was bonded to the nylon and S-13G paint was applied over the nylon. The solar side α_s/ϵ was 0.25/0.90 and the vehicle side was 0.15/0.34. The sail was **stabilized by two rods attached** to the ATM and ropes. The nominal configuration is shown in Figure 2.2.3-43. Additional information on this sail configuration can be found in the Skylab Operational Data Book Appendix B.3.

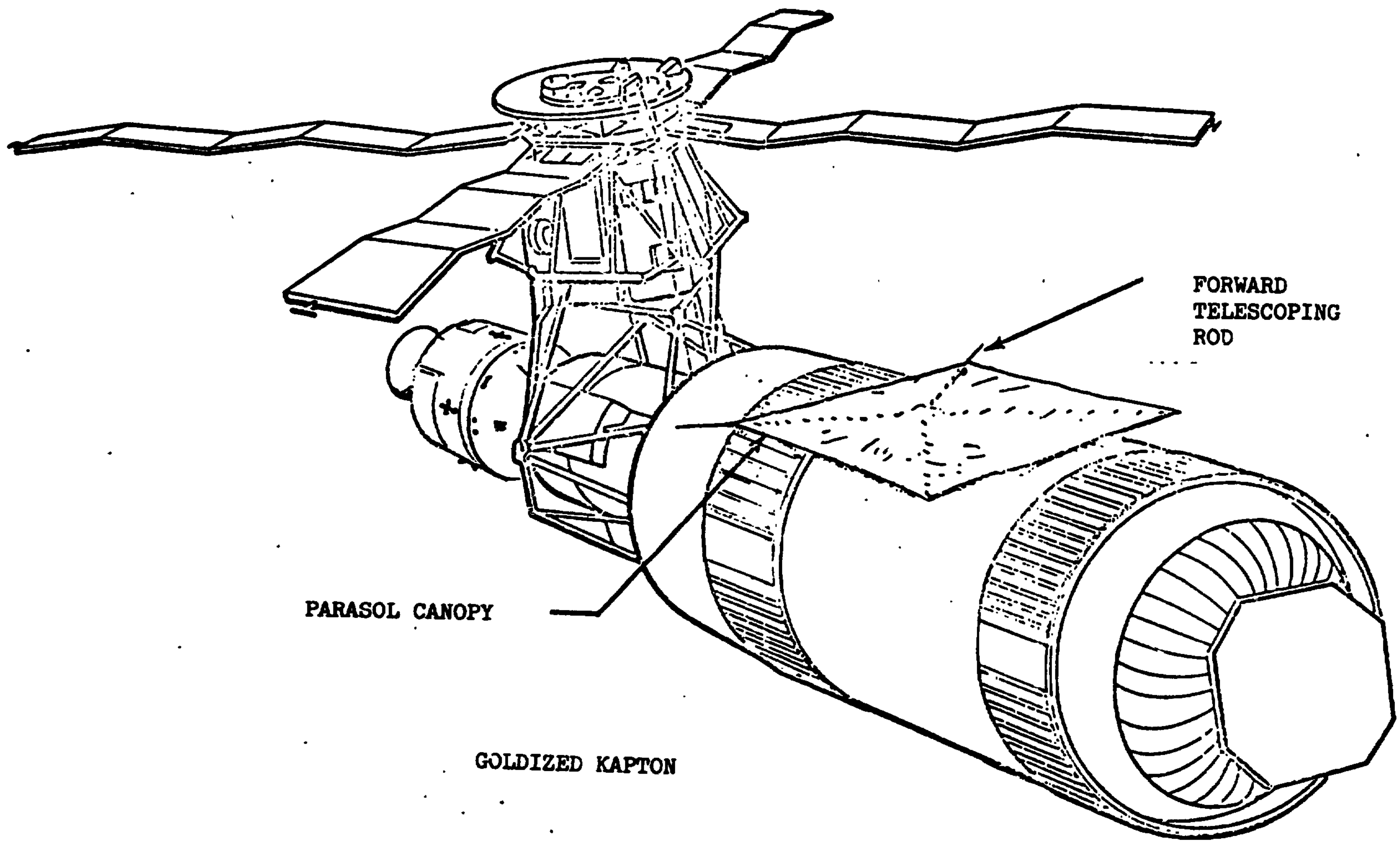


Figure 2.2.3-42 JSC Parasol Configuration

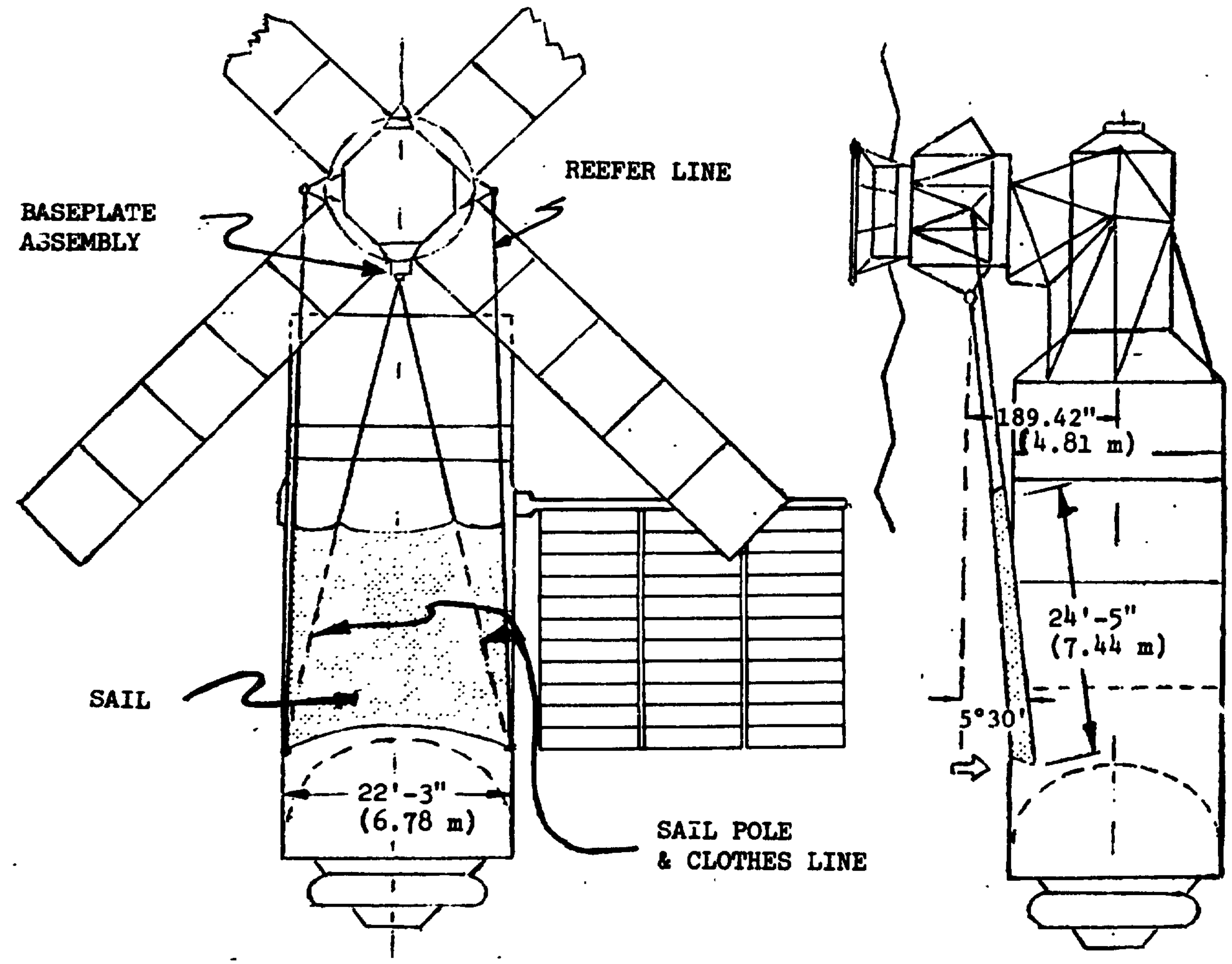


Figure 2.2.3-43 MSFC Solar Sail Configuration

3/ Heat Pipes

The heat pipe design parameters are shown in Table 2.2.3-8. The design temperatures were the maximum and minimum extremes including the OWS shipping environments.

Table 2.2.3-8

HEAT PIPE DESIGN PARAMETERS

Parameter	Shipping	Orbit Nominal	Orbit Tolerance
Temperature, °F (K)	-40 to 200 (233 to 367)	75 (297)	±70 (±39)
Heat loads, watts			
Balsa wood	N/A	31	±11
Water bottles	N/A	23	± 7
Floors	N/A	10	± 5
Logic box	N/A	18	±12

For the OWS applications, heat pipes were designed to provide heat inputs to internal surface areas which would have had minimum temperatures below 55°F (286 K) during habitation. Areas below 55°F (286 K) would have been susceptible to water vapor condensation and would have presented a medium for the growth of bacteria. The basic function of the heat pipes was to provide high thermal conductance paths which allowed heat to be transferred readily from hot areas to raise the temperatures of the cold areas.

An arterial heat pipe (Figure 2.2.3-44) was utilized which had type 304 stainless steel screening for the 0.094-inch (0.24 cm) diameter artery and the wick. The pipe was made of type 6061 aluminum and utilized Freon-22 as a heat transfer fluid.

At each of the four OWS locations shown in Figure 2.2.3-45 there were two rings of heat pipes attached to the internal tank wall. Double rings of heat pipes were necessary since a single heat pipe 67.5 feet (20.6 m) long (the circumference of the OWS) was not available and would have been too long to install in one piece. The double rings provided the cascading effect (heat transfer continuity between overlapping pipes) necessary to transfer thermal energy from one side of the vehicle to the other. Each heat pipe, 160-inches (4.06 m) long, overlapped the adjacent heat pipe by 50 percent to achieve the necessary cascading effect.

Figure 2.2.3-46 shows the forward joint heat pipe installation on the balsa wood. The thermal requirements dictated that the installation of these heat pipes be different between the hot and cold sides of the vehicle. On the hot side, which faced the sun during the solar inertial vehicle orientation, 27 aluminum heat shorts and 1/8-inch (0.318 cm) thick aluminum mounting plates were installed to conduct the required heat

SKYLAB - ORBITAL WORKSHOP OWS HEAT PIPE WICK CONFIGURATION

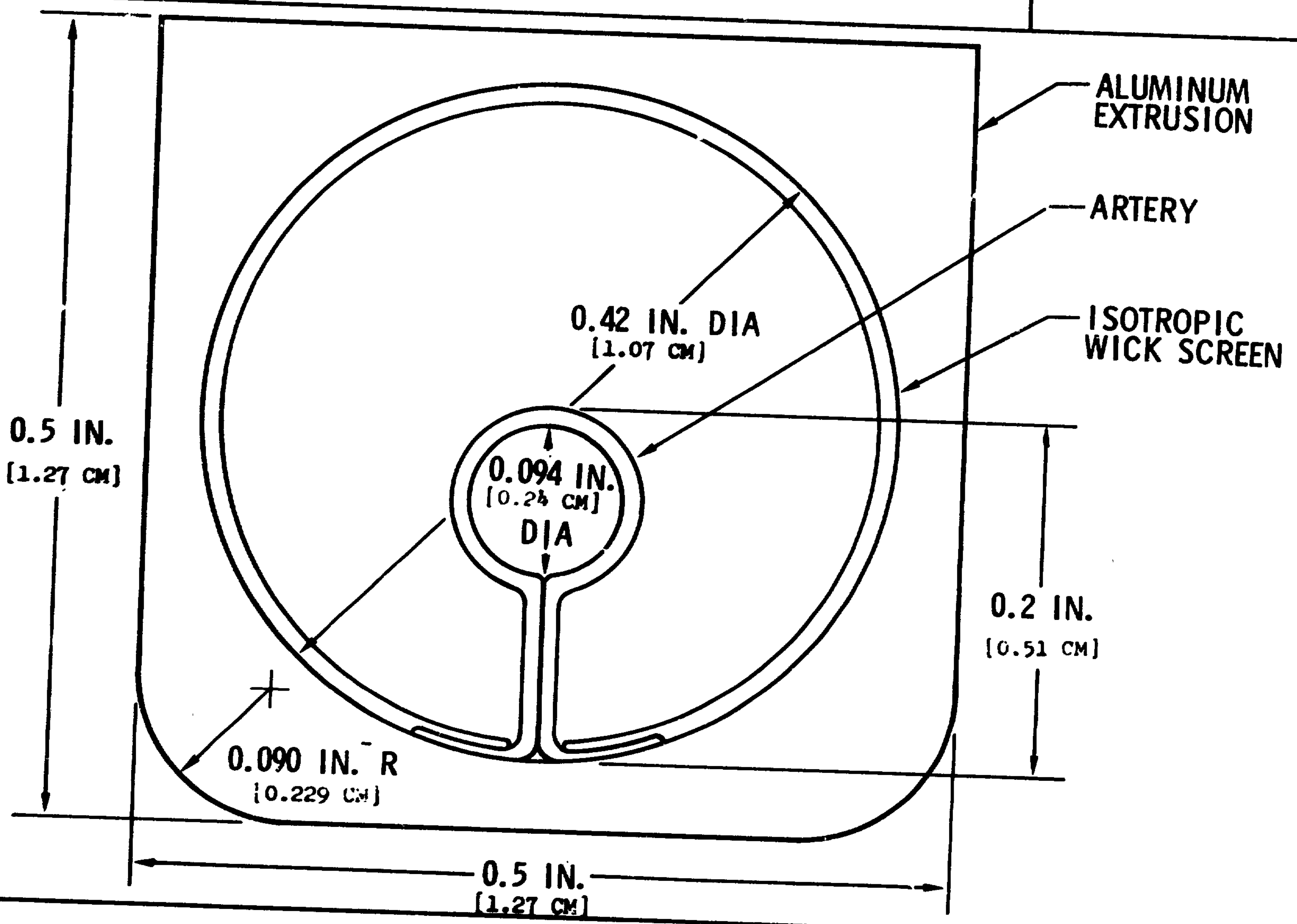


Figure 2.2.3-44

INTERNAL TANK WALL
HEAT PIPE LOCATIONS

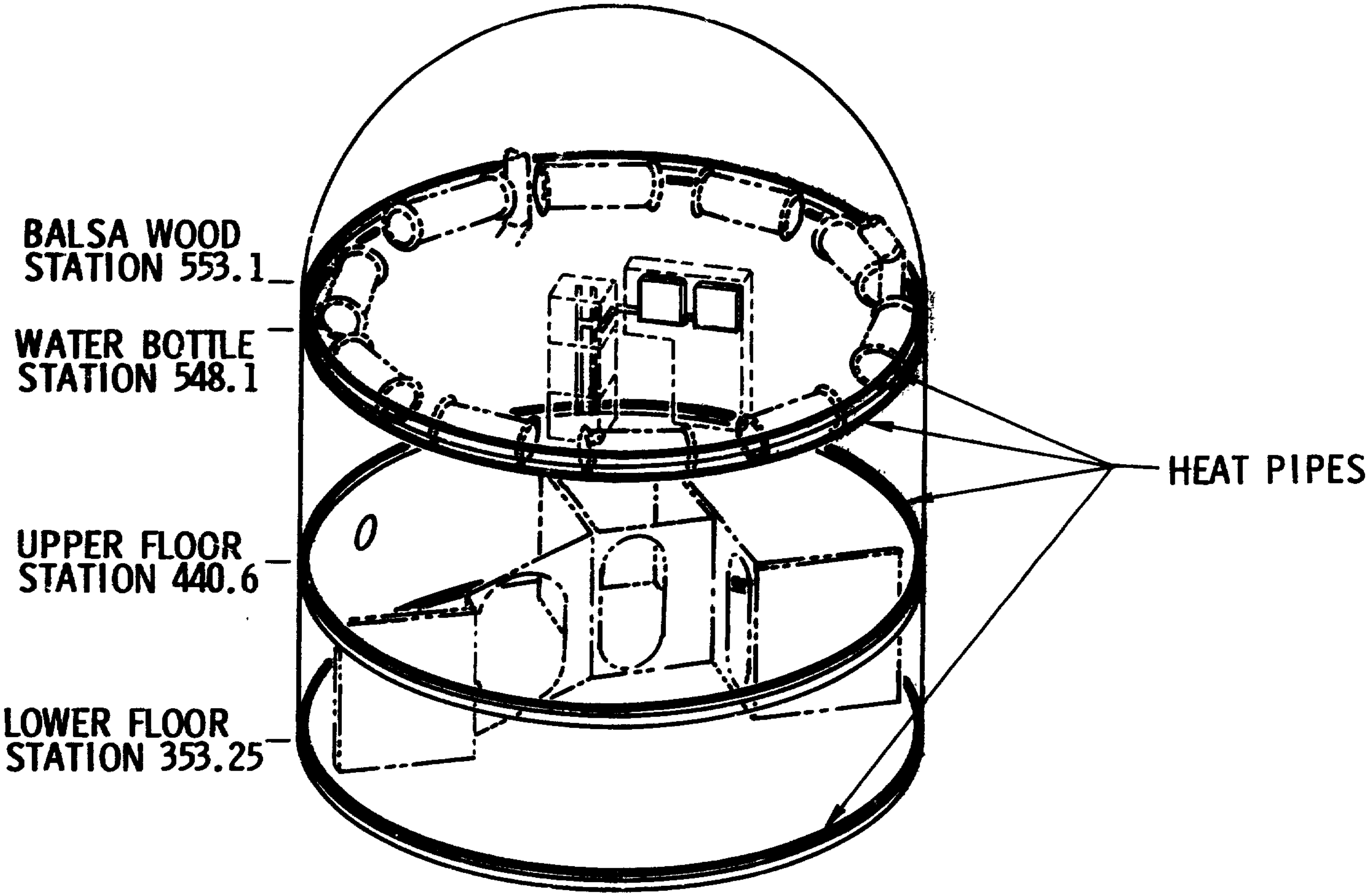


Figure 2.2.3-45

SKYLAB - ORBITAL WORKSHOP
HEAT PIPE INSTALLATION
AT WATERBOTTLE/BALSA WOOD

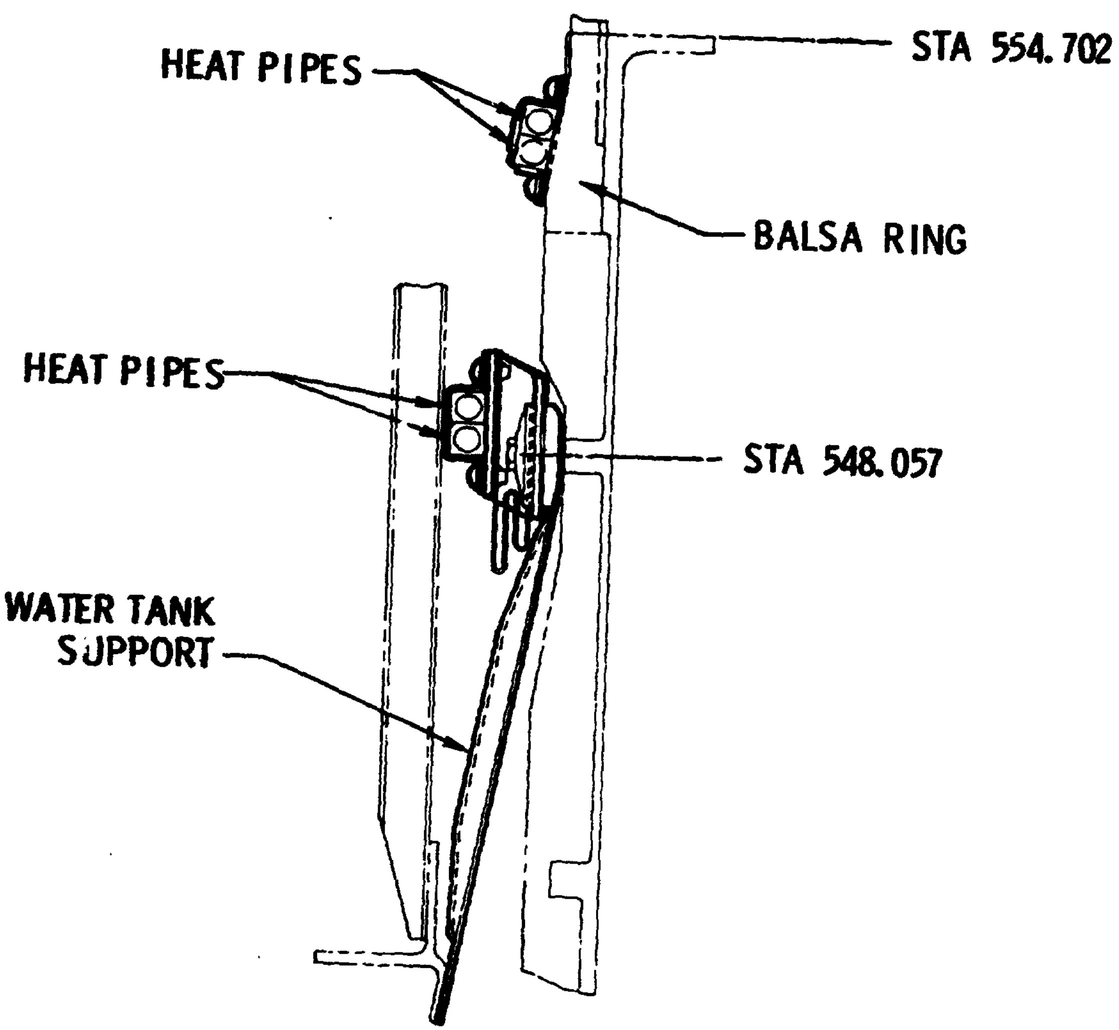


Figure 2.2.3-46

from the relatively hot vehicle structure to the heat pipes. On the cold side which normally faced the earth or deep space, 66 .040-inch (0.102 cm) plates of high conductivity aluminum alloy were used to distribute the heat. The cold side plates covered almost the entire surface where temperatures below 55°F (286 K) were expected without the heat pipes.

The thermal contact resistance between the cascaded heat pipes was minimized by using DC-340 grease on all thermally critical contact areas and by maintaining a contact pressure of 5 psi (3.45 N/cm ^{2}) with a holding clip. The clip had top and side cantilever springs which forced the heat pipes down on the installation plates and against each other. Good thermal contact between the heat pipes was required to permit cascading the heat pipes without impairing overall system performance.

The heat pipe installations behind the water bottles and at the upper and lower floors (Figures 2.2.3-47 and 2.2.3-48) were similar. In all three cases the heat pipes were attached to the interior of the tank wall at 63 waffle intersection points. The requirement of a low thermal resistance, not more than 1.2°F-hr/Btu (2.28 K/w), between the heat pipes and the tank wall dictated a relatively

SKYLAB - ORBITAL WORKSHOP
HEAT PIPE INSTALLATION
AT FLOORS

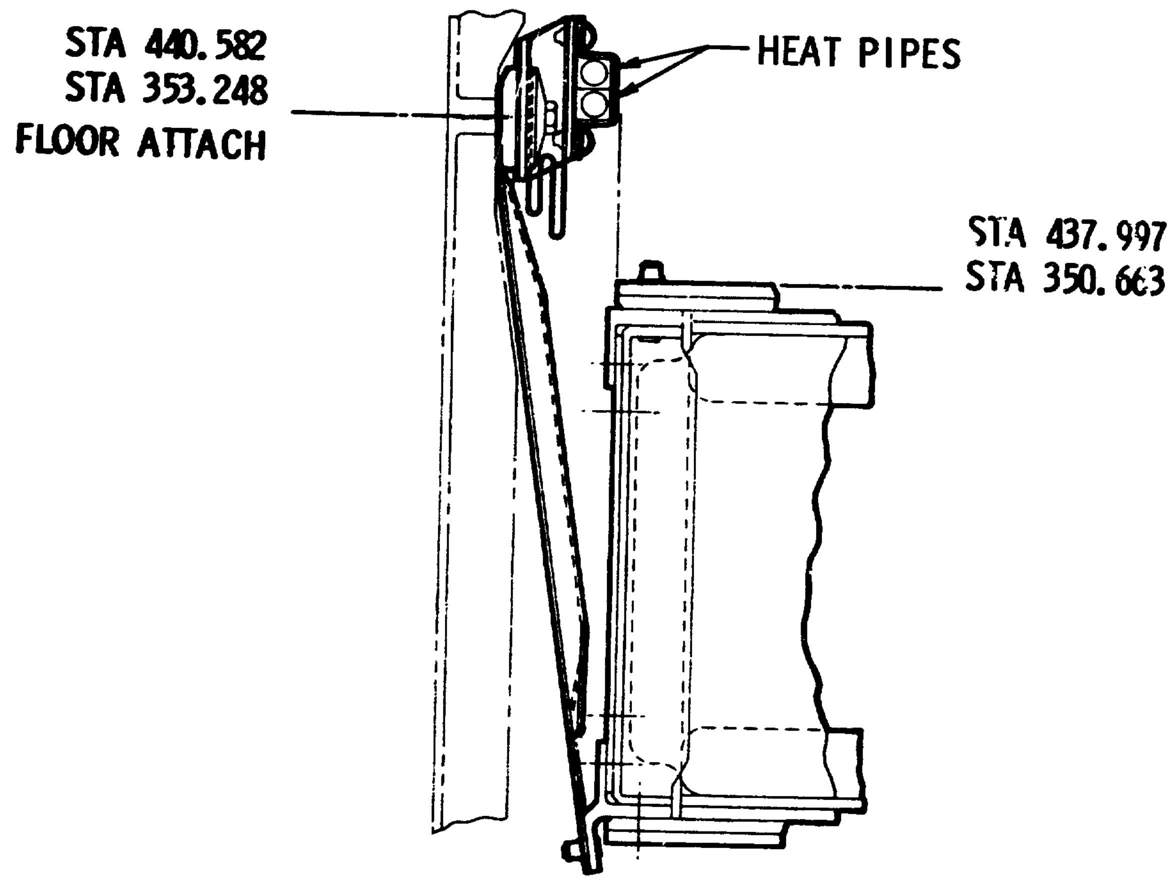


Figure 2.2.3-47

SKYLAB - ORBITAL WORKSHOP
HEAT PIPE SUPPORT ASSEMBLY

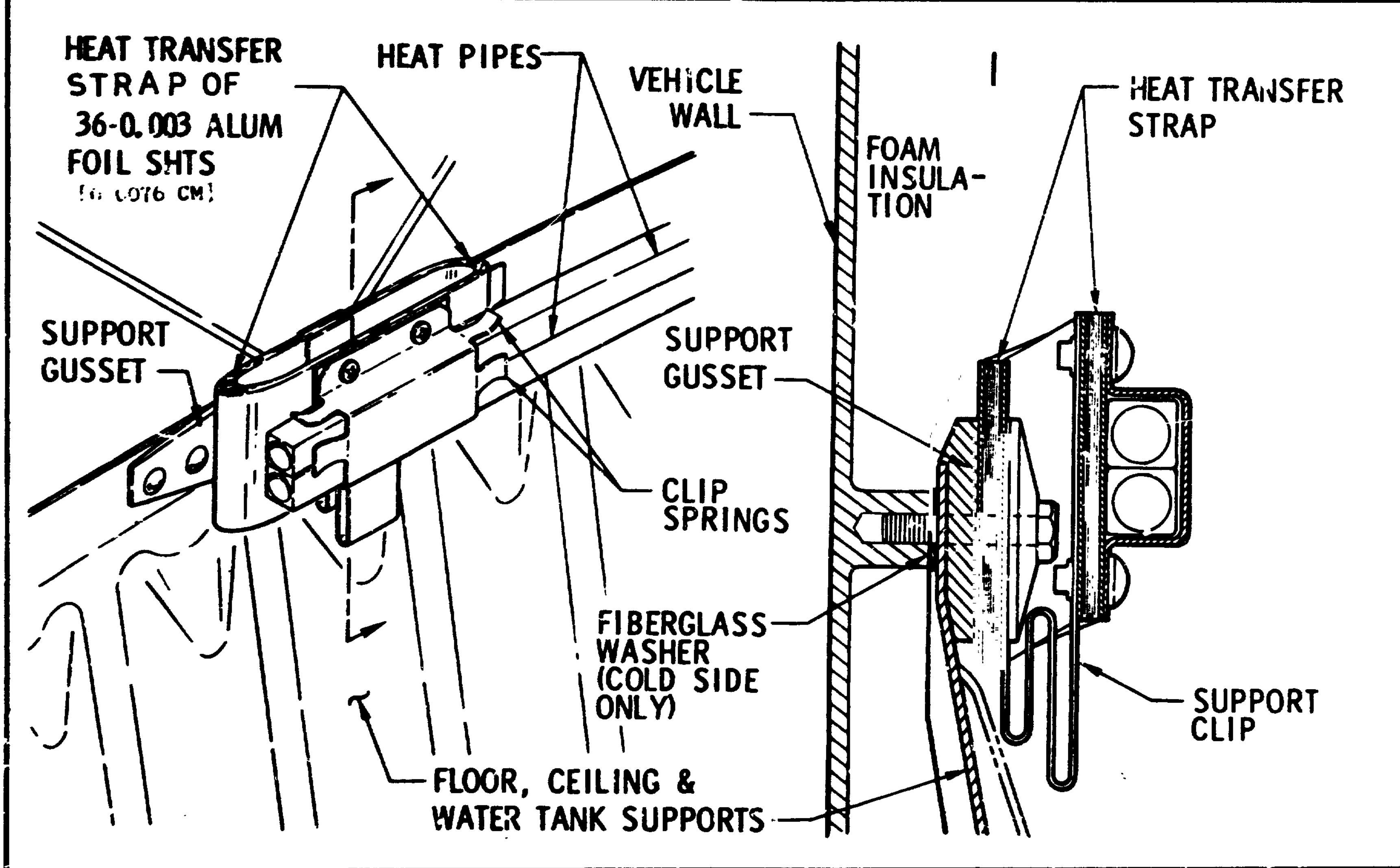


Figure 2.2.3-48

short, thick, high conductivity clip which had to be flexible to accommodate the tank radial expansion during boost. The movement of the heat pipe attach points caused by the changing vibration, pressure and temperature during boost required a low stiffness clip to attach the heat pipes to the tank wall without over-stressing the heat pipes; consequently, the design of the heat pipe support assembly had to accommodate the conflicting requirements of low thermal resistance and minimal stiffness. The heat pipe support assembly is shown in Figure 2.2.3-48. A support clip with three U-bends was used to provide a flexible support between the heat pipes and the tank wall. A low resistance heat transfer path was provided by the heat transfer strap which was made of a stack of 36 soft aluminum foils, 0.003-inch (0.0076 cm) thick. The aluminum foils did not contribute significantly to the bending stiffness of the assembly. The support clip was used because the heat transfer strap was not structurally adequate to support the heat pipes during the launch vibration environment. The heat pipes were attached to the heat transfer strap by sheet metal clips with leaf springs that forced the heat pipes against each other and against the strap for good thermal contact. The thermal contact resistance was minimized by the use of DC-340 compound.

A key factor in the heat pipe system design was the use of phenolic-fiberglas washers on the bolts to thermally isolate the floor, ceiling and water tank supports from the tank wall on the cold side of the vehicle. Electrical grounding requirements prevented the use of fiberglas washers on all the cold side bolts; thus, only about two-thirds of the bolts were so equipped.

Heat pipes were also used to transfer the heat dissipated by the logic unit electronics to the back of the forward compartment food freezer, thereby maintaining that surface above the atmospheric dew point temperature. The logic unit heat pipe installation is shown in Figure 2.2.3-49. The heat pipes were installed on the logic unit containers by eight attachment clips equipped with leaf springs similar to those used on the balsa wood area heat pipes. The heat pipes were held on the back of the freezer by 25 attachment clips. One of the more significant mechanical requirements of this installation was the necessity of bending the heat pipes to a 3-inch (7.6 cm) bend radius. This was needed to provide a continuous thermal path from the logic boxes to the back of the freezer.

