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SPACELAB

PAYLOAD ACCOMMODATION

HANDBOOK

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SPACELAB

PAYLOAD ACCOMMODATION HANDBOOK

MAY 1976

P R E L I M I N A R Y
I S S U E

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FOREWORD

With the advent of the Space Shuttle System, operations in earth-orbital space will become less complex and costly. This system will make launching of payloads into earth orbit virtually a routine event. Most of the physical strain aspects of transportation to and from space will be reduced in the Space Shuttle so that scientists and engineers will be able to participate directly in their research and experimentation activities in orbit.

Such a capability presents a new opportunity to expand research in space and to further the growth and development of science and applications. Key to this venture is the Spacelab.

In a significant step towards internationalizing future manned space programs, Europe has agreed to design and build Spacelab with European funds to joint U.S. and European requirements.

Spacelab is envisioned as a highly versatile general-purpose orbiting laboratory to be used for manned and automated space activities in the fields of science and applications. As a major element of NASA's Space Shuttle System, it offers the international community of users a low-cost, effective means of conducting scientific, applications and technology experiments in near earth orbit.

The contributions by the VFW-Fokker ERNO team, who prepared parts of this preliminary issue of the Spacelab Payload Accommodation Handbook under contract to ESA, are greatly acknowledged.

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1. INTRODUCTION

1.1 Purpose of Document

The purpose of this document is to describe the main characteristics of the Spacelab system and to provide sufficient information on Spacelab capabilities to enable individual experimenters or payload planning groups to determine how their payload equipment can be accommodated by Spacelab. Major Spacelab/experiment interfaces, Spacelab payload support systems and requirements the experiments have to comply with are described to allow experiment design, development and integration up to a level where a group of individual experiments are integrated into a complete Spacelab payload using Spacelab racks/floors and pallet segments (Level III/IV integration). Integration of a complete Spacelab payload with Spacelab subsystems, primary module structure etc. (Level II integration), integration of Spacelab with the Orbiter (Level I integration) and basic operational aspects are briefly outlined.

The procedural aspects of user involvement in Spacelab utilization will be covered in the Spacelab User's Guide. The Space Shuttle System is briefly described in the Spacelab Payload Accommodation Handbook but more complete information is available in the Space Shuttle System Payload Accommodation (JSC-07700, Vol. XIV).

This preliminary edition of the Spacelab Payload Accommodation Handbook reflects the current Spacelab baseline design and is for information only. Potential changes to the baseline design are noted in the text whenever appropriate. The Spacelab Payload Accommodation Handbook will be placed under formal change control at a later stage.

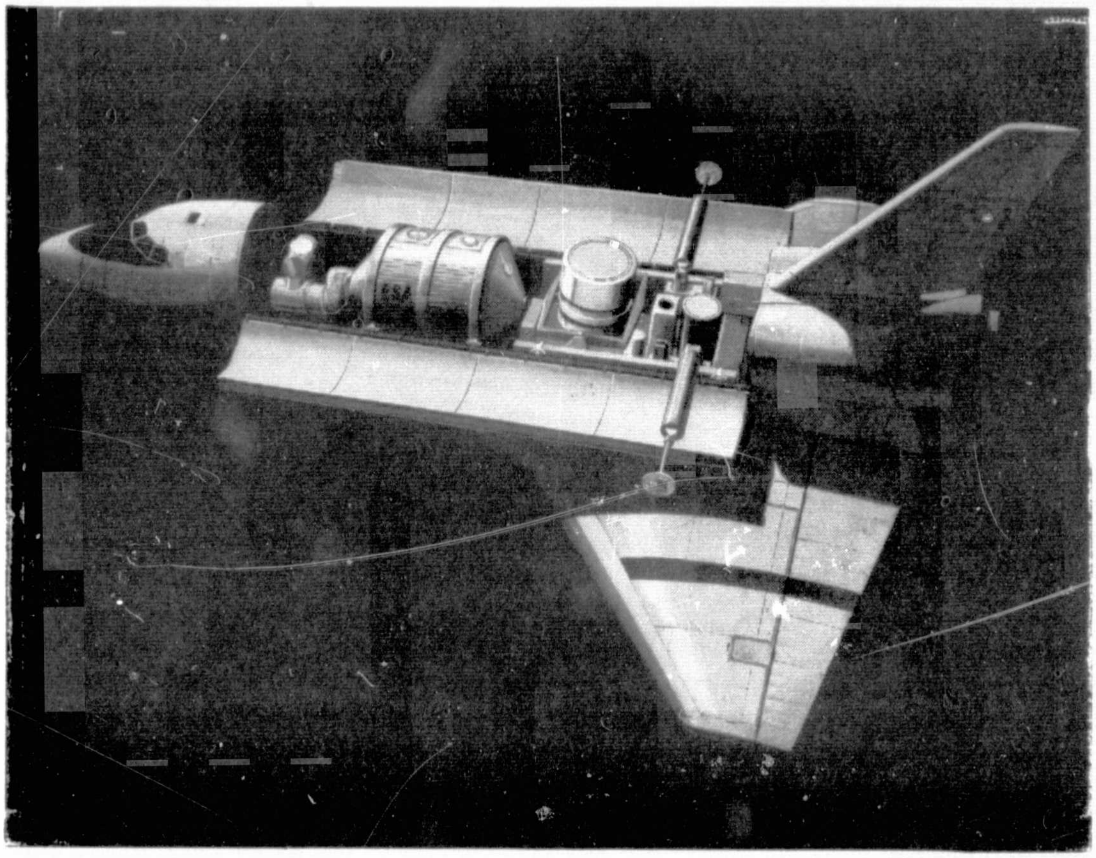


Figure 1-1 Conceptual View of Spacelab in Orbit

1.2 General Spacelab System Description

Spacelab consists of two basic elements - a pressurized module and an unpressurized pallet - which can be used separately or in combination.

The three typical flight configurations of Spacelab are:

- module only
- module plus pallet
- pallet only

The module provides a controlled pressurized environment for the users and their equipment, and supplies basic services such as power, thermal control, and data management, together with certain basic support equipment such as standard racks, airlock etc. which may be used as required. The pallet is an unpressurized platform to which instruments such as telescopes and antennas may be mounted which require direct exposure to space. The pallet provides basic services, such as power distribution, thermal control and data acquisition and commands.

Spacelab is carried to and from orbit by the Space Shuttle. It remains attached to the Orbiter of the Space Shuttle throughout the flight. Figure 1 - 1 shows a module plus pallet configuration of Spacelab in the Orbiter during its orbital stay. After landing, Spacelab is removed from the Orbiter; Figure 1 - 2 shows a possible concept of this operation.