SKYLAB - ORBITAL WORKSHOP

HEAT PIPE INSTALLATION

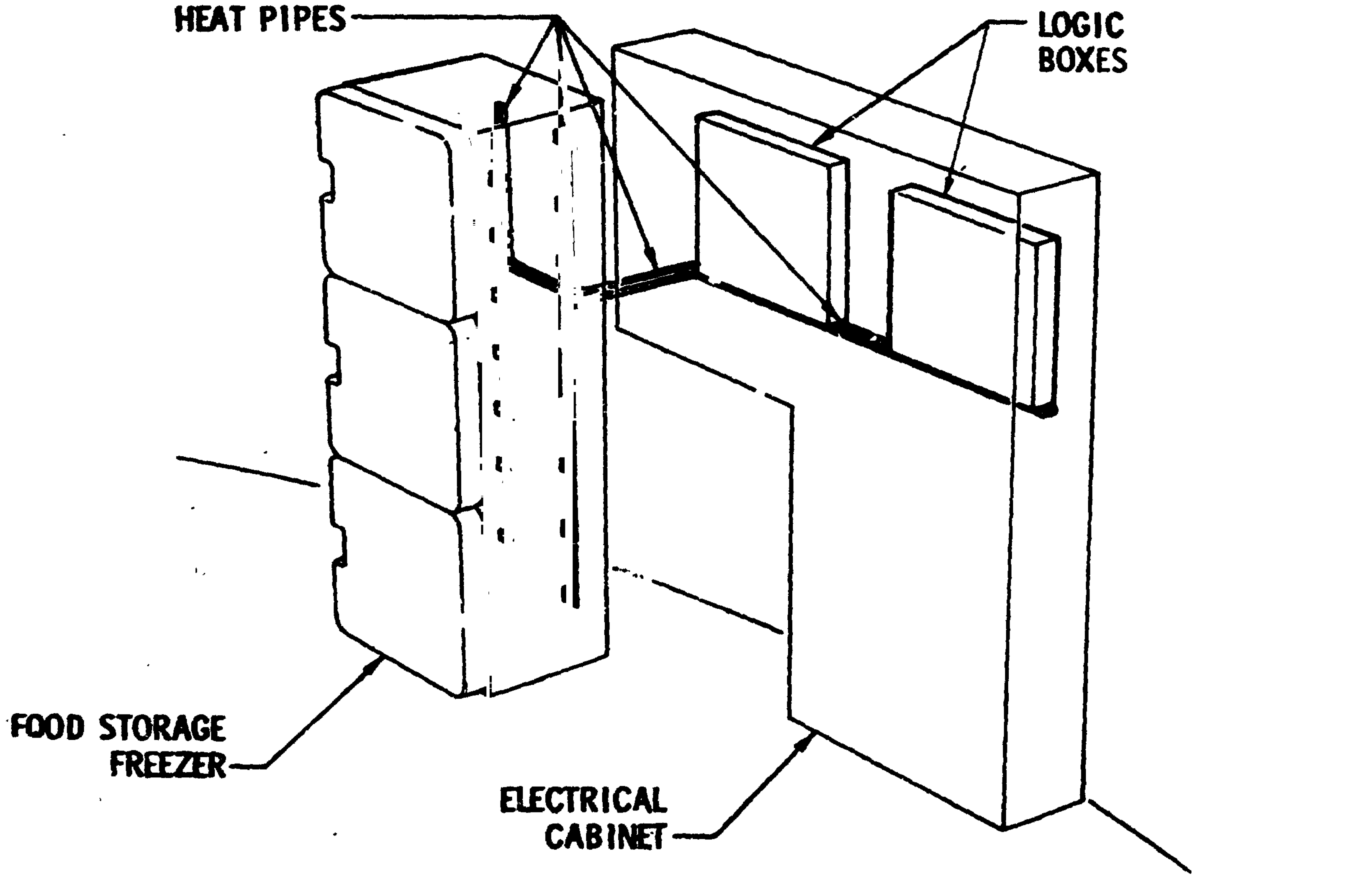


Figure 2.2.3-49

2.2.3.3 Testing

A. Vertical Checkout Laboratory

During the subsystem checkout of the electrical/electronic portion of the Thermal Control Subsystem, several operation problems were identified. Descriptions of these problems and their solutions follow:

1/ Simultaneous OFF-ON Heat Exchanger Commands

During manual cooling mode of operation checkout the heat exchanger OFF and ON commands were being initiated by the logic at the same time.

By analysis and test, it was determined that the spacecraft control system for the TCS was not compatible with the design of the heat exchanger relay driver module within the TCS logic assembly. The fix was to revise the spacecraft control system circuit (Control and Display Panel 617) to prevent a back bias condition within the relay driver module. This condition existed when manually controlling the heat exchangers from Control and Display Panel 617. The "back-bias" condition caused both commands to be initiated at the same time.

2/ Heat Exchanger Driver Relay

In determining the solution of the above problem, a single point failure was identified within the heat exchanger relay driver module. In the original design of the heat exchanger relay driver module, Heat Exchanger 1 ON circuit was the only circuit which removed the heat exchanger OFF command(s). In the original design, Heat Exchanger 1 would always be activated whenever any cooling was required. However, in the automatic mode of operation a failure in this circuit would cause the heat exchanger OFF command to be maintained continuously once made.

The heat exchanger relay driver circuit was modified to allow each of the four (4) heat exchanger ON commands to remove the heat exchanger OFF command(s).

3/ Simultaneous Activation of Duct Heaters and Heat Exchangers

When the subsystem was activated in the automatic "COOLING MODE" of operation (logic, heat exchangers, duct heaters circuit breakers placed to the CLOSED position), the system then activated both the heat exchangers and the duct heaters.

Analysis of the subsystem design identified the problem to be associated with the sequence of activation of the subsystem circuit breakers. The correct activation sequence was determined, and all Skylab personnel concerned with the operation of the TCS were made aware of the required sequence. In essence, the logic circuit breakers were required to be activated after all other breakers are closed.

4/ Bus Power Unbalance In Heating Mode

In the automatic heating mode of operation, the original design provided for all of Duct 1 heaters to be activated before Duct 2 heaters would be activated. This type of control caused a large OWS Bus 1/Bus 2 power unbalance. The duct heater control logic was modified to activate half of Duct 1 heaters, half of Duct 2 heaters, half of Duct 1 heaters then half of Duct 2 heaters in that order if/when more than one heater was required to satisfy the temperature requirement.

B. Qualification Test

During the performance of EC-12 (Thermal Control System Module line item) the 12-Vdc regulator operation was affected by voltage oscillations on the 28-Vdc power buses.

A new filter module was designed, fabricated, and installed in the TCS assembly and solved the oscillation problem (reference TV-DSV7-EE-R6968 for complete results of the Qualification Test).

C. Passive Systems Optical Properties Verification

1/ TCS Optical Property Requirements Verification

Measurements of optical properties of the various passive thermal control surfaces were made at Huntington Beach and KSC to verify that the surfaces met specific thermal control requirements. Table 2.2.3-9 lists these requirements and the verification measurements. Also listed is the Waiver Request number and disposition for the known discrepancies.

TABLE 2.2.3-9

OWS Thermal Control System Optical Properties Requirements Verification

REQUIREMENTS		VERIFICATION MEASUREMENTS		KSC WAIVER/DISPOSITION
DESCRIPTION	SPECIFICATION	FACTORY	KSC ROLLOUT	
8.5* Thermal control (passive)				
1.0 Tank Assembly				
1.1 Gold Surface Emissivity	1.1. 0.04 Max.	0.03	0.03	None
1.2 Coupons				
1.2.2 Hatch Coupon Emissivity	1.2.2 0.10 Max.	S/N 01:0.02 S/N 02:0.02 S/N 03:0.02	0.02	None
1.2.3 FWD Crotch Coupon Emissivity	1.2.3 0.05 Max.	S/N 01:0.02 S/N 02:0.02 S/N 03:0.02	0.03	None
1.2.4 Cylinder Coupon Emissivity				
1.2.4.1 Gold Surface	1.2.4.1 0.04 Max.	0.02	0.025	None
1.2.4.2 S-13G Paint	1.2.4.2 0.85 Min.	0.90	0.92	None
1.2.4.3 Black Paint	1.2.4.3 0.85 Min	0.91	0.91	None
1.2.5 Cylinder Coupon Solar Absorptivity				
1.2.5.1 S-13G Paint	1.2.5.1 0.22 Max.	0.16	0.19	None
1.2.5.2 Black Paint	1.2.5.2 0.85 Min.	0.88	0.96	None

* Numbers indicated are paragraph numbers in the TCRSC

TABLE 2.2.3-9 (Cont'd)

OWS Thermal Control System Optical Properties Requirements Verification

REQUIREMENTS		VERIFICATION MEASUREMENTS		KSC WAIVER/DISPOSITION
DESCRIPTION	SPECIFICATION	FACTORY	KSC ROLLOUT	
1.2.6 FWD Crotch Coupon Emissivity	1.2.6 0.05 Max.	S/N 01 0.02 S/N 02 0.02	0.02	None
1.3 Emissivity of HA Fire Retardant Liner	1.3 0.50 Min.		0.84	None
1.4 Emissivity of FWD Dome HPI	1.4 0.10 Max.	0.04	0.04 to 0.09	None
2.0 FWD Skirt Optical Properties 2.1 Exterior Surfaces (Emissivity)	2.1 White - 0.85 Min. Black - 0.85 Min.		0.91 0.91	None None
2.2 Exterior Surfaces (Absorptivity)	2.2 White - 0.22 Max. Black - 0.85 Min.		0.24 to 0.26 0.91	WR-43. Out of spec values acceptable since emissivity of white paint is 0.91 which maintains approximately same ratio of absorptivity to emissivity, resulting in no max. temperature increase.

TABLE 2.2.3-9 (Cont'd)

OWS Thermal Control System Optical Properties Requirements Verification

REQUIREMENTS		VERIFICATION MEASUREMENTS		KSC WAIVER/DISPOSITION
DESCRIPTION	SPECIFICATION	FACTORY	KSC ROLLOUT	
3.0 Aft Skirt Optical Properties 3.1 Exterior Surfaces (Emissivity)	3.1 White - 0.85 Min. Black - 0.85 Min.		0.92 0.91	None None
3.2 Exterior Surfaces (Absorptivity)	3.2 White - 0.22 Max. Black - 0.85 Min.		0.21 to 0.25 0.93	WR-43. Out of spec values acceptable since emissivity of white paint is 0.92 which maintains approx. same ratio of absorptivity to emissivity resulting in no max. temperature change.
4.0 MS Optical Properties 4.1 Exterior Surfaces (Emissivity)	4.1 White - 0.85 Min. Black - 0.85 Min		0.91 to 0.94 0.90	None None
4.2 Exterior Surfaces (Absorptivity)	4.2 White - 0.22 Max.		0.21 to 0.23	WR-43. Out of spec value acceptable since it will produce only a slight increase in OWS cooling loads. Crew comfort not compromised

TABLE 2.2.3-9 (Cont'd)

OWS Thermal Control System Optical Properties Requirements Verification

REQUIREMENTS		VERIFICATION MEASUREMENTS		KSC WAIVER/DISPOSITION
DESCRIPTION	SPECIFICATION	FACTORY	KSC ROLLOUT	
7.0 Main Tunnel Emissivity of interior stations 225 to 554	7.0 0.05 Max.		0.02	None
8.0 Food Stowage, Emissivity of surfaces facing tank wall	8.0 0.15 Max.		0.07 to 0.13	None
9.0 Film vault, Emissivity of accessible surfaces	9.0 0.50 Min.		0.22	WR-42. Out of spec. value acceptable since OWS minimum dew point provides sufficient margin to preclude condensation behind film vault.
10.0 Thermal Shield (FWD and AFT) Optical properties				
10.1 Exterior Surfaces (Emissivity)	10.1 0.85 Min.		0.92 FWD 0.90 AFT	None None
10.2 Exterior Surfaces (Absorptivity)	10.2 0.85 Min.		0.90 FWD 0.95 AFT	None None

TABLE 2.2.3-9 (Cont'd)

OWS Thermal Control System Optical Properties Requirements Verification

REQUIREMENTS		VERIFICATION MEASUREMENTS		KSC WAIVER/DISPOSITION
DESCRIPTION	SPECIFICATION	FACTORY	KSC ROLLOUT	
11.0 HA Optical Properties 11.1 Compartment Walls (Emissivity)	11.1 0.75 Min.		0.83 to 0.84	None
11.2 Compartment Lockers (Absorptivity)	11.2 0.75 Min.		0.77 to 0.89	None

2/ Wardroom Window Coating

A combination of electrical resistance and surface emittance was specified to meet the design requirements for the gold coating applied to the outer surface of the inner pane. Test measurements from the development and qualification test windows and the window for OWS No. 1 are shown in Figure 2.2.3-50 superimposed on the design requirements. It can be seen that all measurements fell inside the box which defined the requirements and were therefore acceptable.

SKYLAB - ORBITAL WORKSHOP
OWS VIEWING WINDOW
COLD COATING TEST RESULTS

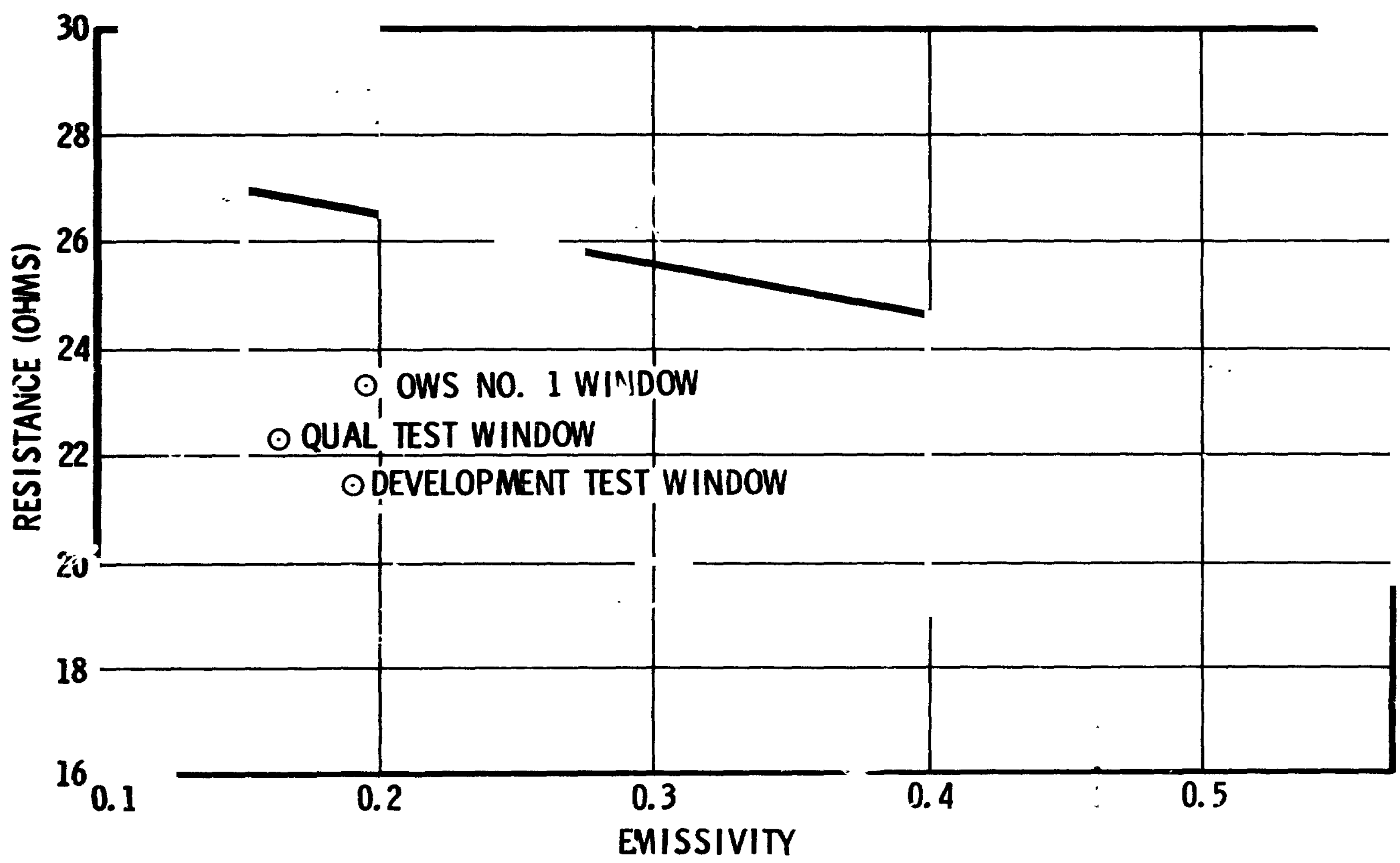


Figure 2.2.3-50

D. Ventilation System Flow Testing

1/ KSC Flow Rate Tests

As part of the KS0045 Test at KSC, comparisons of T/M flow measurements and on board flow meter readings were made at 28 Vdc and 14.7 psia (10.[illegible] cm^2). These conditions were equivalent to 26 Vdc operation for the flight pressure of 5 psia (3.45 N/cm^2). The data obtained are summarized as follows:

FLOW RATES - cfm *

	Duct #1		Duct #2		Duct #3	
	T/M	Meter	T/M	Meter	T/M	Meter
All Fans On	OSH (1)	OSH (1)	671	OSH (1)	683	650
Fan #1 Off	551	555	368	350	445	390
Fan #2 Off	466	530	370	290	455	410
Fan #3 Off	430	400	440	390	452	400
Fan #4 Off	495	435	415	395	452	400
All Off	105	(2)	130	(2)	121	(2)
Duct #3 Off	(2)	(2)	(2)	(2)	140 (3)	(2)

---

(1) OSH is off scale high at 700 cfm

(2) Not recorded

(3) Backflow

2/ MSFC Flow Distribution Tests

MSFC conducted flow distribution tests in the OWS mock-up to determine if the 15 to 100 fpm (4.6 to 30.4 m/min) velocity requirements were met in the various OWS compartments. Tests were conducted with 12 and 8 duct fans having flow rates

* To convert cfm to m^3/min multiply by 0.0283

of 1500 cfm (42.5 m^3/min) and 1000 cfm (28.3 m^3/min), respectively. Average velocity ranges for the various OWS compartments are shown in Table 2.2.3-10.

TABLE 2.2.3-10

Average Flow Velocities Test Data

Condition*	Compartment	Average Velocity fpm	m/min
12 Duct Fans	Forward	20 - 22	(6.10 - 6.71)
	Experiment	26 - 30	(7.92 - 9.14)
	Wardroom	23 - 30	(7.01 - 9.14)
	Sleep	20 - 30	(6.10 - 9.14)
	Waste Management	20 - 25	(6.10 - 7.62)
8 Duct Fans	Forward	19 - 20	(5.79 - 6.10)
	Experiment	18 - 24	(5.49 - 7.32)
	Wardroom	22 - 24	(6.71 - 7.32)
	Sleep	15 - 25	(4.51 - 7.62)
	Waste Management	20 - 25	(6.10 - 7.62)

* WMC fan on, circular diffusers set wide, rectangular diffusers and dampers open.

2.2.3.4 Mission Results

A. Boost Heating

Boost heating analytical simulations are presented in Figures 2.2.3-51 through 2.2.3-54 and compared to the OWS sidewall structure temperature data up to the time of orbital insertion. The analysis considered the effects of the meteoroid shield separation after approximately 63 seconds of flight exposing the gold Kapton tape. The properties $\alpha_s/\varepsilon = 0.2/0.05$ were used for the goldized Kapton external surface. The maximum sidewall temperature resulting from boost heating was approximately 110°F (317 K). The normal temperature decay after the boost heating pulse (140 seconds) did not occur for the sidewall because of the overriding solar heating effect. Sensor C7047 Position Plane III (Figure 2.2.3-53) that faced sunward showed this most clearly by its continuously increasing temperature during the first 600 seconds following liftoff.

The boost temperatures of the forward and aft thermal shields are presented in Figures 2.2.3-55 and 2.2.3-56, respectively. As would be expected because of the relatively low mass of the shields, the temperatures are significantly higher than the sidewall temperatures, reaching a maximum of approximately 230°F (383 K).

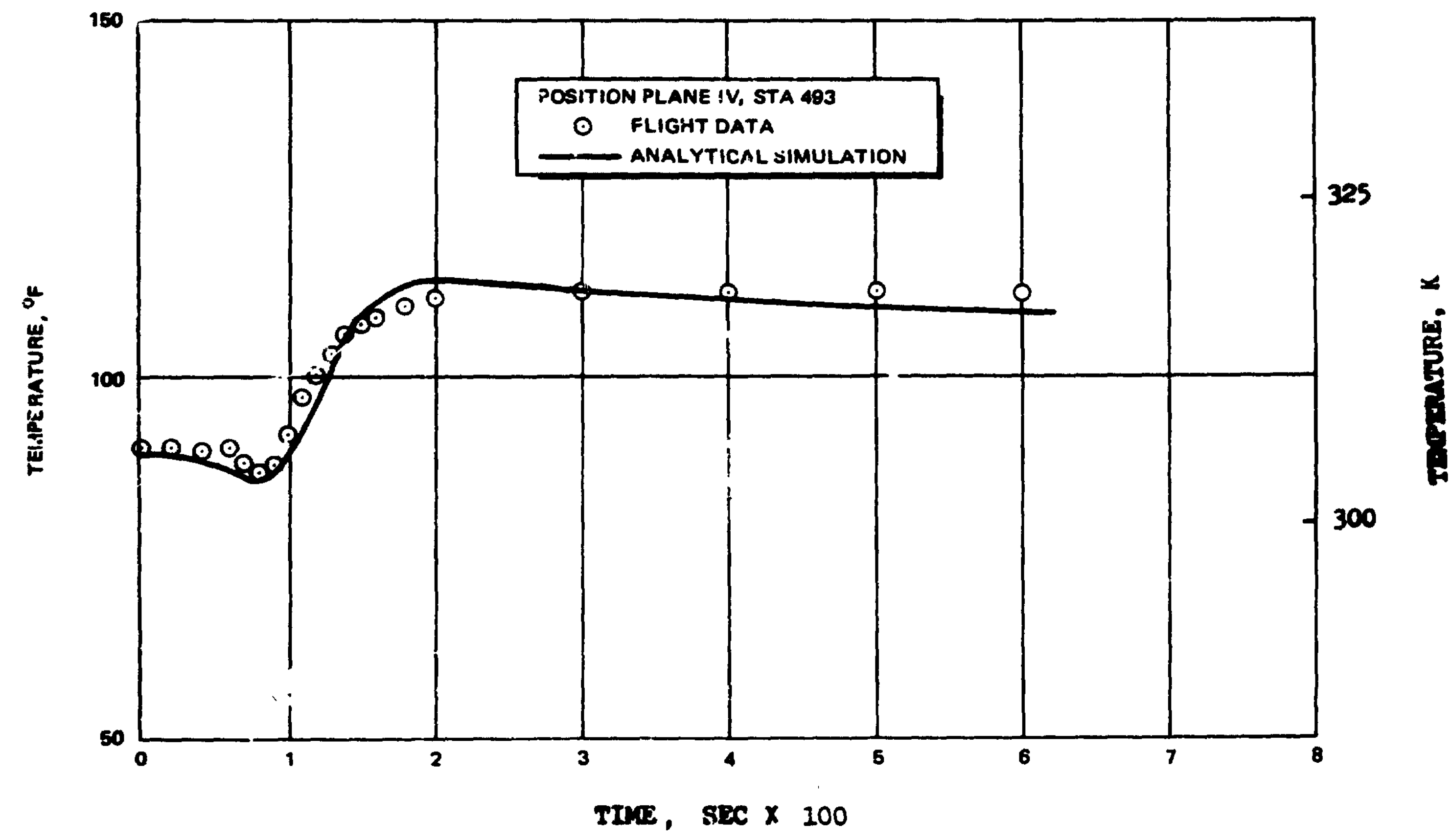


Figure 2.2.3-51. SL-1 Forward Compartment Wall Boost Temperature History, Sensor C7034

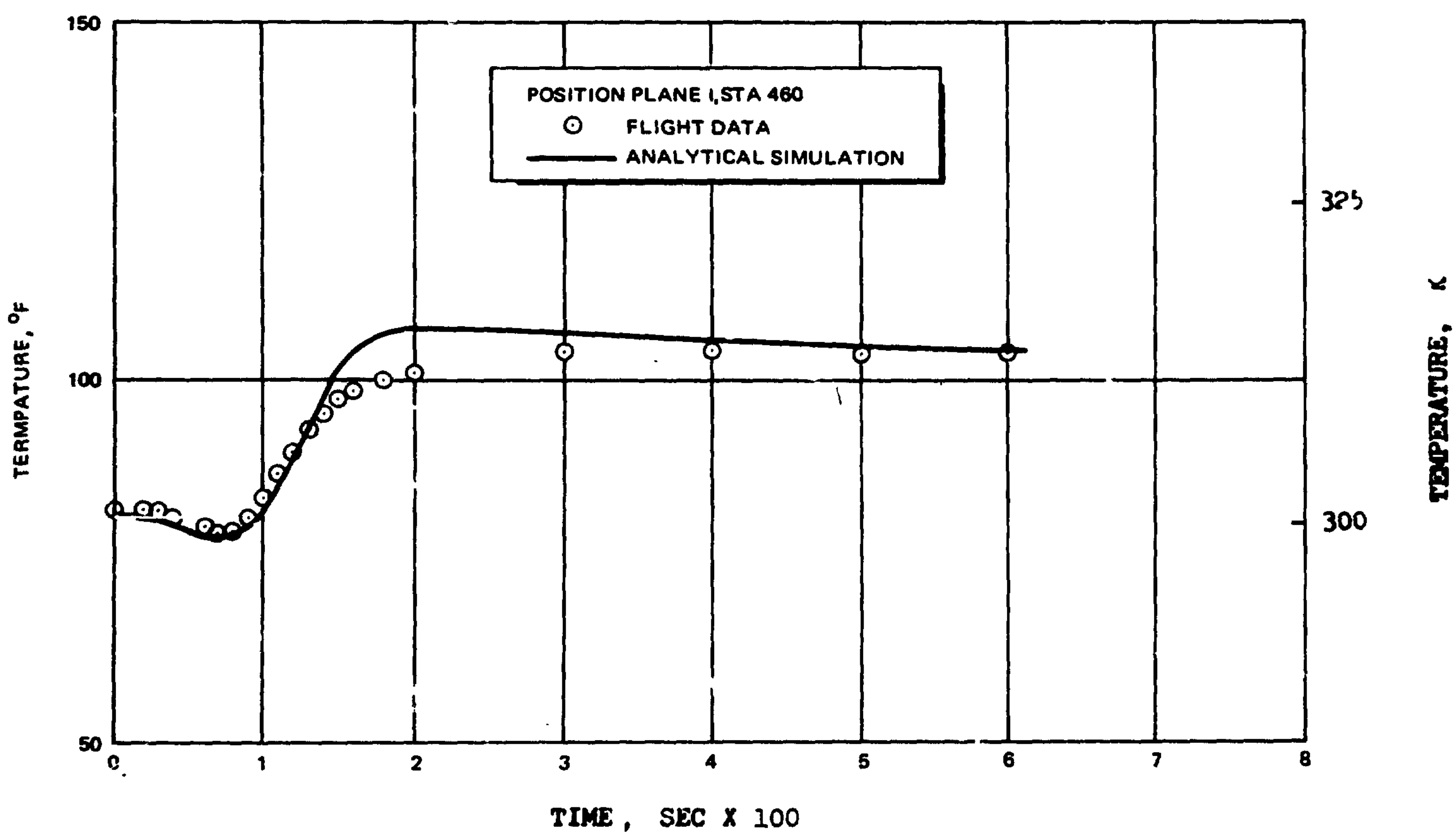


Figure 2.2.3-52. SL-1 Forward Compartment Wall Boost Temperature History, Sensor C7045

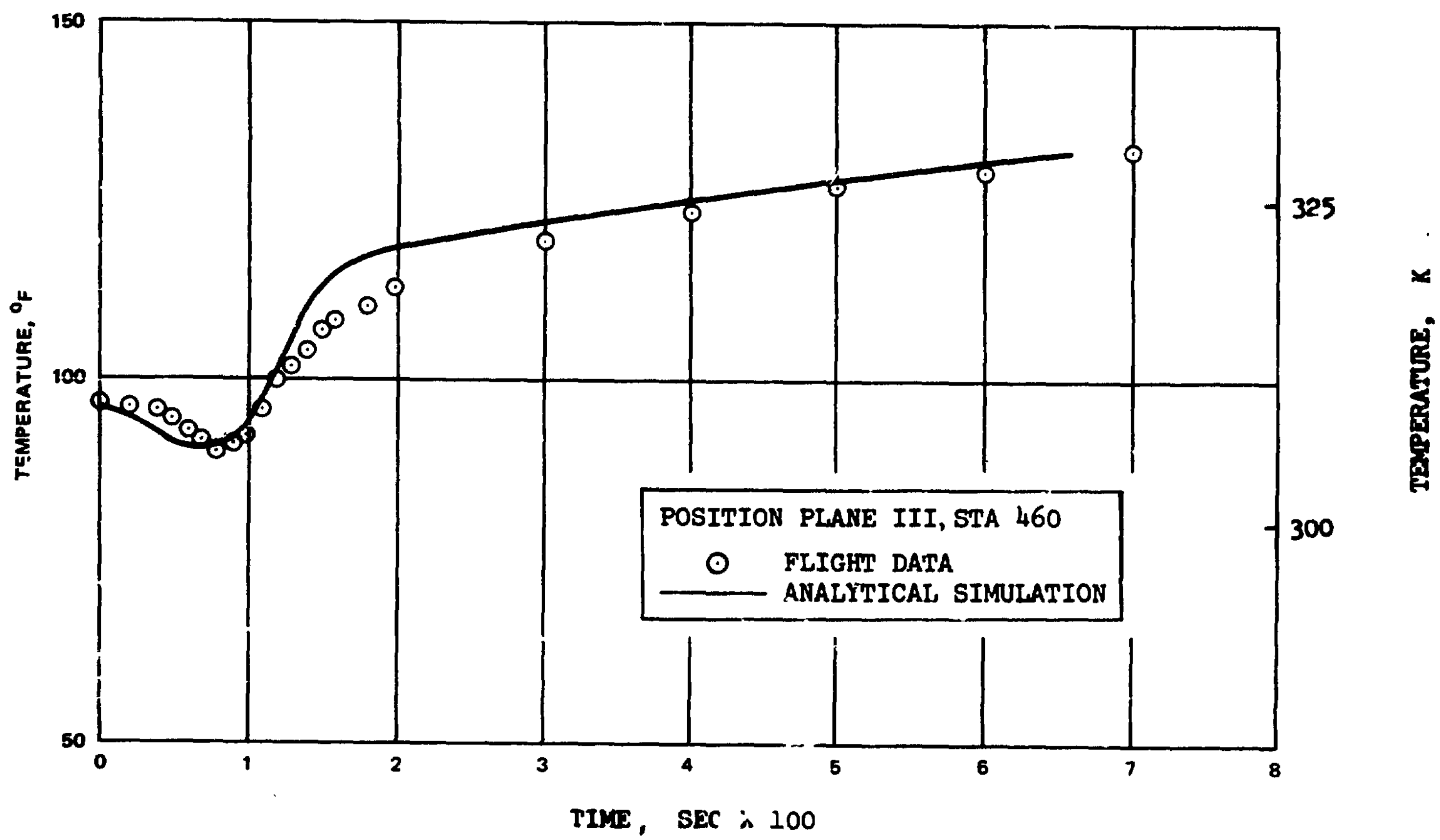


Figure 2.2.3-53. SL-1 Forward Compartment Wall Boost Temperature History, Sensor C7047

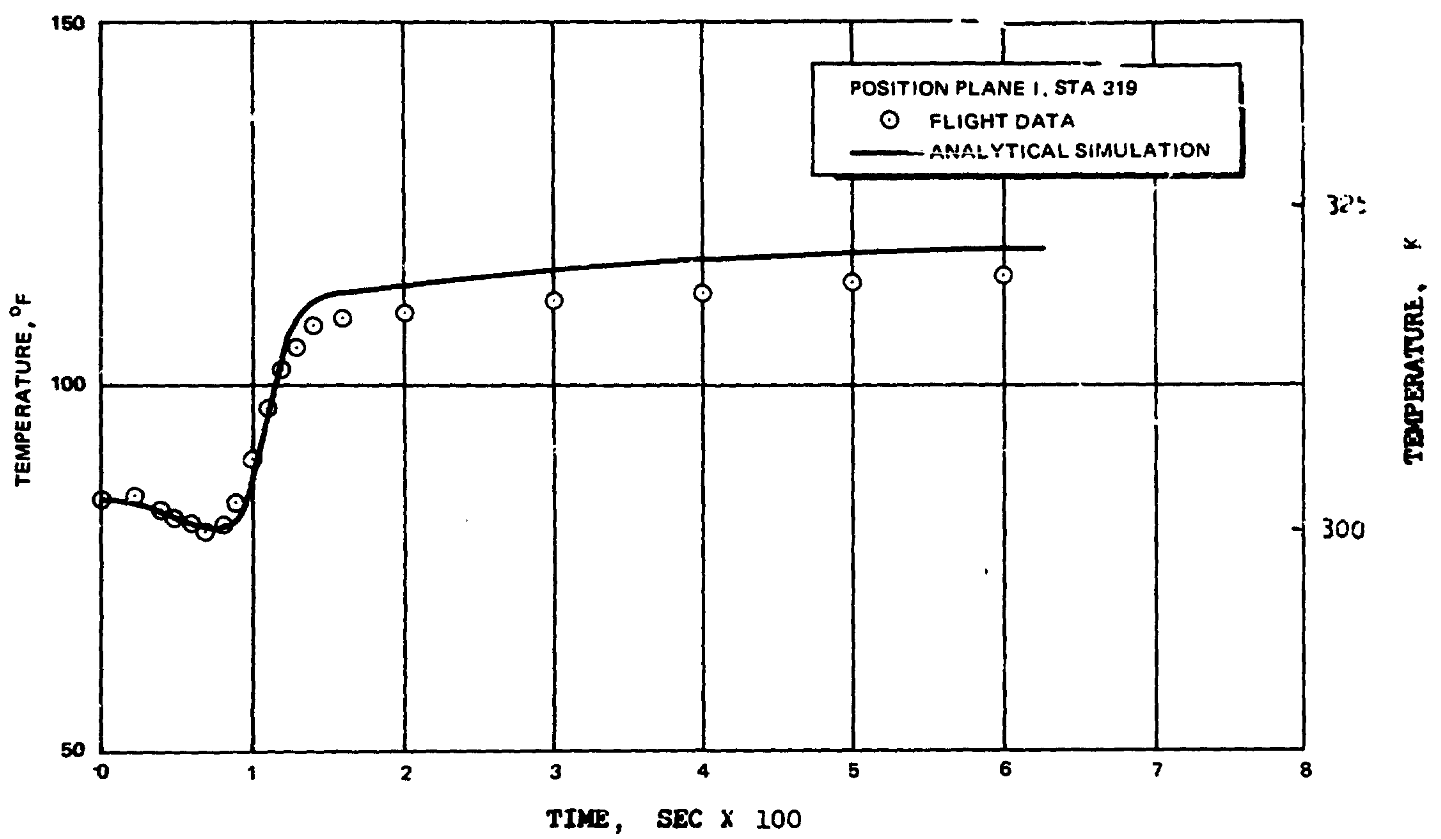


Figure 2.2.3-54 SL-1 Aft Compartment Wall Boost Temperature History, Sensor C7053

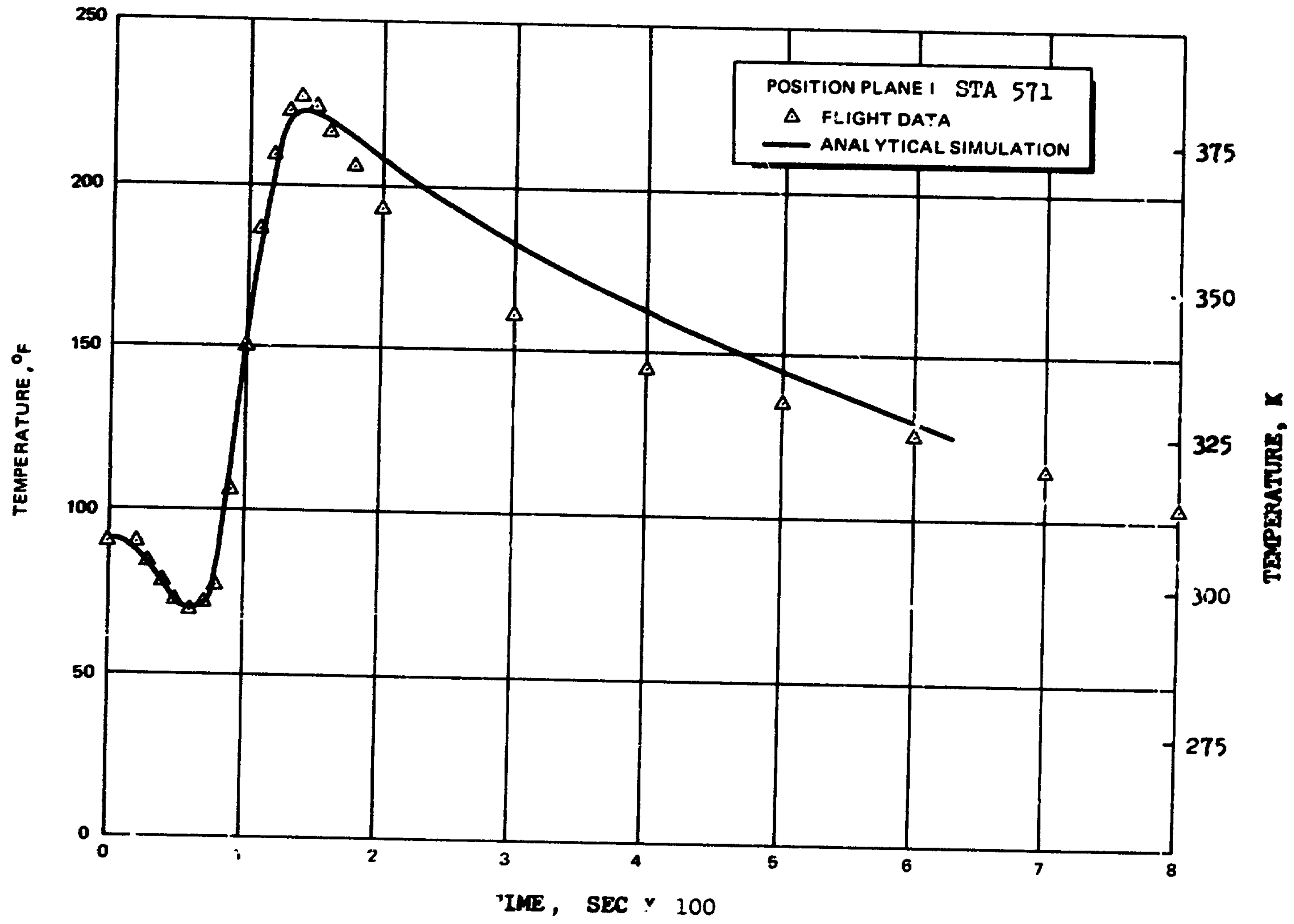


Figure 2.2.3-55. SL-1 Forward Skirt Thermal Shield Boost Temperature History, Sensor C7185

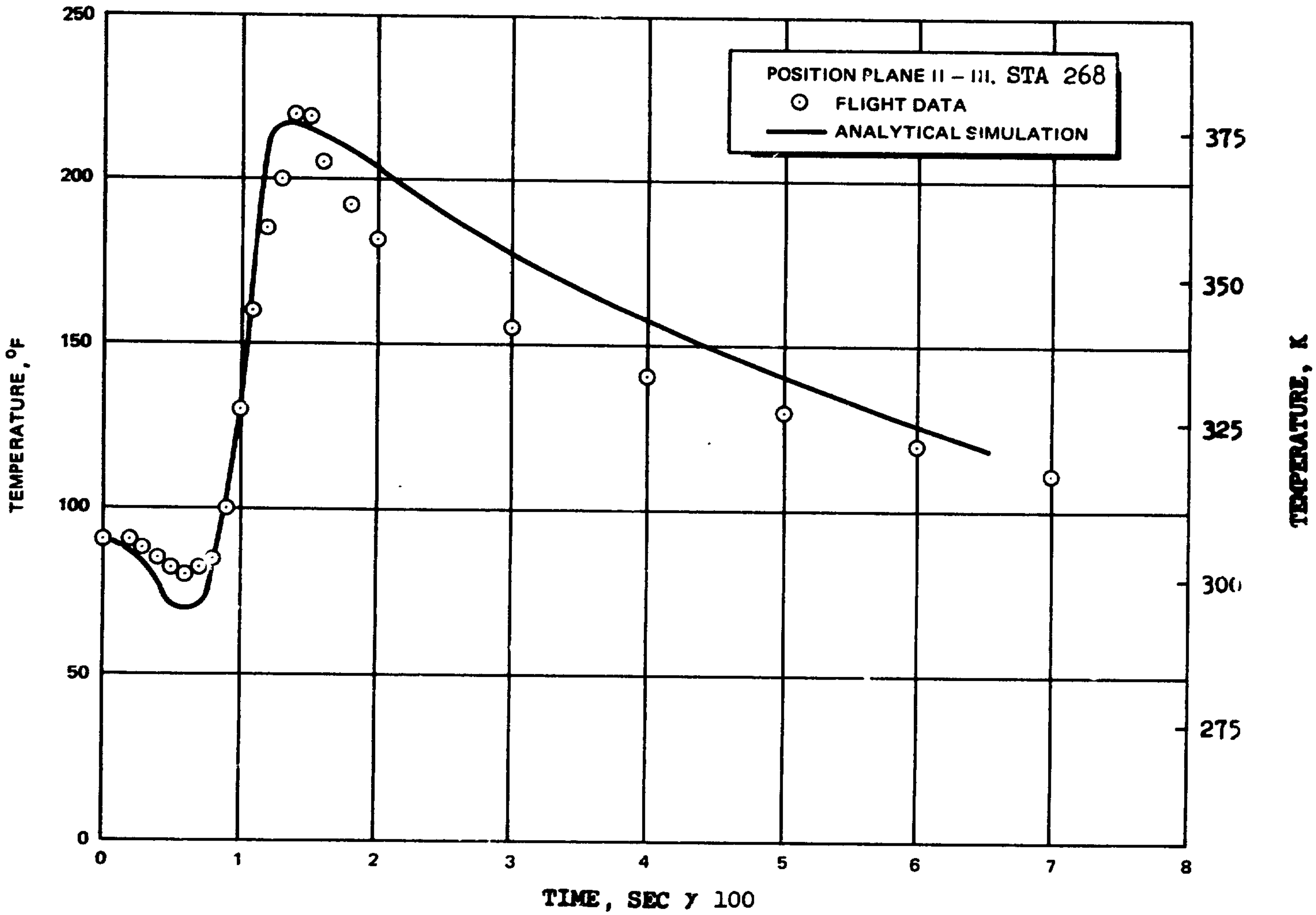


Figure 2.2.3-56 SL-1 Aft Skirt Thermal Shield Boost Temperature History, Sensor C7177

B. Environmental/Thermal Control System Orbital Peformance

1/ Active Subsystems

a. Radiant Heaters

The eight radiant heaters were turned on by DCS command from approximately 2015 to 2344 GMT, DOY 134. Telemetry data indicated a load of about 1270 watts (4335 Btu/hr) at 28 Vdc. This load was about 15 percent below the 1500 watts (5120 Btu/hr) determined for the same voltage by ground test measurements. The energy dissipation during this period was 4424 W·hr (15100 Btu). The radiant heaters were turned off after the noted period and not reactivated since the increased environmental heating resulting from loss of the meteoroid shield eliminated the need for radiant heater operation.

b. Convective Duct Heaters

The increased orbital environmental heating to the habitation area even with both the parasol and sail sunshades deployed was such that the duct heaters were never activated.

c. Wardroom Window Heater

The Wardroom window temperatures were influenced primarily by the OWS exterior thermal environment which varied with the orbit beta angle. The maximum external heat inputs occurred at orbital noon for low beta angles when the the incident albedo energy on the window was the greatest. The minimum external heat inputs occurred at high beta angles when the albedo inputs were minimal. The interior environment was represented primarily by the mean radiant temperature of the Wardroom. The window temperature was also dependent on the window heater and the use of the window shade and the tempered glass or aluminum protective window cover.

Temperature sensors C7293 and C7294 were used to monitor the window structure temperatures. Neither sensor measured the glazing temperature directly. Sensor C7293 was mounted on the interior window retainer which was separated by circumferential seals from the inner glazing. Sensor C7294 was located on the window doubler which was separated from the outer glazing by a retainer between two sets of seals. A layer of foam insulated the doubler from the OWS internal environment. Because of the weak thermal coupling between the sensor locations and the glazing, the data from these sensors proved to be unsuitable for assessing heater operation. The crew of SL-2 reported that the window heater was turned off from about 2300 GMT on DOY 168 to 0800 GMT on DOY 169, a sleep period during which condensation formed on the inner surface of the metal window cover then in place. The crew reported that the heater was turned on and the condensate evaporated. The fact that the heater was functioning is supported by the evaporation of condensate when the beta angle was above 54 degrees and increasing. If the heater were not functioning, the environmental conditions would result in decreased temperatures and increased condensation.

The flight data indicate the sensitivity of the window region to the external heat flux which varied with orbit beta angle. This is illustrated in Figure 2.2.3-57 where the daily temperature extremes experienced by the two window measurements are plotted for the second storage period. The window heater was off and there were no changes in the window shade configuration during this time period, thereby removing their effects from the temperature monitored. The Wardroom ceiling temperature (C7123), which was representative of the mean radiant interior environment of the window, and the orbit beta angle history are also presented. The results which were typical of the response to beta angle, show that the window temperatures decreased as the absolute value of the beta angle increased. From DOY 280 to 294, this was true even though the interior mean radiant temperature of the Wardroom was increasing.

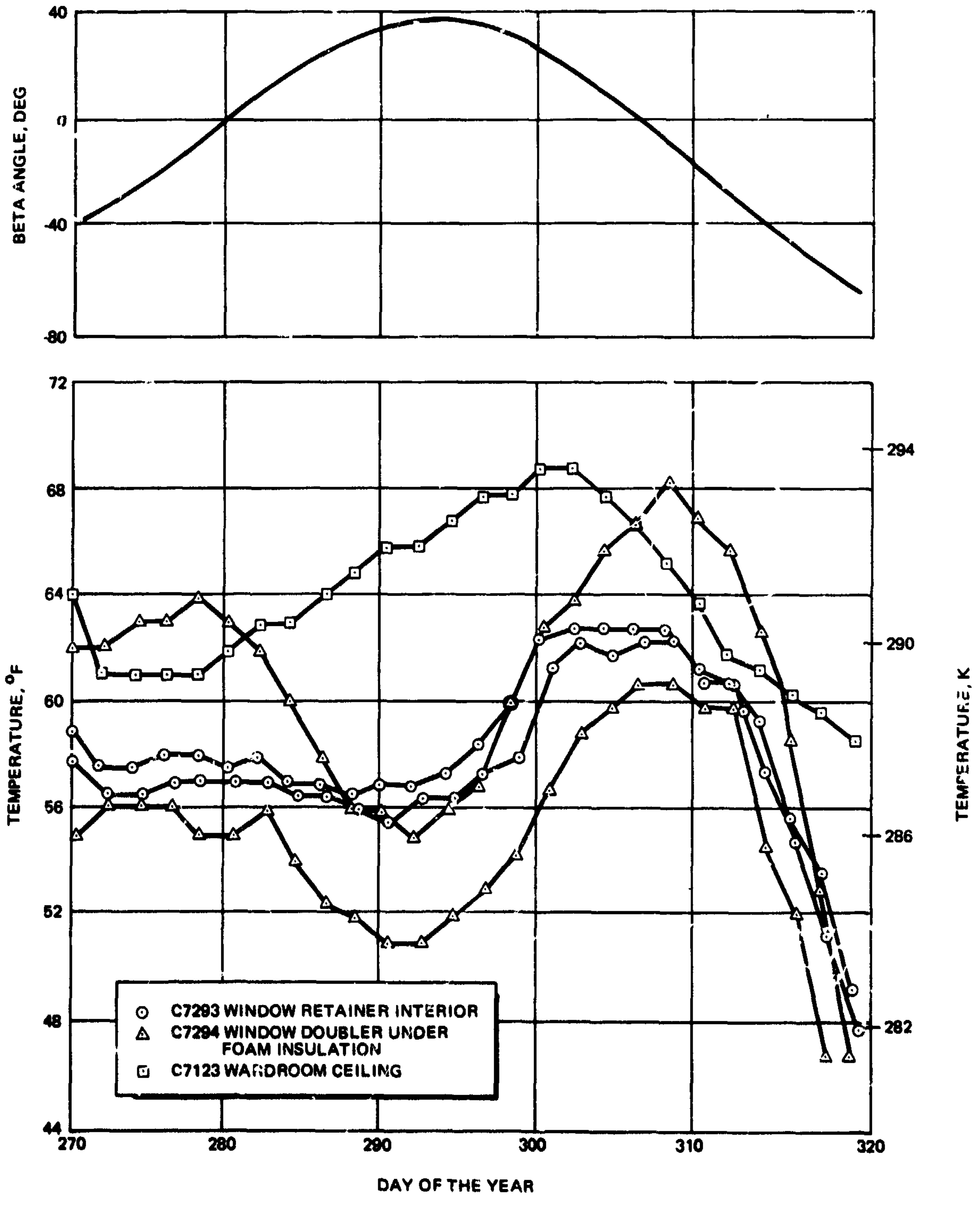


Figure 2.2.3-57. Wardroom Window Daily Temperature Extremes

d. Ventilation Subsystem

1. SL-2 Habitation Period (DOY 145 to 173)

The OWS Duct 1 and Duct 2 fans were activated on DOY 146 in accordance with the SL-2 activation checklist and the duct flows indicated by telemetry (T/M) were 630 and 450 cfm (17.8 and 12.7 m^3/min), respectively. The Duct 3 fans were not activated at this time in order to conserve power.

On DOY 147 the T/M flow measurement for Duct 1 failed. The on-board display reading of 600 cfm (17.0 m^3/min) verified that it was a T/M flow measurement failure and not fan degradation or failure. Also, on DOY 147 the on-board display reading of 550 cfm (15.6 m^3/min) for Duct 2 indicated that the accuracy of the T/M flow measurement for that duct, 450 cfm (12.9 m^3/min) was degraded.

The Duct 3 fans were activated on DOY 149 in order to insure ground verified compliance with the flight mission rule requiring operation of a minimum of six duct fans.

Table 2.2.3-11 summarizes the duct flow measurement data taken during the SL-2 habitation period.

TABLE 2.2.3-11

SL-2 VENTILATION DUCT FLOW SUMMARY

Time DOY:GMT	Flow Measurement, cfm (m^3/min)					
	Duct 1		Duct 2		Duct 3	
	T/M	Display	T/M	Display	T/M	Display
Prelaunch (KS 0045)	OSH	OSH	671 (19.0)	OSH	683 (19.3)	650 (18.4)
146:2000 (Activation)	630	-	450	-	*	*
147:0455	OSL	600 (17.0)	451 (12.7)	550 (15.6)	118* (3.34)	75* (2.12)
159	OSL	-	455 (12.9)	-	560 (15.8)	-
169	OSL	520 (14.7)	452 (12.8)	5[illegible]0 (14.4)	595 (16.8)	550 (15.6)

OSL = Off Scale, Low

OSH - Off Scale, High

*Duct [illegible] was off Duct 3 flow was apparently toward mixing chamber (back flow)

T/M Measurement Numbers: Duct 1, F7000

Duct 2, F7001

Duct 3, F7002

Before and after the EVA periods it was necessary to remove and then reinstall the gas interchange duct from the OWS hatch opening. The crew reported that the two Calfax fasteners which secured the center of the duct did not mate with the attachment fitting on the hatch ring. However, the duct attachments at the OWS mixing chamber and the AM duct provided sufficient support.

The crew reported that there were no dead flow spots in the OWS and that the ventilation system was especially useful in collecting lost items at the mixing chamber inlet screen. The housekeeping task of cleaning the screen was necessary and was performed approximately every three days. The noise level of the system was very low as reported by the SL-2 crew. They could not hear the operation of the fans unless they were close to a fan cluster.

During the first few days of SL-2, one portable fan was mounted in the OWS entry hatch to circulate additional hot gas from the OWS into the AM toward the aft cabin heat exchangers. The crew felt this configuration provided additional cooling for the OWS. A portable fan was also occasionally used to provide additional circulation in the forward compartment.

The WMC fan operated satisfactorily throughout SL-2. The filters were cleaned and replaced as planned. The crew did not report any odor problems which indicated that the subsystem had sufficient odor removal capacity.

The adjustable circular diffusers in the crew quarters were set in the wide pattern at launch and were not changed by the crew. The dampers were left in the open configuration except for reducing flow in a particular area for short periods. The rectangular sleep diffuser settings were adjusted by the crewmen to direct flow either on or away from them depending upon individual preferences.

2. SL-3 Habitation Period (DOY 209 to 268)

During SL-3 the flow rates taken from T/M were typically 400 to 440 cfm (11.3 to 12.5 m^3/min) for Duct 2 and 560 to 600 cfm (15.8 to 17.0 m^3/min) for Duct 3. The only crew readout of the on-board meter for Duct 1 was 540 cfm (15.3 m^3/min) on DOY 228.

The Duct 1 T/M flow meter had failed on SL-2. The Duct 2 T/M flow meter dropped from 440 to 340 cfm (12.5 to 9.6 m^3/min) in 8 seconds on DOY 228. The on-board meter reading at this time was 500 cfm (14.1 m^3/min). Within 24 hours the T/M flow meter for Duct 2 was again reading 400 to 440 cfm (11.3 to 12.5 m^3/min). This T/M flow meter had been reading lower than the on-board meter since the beginning of SL-2 and apparently experienced a malfunction resulting in a degraded and erratic output. A summary of the duct flow rates for SL-3 is given in Table 2.2.3-12.

TABLE 2.2.3-12
SL-3 VENTILATION DUCT FLOW SUMMARY

Time DOY:GMT	Flow Measurement, cfm (m^3/min)					
	Duct 1		Duct 2		Duct 3	
	T/M	Display	T/M	Display	T/M	Display
228:1548	OSL	-	437 (12.4)	-	575 (16.3)	-
228:1549	OSL	540 (15.3)	321 (9.1)	500 (14.2)	575 (16.3)	540 (15.3)
243:1938	OSL	-	166* (4.7)	-	100* (2.8)	-
244:0206	OSL	-	422 (11.9)	-	585 (16.6)	-

OSL: Off Scale, Low

* Duct 3 fans and two fans in Duct 2 are off.

Several M509 and T020 experiments were performed during SL-3. These maneuvering experiments required minimum gas velocities to reduce the flow effect on the experiment results. The four Duct 3 fans and two Duct 2 fans were turned off during these experiments. At this time typical readings were 190 cfm (5.4 m^3/min) for Duct 2 and 100 cfm (2.8 m^3/min) for Duct 3. The 100 cfm (2.8 m^3/min) reading for Duct 3 was probably back flow since all the fans were off. With this configuration, the net flow through the floor diffusers was approximately 600 cfm (17.0 m^3/min).

As for SL-2, the crew reported no dead flow spots in the OWS. The housekeeping task of cleaning the mixing chamber inlet screen was again performed on the average of every three days.

The SL-3 crew used the portable fans in three locations. Prior to deployment of the MSFC sunshade, one was placed in the OWS hatch to circulate more gas toward the aft cabin heat exchangers, and another was used to provide convective cooling of a crewman using the ergometer. The third fan was mounted in the MDA during SL-3 deactivation to provide contingency cooling of the rate gyro six-pack should a heater-on malfunction occur during storage.

The WMC fan operated satisfactorily throughout SL-3. The filters were cleaned and replaced as planned. The crew did not report any odor problems which indicated that odor removal continued to be sufficient.

As in SL-2, the adjustable circular diffusers were left in the wide pattern. The dampers were left in the open configuration except for reducing flow in a particular area for short periods. The rectangular sleep diffuser settings were adjusted by the crewmen to direct flow either on or away from them depending upon individual preferences.

3. SL-4 Habitation Period (DOY 320, 1973 to DOY 39, 1974)

The ventilation subsystem performance was normal throughout SL-4. The Duct 2 T/M flow meter data from sensor F7001 continued to be degraded and erratic. Early on DOY 30 the T/M reading for Duct 2 flow dropped to zero and remained so through the end of the mission. On board checks made when T/M data indicated zero flow showed the Duct 2 flow to be normal.

2/ Passive Subsystems

a. High Performance Insulation

Conductance values for the 48-layer sections of the forward dome high performance insulation (HPI) were determined from flight temperature measurements of the HPI exterior surface, the wall structure separating the HPI and internal foam insulation, and the interior surface of the foam. There was one set each of temperature sensors on PP I and PP III identified as follows:

Location	HPI Exterior	Forward Dome	Foam Interior
PP I	C7100	C7162	C7106
PP III	C7101	C7163	C7107

The applied value of the foam conductance, 0.48 Btu/hr-ft^2-°F (2.73 W/m^2·K), was based upon a conductivity of 0.02 Btu/hr-ft-°F (0.035 W/m·K) and a thickness of 0.5 inch (1.27 cm). The heat flux, the product of the foam conductance and surface temperature difference across the foam (ΔT_f*) divided by the temperature difference between the HPI surfaces (ΔT_m**) yielded the HPI conductance. The comparatively low conductance of the HPI results in small ΔT_f's, roughly 1 to 2°F (0.56 to 1.11 K),

* ΔT_f = Forward Dome Temperature - Foam Interior Temperature

** ΔT_m = HPI Exterior Temperature - Forward Dome Temperature the accuracy of which largely determined the accuracy of the conductance evaluations. The accuracy of ΔT_f and ΔT_m were limited by instrumentation-telemetry sensitivity (one data bit represented about 0.48°F (0.27 K) for the foam surface (wall and interior) temperatures and nearly 1.6°F (0.88 K) for the exterior temperatures) and by use of different multiplexers for the foam temperature measurements on PP I. With the low heat fluxes of 1 Btu/hr-ft^2 (3.15 W/m^2) or less, the wall thermal capacity was sufficient to require that the temperature data used for HPI evaluation be taken when essentially steady state heat transfer conditions prevailed. Such conditions were found at extreme values of beta angle (±65 deg or more) when, in the solar inertial attitude, the external thermal environment changes were small over a period of several days. Because of the relatively large change in temperature represented by one data bit, a large number of readings, such as were made during habitation periods, was desired in order to determine the critical foam temperature difference with some degree of accuracy. A further limitation was the availability of suitable data resulting from the intermittent operation of low level Multiplexer B through which data from Sensors C7101 and C7162 were transmitted.

Suitable data were obtained for nearly all of DOY 176 at the beginning of the first storage period and for several periods totalling about 72 hours from DOY 325 through 330 during SL-4. The average conductance values from these data were as follows:

Location	DOY	Average Conductance Btu/hr-ft^2-°F	Average Conductance W/m^2·K	Average Temperature Difference ΔT_f °F	ΔT_f K	ΔT_m °F	ΔT_m K
PP I	176	0.010	0.058	1.45	0.81	69.6	38.7
PP I	325-330	0.0061	0.035	0.88	0.49	69.1	38.4
PP III	176	0.0064	0.037	1.73	0.96	130	72.2
PP III	325-330	0.0077	0.044	1.75	0.97	109	60.5

The conductances at PP I and PP III could differ significantly, because conduction was sensitive to local compression, nearby joints, button fasteners, and penetrations which comprised over 90 percent of the heat transferred by the HPI. The CEI specification called for a conductance of not more than 0.02 Btu/hr-ft^2-°F (0.11 W/m^2·K) or twice the largest value above.

b. External Coating Degradation

1. S-13G Paint

(a) Installation

The S-13G white silicone paint is a low α_s/ε paint (nominally, α_s = 0.22, ε = 0.9, α_s/ε = 0.25) developed by IIT Research Institute. It is suitable for spacecraft thermal control application because of the relative stability of α_s under solar exposure in space environments. The aft three feet of the OWS aft skirt (Figure 2.2.3-58), was painted with the S-13G paint to provide passive thermal control of electrical and attitude control equipment contained in this non-pressurized region of the OWS. In selected areas of the white aft skirt, black Cat-a-lac epoxy enamel paint stripes ($\alpha_s/\varepsilon=1.0$) were used for further thermal control.

The S-13G paint was spray coated with successive wet coats of approximately 2.5 mils (0.0064 cm) until a dry film thickness of 8 mils (0.020 cm) was achieved. Successive one mil (0.0025 cm) wet coat thicknesses of Cat-a-lac black were used to attain a dry film thickness of approximately 3 mils (0.0076 cm). The OWS aft skirt surface on which these paints were applied consisted of the following:

Primary Structure:	Anodized 7075-T6 Aluminum
F.R. Coating	0.75 to 1.3 mil (0.0019 to 0.0033 cm) epoxy
Silicone Primer	0.1 to 0.35 mil (0.00025 to 0.0089 cm) silane

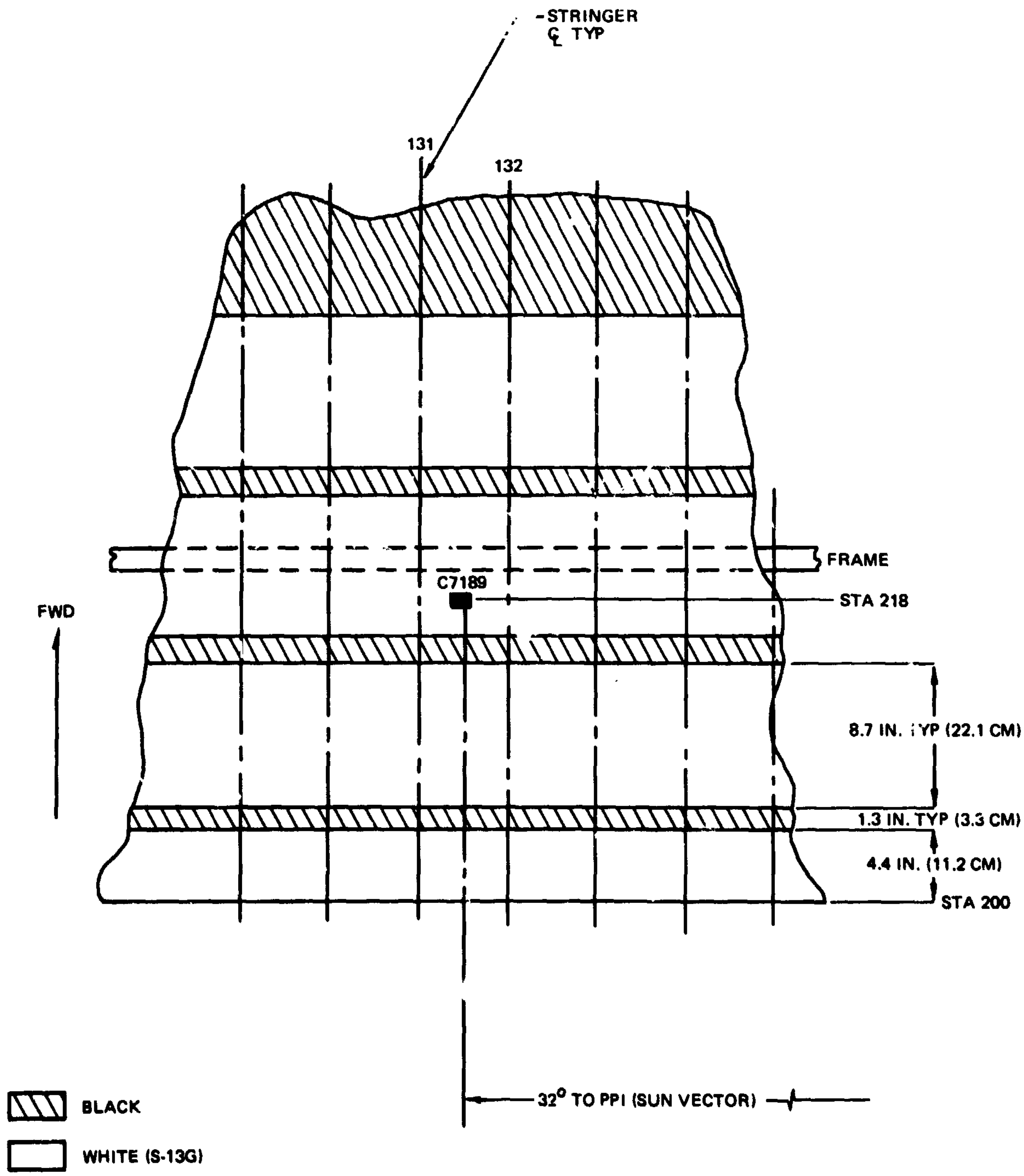


Figure 2.2.3-58. OWS Aft Skirt Transducer C7189 Location

The coatings were cured at ambient temperature for over 40 hours, then for more than 24 hours at approximately 160°F (344 K) to meet the NASA specified outgassing requirements.

The optical property measurements of the S-13G after the curing were 0.2 and 0.9 for α_s and ε, respectively. The optical properties of the Cat-a-lac black paint were not measured at installation.

(b) Prelaunch Measurements

Optical property measurements of the black and white surfaces were made to assure proper thermal control of the associated electronic components during orbit. The following table summarizes optical property specifications and measured values.

Description	Parameter	Specification Value	KSC Measurement
S-13G White Paint	Emittance	0.85 min	0.92
	Solar Absorptance	0.22 max	0.21 to 0.25
Cat-a-lac Black Paint	Emittance	0.85 min	0.91
	Solar Absorptance	0.85 min	0.93

The range of solar absorptance values were higher than the allowable specification for the S-13G white paint. However, these conditions were determined to be acceptable since the emittance of the white paint was 0.92 which maintained approximately the same α_s/ϵ ratio, resulting in no significant change in the desired thermal control range.

(c) Orbital Insertion Data

Orbital temperature data from a temperature measurement on the aft skirt located 32° from the normal solar vector was utilized to evaluate the ultra-violet and proton degradation effects on the white paint solar absorptance. A thermal model simulating the aft skirt structure in the vicinity of the transducer was set up to assess the effects of paint degradation.

The retro-rocket firing to effect stage separation resulted in plume contamination of the S-13G paint on the aft skirt. The contamination consisting of particle deposition was visibly evident in photographs taken of the white painted surface during fly-around maneuvers prior to the docking of the first Skylab crew. The contrast between the white painted surface under the SAS beam fairing number 1 that was protected during retro-fire, and the skirt surfaces around it was clearly visible. The plume contamination primarily affects the α_s while the ε, being initially high, is not substantially increased.

Plume contamination effects on the white paint cause degradation that is dependent upon the plume flow impingement angle and the separation distance during staging. Assuming the aft skirt surfaces could be approximated locally by a flat plate surface parallel to the plume flow, the AEDC test data[1] and Figure 2.2.3-59 can be used to predict the severity of the degradation. On this basis, the plume degradation effect on transducer C7189 located at Station 218 resulted in an increase of α_s from the initial range of 0.21 to 0.25 to a value of $\alpha_s = 0.34$.

Orbital flight data taken within four hours after orbital insertion showed further α_s degradation to a value of 0.37. This condition is consistent with test data of Zerlaut[2] that showed a high proportion of the UV and proton damage to S-13G white paint occurred within five hours of exposure to simulated solar sources.