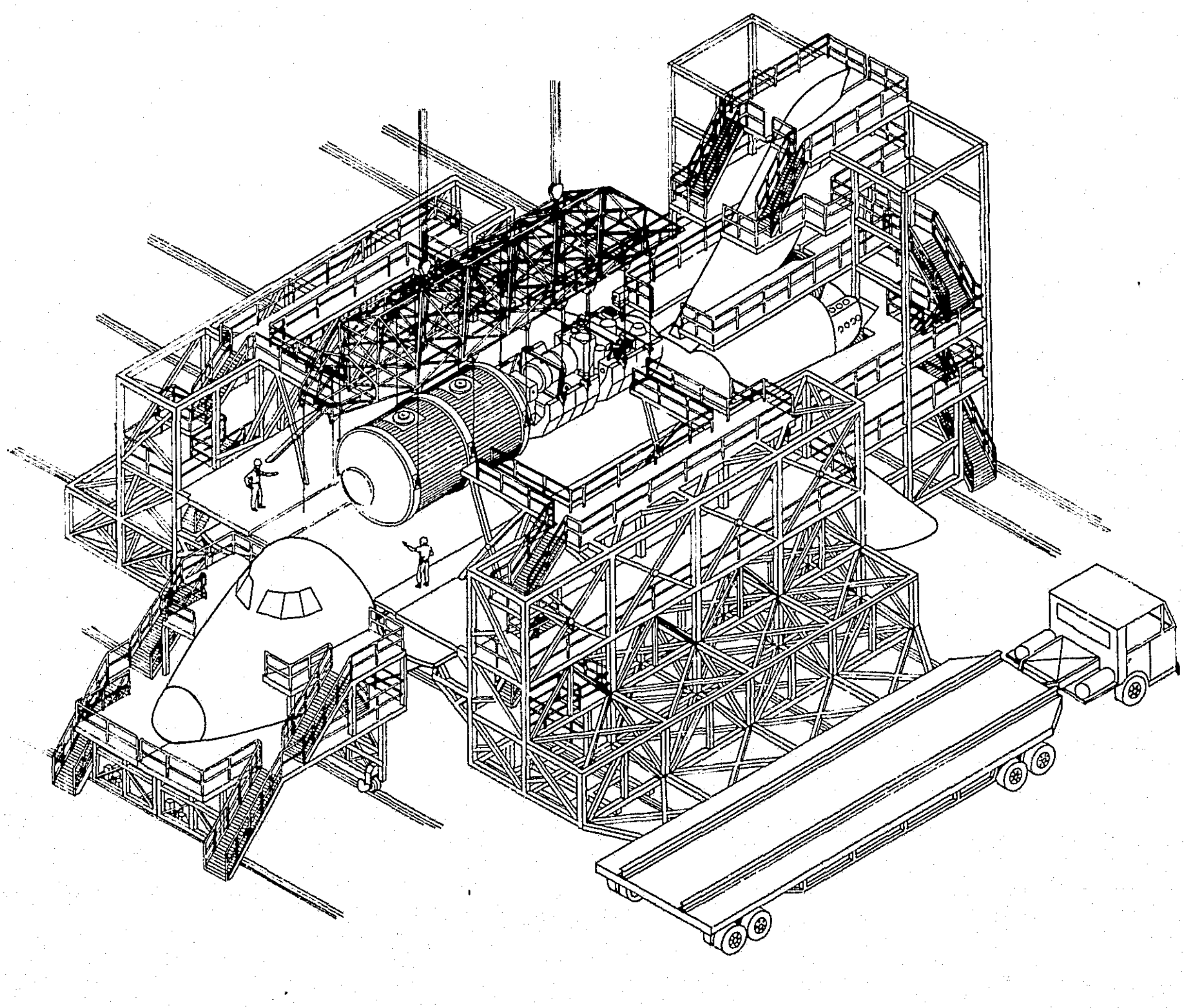


Figure 1-2 Possible Concept of Spacelab Removal From Orbiter

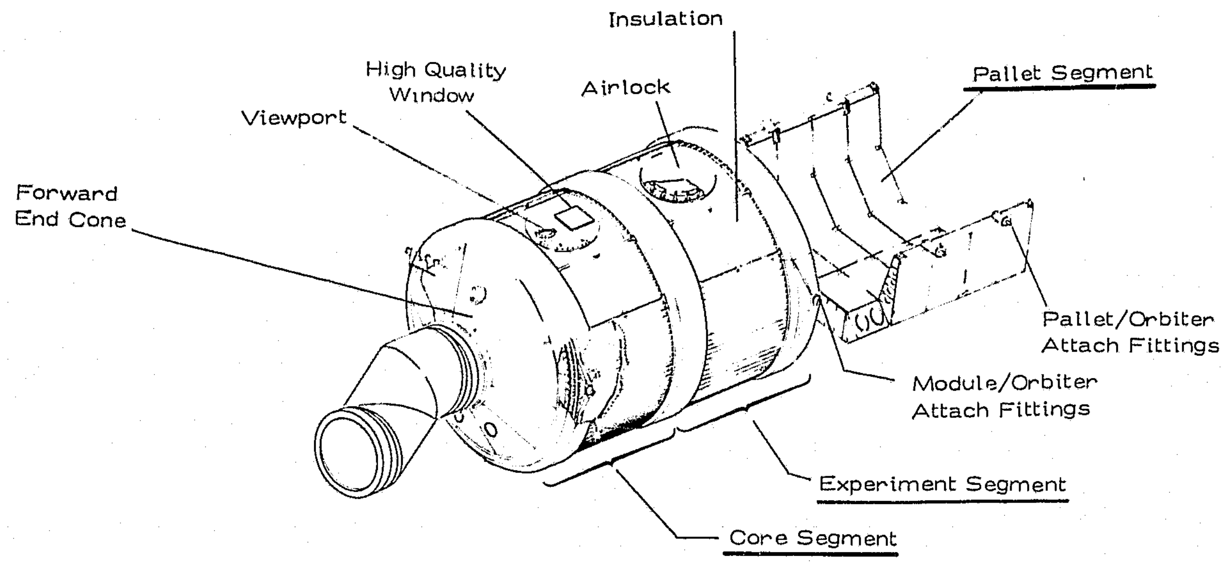


Figure 1-3: Spacelab External Configuration

Major external design features of Spacelab in a typical module plus pallet configuration are shown in Figure 1 - 3. The presented configuration consists of a two-segment module and one pallet segment. In general, the module consists of either a single cylindrical segment (core segment) or two segments (core plus experiment segment). The pallet only configuration may contain up to five pallet segments to accommodate experiments which require up to 15 m mounting length. The module can also be combined with pallet segments. Up to three pallet segments can be accommodated with a short module (core segment only) and up to two pallet segments can be combined with a long module (core and experiment segment). The module diameter is slightly over 4 meters and each cylindrical segment is approximately 2.7 meters long. The pallet segments are approximately 3 meters long and 4 meters wide.

The module itself is formed of a cylindrical pressure shell and cone-shaped end closures (end cones). It is covered with high-performance insulation. The module is structurally attached to the Orbiter by attach fittings located on the main ring frames of the module cylindrical segments. Signal, power and other utility lines between the module and the Orbiter are routed from the forward end cone. The forward located module segment (core segment) contains subsystem equipment and crew work space, but also leaves about 60 percent of the rack volume for experiment installation. The experiment segment is dedicated entirely to experiment installation and operations.

The center of gravity of the Orbiter with the integrated Spacelab must lie within certain limits which result from constraints during reentry and landing. For this reason the Spacelab module cannot be located at the very forward end of the Orbiter cargo bay. A tunnel is provided for crew and equipment transfer between the Orbiter and the Spacelab module. In addition a tunnel adapter/airlock combination is attached to the Orbiter forward bulkhead. Extra-vehicular activity (EVA) can be performed through the tunnel adapter/airlock. The design of this unit is such that access to Spacelab from the Orbiter is not interrupted during EVA.

The top of each module segment contains provisions for mounting either a high quality window/viewport assembly or an airlock. However, for operational reasons the airlock might only be flown in the experiment segment. A second viewport is located in the aft end cone to give an unobstructed view of the pallet. The aft end cone also provides for installation of an airlock and utility feedthough panels.

The U-shaped pallet segments are covered with aluminum honeycomb panels. They are integral parts of the pallet structure, but can also be used for mounting of light weight payload equipment. A series of hardpoints attached to the main structure of a pallet segment is provided for mounting of heavy payload equipment. The pallet segments are mounted to the Orbiter with a set of attach fittings. Up to three pallet segments may be structurally linked together to form a pallet train and attached to the Orbiter by a single set of attach fittings. Up to five pallets may be flown on a single mission.

The module and pallet are interconnected by a utility bridge which carries power-, signal- and other utility lines for subsystem and payload equipment on the pallet.

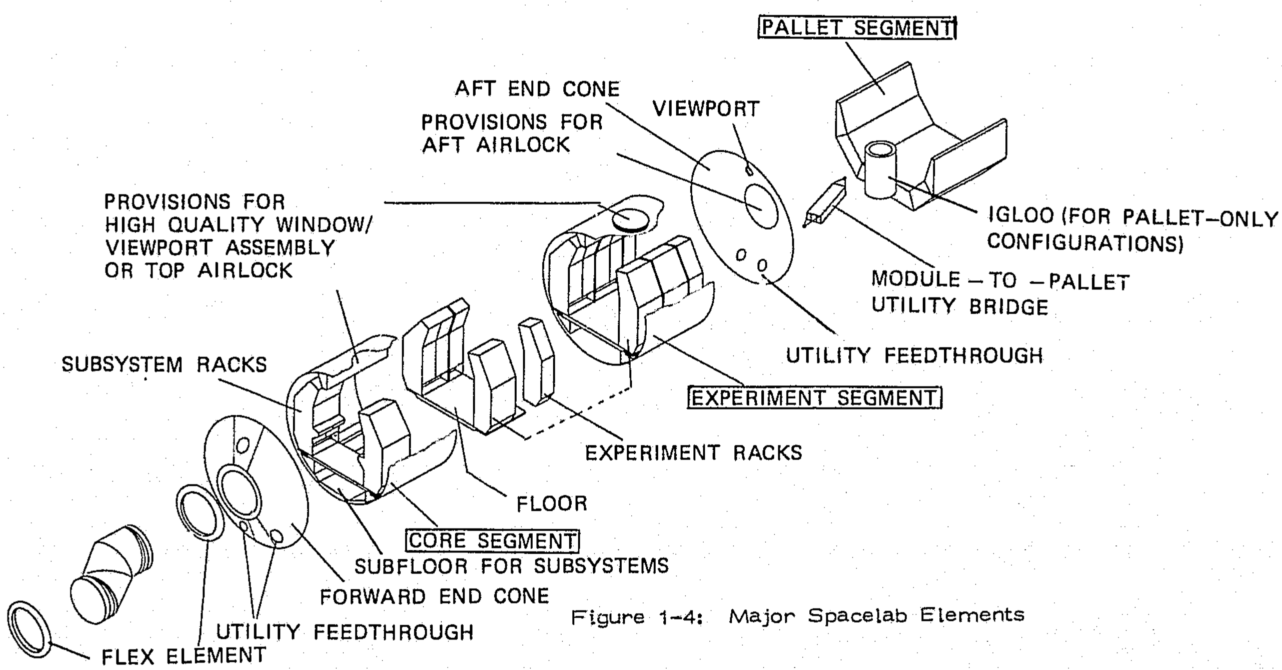


Figure 1-4: Major Spacelab Elements

Figure 1 - 4 shows Spacelab in an exploded view. The interior design is modular and provides flexibility to the user. Racks are arranged in single and double rack assemblies for mounting of equipment. The floor is segmented. The most forward floor segment in the core segment provides support for the sub-

system double rack assembly on each side. A second floor segment provides for support of rack assemblies for experiments.

The core segment can accommodate one single and one double rack assembly for experiments while the experiment segment can accommodate one single and two double rack assemblies for experiments,on each side. The sequence of single and double racks must be as indicated in Figure 1- 4 .The rack assemblies are independently attached to the floor and overhead structure so that as many as necessary may be installed for a given mission. If some racks are not required, other special experiment equipment may be attached in their place.