1. Muse, W. W., "Full-Scale Simulated Altitude Investigation of the Centaur-Payload Surface and Functional Degradation Resulting from the Saturn S-IVB Retro-Rocket Exhaust Contaminants", ARO, Report No. AEDC-TR-66-57, May, 1966
2. Zerlaut, G. A. and Gilligan, J. E., "Study of in Situ Degradation of Thermal Control Surfaces", IIT Research Institute Report No. IITRI-U6061-17, March 7, 1969

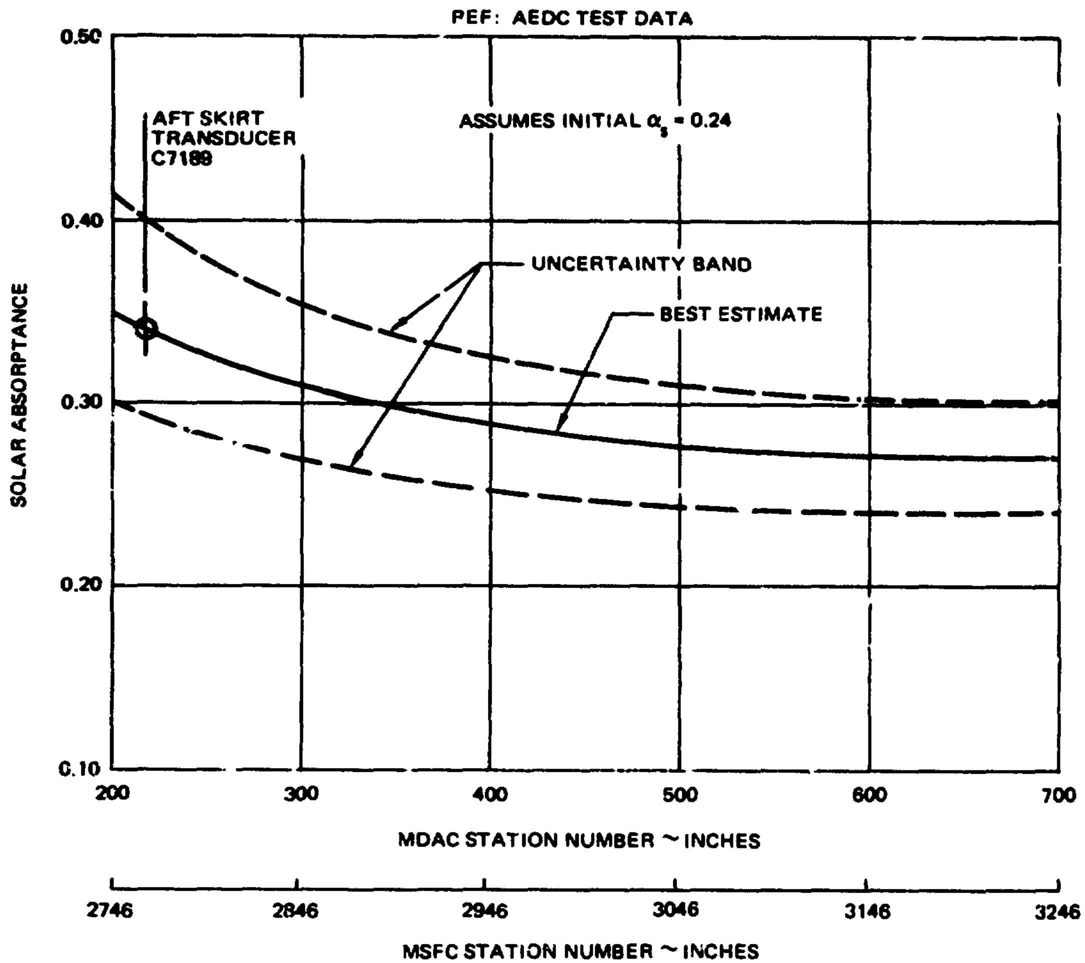


Figure 2.2.3-59. Estimated Retro-Rocket Plume Contamination

C6

(d) Post Orbit Insertion Data

Orbital temperatures for surfaces viewing the sun will, in general, follow a cyclic temperature profile. As shown in Figure 2.2.3-60, the aft skirt thermal model closely simulated the flight data for September 7, 1973, ($\beta = 0^\circ$). Peak temperatures for $\beta = 0^\circ$ conditions starting at June 6, 1973, through December 9, 1973, were plotted in Figure 2.2.3-61 to determine the optical property degradation and seasonal solar intensity fluctuation effects. The flight temperatures were compared with calculated temperatures for white paint with α_s = 0.38, 0.40, and 0.42, generated for the following seasonal variations:

	$\dot{q}_{solar}$		$\dot{q}_{ir}$	
	Btu/hr-ft^2	W/m^2	Btu/hr-ft^2	W/m^2
Summer Solstice	415	1310	72.6	229
Autumnal Equinox	429	1350	75.1	237
Winter Solstice	444	1400	77.7	245

For a constant value of α_s the analytical seasonal temperature variations is seen in Figure 2.2.3-61 to be substantially less than that of the flight

CORRELATION OF TEMPERATURE DATA FOR S-13G PAINTED AFT SKIRT

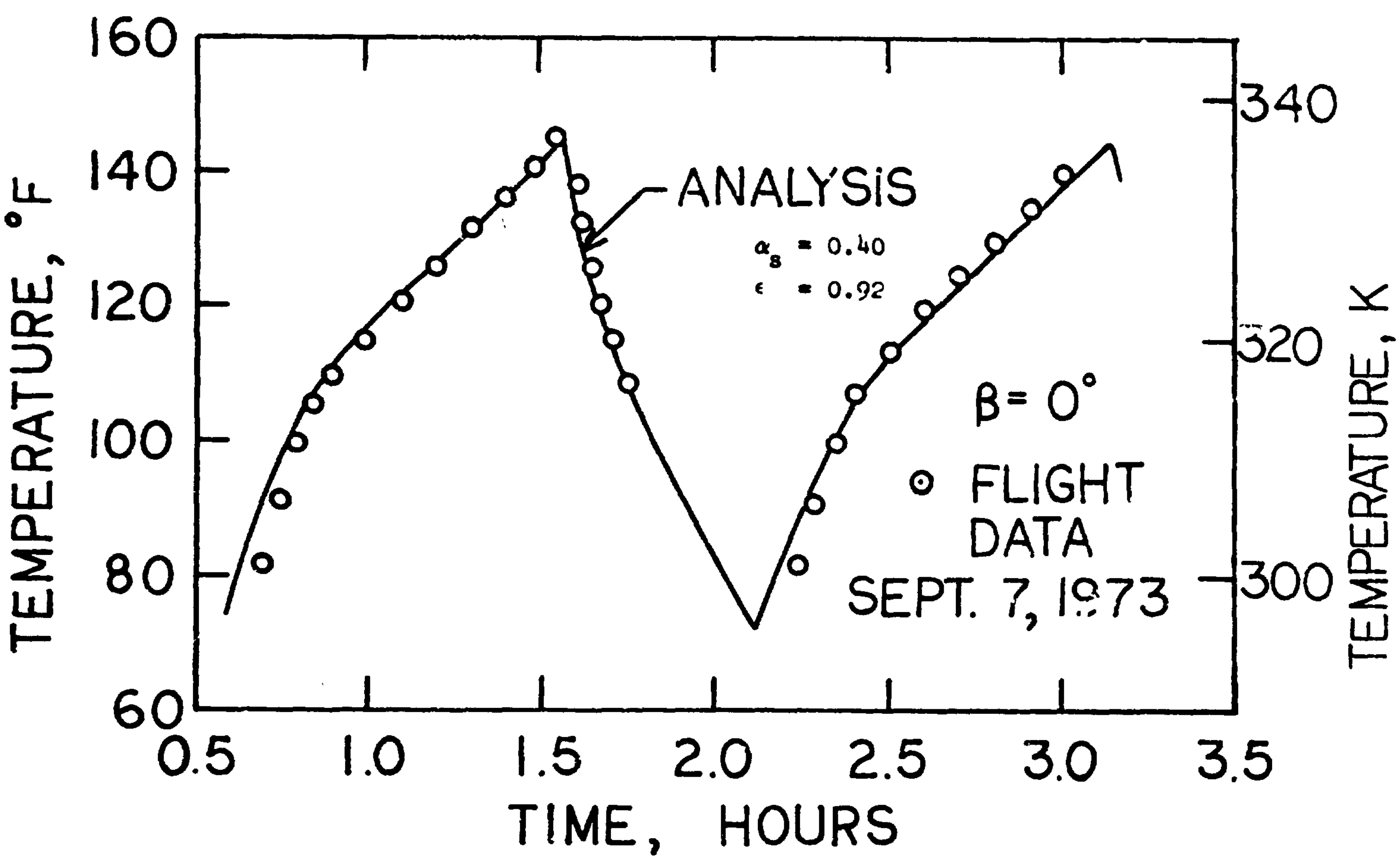


Figure 2.2.3-60

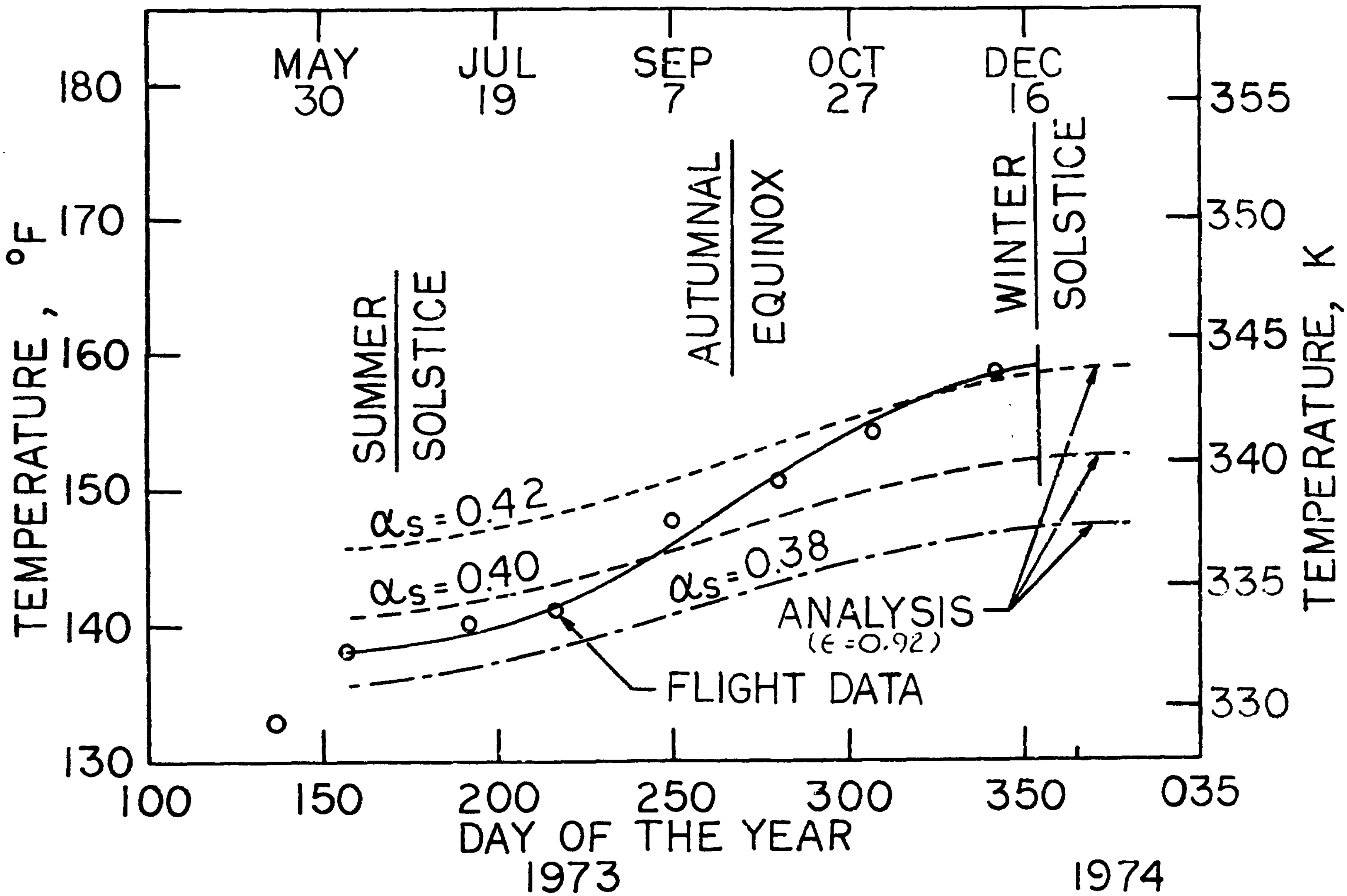


Figure 2.2.3-61 AFT SKIRT MAXIMUM ORBITAL TEMPERATURES AT BETA= 0 DEG. SENSOR C7189

data. The flight data clearly showed that there was a solar absorptance degradation during this time period. Starting at the summer solstice, the analytical data indicated:

	α_s
Summer Solstice	0.39
Autumnal Equinox	0.41
Winter Solstice	0.42

Considering an orbital insertion value for the white paint absorptance of 0.34, reflecting the retro-rocket plume contamination, the degradation effect (α_s/α_s') normalized to orbital insertion is shown in Figure 2.2.3-62 in terms of solar exposure time. Compared with design test data, the flight data are seen to compare well to the test data of Steube[3], but are lower than the degradation rate given in the previously mentioned data of Zerlaut. It should be noted, however, that the design test data are for an S-13G surface while the flight data are for a plume contaminated S-13G surface.

3. Steube, K. E. and Linford, R.M.F., "Long-Duration Exposure of Spacecraft Thermal Coatings to Simulated Near-Earth Orbital Conditions, "AIAA 6th Thermophysics Conference, Paper No. 71-454, April 26-28, 1971

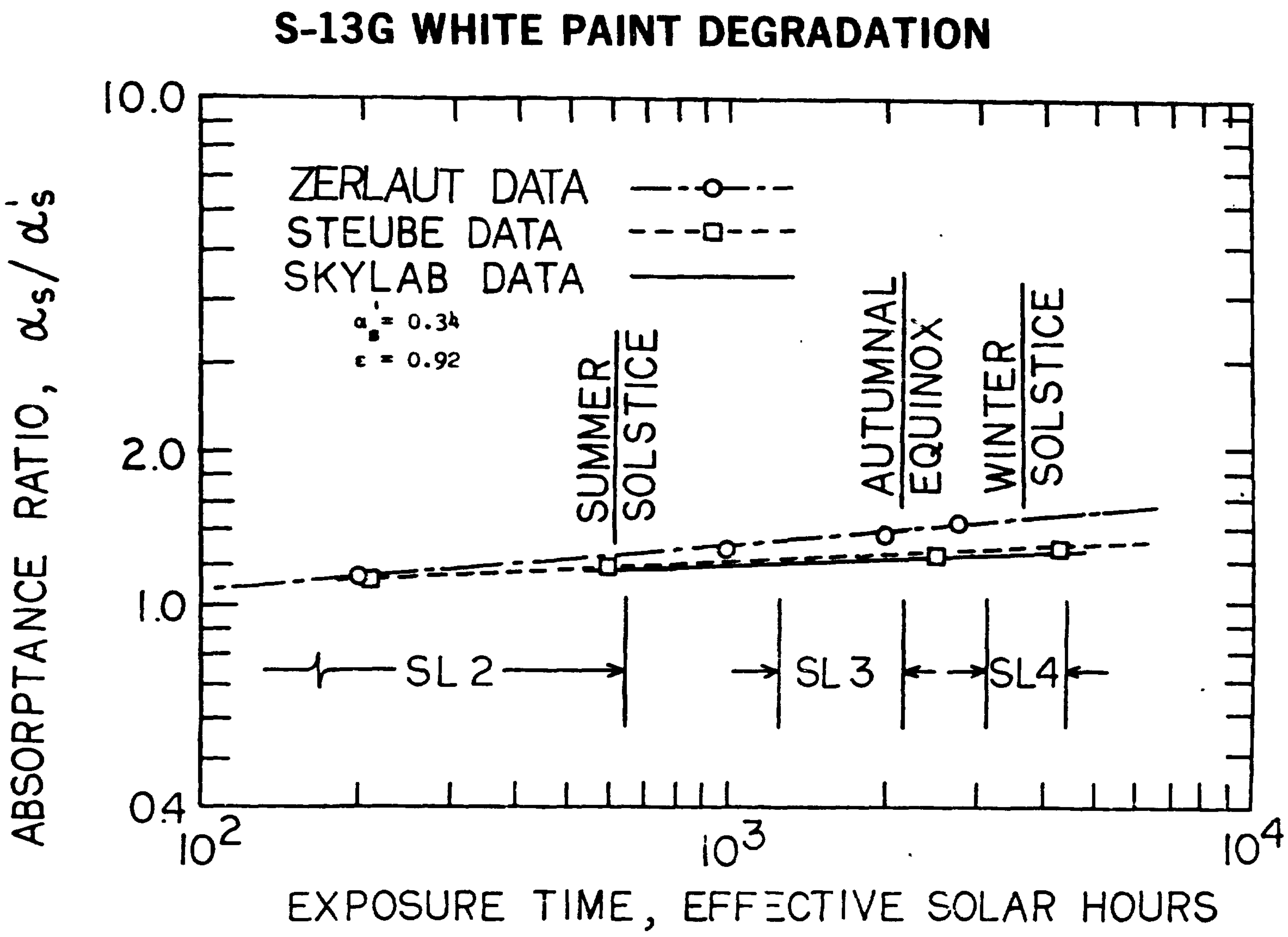


Figure 2.2.3-62

2. Goldized Kapton Tape

(a) Installation

The goldized Kapton Tape was installed on the external surface of the habitation area sidewall to provide a low emittance (ε) surface. This surface in combination with the black-and-white painted meteoroid shield originally covering it, was to have provided the desired heat balance to meet astronaut comfort and other thermal control requirements within the habitation area. The six-inch (15.2 cm) wide tape, Mystic 4017, consisted of 680 Angstroms of gold on a one mil (0.0025 cm) Kapton film backed with silicone adhesive and was installed in butted circumferential bands on the habitation area sidewalls. The tape was installed by a controlled application procedure which included air bubble removal between the tape and the sidewall surface. The gold surface of the tape was protected by a plastic film until just prior to installation of the meteoroid shield in October, 1973.

Extensive measurements were made of the gold tape ε after installation on the habitation area sidewall. The average value from 50 measurements was 0.03.

(b) Prelaunch Measurements

The measurements of the installed goldized Kapton were made at KSC from March 31, 1973, to April 13, 1973. Measurements were made at four general locations and values obtained were 0.022, 0.022, 0.036 and 0.040. An average of the four measurements gives 0.030 which agrees well with data obtained from new material as well as that obtained shortly after installation. No data were taken for solar absorptance (α_s) since the Kapton surface was not designed to be exposed to direct solar impingement or albedo. Measurements of α_s for samples of the goldized Kapton gave a value of approximately 0.15. Therefore, a reasonable estimate of the optical properties at lift-off is $\alpha_s/\varepsilon = 0.15/0.03$.

(c) Orbital Optical Properties Prior to Sunshade Deployment

With the loss of the meteoroid shield approximately 63 seconds after lift-off on DOY 134, the Position Plane I side of the habitation area sidewall was subjected to continual direct solar exposure after the vehicle was inserted into orbit and attained a solar inertial attitude. As a result of exposure of the gold surface with a high α_s/ε, temperatures of the habitation area on the solar side rose rapidly attaining an estimated maximum temperature of 300°F (422 K) at the external surface and 200°F (367 K) on the internal surface of the one-inch (2.5 cm) thick polyurethane foam. After approximately one and a half days in orbit, a series of pitch maneuvers was performed to reduce the solar incidence angle of the gold surface, and lower the sidewall temperatures. On DOY 142, the habitation area temperature was stabilized to a mean internal value of 125°F (325 K) with a maximum external temperature of approximately 200°F (367 K).

Following orbital insertion, the main complication in evaluating the gold tape optical properties was that the temperature instrumentation on the sunside of the vehicle was off-scale high because of the higher than expected heat fluxes, resulting in ill-defined boundary temperatures for the OWS heat balance. Two evaluation methods were employed to determine optical properties for the period immediately after insertion on DOY 134. A small scale thermal model was used to analyze the large transient reponse of the Position Plane I internal and external temperature data before their respective maximum temperature scales of 120°F (322 K) and 190°F (361 K), were exceeded. Also, the OWS CINDA model was run to correlate the temperature readings remaining on-scale.

For the transient analysis of Position Plane I, temperature data taken at MDAC Stations 319, 420 and 460 approximately 75 minutes after lift-off were utilized to estimate the optical property values. The data consisted of the outboard and inboard surface temperature responses of the tank sidewall foam insulation on the Position Plane. During the time period chosen, sunlight was directly incident on the area. The thermal model was defined to solve for the tank sidewall temperatures for a given set of external surface optical properties.

Utilizing a solar flux of 419 Btu/hr-ft^2 (1325 W/m^2) and an albedo of 0.3, it was found that at MDAC Station 420 an α_s and ε of 0.175 and 0.035, respectively, best matched the flight data as shown in Figure 2.2.3-63. Similar analyses of the temperature data at MDAC Stations 460 and 319 gave α_s/ε values of 0.165/0.03 and 0.21/0.05,

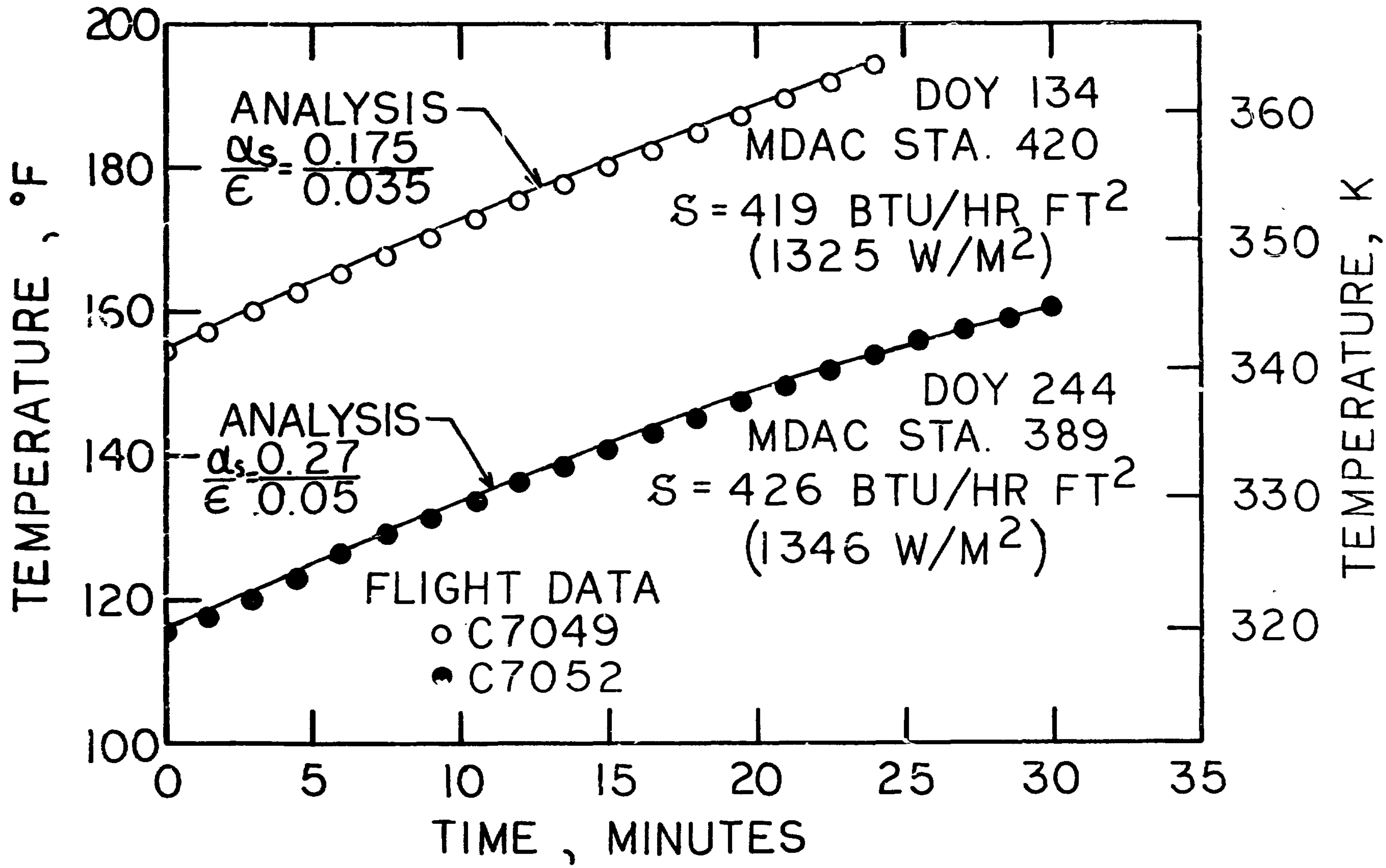


FIGURE 2.2.3-63 TEMPERATURE RESPONSE OF GOLD TAPED SIDEWALL TO DIRECT SOLAR EXPOSURE

respectively. The optical property variation with the longitudinal station was indicative of a decreasing degradation effect from retro-rocket plume contamination with increasing distance from the plume source.

Utilizing the OWS CINDA model correlations of the flight temperature data for the first one and a half days and three days after orbit insertion were made. The analyses of the on-scale internal temperatures during these time periods indicated a gold tape α_s of 0.20 and an ϵ of 0.04.

The results of the transient response analyses of Position Plane I as well as the overall heat balance analyses using the CINDA model were indicative of a degradation in the gold tape α_s and ϵ. Surface degradation of the tape was verified by photographs which indicated contamination from the retro-rockets (fired during separation of the OWS from the Saturn S-II Stage), scratches from the meteoroid shield and bubbling on the Position Plane I side of the vehicle, probably resulting from the high temperatures which occurred before deployment of the sunshade.

(d) Orbital Optical Properties after Sunshade Deployment

On DOY 147, the first Skylab crew erected a parasol sunshade, which in conjunction with the operation of the active gas cooling system reduced the OWS mean internal temperatures to a range of 78°F (299 K) to 90°F (305 K) making the environment suitable for habitation and equipment operation. This was followed on DOY 218, by the deployment of a sail sunshade over the parasol to provide additional shading and reduce the habitation area temperatures further.

The evaluation of the gold tape properties following sunshade deployment was difficult in that there were many parameters strongly influencing the habitation area heat balance which could not be individually assessed in any given period. The changes during the mission in orbital sunlight fraction, angle between the orbit plane and the solar vector (β), habitation area waste heat loads and cooling, shading from different sunshade configurations, and seasonal variations in the solar flux lead to a very complex set of conditions to analyze.

Utilizing the OWS CINDA model and the flight temperature data from DOY's 225 and 234, gold tape α_s and ε values of 0.21 and 0.04, respectively, were calculated using a solar flux of 424 Btu/hr-ft^2 (1340 W/m^2) and an albedo of 0.3. These values were essentially the same as the values calculated at orbital insertion. The results are questionable, however, because the exact sunshade coverage which strongly influences the heat balance was not known.

Based on temperature data taken on DOY 244 during an Earth Resources maneuver which resulted in large temperature transients in the Position Plane IV area, the gold tape α_s and ε at MDAC Station 389 were calculated to be 0.27 and 0.05, respectively, (See Figure 2.2.3-63). For this analysis a solar flux of 426 Btu/hr-ft^2 (1346 W/m^2) was used with an albedo of 0.3. A similar analysis of temperature data taken during a maneuver to photograph a barium cloud on DOY 331, was indicative of the same calculated optical properties.

An analysis of temperature data taken at MDAC Stations 319 and 389 in the Position Plane II area during two consecutive Earth Resources passes on DOY 258, indicated gold tape optical properties at Station 319 of 0.35 and 0.10 for α_s and ε, respectively, and 0.30 and 0.06, respectively, for Station 389 as indicated by the correlation in Figure 2.2.3-64. A solar flux of 428 Btu/hr-ft^2 (1352 W/m^2) and an albedo of 0.3 were used in the analysis.

In late December 1973, and in January 1974, a series of Kohoutek comet viewing maneuvers and Earth Resources maneuvers were performed. These maneuvers again exposed the Position Plane II area to direct sunlight, resulting in large temperature transients because of the high β angles and associated large orbital sunlight fractions. The analyses of the temperature transient at MDAC Station 319 indicated an α_s of 0.35 and ε of 0.10. For Station 389 the α_s and ε were 0.30 and 0.06, respectively. A solar flux of 441 Btu/hr-ft^2 (1394 W/m^2) and an albedo of 0.3 were assumed. The data correlation for DOY 009 (1974) is shown in Figure 2.2.3-65.

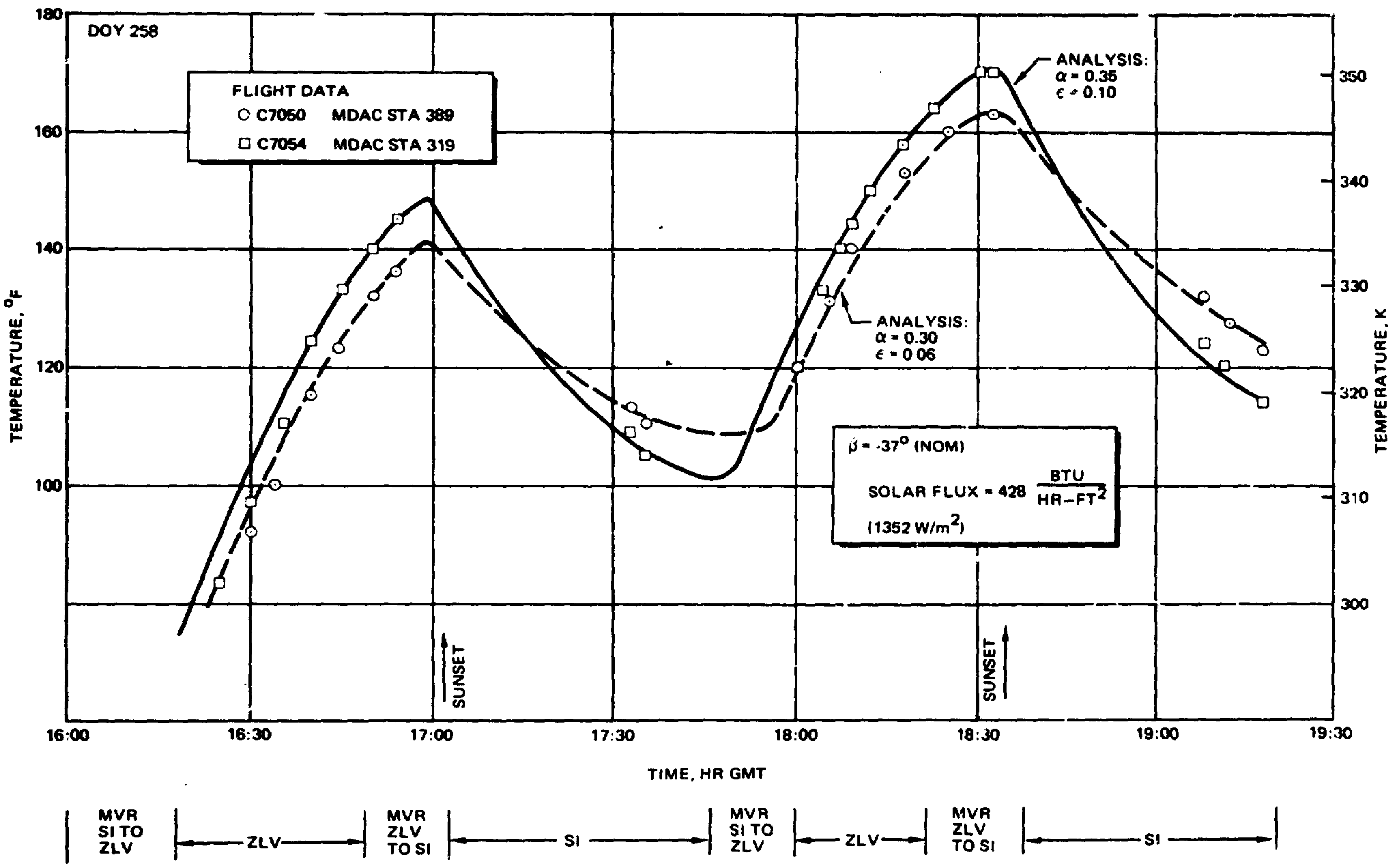


Figure 2.2.3-64. OWS External Wall Temperature Simulation for EREPS 31 and 32

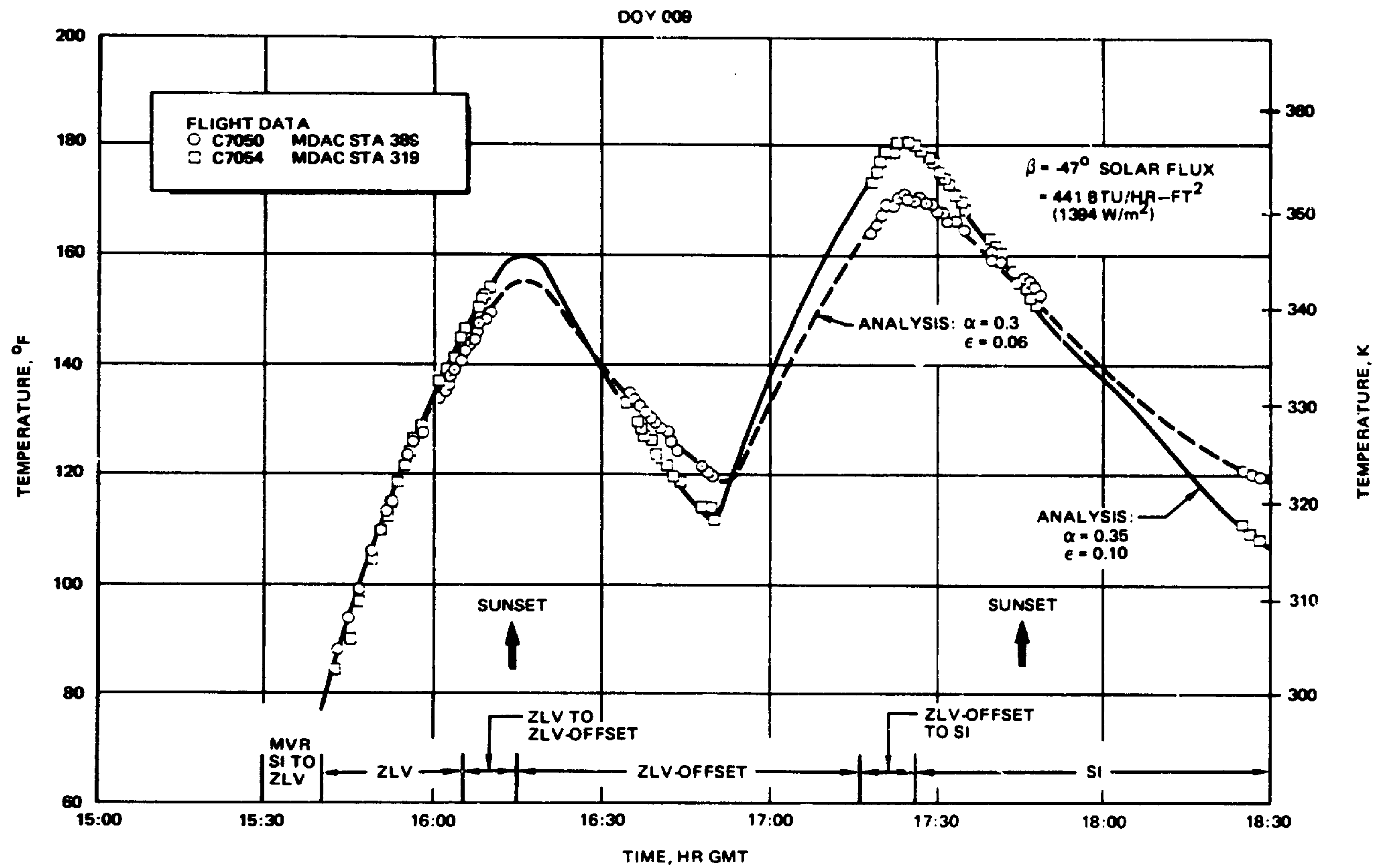


Figure 2.2.3-65. OWS External Wall Temperature Simulation for EREP 24

(e) Data Summary

The calculated optical properties cannot be considered as exact values because of the potential errors associated with accuracy of the temperature data, sunshade coverage and vehicle attitude history during maneuvers. However, the results are valid for determining variations and establishing trends in the optical properties over a period of time. The computed optical properties are also sensitive to the orbital heat flux. The solar flux used was the nominal value based on the distance from the earth to the sun for the particular season of the year being considered. An albedo of 0.3 was assumed which was significant only for the average sidewall calculations using the CINDA model.

The gold tape optical property data are summarized in Figure 2.2.3-66 and Table 2.2.3-13. The results show the effect of two parameters, retro-rocket plume contamination and exposure time to the orbital environment. The results presented for DOY 134 were based on temperature data read from

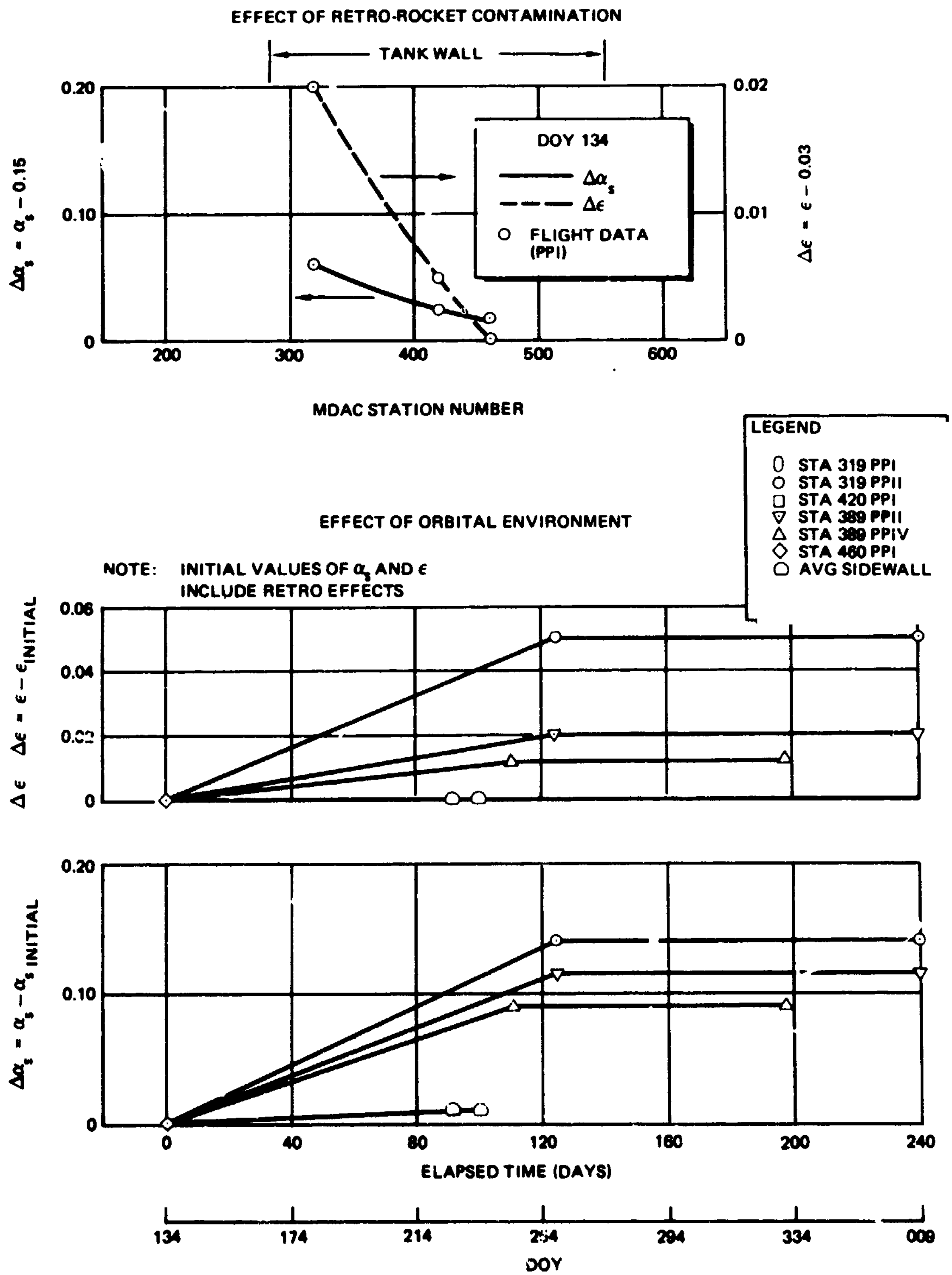


Figure 2.2.3-66. Tank Wall Optical Properties

Table 2.2.3-13

Gold Tape Optical Properties

DOY	Mission Phase	Orbital Solar Flux $Btu/hr\text{-}ft^2$	Orbital Solar Flux $(watt/m^2)$	Location PP/MDAC Sta. Number	α_s	ε	α_s/ε
134 (1973)	Prelaunch	N/A		Average Sidewall	0.15*	0.03**	5.0
134	Orbital Insertion	419	1324	I/460	0.165	0.03	5.5
134	Orbital Insertion	419	1324	I/420	0.175	0.035	5.0
134	Orbital Insertion	419	1324	I/319	0.21	0.05	4.3
134	Orbital Insertion	419	1324	Average Sidewall	0.20	0.04	5.0
225	On-Orbit	424	1340	Average Sidewall	0.21	0.04	5.3
234	On-Orbit	424	1340	Average Sidewall	0.21	0.04	5.3
244	On-Orbit	426	1346	IV/369	0.27	0.05	5.4
258	On-Orbit	428	1352	II/319	0.35	0.10	3.5
258	On-Orbit	428	1352	II/369	0.30	0.06	5.0
331	On-Orbit	439	1387	IV/369	0.27	0.05	5.4
9 (1974)	On-Orbit	441	1394	II/319	0.35	0.10	3.5
9 (1974)	On-Orbit	441	1394	II/369	0.30	0.06	5.0

* Based on measured α_s for new material.

** Average value based on measured ε values at 4 sidewall locations.

three longitudinal stations on PP I. They indicated the degradation effect of the retro-rocket plume. Both α_s and ε were increased by the contamination. The results presented for DOY's 134, 244, 258, and 9 showed the effect of orbit time on the optical properties. The trend indicated was an increase in α_s and ε for the first 100 to 125 days and essentially constant values thereafter.

c. Common Bulkhead Heat Leak

The common bulkhead heat leak values determined from flight data at various conditions, and analytical predictions at beta angles (β's) of zero and -73 degrees are presented in Table 2.2.3-14. The flight data from four pairs of temperature sensors located on the bulkhead insulation surfaces (defining the temperature differences across the insulation) and thermal conductivity values obtained from experimental data were used in the heat leak calculations. The analytical predictions for the maximum heat leak through the 500 ft^2 (46.5 m^2) of common bulkhead area during habitation were 236 Btu/hr (69.0 watts) greater than the average value from flight data at $\beta = 0°$, and 175 Btu/hr (51.4 watts) greater than the

TABLE 2.2.3-14

COMMON BULKHEAD HEAT LEAK

Time DOY (5)	Beta Angle, deg	Source of Data	Average Temperatures Plenum Gas(2) °F	K	Insulation OWS Face (3) °F	K	Average Insulation Temperature Difference (3) °F	K	Heat Leak (4) Btu/hr	Watts
217	0	Flight	76	298	75	297	14.5	8.06	524	154
235	62	↑	72	296	72	296	12.0	6.67	432	126
265	-49	│	71	295	71	295	13.7	7.61	490	144
280 (6)	0	│	55	286	54	286	6.0	3.33	209	62
307 (6)	0	│	60	289	57.5	288	6.5	3.61	228	67
364	0	↓	72	296	72	296	11.0	6.11	396	116
016	-71	Flight	79	299	78.5	299	8.0	4.44	292	85
(1)	0	Analysis	69	294	65	292	20.0	11.1	696	204
(1)	-73	Analysis	84	302	61.5	301	12.9	7.17	467	137

NOTES:
(1) Analytical prediction for habitation period.
(2) Average plenum gas temperature is average from sensors C7144 and C7256 (flight data only).
(3) Insulation temperatures from sensor pairs: C7181-C7179, C7095-C7097, C7182-C7180, C7091-C7092 (flight data only)
(4) Based on an insulation thickness of 3 in. (7.62 cm) and the following insulation conductivity:
    0.016 Btu/hr-ft-°F (0.027 W/m·K) at 0°F (256 K)
    0.019 Btu/hr-ft-°F (0.033 W/m·K) at 100°F (311 K).
(5) MSFC Sun Shade deployed after 1730 GMT on DOY 218.
(6) DOY's 280 and 307 occur during the storage period between SL-3 and SL-4.

flight data value at near maximum negative values of β. The maximum difference between the analytical and flight data heat leak values represented approximately 5 percent of the total OWS heat leak.

The smaller bulkhead heat leak values during storage primarily resulted from the duct fans being turned off, reducing the convection heat transfer in the plenum region at the insulation surface to essentially zero. A comparison of the data from the table shows the storage period heat leak rates on DOY's 280 and 307 to be roughly one half of the corresponding habitation period heat leak values for DOY's 217 and 364. The significance of convection heat transfer to the common bulkhead during habitation is also shown by the difference in the average gas-side insulation surface temperature between the habitation and storage periods. During the habitation period, the gas-side surface of the insulation follows the plenum gas temperature closely for the temperature range encountered (71 to 79°F or 295 to 299 K). During storage, the plenum gas temperature measurements reflected the mean radiant temperature of the floor and plenum area which was 15 to 20°F (8 to 11 K) colder than during habitation.

d. Heat Pipes

The combination of the sunshades and the gold external surface on the habitation area provided a significant reduction of the habitation area sidewall circumferential temperature gradients over that anticipated from the design condition with the meteoroid shield passive control. Consequently, the sidewall heat pipe loads were so small as to preclude a meaningful evaluation. This was shown by measurements*on DOY 227 of heat pipe temperatures of 74°F (297 K) and 71°F (295 K) at PP I and between PP II and PP III, respectively, at the forward compartment floor. Also at the experiment compartment floor station the heat pipe temperatures were 71°F (295 K) and 73°F (296 K) at PP I and between PP III and PP IV, respectively. The heat load was small for the set of heat pipes which thermally coupled the logic unit to the food freezer. The heat pipe temperatures of 67°F (293 K) at the freezer and 73°F (296 K) at the logic unit measured on DOY 227 differed by only a few degrees as expected and were in a correct relationship with each other and the local environment temperature of about 71°F (295 K).

For the food freezer application, the heat pipes had a much larger heat transfer capability than required and

* Hand-hold probe

and could perform their design function with the artery (Figure 2.2.3-44) not being filled with liquid returning to the evaporator section of the heat pipe. The heat transfer capability of these heat pipes in the non-arterial mode is approximately 13 watts whereas the operational requirement was only about 9 watts. Thus, the temperature relationships noted in the previous paragraph for these heat pipes was not a demonstration of arterial mode operation.

In summary, the heat pipe data obtained during the SL-2 mission did not answer the basic questions concerning the operation of the heat pipes. Very little information could be gleaned from the circumferential heat pipe data. The data obtained from the logic unit/food freezer installation indicated that the heat pipes were operating, but the loads were so small that operation in an arterial mode was not demonstrated. For these reasons it was concluded that one or more experiments would have to be performed on the heat pipes during the SL-3 mission if a performance verification was to be obtained. A test procedure was written for the SL-3 crew to determine if one of the heat pipes in the logic unit/freezer system would operate in the arterial mode. The procedure (which was not implemented because of activity timeline considerations) involved removal of one heat pipe from the back of the freezer, immersing one end of the pipe in a bag of cold water and the other end in hot water, and then measuring the temperature responses of the system.

3/ Overall System Performance

a. OWS Temperatures

1. SL-1 Insertion through SL-2 Habitation Period (DOY 134 to 173)
The OWS mean internal temperature history and the temperature histories of the ambient food and film vault are shown in Figures 2.2.3-67, -68 and -69, respectively, from orbital insertion (DOY 134) until deployment of the parasol sunshade (DOY 147). The temperature histories were estimated using the empirical relationships to Sensors C7040 and C7044 noted on the figures since many of the internal temperatures were off-scale high (120°F or 322 K maximum). With the vehicle in a solar inertial attitude and the gold surface of the sidewall exposed to direct solar inputs, the internal temperatures rose rapidly for about 1.5 days after orbital insertion. The rate of temperature change was then decreased by a series of pitch maneuvers which were implemented to provide temperature control.

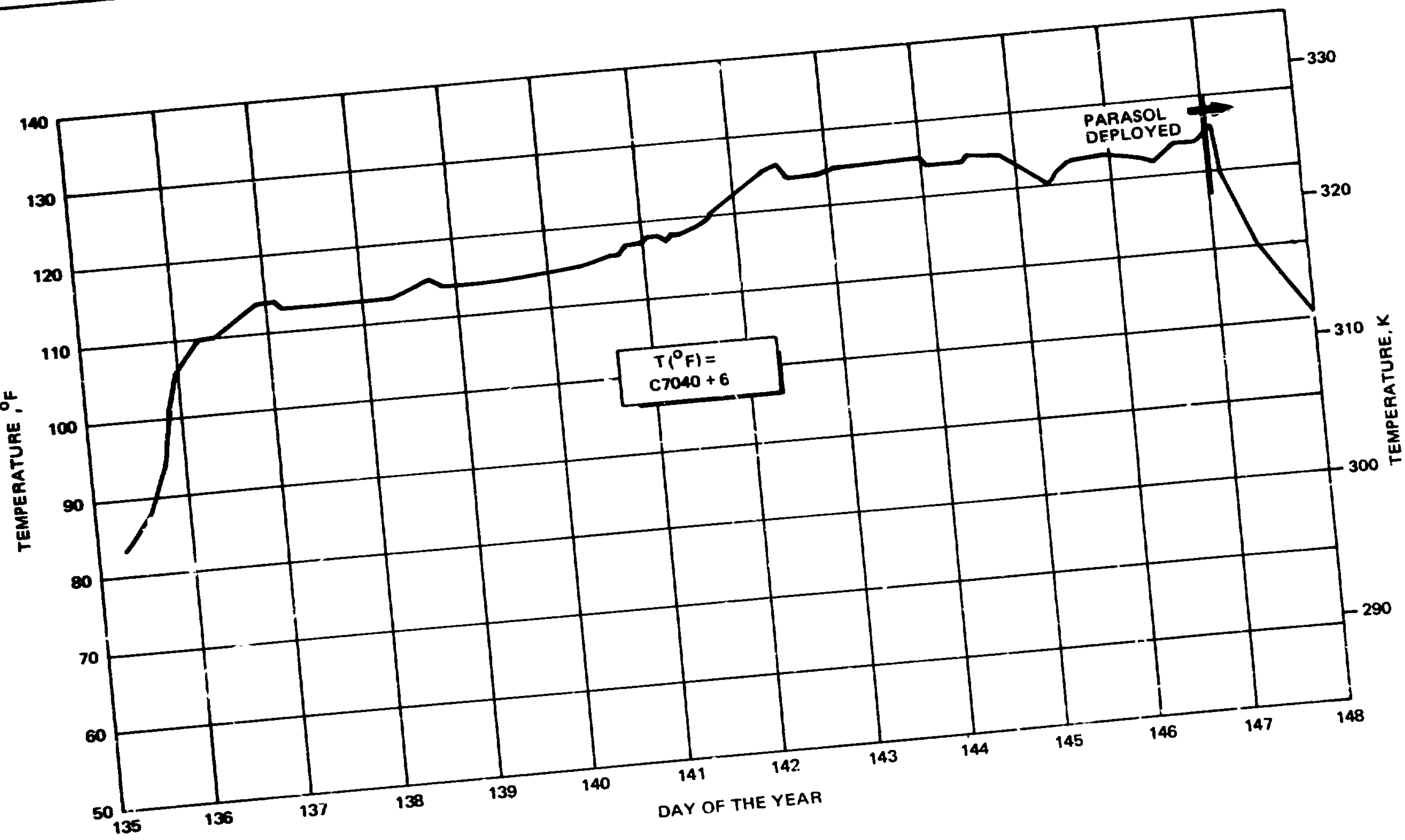


Figure 2.2.3-67. OWS Mean Internal and Floor-Stowed Food Temperature History, DOY 135-147

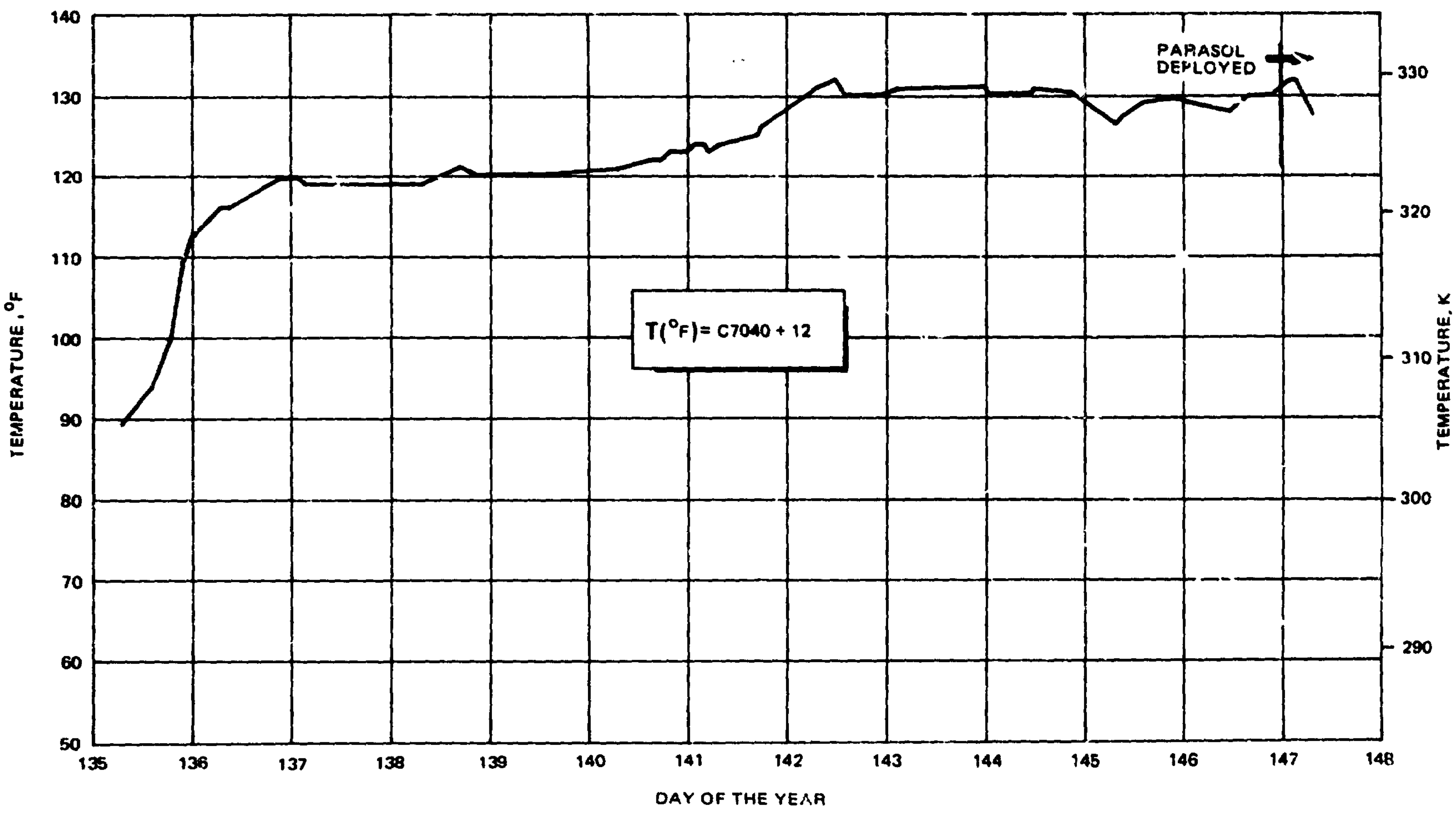


Figure 2.2.3-68. OWS Rack-Stowed Food Temperature History, DOY 135-147

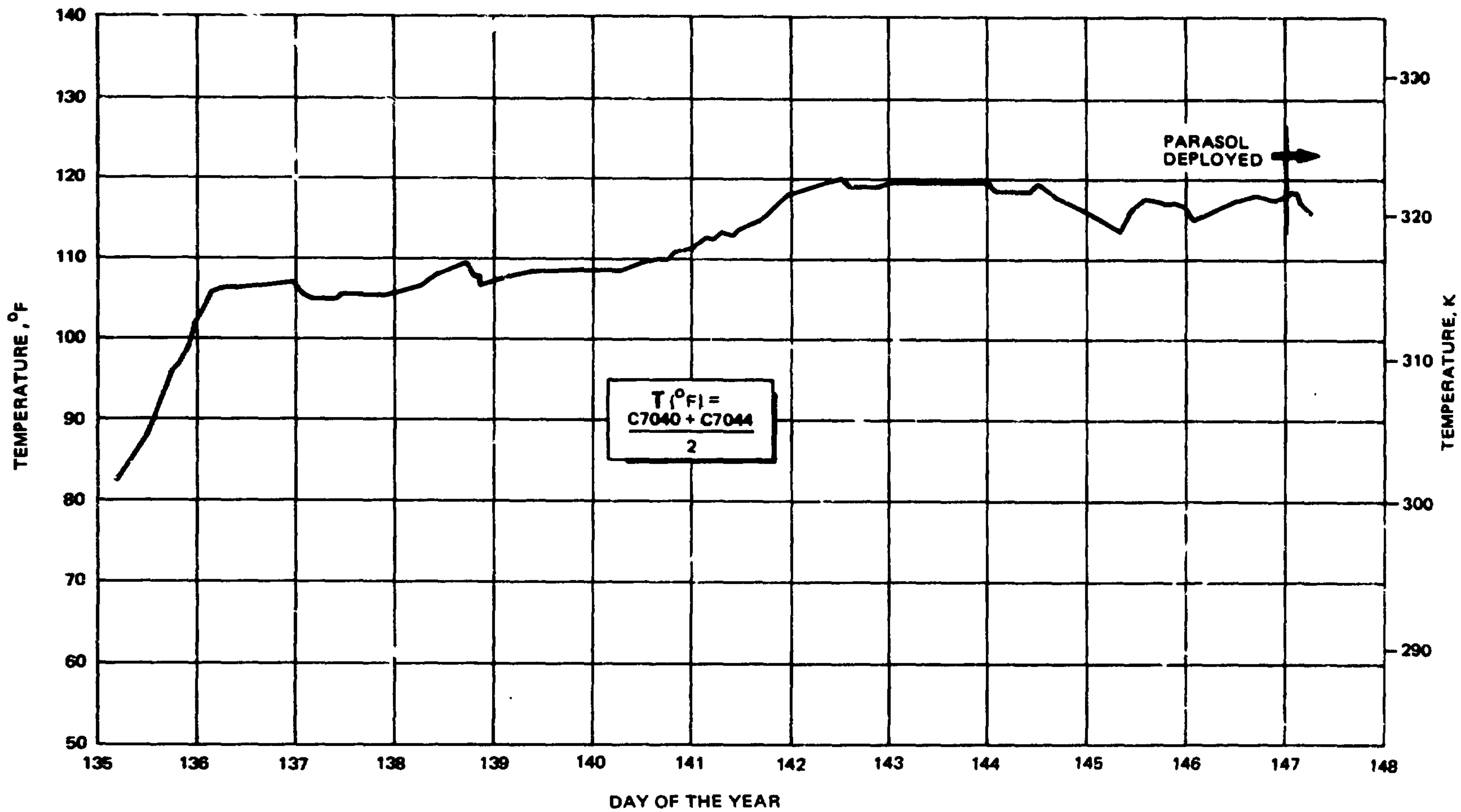


Figure 2.2.3-69. OWS Film Vault Temperature History, DOY 135-147

The mean internal and floor-stowed food temperatures reached a maximum of 126°F (326 K) on DOY 142 and then varied between this value and 120°F (322 K) until deployment of the parasol on DOY 147. The ambient rack-stowed food was estimated to be 6°F (3.3 K) higher during this period since it was influenced more by the hotter sidewall near the Position Plane I side of the habitation area. The film vault was estimated to be approximately 5°F (2.8 K) cooler than the mean internal temperature with a maximum of 120°F (322 K). The vault was near the Position Plane IV wall where the temperatures were moderated. The OWS mean internal, rack-stowed food, and film vault temperature responses are shown in Figures 2.2.3-70, -71, and -72, respectively, during the cooldown period after deployment of the JSC Parasol on DOY 147. The temperatures started to drop immediately upon parasol deployment. By noon on DOY 154 the mean internal temperatures of the crew quarters and forward compartment had just come into the CEI-specified crew comfort box as shown in Figure 2.2.3-73. The average internal surface temperature dropped to about 76°F (298 K) during DOY 156 and continued at that level until DOY 149 when it dropped to about 74°F (297 K).

Analytical simulations of the initial OWS interior cooldown agreed well with flight data as shown in Figure 2.2.3-74. The