The subsystem racks are also detachable but will normally remain installed in the core segment between flights. In operational use, the experiment racks and floors will normally be pre-integrated and checked out as a complete assembly. This assembly will then be rolled into the module shell. The necessary interface connections will then be made with the primary structure and the subsystems in the core segment.

In module only and module plus pallet configurations, Spacelab subsystems are mounted in the subsystem racks and on the subfloor as shown in Figure 1- 4 . In these configurations the module can accommodate crew members for operation of subsystems and experiments. For pallet-only configurations an "Igloo", a pressurized cylinder attached to a pallet, is provided for installation of certain subsystem hardware which is normally mounted within the core segment. In pallet-only configurations, operation of subsystems and experiments will be performed from the Orbiter´s aft flight deck or from the ground.

A prime consideration in designing Spacelab was the provision of as many services as possible for the users within the given constraints. This has led to a modular design of subsystems. A certain part of the subsystem equipment may be selected by the users in order to satisfy the specific need for a flight in an optimal manner. This freely selected subsystem equipment is defined as "mission dependent" equipment.

The Spacelab flight hardware is devided into the following subsystems: Structure, environment control, electrical power and distribution, command and data management, and common payload support equipment.

The environment control subsystem (ECS) comprises elements for environmental control, life support, and passive and active thermal control. Oxygen/nitrogen atmosphere at sea level pressure is provided in the module by this subsystem. Crew habitability support such as food, drink, sleep, hygiene, and waste management facilities is provided by the Orbiter.

The electrical power and distribution subsystem (EPDS) conditions the basic electric power derived from the Orbiter´s fuel cells and distributes it to Spacelab subsystems and Spacelab payloads.

The command and data management subsystem (CDMS) provides support functions, such as data acquisition command, formatting, display and recording. The CDMS includes three identical computers: one dedicated to Spacelab payloads, one dedicated to subsystems and one back-up computer for either of the two dedicated computers. The CDMS subsystem is largely independent from the Orbiter. Communication with ground facilities, either directly to a Spaceflight Tracking and Data Network (STDN) Station or via the Tracking and Data Relay Satellite System (TDRSS), is provided through the Orbiter's communication system.

Common payload support equipment (CPSE) consists of the high quality window/viewport assembly, airlocks, and film storage containers.

The Spacelab program provides also software for operation of Spacelab on orbit and check-out of Spacelab on the ground. Furthermore the program includes mechanical and electrical ground support equipment for integration and checkout of Spacelab.

Coarse pointing of Spacelab payloads is provided by the Orbiter. A Spacelab-supplied instrument pointing subsystem (IPS) permits high precision pointing of Spacelab payloads.

In Figure 1- 5 the typical operation cycles of Spacelab are schematically presented. Pre-integrated equipment of the user(s) is integrated into Spacelab which is subsequently installed in the Orbiter. In the launch configuration the Space Shuttle consists of the Orbiter, a large External Tank which provides propellant to the Orbiter during launch and two Solid Rocket Boosters.

The Solid Rocket Boosters are jettisoned after burn-out and retrieved . The External Tank is jettisoned in the final ascent phase. The nominal flight duration of the Orbiter is seven days. However, the Orbiter/Spacelab is being designed so as not to preclude extended missions of up to thirty days duration. After launch the doors of the Orbiter cargo bay will be opened in order to expose Spacelab to space. Subsequent to completion of check-out operations Spacelab will be activated and operated. Before re-entry and landing the Spacelab systems will be de-activated and the doors of the Orbiter cargo bay will be closed. After landing, Spacelab and the Orbiter will be refurbished as required and prepared for the next flight in separate ground operation cycles.

Editorial note: The possibility of supporting payloads with limited power and cooling during ascent and descent is presently under investigation.

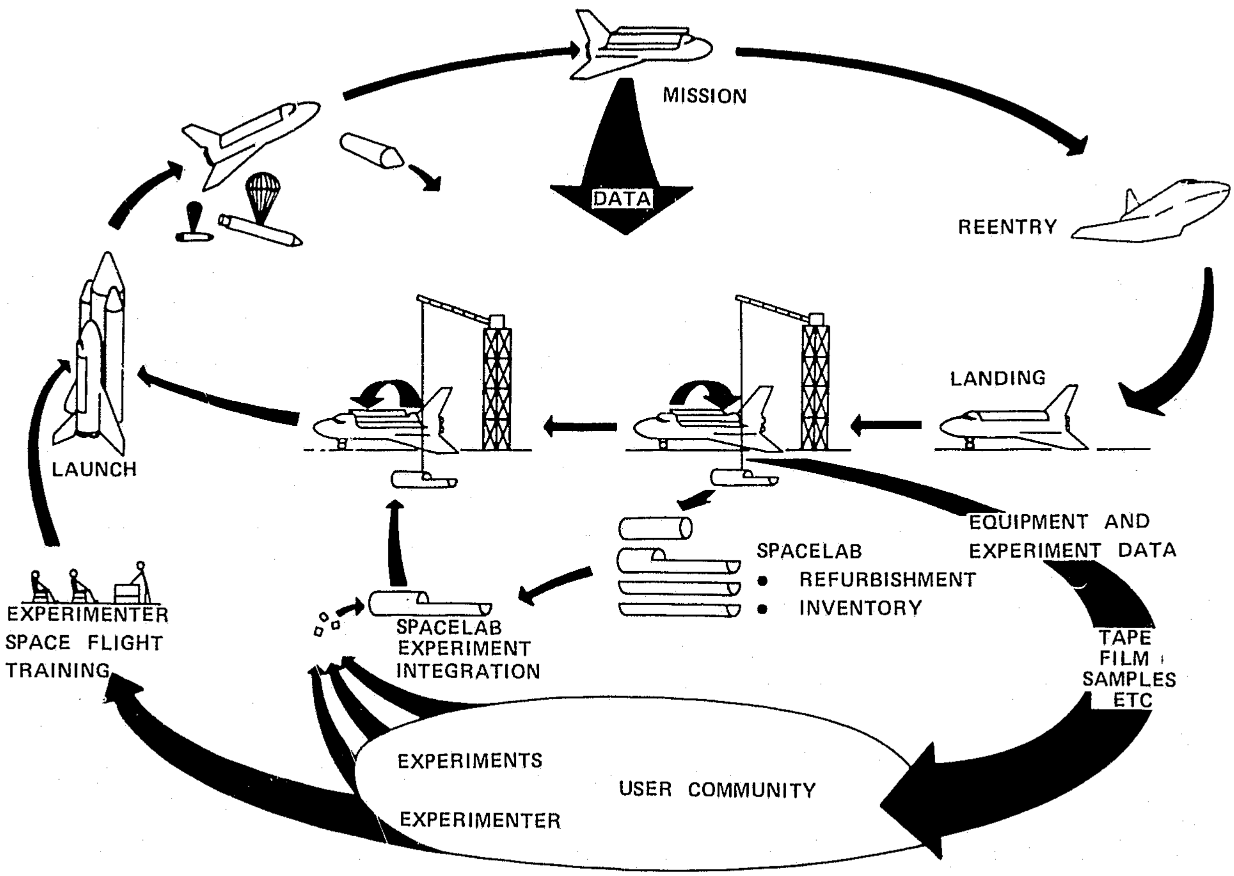


Figure 1 – 5: Shuttle-Spacelab Operational Profile

2 ORBITER/SPACELAB PERFORMANCE AND CONSTRAINTS

In this section information on Orbiter performance and constraints important to the user(s) of Spacelab is presented in summary form. The source of information on this subject is the NASA document "Space Shuttle System Payload Accommodation", JSC 07700, Vol. XIV, Revision D, Nov 26, 1975, including changes dated up to Feb 2, 76. This document will be referenced as "Vol.XIV, Rev. D".

2.1 Coordinate Systems

A large number of coordinate systems is defined in the Shuttle System and Spacelab Programme. In this section those coordinate systems which are of importance for interfaces between the Orbiter and Spacelab are identified.

2.1.1 Orbiter Structural Body Coordinate System

This coordinate system is shown in Figure 2 - 1 and defined as follows:

Type: Orbiter fixed

Origin: In Orbiter plane of symmetry, 400 inches (10.16 m) below the center line of the dynamic envelope of the Orbiter cargo bay. The most forward Yo - Zo plane of the dynamic envelope of the Orbiter cargo bay is at Xo = 582 inches (14.783 m). This Xo coordinate is referred to as "station 582".

Orientation: The Xo - axis is in the Orbiter plane of symmetry, parallel to and 400 inches (10.16 m) below the center line of the dynamic envelope of the Orbiter cargo bay. Positive sense is from the nose of the Orbiter towards the tail.

The Zo - axis is in the Orbiter plane of symmetry, perpendicular to the Xo - axis. Positive sense is upward in landing attitude.

The Yo - axis completes a right handed, orthogonal coordinate system.