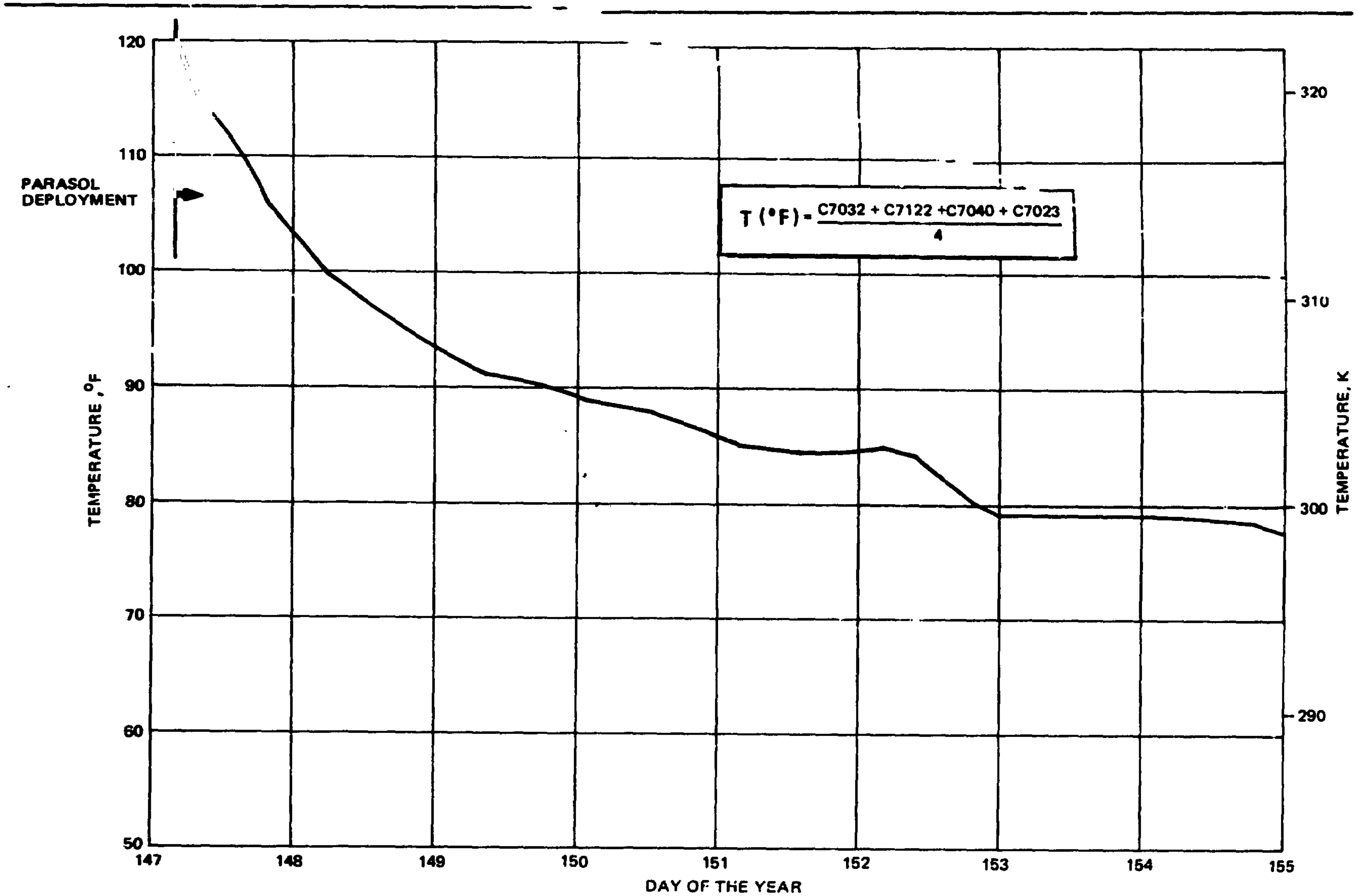


Figure 2.2.3-70. OWS Mean Internal Temperature History, DOY 147-154

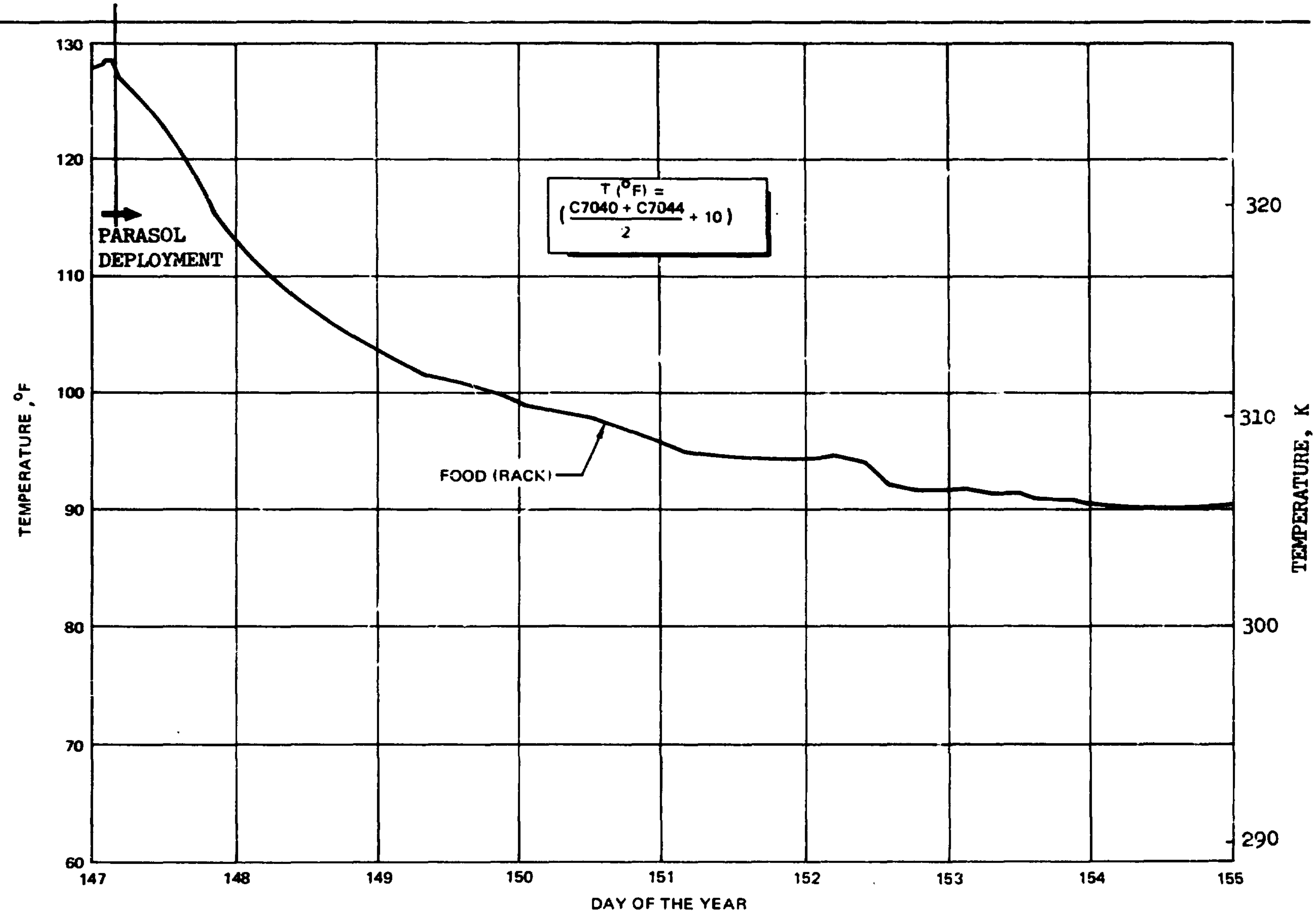


Figure 2.2.3-71. OWS Rack-Stowed Food Temperature History, DOY 147-154

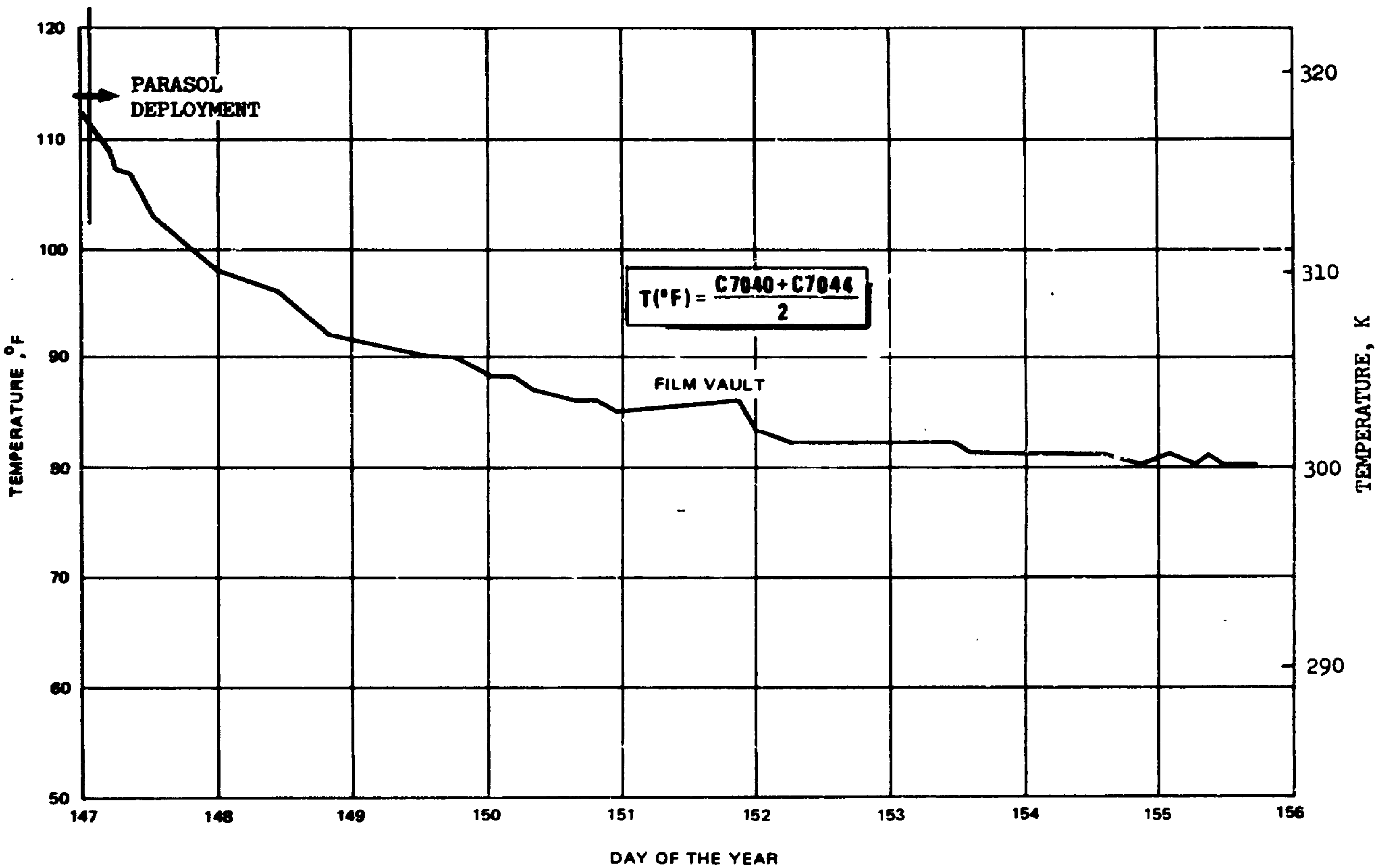


Figure 2.2.3-72. OWS Film Vault Temperature History, DOY 147-155

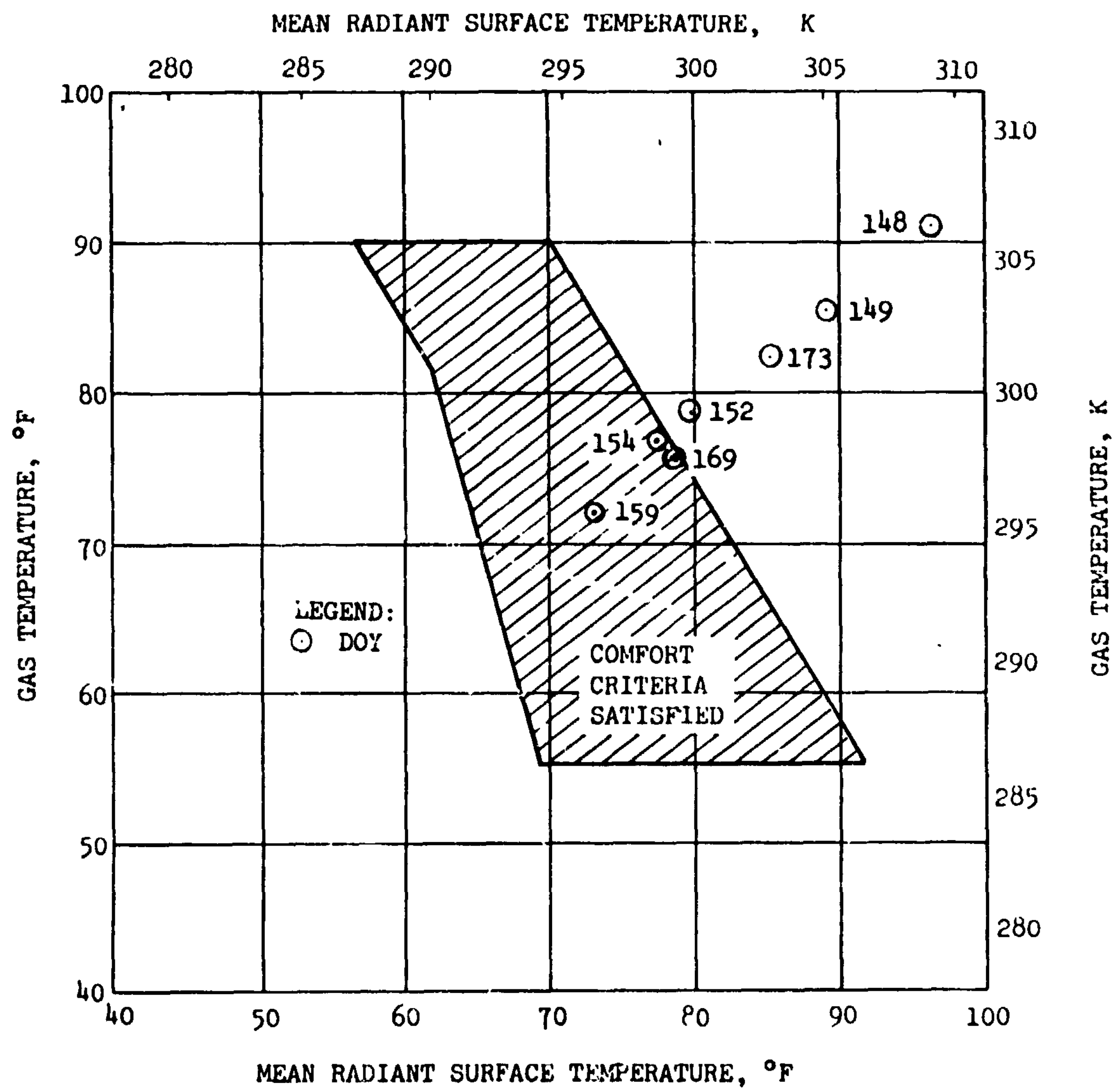


FIGURE 2.2.3-73 SL-2 Crew Comfort Conditions

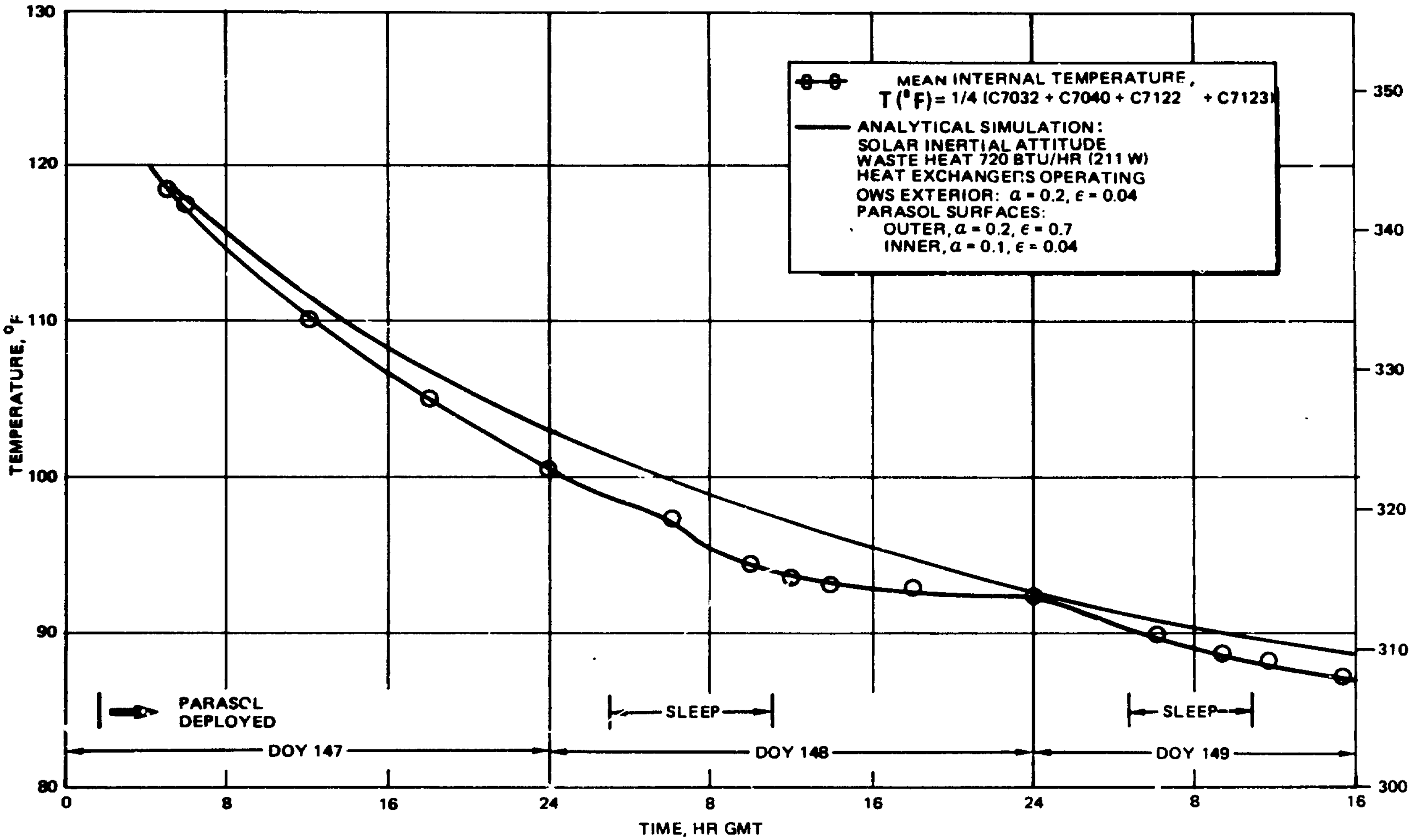
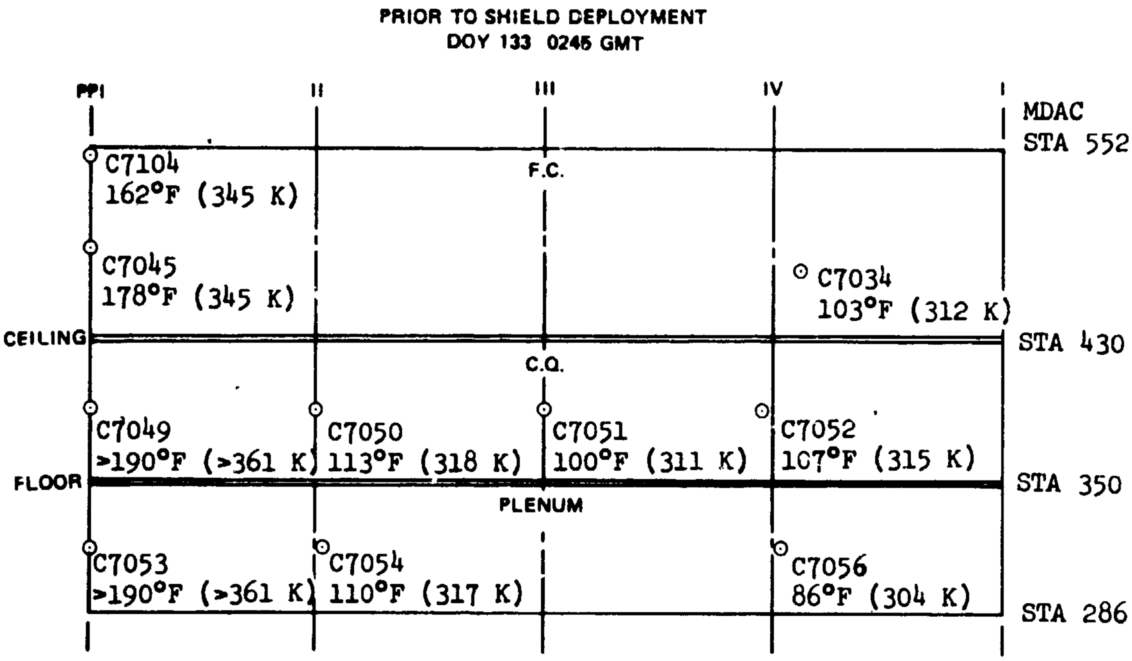


Figure 2.2.3-74. OWS Cooldown After Parasol Deployment

slower cooldown determined from the analytical model was ascribed to uncertainties in sunshade coverage, surface properties, space environment parameters, internal heat loads, and heat exchanger performance. Photographs returned by the crew later indicated that approximately 135 ft^2 (12.6 m^2) of projected gold taped area was exposed to the sun with the vehicle in a solar inertial attitude.

Figure 2.2.3-75 shows typical temperature distributions as measured on the habitation area wall external surface prior to and after parasol deployment. The effect of the shade in reducing temperatures on Position Planes (PP) I, II and IV is quite evident. The anti-solar side was also cooler since heat was being transferred to it from the hotter portions of OWS before parasol deployment. The temperature differential between the solar (PP I) and anti-solar (PP III) side was reduced from approximately 100°F (56 K) to less than 10°F (5.6 K).

On DOY 170 the parasol was rotated, first 25 degrees clockwise viewed from the sun and then 10 degrees counter clockwise. The initial rotation exposed more of the external surface adjacent to the plenum on PP I and also more of the crew quarters area between PP I and II which were monitored by temperature sensors C7053 and C7094, respectively. Both



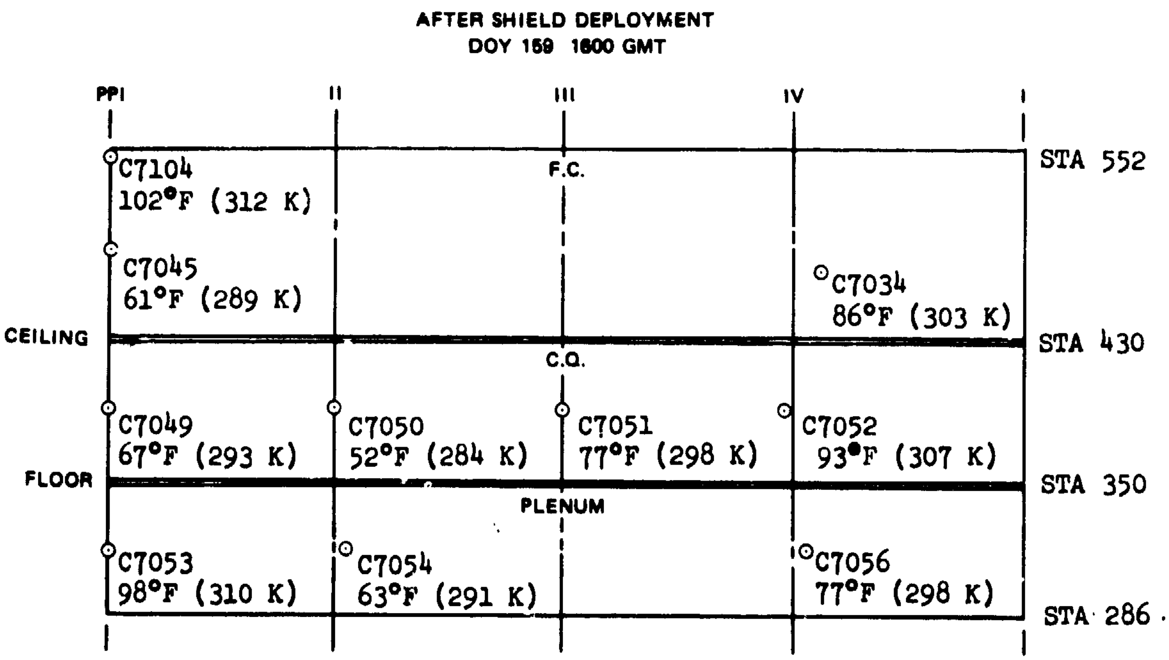


Figure 2.2.3-75 SL-2 External Surface Temperature Distribution

measurements in the noted areas experienced a noticeable temperature increase with the initial rotation and a subsequent decrease when the parasol was rotated back 10 degrees. A review of photographs returned by the astronauts indicated that the support rod over the Position Plane IV crew quarters area was restrained from moving during the clockwise rotation. This resulted in a compression of the sunshade material between the two aft rods thus exposing more tank sidewall area to direct solar impingement.

As shown in Figure 2.2.3-76, after the cooldown following the parasol deployment, there was a warming trend which began during DOY 159. This resulted from the change in beta angle and the higher internal heat dissipation as the power usage increased after SAS Wing 1 was deployed.

For the asymmetric vehicle configuration, it was noted from flight data that higher external heat inputs resulted as the beta angle decreased from -50 degrees and approached the maximum positive value of 73.5 degrees. This trend was verified by data obtained later in the mission on DOY's 275 to 295.

The mean internal temperature started rising at an increasing rate when full sunlight was reached on DOY 172 at a beta angle of 69.5 degrees. The OWS temperatures were sensitive to beta angle particularly near and during full sunlight because of the gold surface exposed as the result of incomplete deployment of the parasol. At the end of the SL-2 mission on DOY 173, the mean internal temperature was 86°F (303 K) and rising rapidly.

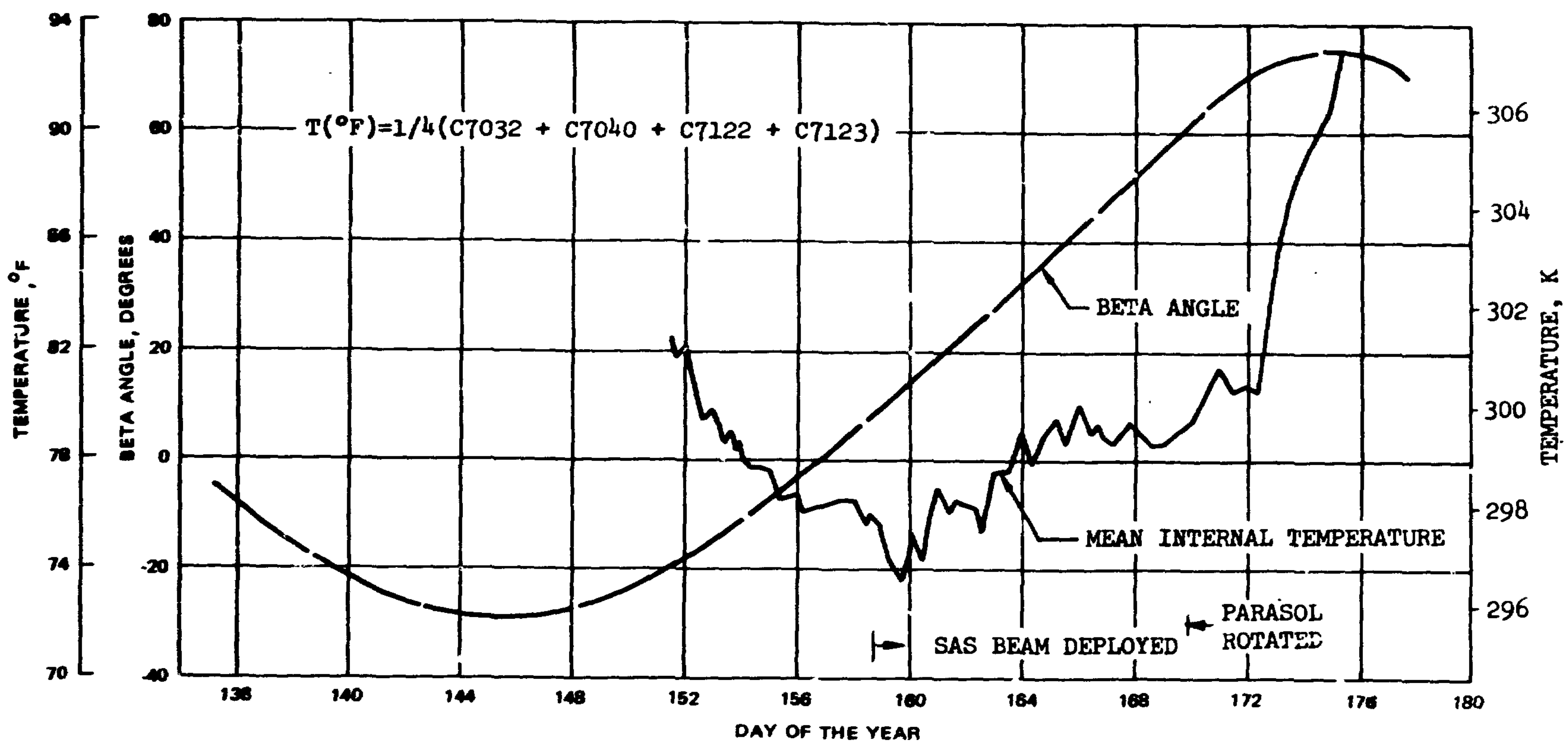


Figure 2.2.3-76 OWS Mean Internal Temperature History, DOY 148-174

2. First Storage Period (DOY 173 to 209)

All active elements of the E/TCS were kept inoperative during storage periods and control was exercised by the passive elements only. Figure 2.2.3-77 shows the histories of the mean internal surface temperature, the rack-stowed food temperature and the film vault temperature. The values of beta angle are also noted. After SL-3 undocking the temperatures continued to increase until shortly before partial sunlight orbits began on DOY 177. A peak value of 98°F (310 K) was calculated for the mean internal and rack-stowed food temperature. The maximum for the film vault was 103°F (313 K). Thereafter, the temperatures dropped as the beta angle decreased. The temperatures reached minimum values on DOY 200 which were maintained until SL-3 CSM docking on DOY 209. These mean internal/food and film vault minimum values were about 79°F (299 K) and 80°F (300 K), respectively.

3. SL-3 Habitation Period (DOY 209 to 268)

The daily mean internal temperature extremes for the SL-3 habitation period are presented in Figure 2.2.3-78. The amplitude of the daily temperature fluctuations of aoproximately 2°F (1.1 K) are caused by the variations in the waste heat profile between the active and sleep periods. In general the minimum temperature occurred at the end of

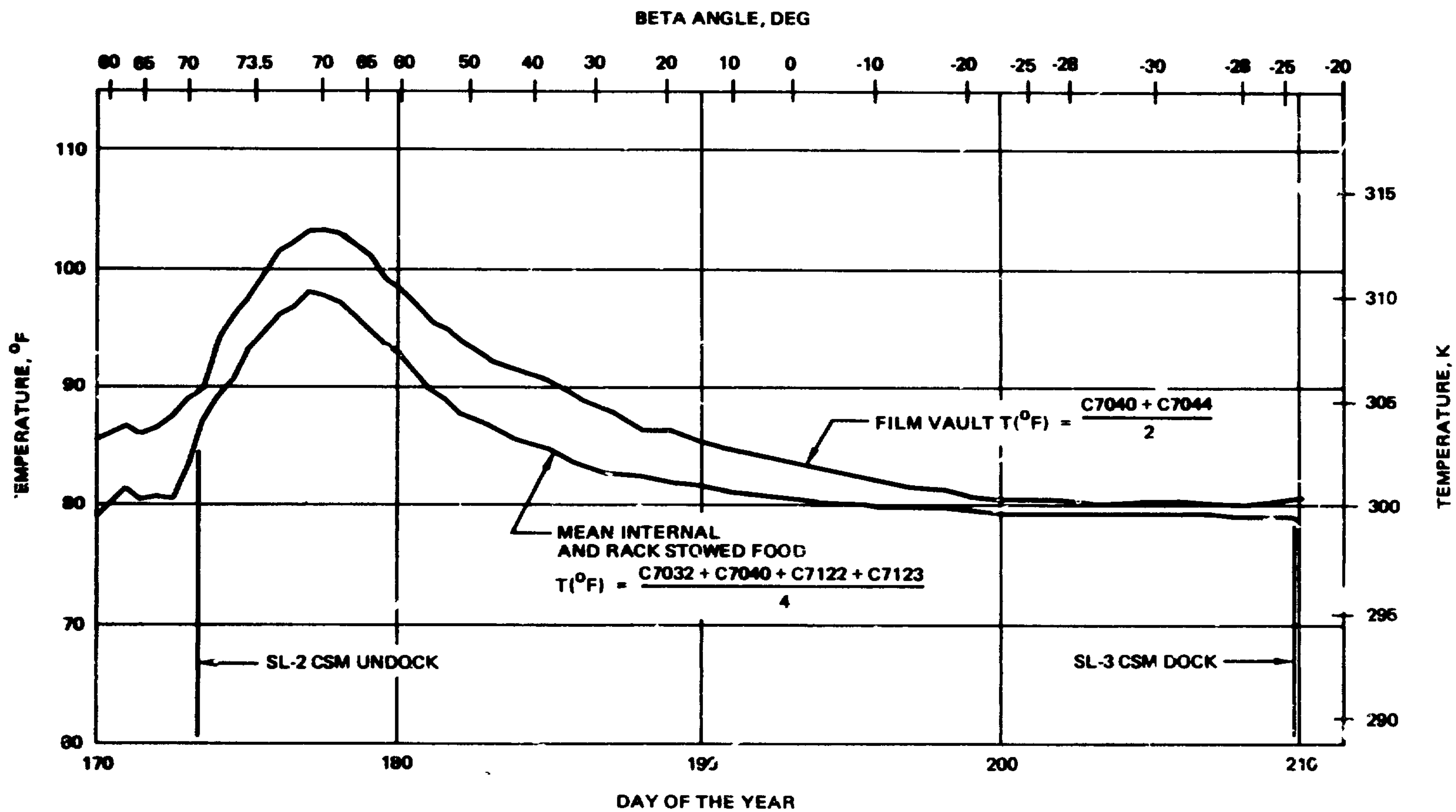


Figure 2.2.3-77. OWS Temperatures During the First Storage Period, DOY 173-209

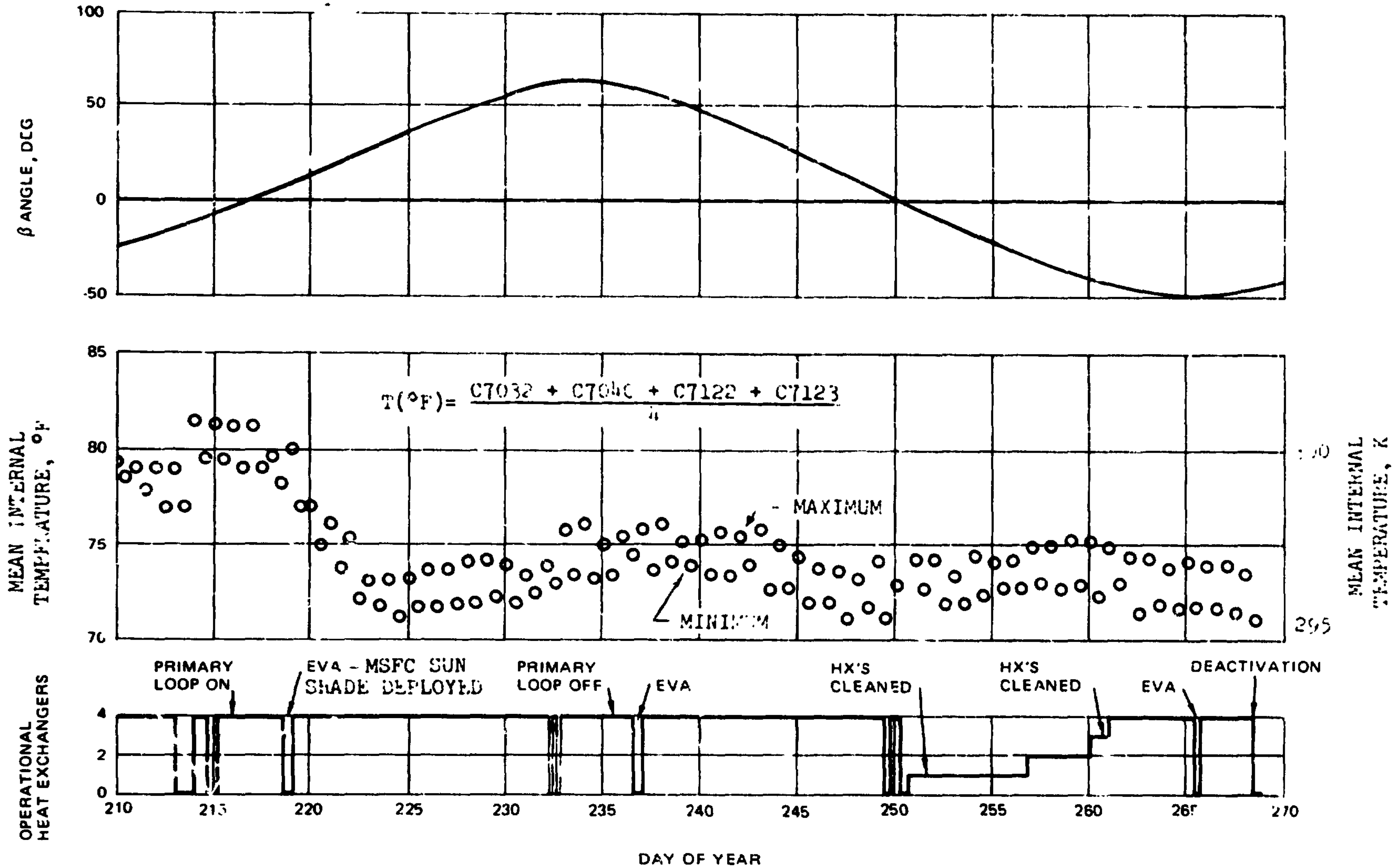


Figure 2.2.3-76 OWS Maximum and Minimum Mean Internal Temperatures, DOY 210-269

the sleep period. The orbit beta angle history and key events affecting the TCS performance are also listed. The activation period commenced on DOY 210 when four cooling heat exchangers (HX's) were turned on. This caused the internal temperatures to drop 2°F (1.1 K) until DOY 213 when the HX's were deactivated to prevent moisture from condensing in them. They remained off approximately 18 hours during which time the temperature increased to 81°F (301 K). They were turned on again and by DOY 218 the temperatures had decreased 1.5°F (0.8 K). At this time the MSFC Sunshade was deployed to provide additional shading of the OWS from direct solar radiation. The additional shielding resulted in a 7°F (3.9 K) temperature decrease during the next six-day period allowing the crew comfort criteria to be met for the remainder of the flight. This is shown in Figure 2.2.3-79 where the mean internal and gas tempesature combinations are plotted for the flight in terms of the comfort requirements.