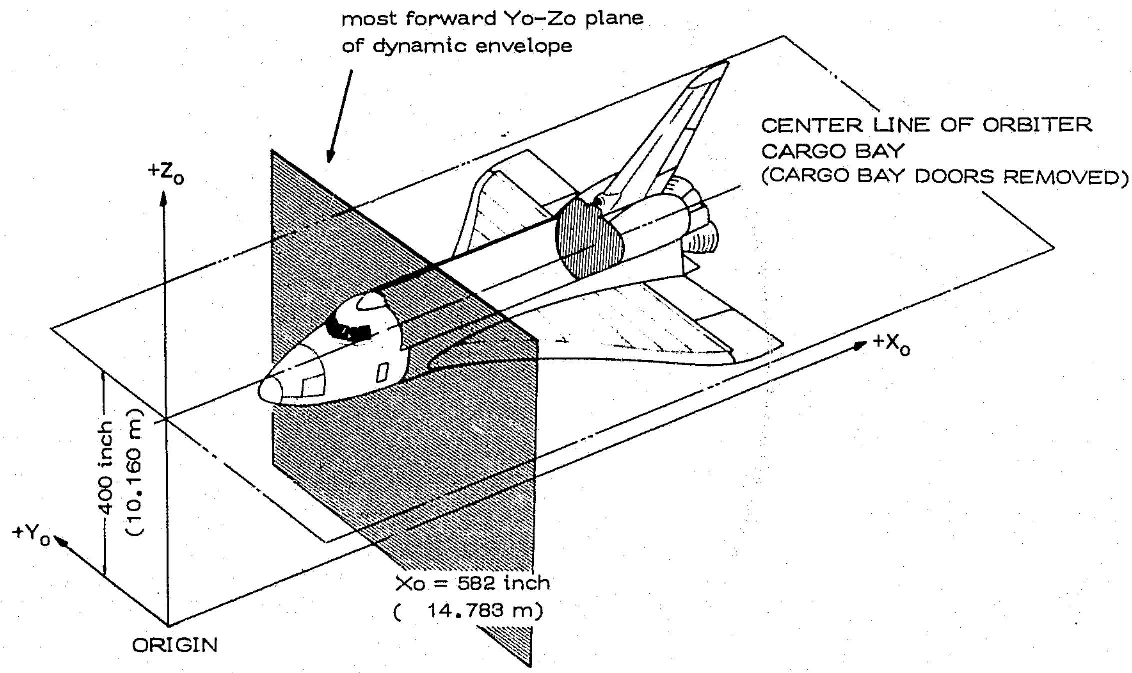


Figure 2-1: Orbiter Structural Body Coordinate System

2.1.2 Orbiter Body Axis Coordinate System

This coordinate system is shown in Figure 2-2 and defined as follows:

Type: Orbiter fixed and related to location of center of gravity (without Orbiter payload)

Origin: Center of gravity

Orientation: X_{BY} axis is parallel to the Orbiter structural body X_0 axis; positive toward the nose.

Z_{BY} axis is parallel to the Orbiter plane of symmetry and is perpendicular to X_{BY}, positive down with respect to the Orbiter fuselage.

Y_{BY} axis completes the right-handed orthogonal system.

Characteristics: L, M, N: Moments about X_{BY}, Y_{BY}, and Z_{BY} axes, respectively.

ϕ, θ, ψ: rotation angles around roll, pitch and yaw axes.

p, q, r: Body rates about X_{BY}, Y_{BY}, and Z_{BY} axes, respectively.

$\dot{p}, \dot{q}, \dot{r}$: Angular body acceleration about X_{BY}, Y_{BY}, and Z_{BY} axes, respectively.

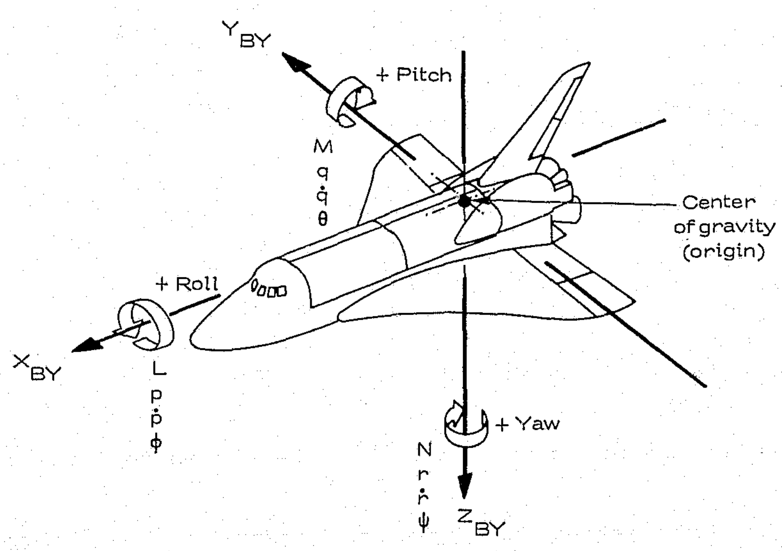


Figure 2-2: Orbiter Body Axis Coordinate System

2.1.3 Spacelab System Coordinate System

This coordinate system is shown in Figure 2-3 and defined as follows:

Type: Fixed with respect to Orbiter cargo bay

Origin: On the center line of the dynamic envelope of the Orbiter cargo bay; the most forward Yo – Zo plane (section 2.1.1) is located at X_L = 10 m exactly.

Orientation: The X_L-axis is identical with the center line of the dynamic envelope of the Orbiter cargo bay. Positive sense is from the nose of the Orbiter toward the tail.

The Z_L-axis is in the Orbiter plane of symmetry, perpendicular to the X_L-axis. Positive sense is upward in landing attitude.

The Y_L-axis completes a right handed orthogonal, coordinate system.

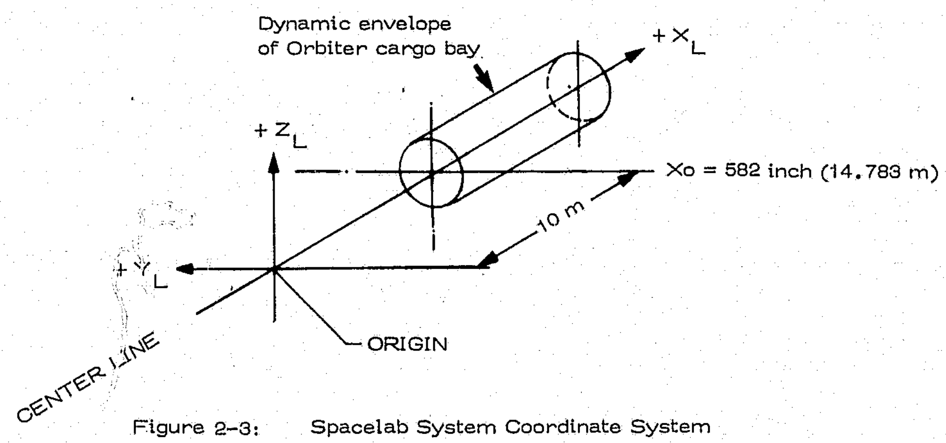


Figure 2-3: Spacelab System Coordinate System

2.2 Dimensional and Physical Data

2.2.1 Dynamic Envelope

Figure 2 – 4 illustrates the dynamic envelope for Spacelab and its payload in the Orbiter cargo bay and the location of this envelope within the Orbiter. The dynamic envelope is that envelope which must not be exceeded by any Spacelab or payload hardware in launch or landing configuration (except for interface

connections) under the maximum predicted dynamic environment, excluding Orbiter crash landing loads.

The dynamic envelope is of cylindrical shape with a diameter of 4.572 m (15 feet) around a center-line parallel to the Orbiter Xo-axis at Orbiter stations Yo = 0 and Zo = 400 inches (10.16 m). The length of the dynamic envelope is 18.288 m (60 feet), extending from Orbiter station Xo = 582 inches (14.783 m) to Orbiter station Xo = 1302 inches (33.071 m).

Technical drawings of the dynamic envelope in the Orbiter cargo bay are given in Figure 2-5. The dimensions are given in inches; figures in brackets are the dimensions in meters.

Particular attention of the users is drawn to the fact that transportation envelopes for various ground transportation modes may impose more severe constraints than the dynamic envelope of the Orbiter cargo bay. The transportation envelopes are outlined in Section 4.7.