The mean internal temperatures rose approximately 3°F (1.7 K) between DOY's 224 and 233 as the beta angle increased to a maximum of 62 degrees and then remained relatively constant until DOY 243 even though the beta angle had decreased by this time to 32 degrees. The temperatures began to decrease

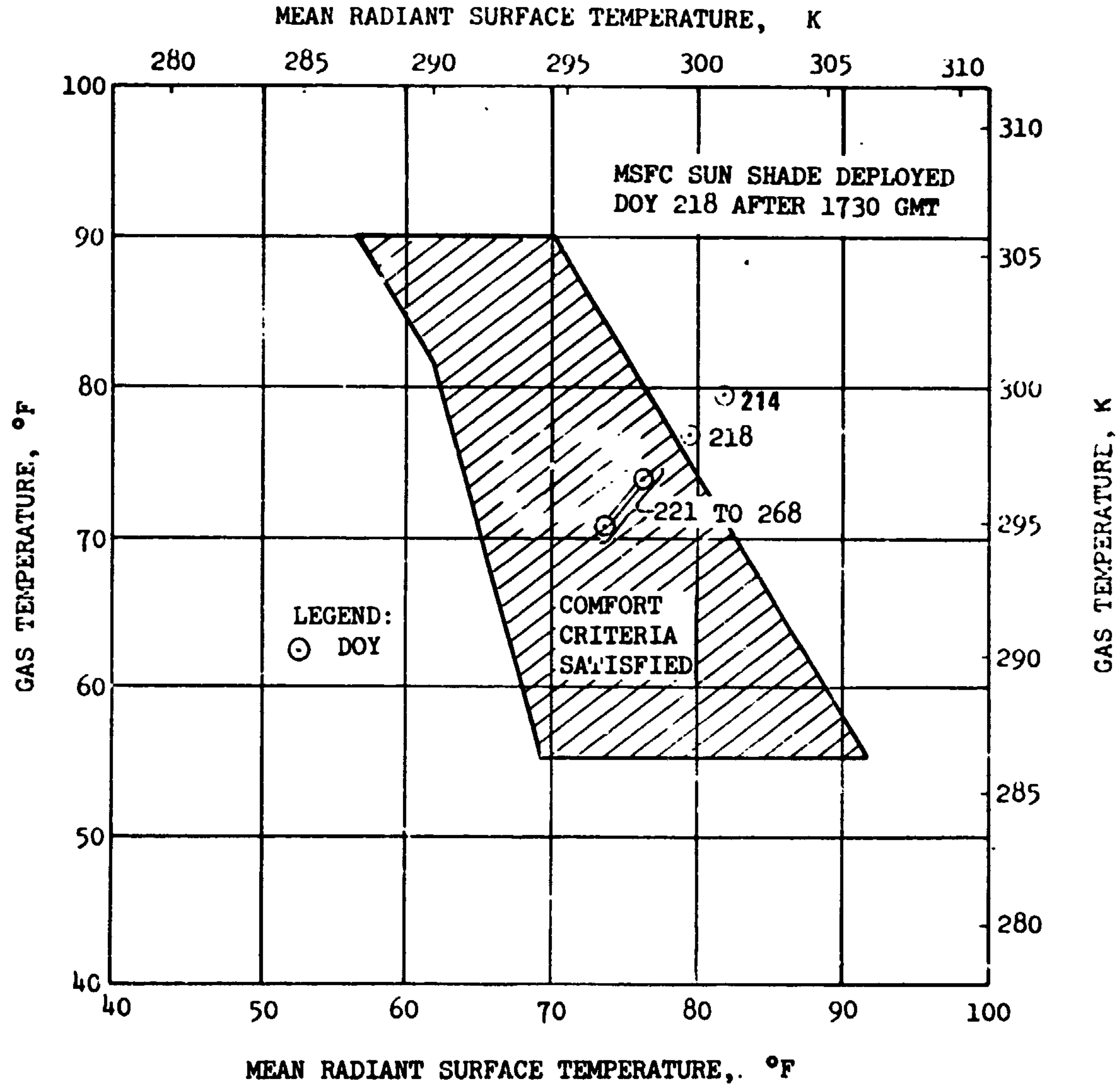


FIGURE 2.2.3-79 SL-3 Crew Comfort Conditions

on DOY 243 and the downward trend continued until DOY 250 when all of the aft cabin heat exchangers were turned off by the Thermal Control System logic unit. At this time the mean internal temperature was 72°F (296 K). The temperatures began to rise due to absence of active cooling. One heat exchanger came on later in DOY 250. The heat exchangers were cleaned on DOY 251 which resulted in additional active cooling. The second, third, and fourth heat exchangers came on automatically on DOY 256, 260, and 261, respectively. From DOY 251 through SL-3 OWS closeout on DOY 268, the mean internal temperatures varied between 71 and 75°F (295 and 297 K) and the gas temperatures between 70 and 74°F (294 and 296 K).

EREP maneuvers were performed 41 times during SL-3. Each maneuver caused local areas of the sidewall normally shadowed by the sunshade to be exposed briefly to direct solar heating. The maneuvers had no significant effect on the OWS mean internal temperatures. Local exterior surface temperatures were of concern, however, since the foam insulation bond line temperature was the same as the exterior surface temperature and an outgassing temperature limit of 200°F (367 K) maximum had been established for the foam.

During the EREP maneuvers, PP II was rolled into the sun for negative values of beta and PP IV for positive values.

Consequently, the appropriately located sensors (see Figure 2.2.3-75) provided significant responses to the maneuvers. Figure 2.2.3-80 depicts the responses for temperature sensors C7034 and C7052 on PP IV to an EREP maneuver on DOY 244 at a beta angle of +27 degrees. The maximum temperature was 165°F (347 K) and in the eight hours of solar inertial vehicle attitude following the maneuver, the temperature decayed to approximately their initial values. The transient temperature response of the exterior wall surface to two consecutive maneuvers on DOY 258 is presented in Figure 2.2.3-81. This figure shows peak temperatures of 145 and 170°F (336 and 350 K) for EREP's 31 and 32, respectively. The rate of temperature change from these sensors was used to determine the α_s and ε of the goldized tape which was discussed previously in the section on external coating degradation.

4. Second Storage Period (DOY 268 to 320)

As in the first storage period, the active TCS elements were deactivated, with thermal control provided by passive means only. There was one basic difference influencing heat transfer in the OWS, however, between the first and second storage periods. The MSFC sail was in place over the JSC Parasol providing additional shading with a resultant lowering of internal temperatures.

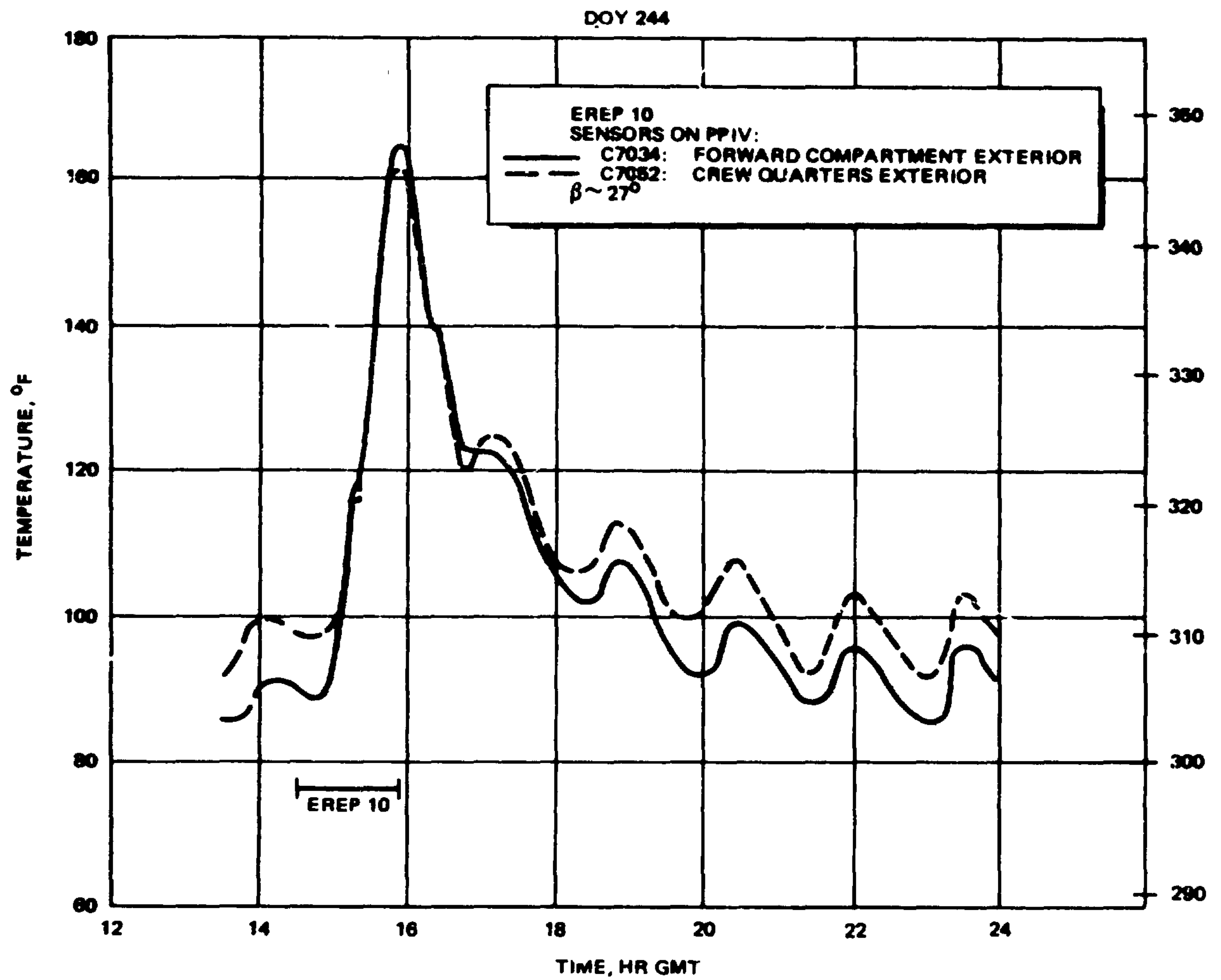


Figure 2.2.3-80. OWS Surface Temperatures for a Single EREP Maneuver

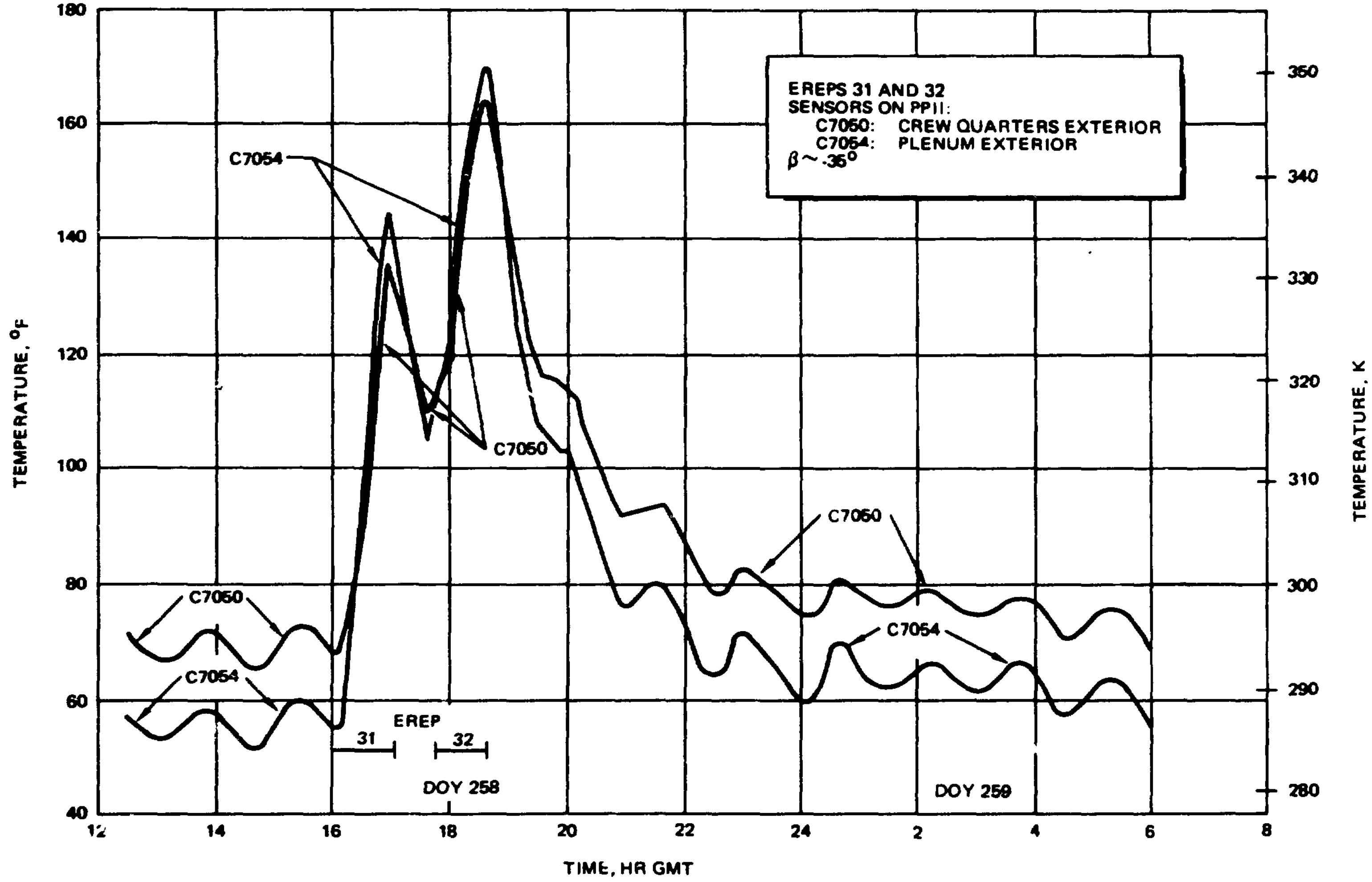


Figure 2.2.3-81. OWS Surface Temperatures for Back-to-Back EREP Maneuvers

The mean internal, rack-stowed food and film vault temperature histories are shown in Figure 2.2.3-82 for the storage period beginning on DOY 268. The initial cooling trend was due to the reduced internal heat generation from the preceding habitation period and to a lesser degree to the decreasing absolute value of beta angle (β). A warming trend began about five days before β reached zero and continued for approximately three days after the maximum positive β occurred. The temperatures then decreased until large values of negative β were approached. This was attributed to the asymmetrical external configuration. The remaining segment of the meteoroid shield, in combination with the main tunnel, shielded approximately a six feet wide segment of the tank wall near PP II. This shielding and the partial blockage of albedo by the SAS wing were most effective at moderate negative β's. From Figures 2.2.3-76 and 2.2.3-82, it can be inferred that the OWS external heat inputs increased as β increased positively from -50 degrees to +73.5 degrees and also as β increased negatively from -50 degrees to -73.5 degrees. For large negative values of β, the increased time in the sun offset the albedo blockage and the OWS warmed up due to the increased direct solar radiation. This latter point was illustrated by the temperature rise at the end of the storage period as β went from -50 to -65 degrees.

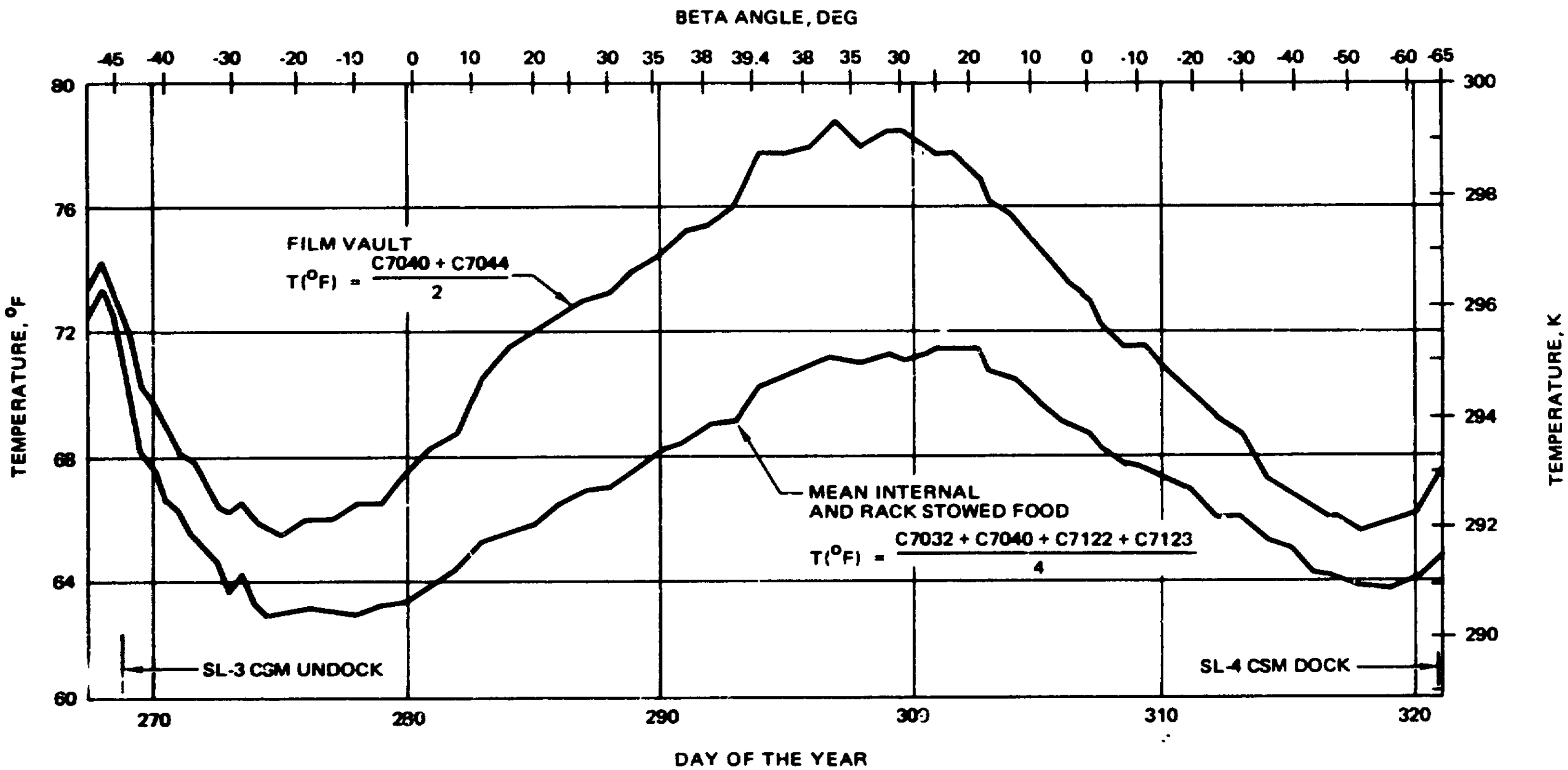


Figure 2.2.3-82. OWS Temperatures During the Second Storage Period, DOY 268-320

5. SL-4 Habitation Period (DOY 320 to 038)

The mean internal temperature extremes, orbit beta angle, and duty cycle for the cooling heat exchangers during SL-4 are presented in Figure 2.2.3-83. The temperatures were daily minimum and maximum values that resulted from the variation of the waste heat profile throughout each day. This temperature fluctuation averaged approximately 2°F (1.1 K) per day and was comparable to that observed for SL-2 and SL-3.

The activation period commenced on DOY 321. Internal temperatures rose rapidly from the storage temperature level of 64°F (291 K) to 76°F (295 K) two days later (DOY 323) when the four cooling heat exchangers were activated. The rapid rise resulted from the increasing orbital sunlight fraction as well as the increased internal heat loads. The temperatures continued to rise, peaking out at 80°F (300 K) as the vehicle orbit passed through the range of high beta angles. The temperatures then dropped to 73°F (296 K) on DOY 332. At this time the heat exchangers were turned off for approximately one day. They were turned back on briefly and then were allowed to remain off for approximately five days from DOY 335 to DOY 340 during which time the temperature rose to 77°F (298 K). The temperatures ranged between 70°F (294 K)

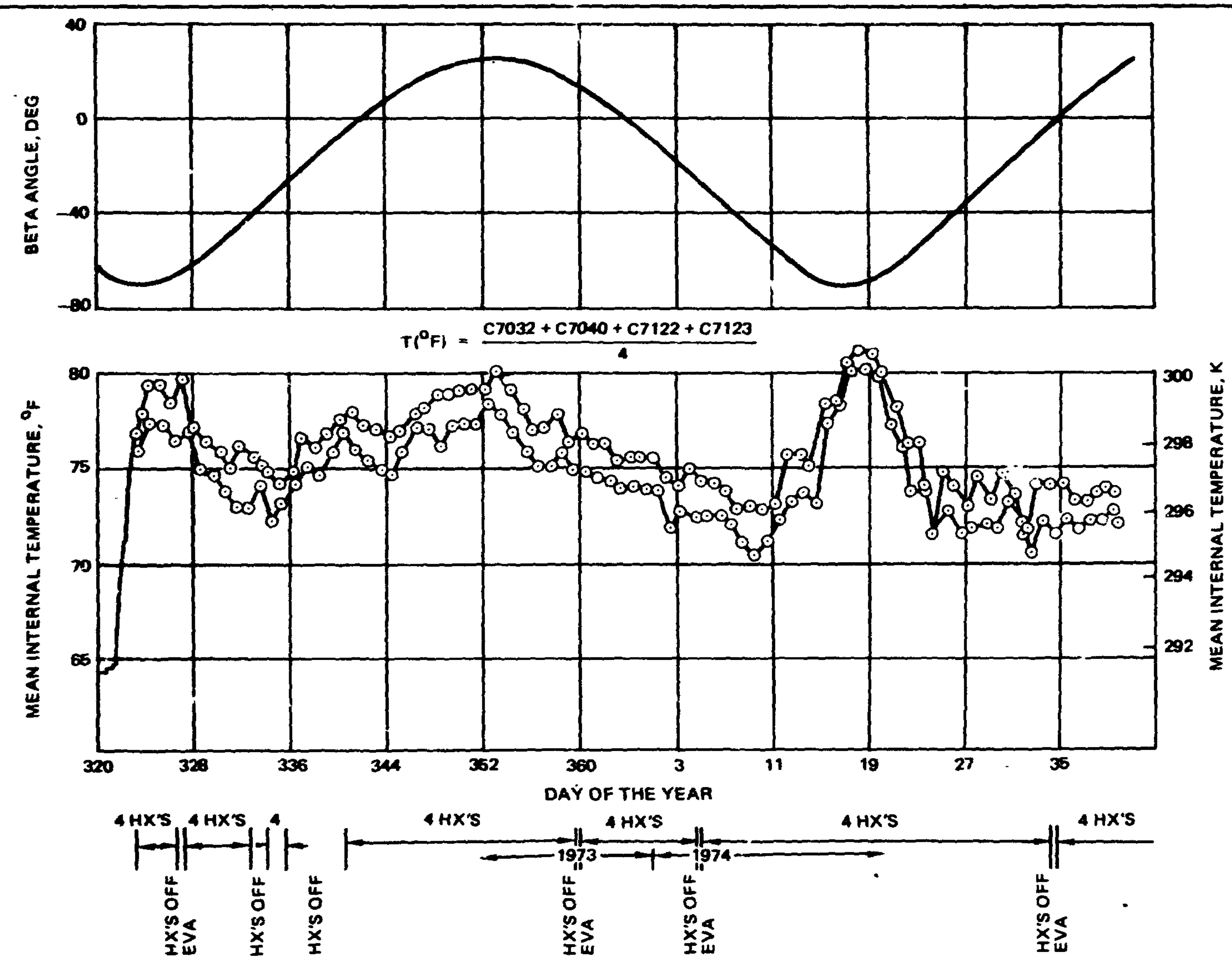


Figure 2.2.3-83. SL-4 Maximum and Minimum Mean Internal Temperatures

and 80°F (300 K) during the next 34 day period. On DOY 9 (1974) the temperatures started to rise rapidly as the vehicle orbit again passed through a range of high beta angles. The temperature peaked out at 81°F (300 K) on DOY 17. This was the highest temperature experienced during SL-4. The temperature dropped to approximately 72°F (295 K) on DOY 24 and remained between 72°F (295 K) and 75°F (297 K) for the remainder of the mission.

The crew comfort criteria were satisfied during most of SL-4 as shown in Figure 2.2.3-84. Crew quarters temperatures exceeded the comfort criteria during DOY's 324 through 327, 349 through 354 and 15 through 20. The high temperatures on DOY's 324 through 327 and 15 through 20 resulted from the high sunlight fraction orbits at the high beta angle conditions. The high temperature period from DOY 349 to 354 occurred when the beta angle was 26 degrees and the percentage time in the sun for the vehicle was slightly less than 63 percent. The increased temperature levels resulted in part from the heat rejection lost during the time period from DOY 335 to 340 when the heat exchangers were off. During this time the internal temperatures increased 3°F (1.7 K) from 74°F (296 K) to 77°F (285 K). The high temperatures during DOY's 349 to 354 also resulted from the extensive maneuvers performed by the vehicle during the Kohoutek Comet observations and back-to-back EREP passes.

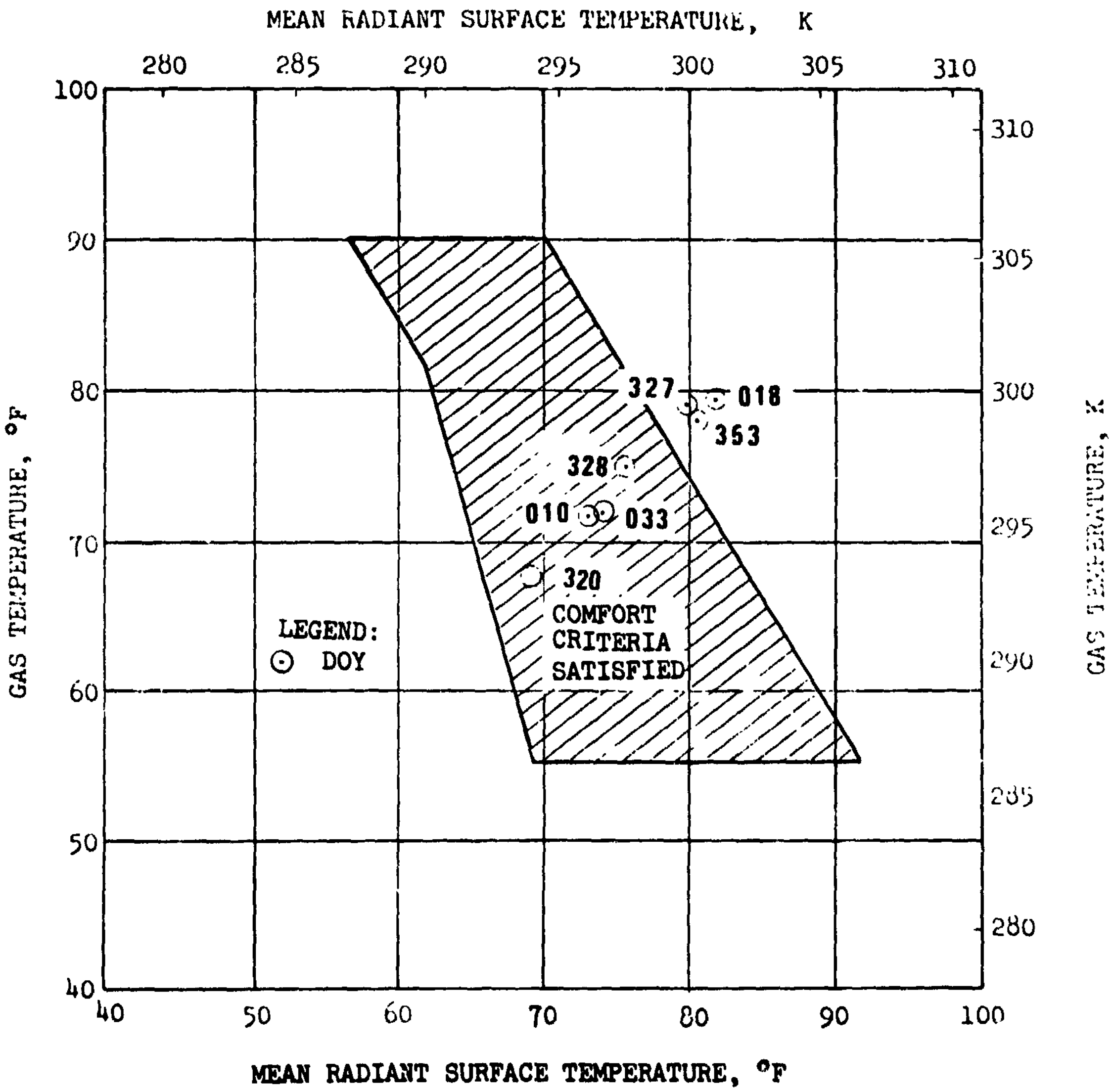


FIGURE 2.2.3-84 SL-4 Crew Comfort Conditions

The temperature control unit was designed to maintain the OWS gas temperature within ±4°F (±2.2 K) of the temperature selected by the astronauts. During SL-4 the temperature control unit maintained the thermal control system in the full cooling mode (four heat exchangers) except during two time periods (DOY 332 and DOY 335) when the heat exchangers were shut off by raising the setpoint temperature. In general the TCS setpoint temperature G7002 was maintained between 60°F (289 K) and 65°F (291 K) during the SL-4 except during the time period from DOY 320 to 341. During this time the setpoint was adjusted at various times to a range between 65°F (291 K) and 75°F (297 K).

The temperature levels during SL-4 habitation period were in general 1°F (0.5 K) to 5°F (2.8 K) warmer than those recorded during SL-3 after the MSFC sunshade was deployed. This was due to the higher beta angles, higher seasonal solar heat flux, and the greater frequency of vehicle maneuvers experienced by the SL-4 vehicle. For SL-4 beta angles exceeding 69.5 degrees (complete sunlight orbits) were encountered during two periods compared to one time period during SL-3 when the maximum beta angle was +62 degrees (77 percent sunlight). Since the MSFC sunshade provided incomplete coverage of the vehicle side-wall, the increased direct solar radiation resulted in higher tank wall temperatures. Also the average solar flux during SL-4 was estimated to be approximately 3-1/2 percent higher than SL-3, increasing further the solar radiation incident on the vehicle. The higher solar flux resulted from the reduced earth-sun distance during the SL-4 time period.

The vehicle maneuver frequency during SL-4 was greater than that of SL-3. Many maneuvers were performed during SL-4 for viewing the comet Kohoutek. Also the frequency of back-to-back EREP passes, and daily combinations of multiple maneuvers was far greater.

The longer duration of the back-to-back EREP maneuvers at higher beta angles resulted in local sidewall temperatures along PP II and PP IV which were within the revised insulation outgassing temperature limit of 300°F (422 K). Maximum sidewall temperatures could not be obtained from the temperature sensors since the temperatures during these maneuvers exceeded the 190°F (360 K) calibration limits. This is illustrated in Figure 2.2.3-85 where temperatures recorded during two consecutive EREP passes on DOY 14 at a beta angle of 67.5 degrees are presented. Analytical correlations of the data and predicted local wall temperatures at the hottest location are also presented. The predicted maximum insulation temperature was 290°F (417 K).

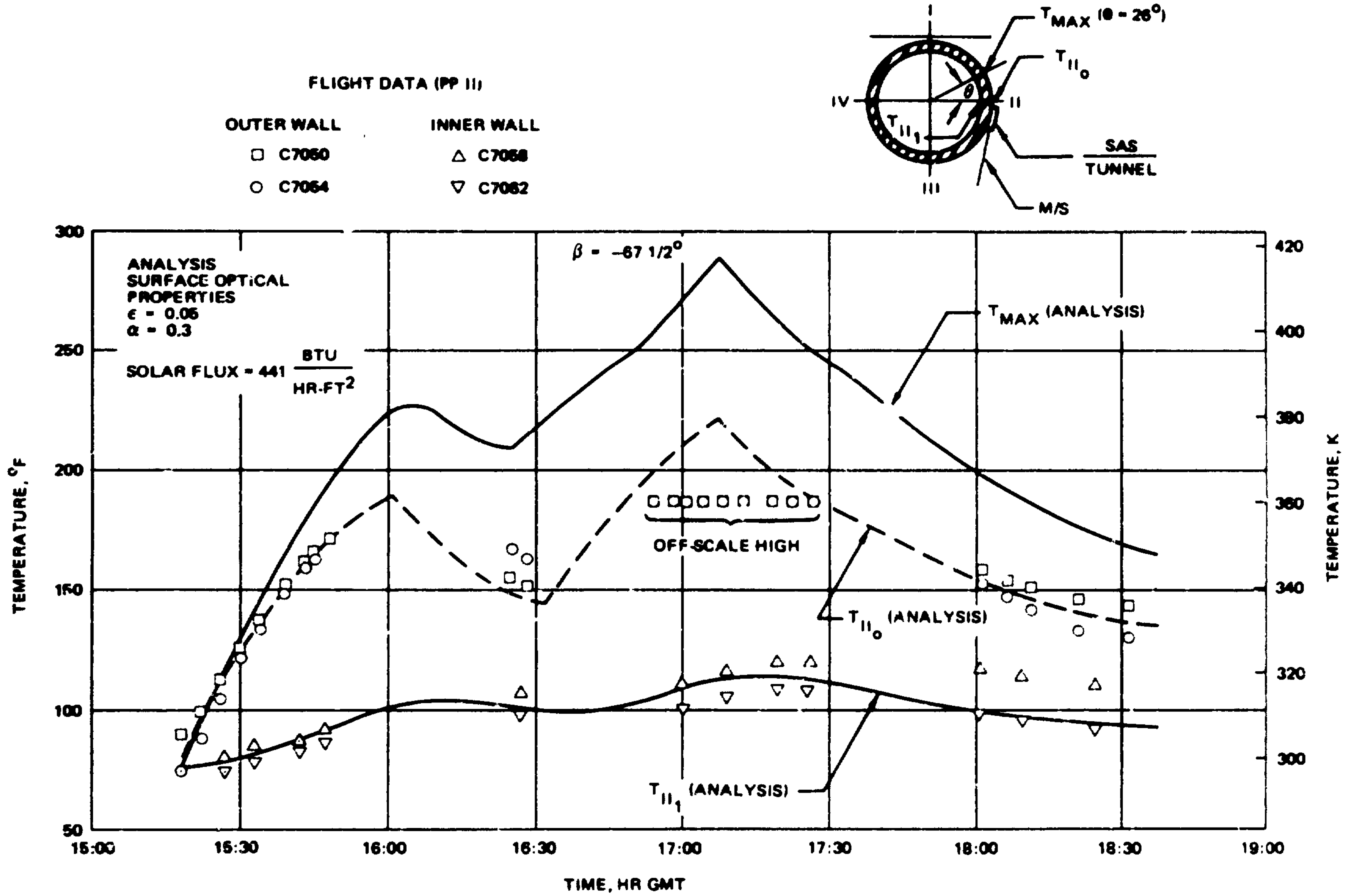


Figure 2.2.3-85. OWS Tank Wall Temperature Response During EREP 29 and 30 (DOY 014)

6. Post SL-4 Storage Period

A thermal analysis was performed to estimate the maximum OWS temperatures that could occur during post SL-4 storage of the vehicle in a gravity gradient orientation. The temperatures were required to assess the structural integrity of various internal components. The results of the analysis were as follows:

OWS Location	Maximum Temperature °F	Maximum Temperature K
External Wall	415	486
Internal Wall	305	425
Hottest Water Bottle	250	394
Hottest Heat Pipe	295	419
Mean Internal	240	389

The temperatures are steady state values for the OWS in a gravity gradient orientation. A beta angle of 73.5 degrees and a solar constant of 429 Btu/hr-ft^2 (1355 watts/m^2) were assumed. Sidewall gold external optical properties of α_s = 0.27 and ϵ = 0.05 were used. It was assumed that the MDA was pointed toward the earth and that the sunshade side of the vehicle was pointed in the antisolar direction continuously.

b. OWS Heat Loads

As previously indicated the use of the parasol and sail sunshades provided habitable environments, but the external heat loads were higher than those for the design conditions with the meteoroid shield. This shift from the design heat loads negated the need for the OWS convective heaters during habitation. To maintain temperatures within or near the comfort box, there was a continual reliance on the cooling delivered from the Airlock Module heat exchangers. In Table 2.2.3-15, the electrical waste heat removal capability of the sun-shaded OWS, ($\dot{Q}_c$) calculated from flight data is compared with the electrical waste heat load ($\dot{Q}_1$) for the two sunshade configurations flown.

The design requirements of 1300 watts and 850 watts for absolute values of beta angles less than 60 degrees and greater than 60 degrees, respectively, were based on anticipated OWS electrical waste heat loads two to three times higher than resulted from actual orbital operation. To evaluate the relative thermal performance of the sunshaded system then, $\dot{Q}_c$ should be compared with $\dot{Q}_1$ and the margin or deficit ($\dot{Q}_c$ - $\dot{Q}_1$) related to comfort box temperatures. This relationship can be approximated as follows: °F within the comfort box = 0.0284 ($\dot{Q}_c$ - $\dot{Q}_1$), where $\dot{Q}_c$ and $\dot{Q}_1$ are in watts.

TABLE 2.2.3-15

OWS ELECTRICAL WASTE HEAT REMOVAL CAPABILITY (1)

FLIGHT PERIOD	SUNSHADE CONFIGURATION	BETA ANGLE, β, DEG	HEAT LOAD $\dot{Q}_1$, WATTS	HEAT REMOVAL CAPABILITY (2) $\dot{Q}_c$, WATTS
SL-2, -3	JSC Parasol	-30 to 60	320 to 400	400
SL-2	JSC Parasol	70	380	200
SL-3	MSFC Sail over Parasol	-49 to 60	380 to 410	560 to 600
SL-4	MSFC Sail over Parasol	0 to -60	290 to 440	380 to 580
SL-4	MSFC Sail over Parasol	0 to +30	350 to 365	330 to 350
SL-4	MSFC Sail over Parasol	-71	265	120

(1) Design values with M/S: $|\beta| < 60$ deg, 1300 W; $|\beta| > 60$ deg, 850 W. 1000 Btu/hr (294 W) sensible metabolic heat load.

(2) Flight capability with 700 Btu/hr (205 W) sensible metabolic heat load

1. JSC Parasol

Power management was utilized early in the SL-2 mission to effect OWS cooldown after the JSC Parasol deployment on DOY 147 and to reduce the battery drain prior to SAS Wing 1 deployment. After cooldown and SAS Wing 1 deployment, the estimated heat removal capability was 400 W as shown in Table 2.2.3-15 for beta angles up to 60 degrees. Near the end of the SL-2 mission, β increased rapidly and the vehicle began complete sunlight orbits on DOY 172. Since the temperatures were increasing rapidly during this time, only an apparent $\dot{Q}_c$ can be calculated. This value was estimated to be 200 watts.

Early in the SL-3 mission for low negative values of β, the heat dissipation capability was estimated to be 400 W.

2. MSFC Sail Over JSC Parasol

After the MSFC sail was erected over the JSC Parasol, the additional shading decreased the external heating. This increased the heat removal capability by 100 to 200 W to the 500 to 600 W range for β = -49 to +60 degrees (Table 2.2.3-15) encountered during the remainder of SL-3.

During SL-4 the capability was reduced by the increased solar flux as the winter solstice was approached and by the increased frequency of maneuvers previously discussed. For β = 0 to -60 degrees, the reduction was 20 to 120 W.

For β = 0 to +30 degrees, the reduction was considerably larger at 170 to 250 W. Some of the difference between the $\dot{Q}_c$'s for negative and postive β's can be attributed to shading by the SAS and a piece of meteoroid shield remaining near PP II. For negative β, PP II was rolled toward the Earth and was partially shaded from albedo and Earth IR fluxes which resulted in higher heat removal capability. Although there was incomplete shading of the gold surfaces from direct solar impingement, which was verified by photographs, some capability remained in full sunlight as shown for β = 71 degrees. High positive β's were not encountered during SL-4.

C. Temperature Instrumentation

Program considerations dictated that telemetry instrumentation be limited. Consequently the number of temperature measurements required for the Thermal Control System performance evaluation was minimized. Recognizing this basis, the flight temperature instrumentation is discussed in terms of its usefulness in providing system performance and flight status. The locations of the transducers used to monitor structural temperatures are shown in Figure 2.2.3-86.

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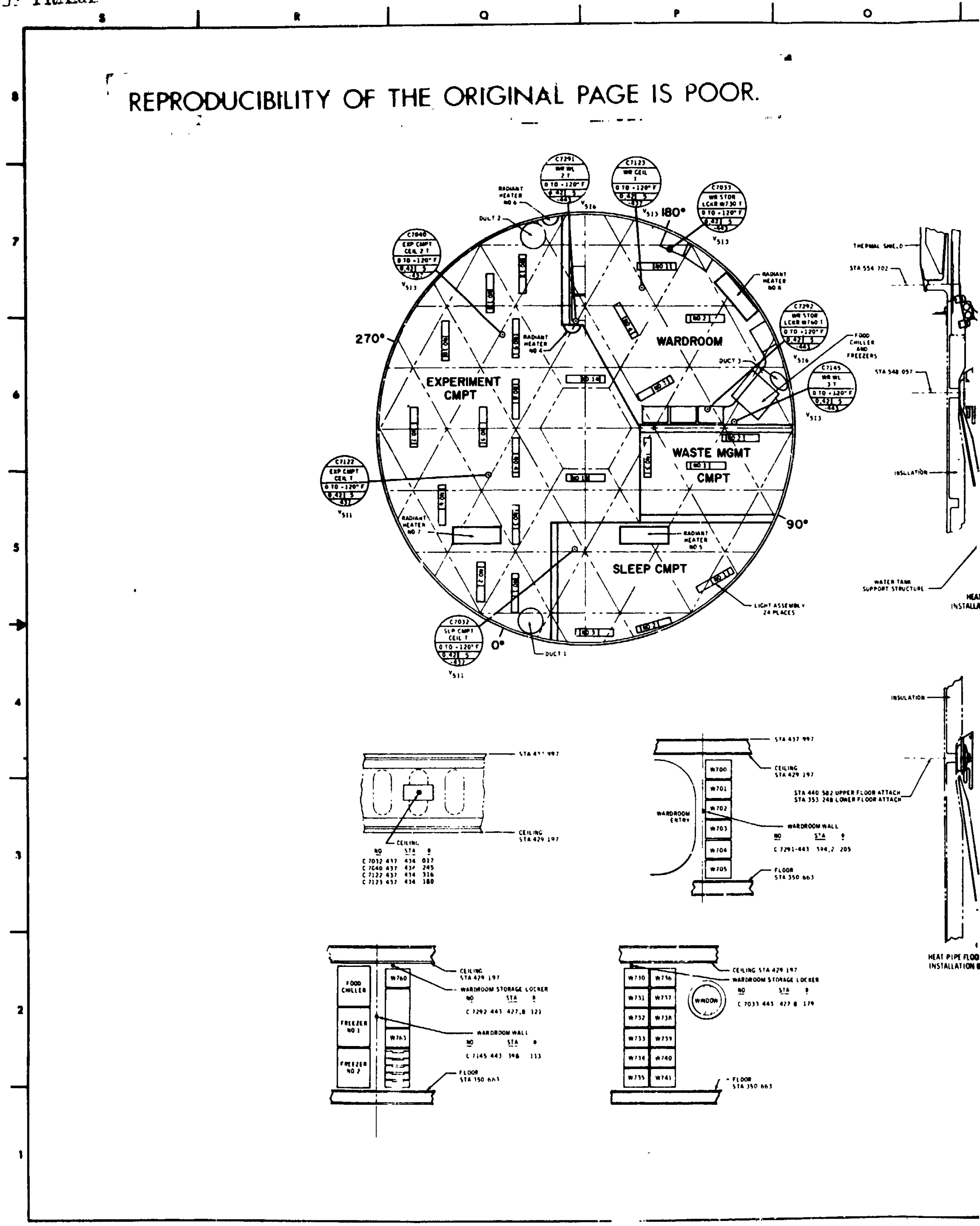

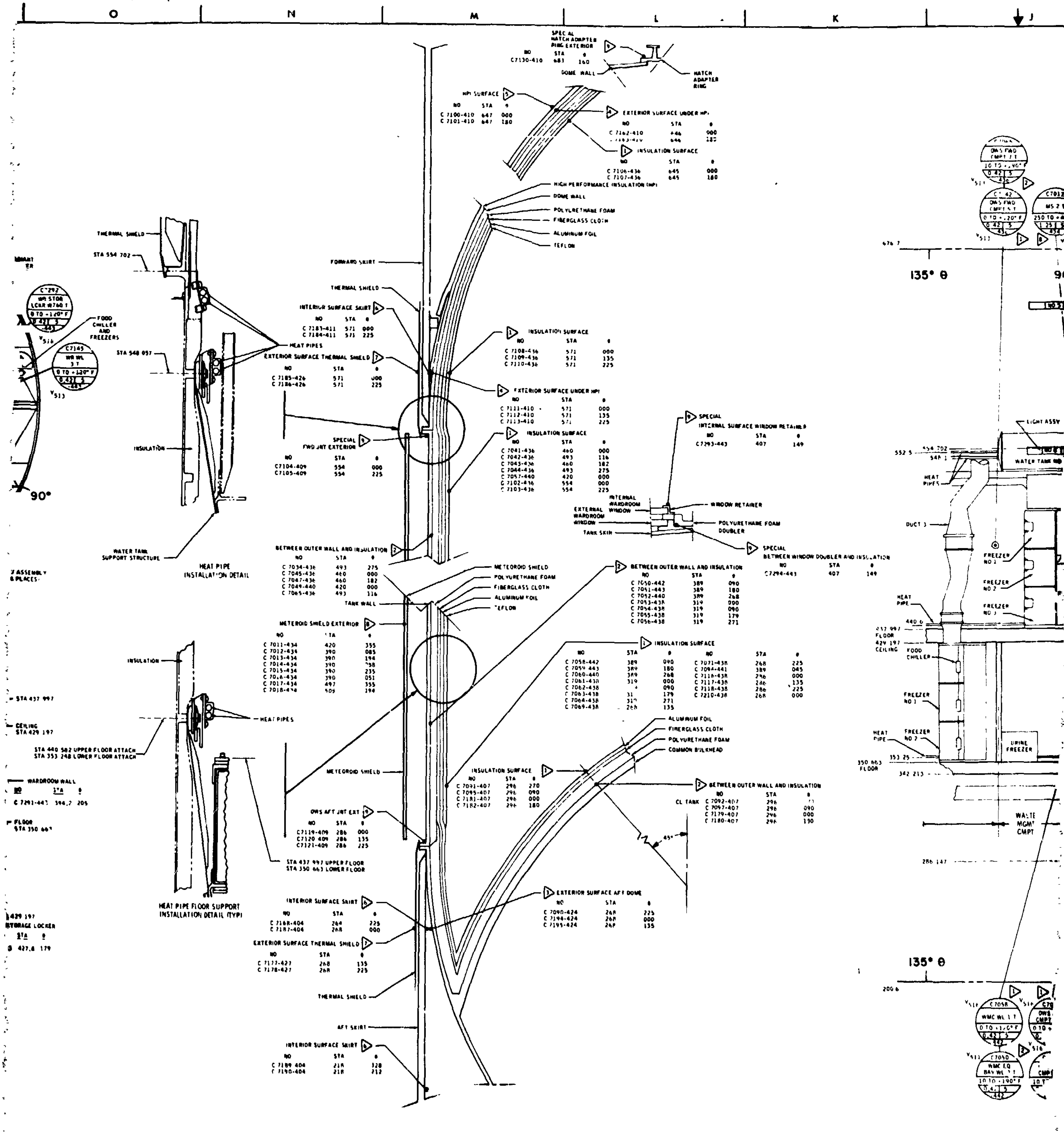

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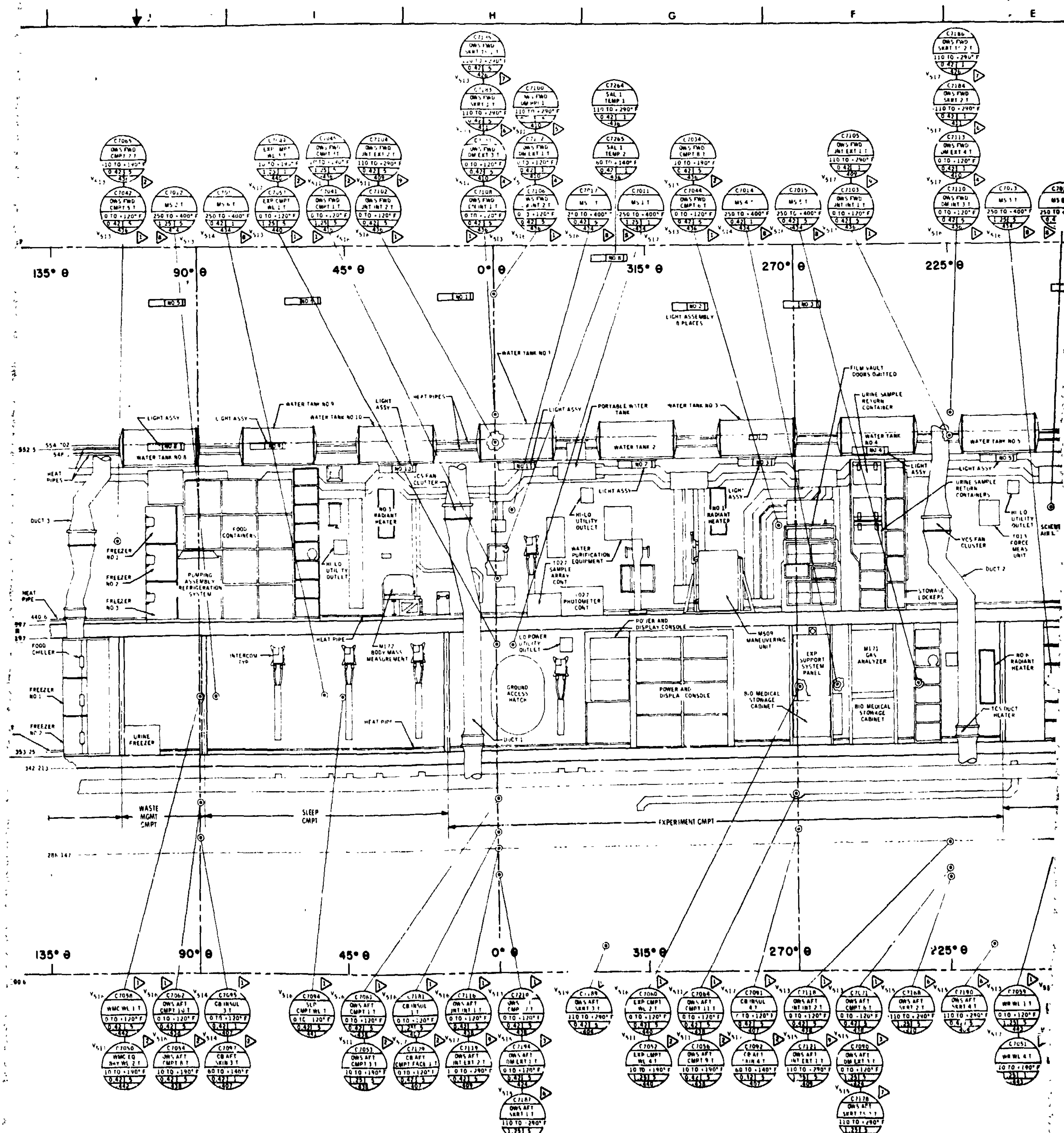
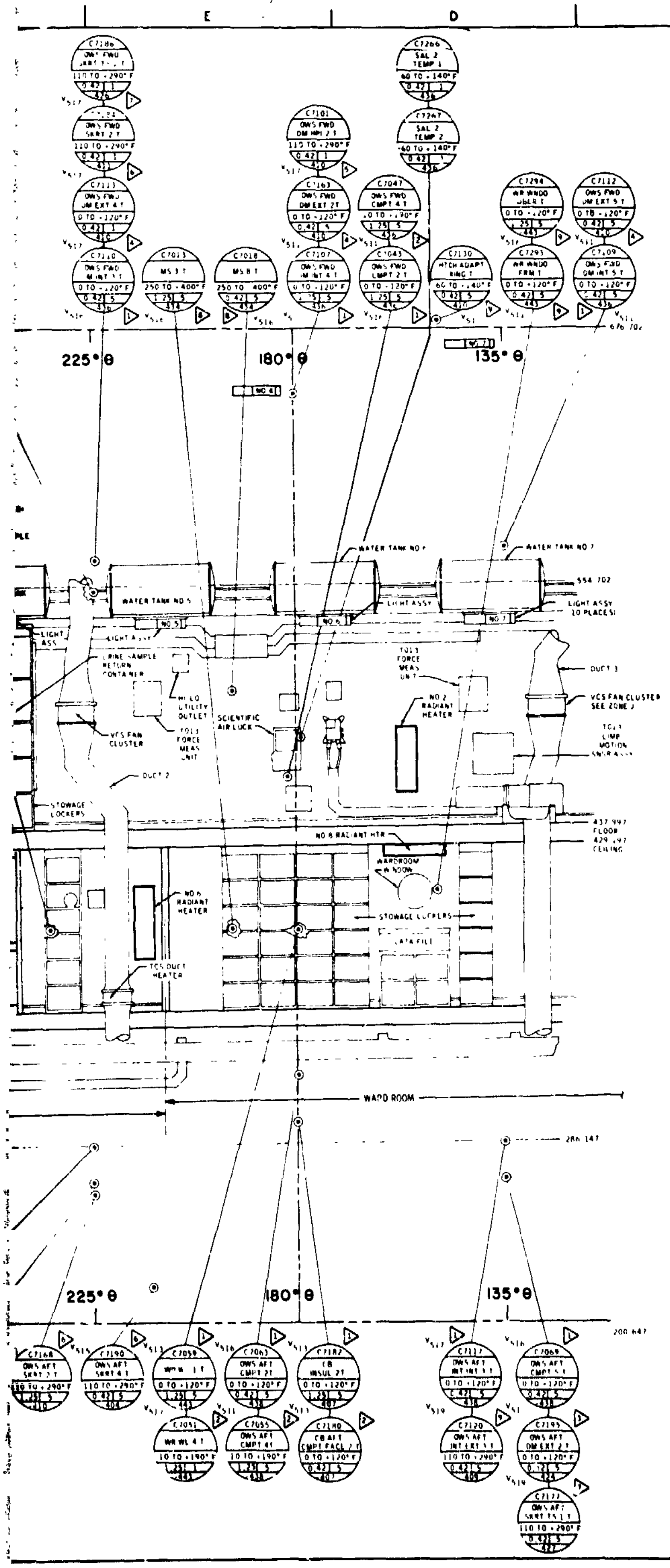


H
G
F
E
135° θ
90° θ
45° θ
0° θ
315° θ
270° θ
225° θ
WATER TANK NO 1
WATER TANK NO 9
WATER TANK NO 10
WATER TANK NO 8
WATER TANK 2
WATER TANK NO 3
WATER TANK NO 4
WATER TANK NO 5
PORTABLE WATER TANK
LIGHT ASSY
LIGHT ASSEMBLY 8 PLACES
HEAT PIPES
FILM VAULT DOORS OMITTED
URINE SAMPLE RETURN CONTAINER
URINE SAMPLE RETURN CONTAINERS
VCS FAN CLUSTER
NO 3 RADIANT HEATER
NO 1 RADIANT HEATER
NO 4 RADIANT HEATER
HI-LO UTILITY OUTLET
WATER PURIFICATION EQUIPMENT
T027 SAMPLE ARRAY CONT
T027 PHOTOMETER CONT
FOOD CONTAINERS
FREEZER NO 2
FREEZER NO 3
FREEZER NO 1
PUMPING ASSEMBLY REFRIGERATION SYSTEM
DUCT 1
DUCT 2
DUCT 3
STOWAGE LOCKERS
T013 FORCE MEAS UNIT
FOOD CHILLER
URINE FREEZER
INTERCOM TYP
M172 BODY MASS MEASUREMENT
LO POWER UTILITY OUTLET
GROUND ACCESS HATCH
POWER AND DISPLAY CONSOLE
M509 MANEUVERING UNIT
EXP SUPPORT SYSTEM PANEL
M171 GAS ANALYZER
BIO MEDICAL STOWAGE CABINET
TCS DUCT HEATER
HEAT PIPE
WASTE MGMT CMPT
SLEEP CMPT
EXPERIMENT CMPT



9. MEASUREMENTS ARE LOCATED AS SHOWN ON THE DETAIL DRAWINGS
8. MEASUREMENTS ARE LOCATED ON THE EXTERIOR SURFACE OF THE METEOROID SHIELD
7. MEASUREMENTS ARE LOCATED ON THE EXTERIOR SURFACE OF THE FORWARD OR AFT THERMAL SHIELD
6. MEASUREMENTS ARE LOCATED ON THE INTERIOR SURFACE OF THE FORWARD OR AFT SKIRT
5. MEASUREMENTS ARE LOCATED ON THE EXTERIOR SURFACE OF THE DOME HPI
4. MEASUREMENTS ARE LOCATED BETWEEN THE HPI AND THE EXTERIOR SURFACE OF THE FORWARD DOME
3. MEASUREMENTS ARE LOCATED ON THE EXTERIOR SURFACE OF THE AFT DOME
2. MEASUREMENTS ARE LOCATED BETWEEN THE INSIDE OF THE TANK WALL AND THE INSULATION
1. MEASUREMENTS ARE LOCATED ON THE INTERIOR SURFACE OF THE INSULATION

NOTES

OWS THRM

SIGNATURES	DATE	NATIONAL AERONAUTICS & SPACE ADMINISTRATION MANNED SPACECRAFT CENTER · HOUSTON TEXAS
DR		
DSGN		
QC		OWS STRUCTURAL TEMP
ENGR		
ENGR		
APP		
APP		SWS — DWG NO 5.6
STC		104 5 4 34 — PAGE 5 6 — SHEET OF

Figure 2.2.3-86

The temperature sensors installed in the OWS to evaluate the performance of the Thermal Control System produced the expected data quality. They were, for the most part, properly located for their intended purpose and were in sufficient number to provide the necessary data. For purposes of determining temperature levels, the instrumentation accuracy (Table 2.2.3-16) was adequate. For purposes of determining heat flux utilizing the temperature difference (ΔT) of adjacent sensors, the instrumentation yielded errors as large as 60 percent using the ΔT across the sidewall foam. Temperature sensors used as a set to calculate ΔT's were placed on the same multiplexer where possible in order to reduce the relative error for determining the temperature difference. This cut the error range across the foam insulation by approximatley one-half, for example. For purposes of determining habitation area heating rates, it would have been desirable to have had a number of ΔT sensors and/or heat flux gauges to supplement the temperature sensors. This was particularly true for the habitation area sidewall and forward dome where ΔT's across the foam insulation were generally less than 10°F.

The loss of the OWS meteoroid shield during launch caused high temperature conditions in orbit beyond that which could resonably be designed into the instrumentation. This caused items such as the film vault, ambient food storage lockers and internal insulation to exceed their design temperatures. Therefore, it became necessary

TABLE 2.2.3-16

TEMPERATURE INSTRUMENTATION ERROR SUMMARY

Sensor Location	Total System Error (%)	Temperature Range °F, (K)	Temperature Error °F, (K)
Meteoroid Shield	± 2.16	-250 to 400 (117 to 479)	± 14.04 (± 7.8)
Habitation Area Sidewall (Foam Outboard)	± 2.32	- 10 to 190 (250 to 361)	± 4.64 (± 2.57)
Habitation Area Sidewall (Foam Inboard)	± 2.35	0 to 120 (256 to 322)	± 2.82 (± 1.57)
HPI Exterior	± 2.17	-110 to 290 (184 to 416)	± 8.68 (± 4.82)
Forward Dome Exterior	± 2.35	0 to 120 (256 to 322)	± 2.82 (± 1.57)
Forward Dome Interior	± 2.35	0 to 120 (256 to 322)	± 2.82 (± 1.57)
Common Bulkhead Interior Surface	± 2.35	0 to 120 (256 to 322)	± 2.82 (± 1.57)
Common Bulkhead/ Foam Interface	± 2.35	0 to 120 (256 to 322)	± 2.32 (± 1.57)

to approximate the temperature of these items with the temperature instrumentation which remained on-scale. It is apparent from data requirements during this period as well as after sunshade deployment (when measurements were back on-scale) that it would have been highly desirable to provide temperature instrumentation directly on the temperature critical film vault and the ambient food racks rather than estimating these temperatures from wall and ceiling measurements. Installation of the Wardroom window sensors C7293 and C7294 in closer proximity to the window heater element would have allowed these sensors to be more useful in determining the operation of the window heater. Such an installation, however, was precluded by viewing considerations. There was no temperature instrumentation for heat pipe performance evaluation. It is highly unlikely, however, that a preflight designed temperature instrumentation system would have given the desired performance data, since the loss of the meteoroid shield significantly altered the anticipated temperature distributions. The structural temperature gradients around the tank sidewall were quite low and there was no vehicle hot side and cold side as such.

2.2.3.5 Conclusions and Recommendations

A. Active Subsystem

All components of the Active Thermal Control Subsystem performed as expected and met design requirements.

1/ The Thermal Control Assembly (TCA) performed as expected; however, for future design a tighter control band is recommended since the astronauts used the thermostat to obtain manual control.

2/ The radiant heaters were not required to maintain storage temperatures because the use of the sunshades rather than the meteoroid shield passive system resulted in sidewall external heat inputs higher than design levels. The radiant heaters were turned on in accordance with the original flight plan on DOY 134 at 2014 GMT, but were turned off approximately three hours later when it was realized that the meteoroid shield was lost and the OWS was warming up rapidly due to exposure of the goldized Kapton to direct solar heating. Electrical power consumption from telemetry data showed that the heaters were performing during this period.

3/ The duct heaters were not required or used during habitation for the same reason that the radiant heaters were not used during storage.

4/ Based on the duct flow measurements, the duct fan performance met the 15 to 100 fpm (4.6 to 30.5 m/min) velocity requirement in the crew quarters without using the portable fans. One portable fan was used periodically to provide additional flow to the crew member using the ergometer.

B. Passive Subsystem

1/ The meteoroid shield was an integral part of the Passive Thermal Control Subsystem. Its loss caused habitation area walls, food and film temperatures, and all internal environment temperatures to exceed their maximum temperature limits. Acceptable habitation area environments were restored with the astronaut deployed parasol and the MSFC sail.

2/ The forward dome high performance insulation (HPI), the internal foam insulation and the remaining control coatings (excluding the lost meteoroid shield coatings) functioned within acceptable levels. The calculated conductance for the HPI was less than the 0.02 Btu/hr-ft^2-°F (0.035 W/m^2·K) allowable design value. The solar absorptance (α_s) degradation of the S-13G white paint on the aft skirt followed the test data of Steube (Figure 2.2.3-62). Although the α_s was higher than used for design thermal analyses, no component problems due to the high α_s were encountered.

The OWS goldized Kapton α_s and ε were determined at various times in the mission to assess degradation. A substantial portion of the degradation occurred after orbital insertion. The α_s degraded from a mean value of 0.18 after retro-firing to 0.3 late in the mission. The ε increased from 0.04 to 0.06 in the same period.

C. Overall System Performance

Shading from direct solar inputs was required to produce acceptable E/TCS performance. After deployment of the JSC Parasol, crew comfort was maintained in the OWS until β exceeded 60 degrees on DOY 170. The estimated electrical waste heat removal capability ($\dot{Q}_c$)* for $\beta <$ 60 degrees was 400 watts versus an electrical waste heat load ($\dot{Q}_1$) of 320 to 400 watts. During SL-3 with the MSFC sail deployed $\dot{Q}_c$ increased and crew comfort was maintained with a $\dot{Q}_1$ of 500 to 600 watts. During SL-4,performance degraded due to increased solar flux, increased frequency and duration of maneuvers, and higher absolute values of β. For these conditions, $\dot{Q}_c$ varied with β and ranged from 120 to 580 watts for $\dot{Q}_1$'s of 265 to 440 watts. This produced temperatures outside the comfort box during portions of the SL-4 mission. For future design it is recommended that the thermal control system be independent of deployable systems or a redundant system capable of operation with any type of failure be utilized.

D. Instrumentation

The temperature sensors installed in the OWS to provide TCS performance data produced the expected data quality. The data were used extensively to assess real time problems through the

*Q_c is the electrical waste heat which can be added to the OWS internal environment without producing temperatures exceeding the comfort box criteria.

entire flight period. The intermittent operation of low level Multiplexer B had no significant impact on the TCS instrumentation since data were available in sufficient quantities for all required thermal evaluations.

For future applications, heat flux meters and/or differential temperature measurements should be used for determining heat flow. Using differences in absolute temperature measurements does not give the desired accuracy.

2.2.3.6 **Development History**

The Environmental/Thermal Control System (E/TCS) as defined by MSFC in December, 1967, for the Saturn I (Wet) Workshop provided control by fan circulated gas in eight evenly-spaced ducts. These ducts were formed by a series of thermal curtains and rails around the periphery of the habitation area. This system gave gas temperatures in the range of approximately 55°F (286 K) to 105°F (314 K). The design was based on a gravity gradient vehicle orientation at a 28.5 degree orbital inclination. A meteoroid shield with a black painted external surface (α_s/ϵ = 0.9/0.9) was assumed with a moderate resistance to heat transfer (no gold) between the meteoroid shield internal surface and the tank wall. The minimum temperature for safe astronaut entry after tank passivation was defined as -150°F (172 K) and no active heaters were provided for warmup.

During the latter part of 1967 and in 1968, joint studies undertaken by MSFC and MDAC delineated advantages of controlling heat leaks in the tank sidewall, tank joint regions, the forward dome and the plenum region including the common bulkhead. These studies led to the gold tape on the tank external surface, the forward dome high performance insulation system, the thermal shields on the external joint areas, and foam insulation in the plenum region. (The addition of foam insulation was not implemented, however, until after the change from a wet to a dry Workshop.)

Hydrogen retention in the tank foam insulation was of concern from the standpoint of habitation environment flammability as well as insulation thermal performance. Tests completed in 1968 showed that no problems existed since the passivation sequence allowed sufficient time for insulation outgassing.

By mid-1969, when MDAC began design analysis on the E/TCS at the request of MSFC, the system concept and design had undergone many changes. Crew comfort criteria had been defined as follows:

Atmospheric Temperature	65 to 75°F (291 to 297 K)
Mean Radiant Wall Temperature	65 to 75°F (291 to 297 K)
Humidity	0.018 Specific and 95% Relative
Touch Temperature	55 to 105°F (286 to 314 K)
Atmospheric Velocity	15 to 100 ft/min (4.6 to 30.5 m/min)

The temperatures were to be controlled automatically or manually utilizing cooling delivered from the Airlock Module ECS and 750 watts of heater power (500 watt capability in each of two ducts with fan clusters). All major surfaces were to be between 60°F (289 K) and 80°F (300 K), but localized surfaces accessible to the crew could be as cold as 55°F (286 K) or as hot as 105°F (314 K).

The atmospheric distribution system was to be designed to minimize CO_2 and humidity gradients. Radiant heaters providing a maximum of 1000 watts were to be utilized for warmup to provide a 0°F (255 K) mean internal temperature at pressurization initiation and a 40°F (278 K) minimum internal temperature by the time tank seal and lighting installation was completed.

The wet Workshop requirements were reassessed with the change to a dry Workshop configuration in September, 1969, with the mission being flown in a solar inertial attitude at an orbital inclination of 35 degrees. Before completion of this assessment, a change to a 50 degree orbital inclination was made in early 1970. This meant the E/TCS design had to consider the increased heat loads associated with orbits in 100 percent sunlight whereas the maximum had been 73 percent sunlight. Performance requirements were changed to include an expanded comfort box (which was the final specification comfort box). A minimum waste heat (housekeeping) load of 250 watts and a maximum metabolic (sensible) load of 1000 Btu/hr (293 watts) were defined. Maximum heater power usage for cold conditions was redefined as 825 and 1170 watts for nominal and two sigma conditions, respectively. The minimum electrical waste heat removal was specified as between 600 and 1350 watts, being dependent on beta angle as well as consideration of nominal and two sigma conditions.

The major E/TCS design changes resulting from the preceding requirements were the addition of white paint on the solar-facing side of the meteoroid shield, the change from a two-duct gas flow system to a three-duct gas flow system, the addition of 500 watts of manually controlled heater power in the third duct, and foam insulation added in the plenum region to alleviate potential condensation problems and minimize the heat leak.

In 1971, heat pipes were installed in the Workshop to alleviate potential condensation problems in the regions near the floor and ceiling supports, the wall behind the water bottles, the balsa wood forward joint and the back of the storage freezer in the forward compartment. Also, in this period the Airlock Module cooling delivered to the Workshop was redefined with a resultant 50 watt decrease in the specified minimum electrical waste removal requirement for the E/TCS and an increase in the housekeeping load to 400 watts.

The Airlock Module cooling was again redefined early in 1972 and the minimum housekeeping waste lead load was increased from 400 watts to 525 watts which became the final design value. Based on these changes, the white paint pattern on the meteoroid shield external surface was finalized in February, 1972. No significant E/TCS design changes were made between this time and SL-1 launch.

APPROVAL

MSFC SKYLAB ORBITAL WORKSHOP

FINAL TECHNICAL REPORT

Orbital Workshop Project

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, in its entirety, has been determined to be unclassified.

This document has also been reviewed and approved for technical accuracy.

William K. Simmons, Jr.
Manager, Orbital Workshop Project

Rein Ise
Manager, Skylab Program