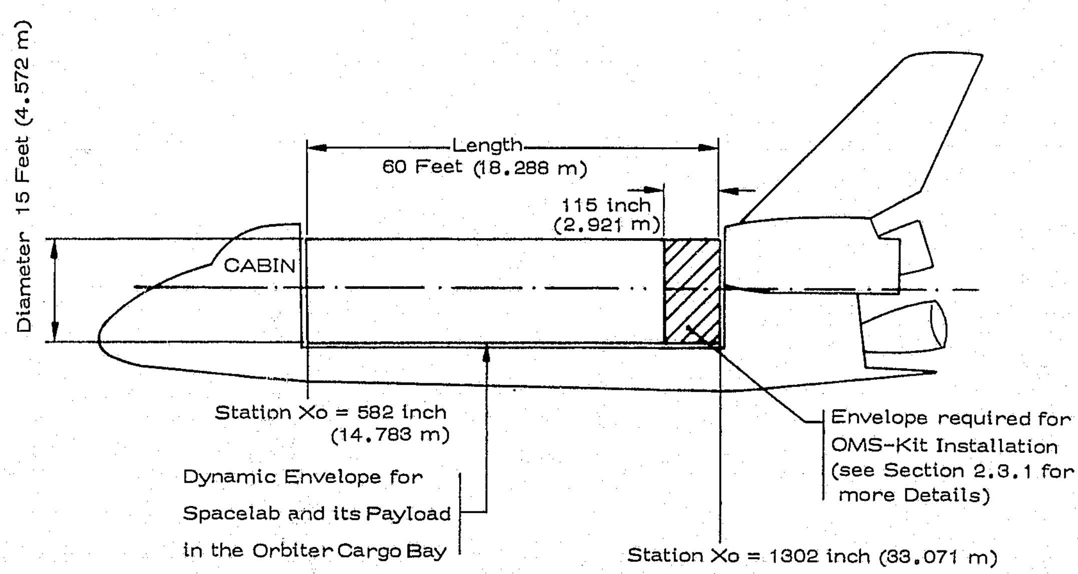
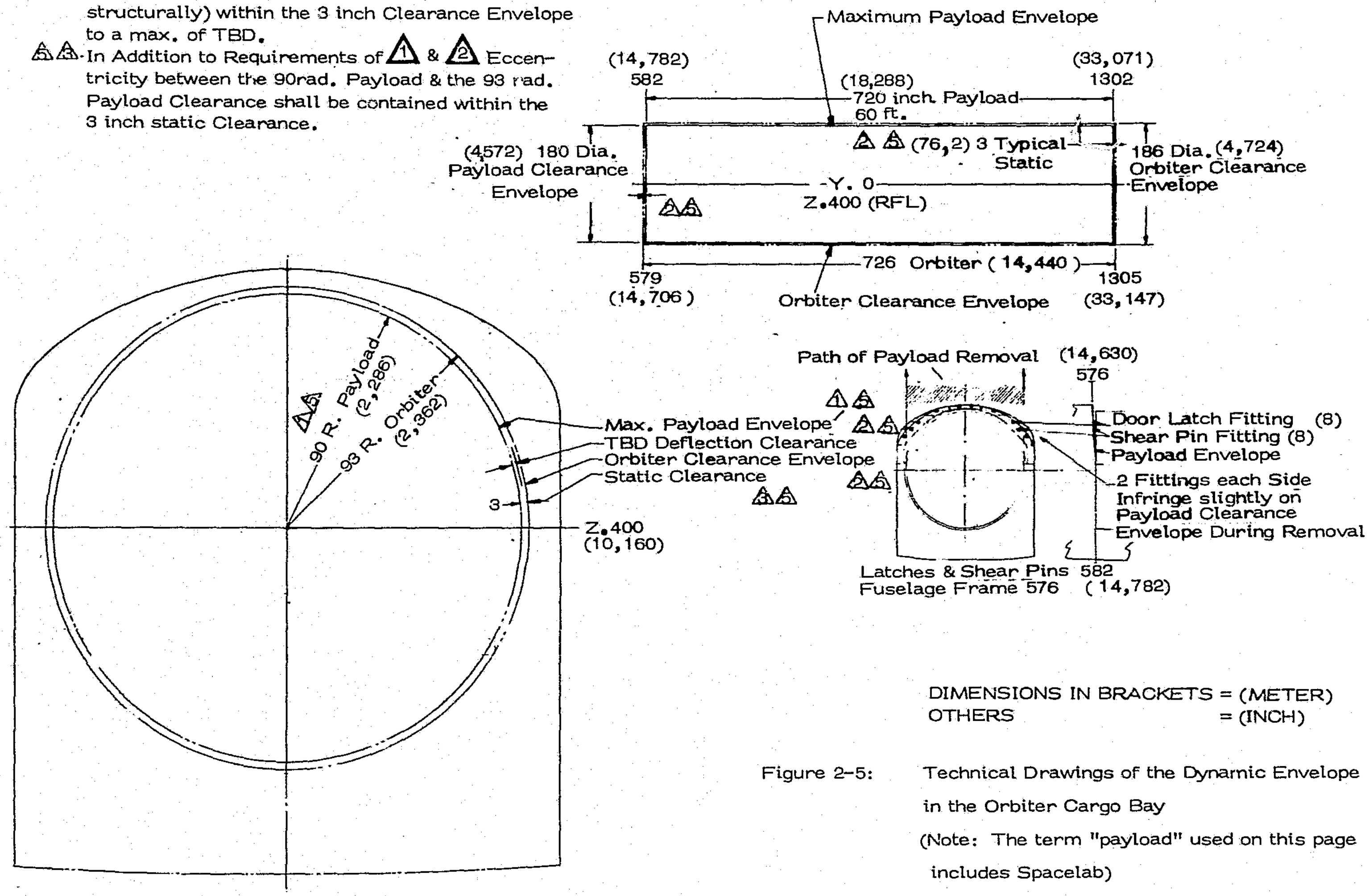


Figure 2-4: Dynamic Envelope for Spacelab and its Payload in the Orbiter Cargo Bay

PAYLOAD ENVELOPE/CLEARANCE

△5 △1 Payload Thermal/Structure Deflections contained
△5 △2 within max. Payload Envelope
Orbiter Structure & Equipment may be installed up to the 93rad. Payload Clearance Envelope Orbiter Structure/Equipment may deflect (thermally and structurally) within the 3 inch Clearance Envelope to a max. of TBD.

△4 △5 In Addition to Requirements of △1 & △2 Eccentricity between the 90rad. Payload & the 93 rad. Payload Clearance shall be contained within the 3 inch static Clearance.



DIMENSIONS IN BRACKETS = (METER)
OTHERS = (INCH)

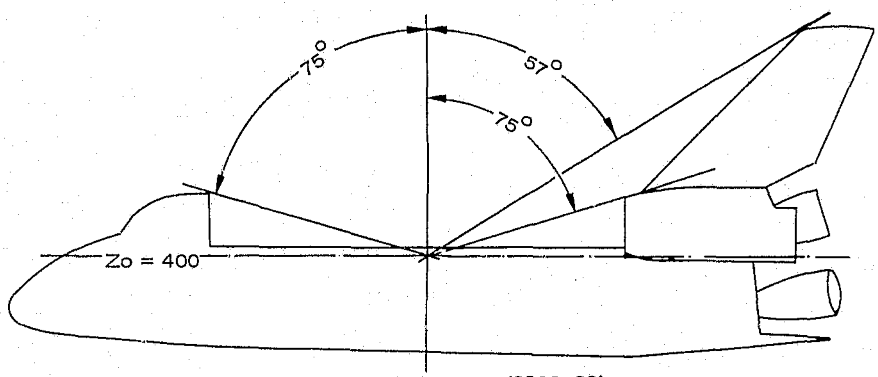
Figure 2-5: Technical Drawings of the Dynamic Envelope in the Orbiter Cargo Bay
(Note: The term "payload" used on this page includes Spacelab)

2.2.2 Field of View of the Orbiter Cargo Bay

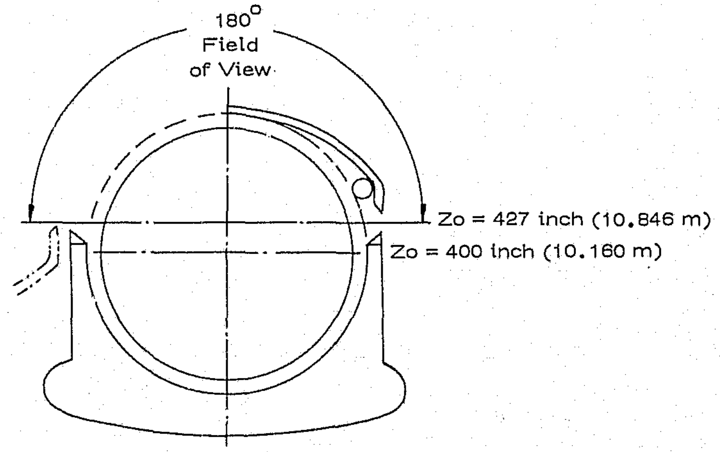
The Orbiter has the capability of exposing the entire length and width of the Orbiter cargo bay to space environment. With the Orbiter cargo bay doors and radiators open, the Orbiter provides an unobstructed 180-degree lateral field of view (except for localized interference due to the manipulator supports and the door hinges) for any point along the line Yo = 0, Zo = 427 (10,845.8 mm) between Xo = 582 (14,782.8 mm) and Xo = 1302 (33,070.8 mm). From the midpoint of the dynamic envelope Xo = 942 (23,926 mm) Yo=0, Zo = 400 (10,160 mm) , the following clearance angles, measured from the Z axis toward the X axis are maintained:

To the forward Orbiter bulkhead	75^{o}	(1.309 radians)
To the aft Orbiter bulkhead	75^{o}	(1.309 radians)
To the vertical stabilizer	57^{o}	(0.99408 radians)

These clearance angles are shown in Figure 2 - 6.



CLEARANCE ANGLES



LATERAL FIELD OF VIEW

Figure 2-6: Field of View and Clearance Angles of Orbiter Cargo Bay

2.2.3 Orbiter Cargo Bay Doors, Radiator and Thermal Surface Properties

The Orbiter cargo bay doors are presently under redesign. Door geometry, geometry of radiators associated with the doors, constraints on obscuring these by deployable payload and surface properties will be supplied as soon as this information is available. Figure 2 - 7 gives a general outline of the Orbiter cargo bay doors and the absorptivity α and emissivity ε of major Orbiter elements to which Spacelab and its payload are exposed.

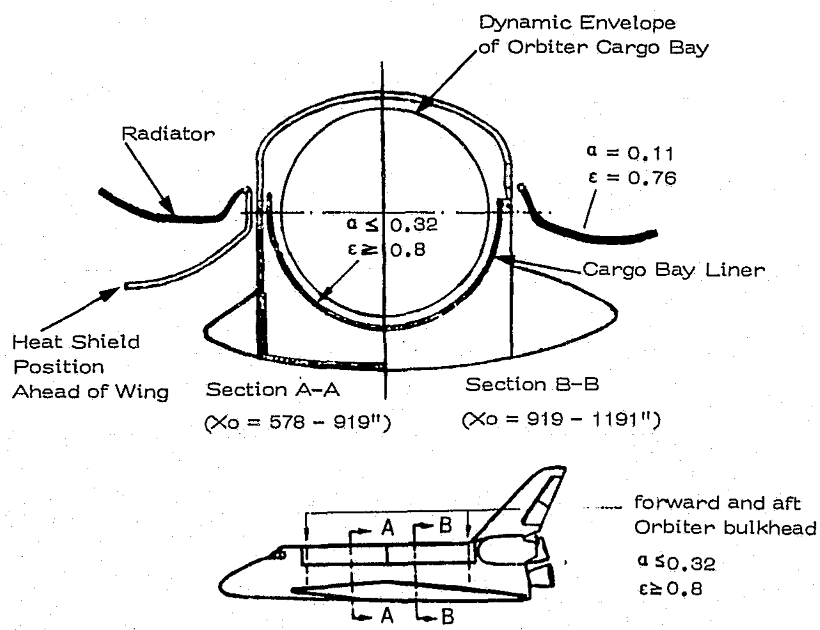


Figure 2-7: Open Orbiter Cargo Bay Doors (under Redesign)

2.2.4 Illumination of Orbiter Cargo Bay

In order to be capable of on-orbit payload handling operations under darkness conditions, the Orbiter will provide lighting in the vicinity of the Orbiter cargo bay. The illumination characteristics are not yet defined in detail, but the followingtentative information is available:

The design criteria for Orbiter cargo bay lighting specifies an average bay light level of 5 foot-candles with a 1 foot-candle minimum along the crew transfer routes. The color of the light will be white with a color temperature of not less than 3200° K. The X and Y coordinates on the cromaticity diagram (CEI scale) shall not be greater than 0.430.

In order to achieve the above light levels, there will be seven fixed lights in the Orbiter cargo bay and one light located on the manipulator arm. All seven fixed lights (six Orbiter cargo bay floodlights and the forward bulkhead light) will be 200 watts and have a 120° cone field of view. In the current baseline, they are located as follows:

1. Xo = − 759 inch Yo = + 56 Zo = + 319	2. Xo = − 750 Yo = − 56 Zo = + 319	3. Xo = − 979.5 Yo = + 56 Zo = + 375
4. Xo = − 979.5 Yo = − 56 Zo = + 375	5. Xo = −1140 Yo = + 56 Zo = + 325	6. Xo = −1140 Yo = − 56 Zo = + 325

The center lines of lights 1, 2, 5, and 6 are approximately 30° above the Yo-axis and the center lines of lights 3 and 4 are along the Yo-axis. The exact location of the forward Orbiter bulkhead light has not been determined but it is planned to be between and above the windows on the forward Orbiter bulkhead.

In order to improve the light distribution in the Orbiter cargo bay, a proposed change is now under consideration. If approved, the Orbiter cargo bay floodlights will be staggered along the Xo-axis rather than the baselined port-starboard symmetrical distribution.

In addition, the remote manipulator system (Section 2.6) provides a light (TBD) for illumination.

2.2.5 Maximum Landing Weight

The Orbiter can de-orbit and land a maximum cargo weight of 14 515 kg (32 000 lbs), although it has the capacity to place up to 29484 kg (65 000 lb) into orbit. The cargo weight comprises the Spacelab and payload weight in the Orbiter cargo bay and all other items charged to Spacelab or its payload, but which are located elsewhere in the Orbiter. In essence, the figure of 14 515 kg constrains the weight of Spacelab including its payload. Under abort and emergency conditions, the Orbiter can return and land with weights up to 29 484 kg (65 000 lb), but no mission should be planned with landing weights exceeding 14 515 kg (32 000 lb). Details of payload weight capabilities of Spacelab are treated in Section 3.

2.2.6 Center of Gravity Constraint

The center of gravity of the assembly Orbiter and Spacelab with its payload must be located within very close tolerances because of aerodynamic effects during re-entry and landing. Therefore, the location of the center of gravity of Spacelab with its payload with respect to the center of gravity of the empty Orbiter has significant constraints. The implications of these contraints are outlined in Section 3.

2.2.7 Mass Properties of the Orbiter

Typical mass properties of the Orbiter in flight configuration, but excluding Spacelab and its payload are presented in Table 2.1

Table 2-1 Typical mass properties of the Orbiter in flight configuration (Spacelab and its payload excluded)

Weight kg (lb)	CENTER OF GRAVITY cm (inch)			MOMENT OF INERTIA $kg \cdot m^2$	(slug $\cdot$ ft^2)	$\cdot 10^{+6}$	PRODUCT OF INERTIA		
	Xo	Yo	Zo	I_{x-x}	I_{y-y}	I_{z-z}	I_{xy}	I_{xz}	I_{yz}
71 214 (157 000)	2819 (1110)	1.0 (0.4)	952 (375)	1.03 (0.76)	7.6 (5.6)	7.8 (5.8)	~0 (~0)	0.22 (0.16)	~ 0 (~ 0)

2.2.8 Space Shuttle Operational Contamination Control

Note: This section does not include contamination effects caused by Spacelab. The term "payload" in this section denotes the Orbiter payload including Spacelab plus Spacelab payload.

2.2.8.1 Prelaunch Phase

Concern for the contamination of payloads originates with the preparations prior to loading the payload into the Orbiter cargo bay. The Orbiter cargo bay liner and the Orbiter surfaces enveloping the payload will be cleaned to the visibly clean-level as defined in NASA SN-C-0005, if requested by payloads. The loading of the payload into the Orbiter cargo bay is to be accomplished within an enclosure with HEPA filtered input air supply guaranteed to be Class 5000 or less. After payload loading and closure of the Orbiter cargo bay doors, purge gases are supplied to the Orbiter cargo bay (see Section 5.4.2).

2.2.8.2 Ascent Phase

At launch and during ascent the Orbiter cargo depressurizes and the payload is generally not subjected to contaminants (except those due to external tank (ET) and solid rocket booster (SRB) which should not generally effect Spacelab) until earth orbit is established and the Orbiter cargo bay doors are opened. For several seconds after lift-off the Orbiter cargo bay vents are closed, thus preventing ingestion of dust particles, and in particular, combustion products of the Orbiter main engines and the solid rocket booster engines.

During ascent and until main engine cut-off the Auxiliary Power Unit (APU) and hydraulic subsystem water boilers are operating with the effluents of large quantities of steam and combustion products of hydrazine. However, these are expected to have no effect on the payloads since during this time the Orbiter cargo bay doors are closed.

Significant quantities of water are emitted by the main flash evaporator system above 42.7 kilometers (140,000 feet) until the Orbiter cargo bay doors are opened and the radiators put into operations.

2.2.8.3 On-Orbit Phase

The major sources of contamination during the on-orbit phase are:

a) the RCS vernier thuster firings which may be required by the payload for attitude control, or for thermal conditioning of the Orbiter and its payload.

b) dumping of potable water by the supplemental flash evaporator for environmental control cooling or because of excess water on board.

c) the release of particulates and outgassed species.

Reaction Control Subsystem (RCS)

Payloads may require either the primary or vernier thrusters for on-orbit operations. Use of the primary thrusters will undoubtedly be limited to mission sequences where contamination sensitive payload measurements are not made. Contamination considerations for these thrusters may be necessary during deployment or retrieval sequences. In general, however, the main RCS contamination considerations will apply to the vernier thrusters since those will undoubtedly be needed for attitude pointing during payload measurements.

The vernier thrusters are located in positions that preclude direct plume impingement on payload surfaces. Their main axis lies along the -Zo, or $\pm$ Yo axis of the Orbiter. Although this orientation precludes direct impingement on Orbiter cargo bay surfaces, reflection of the engine plumes from the wing surface results in plume contributions to the region above the Orbiter cargo bay. This effect has been analytically modeled and calculations of column density and return flux using this model are shown in Table 2-2. These contributions to the contaminant environment are dependent upon thruster firing times, i.e., during the firing of a specific thruster the contribution is present and decays instantaneously after thruster cut-off.

If the contributions to the vehicle environment due to vernier thruster firings are unacceptable for a given payload then the use of payload pointing and stabilization systems may be required or Orbiter flight attitudes must be chosen which minimize thruster firings. One attitude which should provide in excess of 50 minutes observing time is a gravity gradient stabilized mode. In this mode Orbiter remains in an attitude in which the Xo-axis of the Orbiter is local vertical to the surface of the earth.

Flash Evaporator

During orbital operations the flash evaporator operation will dump approximately 141 kg of water overboard per day. The water is dumped in the vapor state through two nozzles located at Xo = 1507 and Zo = 291 on each side of the aft fuselage. The plume expands along the $\pm$ Y-axis of the vehicle. This results in some wing reflection as with the attitude control thrusters, however, the contribution is within the specification requirements. The contributions for this system are also shown in Table 2-2. Operational flexibilities permit manually inactivating this system for short periods of time (11.5 hours) with some possible concurrent heat rejection loss. The water produced during this period must be stored and subsequently dumped.

Other Sources

During orbital operations release of particulate and outgassed species and/or leaks may also contribute to the vehicle external environment. Regarding particulates, systems such as the flash evaporator are being assessed for particle production characteristics and design changes will be recommended if appropriate. General sloughing of particles from vehicle surfaces should be minimized by the use of the Orbiter cargo bay liner. The liner isolates the lower portion of the Orbiter cargo bay from the payload region and thereby limits particle migration between these two regions.

Outgassing of materials used inside the cargo bay and exterior surfaces of the Orbiter exposed to payloads is controlled by selection for low outgassing properties. Early mission exposure of the materials in question will result in reduced outgassing rates on subsequent missions.

2.2.8.4 De-Orbit and Descent Phase

During descent the primary RCS thrusters operate and emit propellant products. At this time the APU and hydraulic subsystems are also in operation. Large quantities of water are emitted by the flash evaporator for heat rejection down to about 30 kilometers (100,000 feet). Below 30 kilometers, heat is rejected by an ammonia boiler. To avoid ingestion of these subsystem effluents, attention has been given to location the Orbiter cargo bay vents at points where the possibility of ingestion into the Orbiter cargo bay is minimal. Further, the Orbiter cargo bay vents are closed prior to de-orbit and remain so until about 23 kilometers (75,000 feet), at which time they must be opened to allow for repressurization of the Orbiter. The forward RCS module is de-activated prior to de-orbit burn. Effluents from the aft RCS operation below 23 kilometers (75,000 feet) will be swept back into the wake and circulate within it. Similarly, the ammonia boiler vent, APU hydrazine exhaust vents, and water boiler vents are located aft of the mid-fuselage to inhibit ingestion of their gaseous contaminants. Circulation of these gaseous effluents within the wake may result in some ingestion into the Orbiter cargo bay with the repressurization air. Quantitative analysis is extremely difficult, hence quantification will have to await experimental results from the first test flight. To inhibit ingestion of particulates into the payload bay during descent, the vent ports leading into the Orbiter cargo bay will be covered with a 35 micron glassbead rated (GBR) filter.

2.2.8.5 Landing Phase

At or near touchdown the vents are closed since repressurization is complete and it is desired to prevent dust, salt, and other particulates from entering the Orbiter cargo bay. Within 30 minutes after rollout, ground support operations will provide HEPA filtered purge gas to the Orbiter cargo bay volume.

Table 2 - 2: Predicted Number Column Density and Return Flux Contributions From Shuttle Orbiter Sources of Contamination

Parameter / Source	Number Column Density (NCD) (molecules/cm^2)			Return Flux (RF) (molecules/cm^2/sec)		
				(Values at 435 km)		
Vernier Reaction Control System	Location/Direction of Thruster			Location/Direction of Thruster		
	Aft/-Zo	Aft/Yo	Fwd/Yo,Zo	Aft/-Zo	Aft/Yo	Fwd/Yo,Zo
LOS 1 A*	4.4×10^{14}	2.0×10^{14}	3.9×10^{12}	7.6×10^{12}	3.4×10^{12}	6.6×10^{10}
LOS 5 A**	1.8×10^{14}	8.1×10^{12}	2.7×10^{12}	3.2×10^{12}	1.4×10^{12}	4.6×10^{10}
Flash Evaporator				700 km	435 km	200 km
LOS 1 A	1.4×10^{12}			8.4×10^{8}	2.4×10^{10}	1.3×10^{12}
LOS 5 A	1.4×10^{12}			8.4×10^{8}	2.4×10^{10}	1.3×10^{12}
Outgassing						
LOS 1 A	$<10^{12}$ after 10 hrs			$<10^{12}$		
LOS 5 A	$<10^{12}$ after 10 hrs			$<10^{12}$		
Leakage from Orbiter Cabin				700 km	435 km	200 km
LOS 1 A	2.2×10^{13}			1.2×10^{10}	3.7×10^{11}	1.9×10^{13}
LOS 5 A	3.5×10^{13}			2.0×10^{10}	5.6×10^{11}	3.1×10^{13}

* LOS 1 A, Zero degree line-of-sight (in the +Zo direction) originating at Xo = 1107 inch

** LOS 5 A, 50° off of +Zo towards -Xo (forward) originating at Xo = 1107 inch

2.3 Orbits

The Space Shuttle provides for transportation of Spacelab to and from earth orbits and utilizes two launch sites. The Eastern Test Range (ETR) located at the Kennedy Space Center (KSC) is used for launches into low inclination orbits and the Western Test Range (WTR) located at the Vandenberg Air Force Base (VAFB) is used for launches into high inclination orbits.

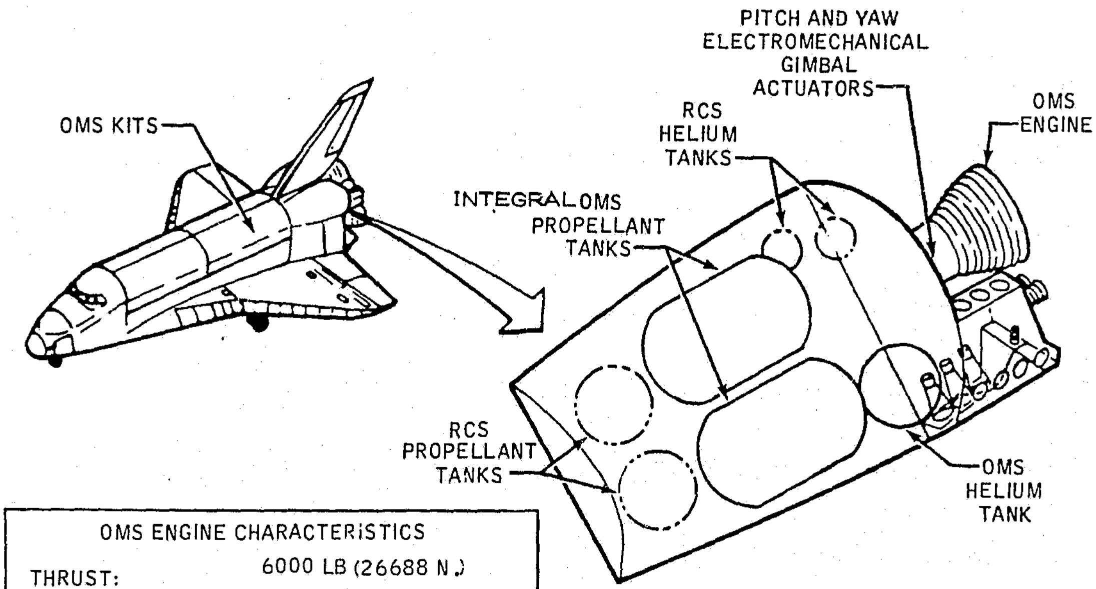
2.3.1 Orbital Maneuvering

The thrust required to accelerate the Orbiter to suborbital velocity is supplied by two Solid Rocket Boosters and the main engines of the Orbiter which are supplied with propellant from an External Tank. The Solid Rocket Boosters and the External Tank are jettisoned during the launch phase. An Orbital Maneuvering Subsystem (OMS) is used to acquire orbital velocity and to place the Orbiter into the desired orbit (Figure 2.8). Furthermore, the OMS provides the propulsive thrust to perform orbit corrections, orbit transfer, rendez-vous and de-orbit maneuvers. The thrust required for Orbiter separation and translational braking is provided by the Reaction Control Subsystem (RCS) which is operated in a special mode for this purpose, although the prime function of this subsystem is attitude control.

The integral OMS tanks of the Orbiter are sized to provide a usable propellant capacity of 11294 kg (24900 lb). The velocity increment which can be imparted to the Orbiter by this amount of propellant is 304.8 m/sec (1000 ft/sec) for a 29484 kg (65000 lb) and about 366 m/sec (1 200 ft/sec) for a 14 515 kg (32 000 lb) cargo weight (para. 2.2.5), respectively. Up to three extra propellant tanks, referred to as OMS kits can be installed in the Orbiter cargo bay for increased operational flexibility. These extra OMS kits are installed at the aft end of the Orbiter cargo bay (Figure 2-4). The detailed envelope required for the installation of one, two and three OMS-kits are described in Figure 2-9, 2-10 and 2-11, respectively. The envelopes shown include structural plus thermal deflections of the OMS-kits and all other associated Orbiter parts.

The dry- and wet weight, as well as the velocity increment which can be imparted to the Orbiter with various cargo weights (para. 2.2.5), are summarized in Table 2-3.

The dry- and wet weight of the OMS-kits will be charged to the landing and launch weight of the Spacelab payload, respectively (Sect. 3). These weights have to be duly accounted for in mission planning and in the assessment of the center of gravity. The velocity increments outlined in Table 2-3 indicate that the OMS-kits are not intended to perform significant inclination changes e.g. from 28 to 0 degree inclination, but to perform orbit corrections or transfer maneuvers in the orbital plane. The maximum achievable inclination change per OMS-kit is about 2 degrees. The use of OMS-kits to obtain orbits with high attitude is shown in Figure 1-12.



OMS ENGINE CHARACTERISTICS	
THRUST:	6000 LB (26688 N) VACUUM
SPECIFIC IMPULSE:	313 SEC
CHAMBER PRESSURE:	125 PSIA (861875 N/M^2)
MIXTURE RATIO:	1.65:1
GIMBAL CAPABILITY:	± 4° PITCH ± 8° YAW

OMS TANKAGE CAPACITY FOR 305 m/SEC (1000 FT/SEC) VELOCITY CHANGE -	
FUEL (MMH) WEIGHT:	9396 LB (4262 Kg)
OXIDIZER (N_2O_4) WEIGHT:	15 504 LB (7032 Kg)

Figure 2 - 8: Orbital Maneuvering Subsystem
(Tank Arrangement and Engine Characteristics)
(Reference, Vol. XIV, Rev. D, Figure 3 - 1)

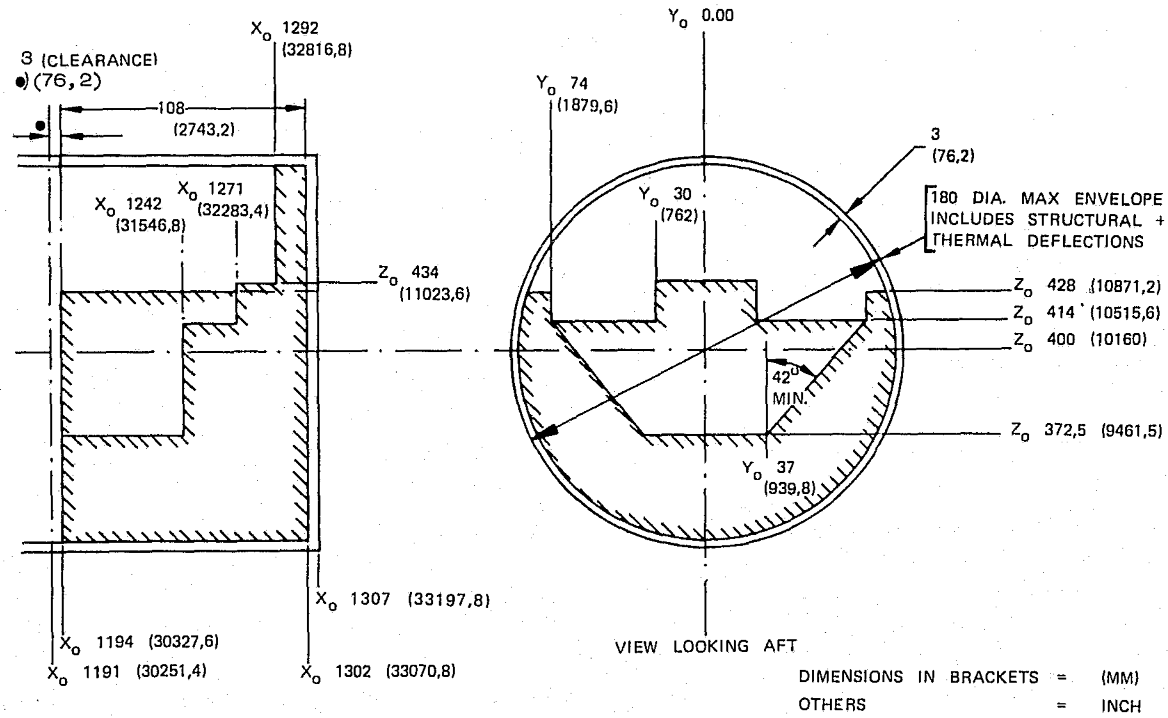


FIGURE 2 – 9: ENVELOPE FOR ONE OMS KIT

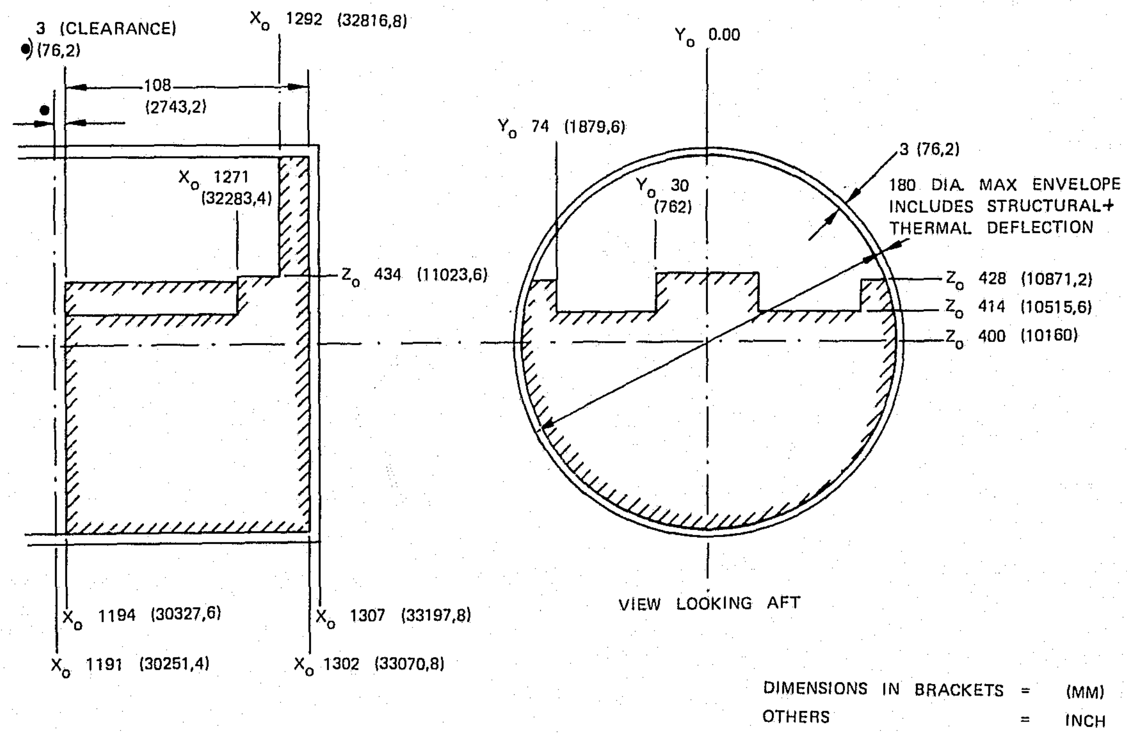


FIGURE 2 – 10: ENVELOPE FOR TWO OMS KITS

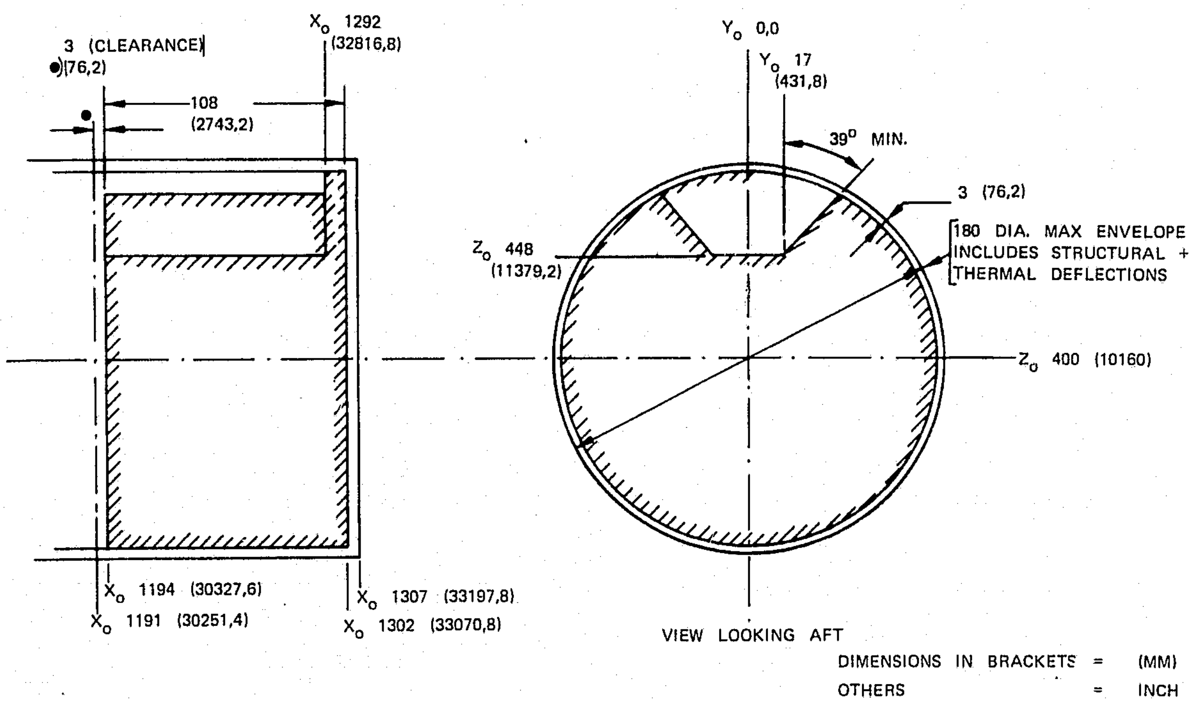


FIGURE 2 – 11: ENVELOPE FOR THREE OMS KITS

Table 2-3 Weight and Velocity increments of OMS-Kits

(Weight Figures are Tentative)

Number of OMS-kits	Dry Weight kg (lb)	Wet Weight kg (lb)	Velocity increments m/sec (ft/sec)	
			for 14 515 kg (32 000 lb) cargo weight*	for 29 484 kg (65 000 lb) cargo weight*
1	1422 (3135)	7034 (15508)	~ 183 (~ 600)	152.4 (500)
2	1857 (4095)	13082 (28841)	~ 366 (~1200)	304.8 (1000)
3	2333 (5145)	19171 (42264)	~ 549 (~ 1800)	457.2 (1500)

* for definition see para 2.2.5

2.3.2 Achievable Orbits

In Figure 2 – 12 typical ranges of circular orbits attainable for Spacelab missions are presented. This figure is based on a total Spacelab weight, including Spacelab payload, of 14 515 kg (32 000 lb). It is assumed that launch takes place from KSC for inclinations between 28° and 57° and from VAFB for inclinations between 56° and 104°.

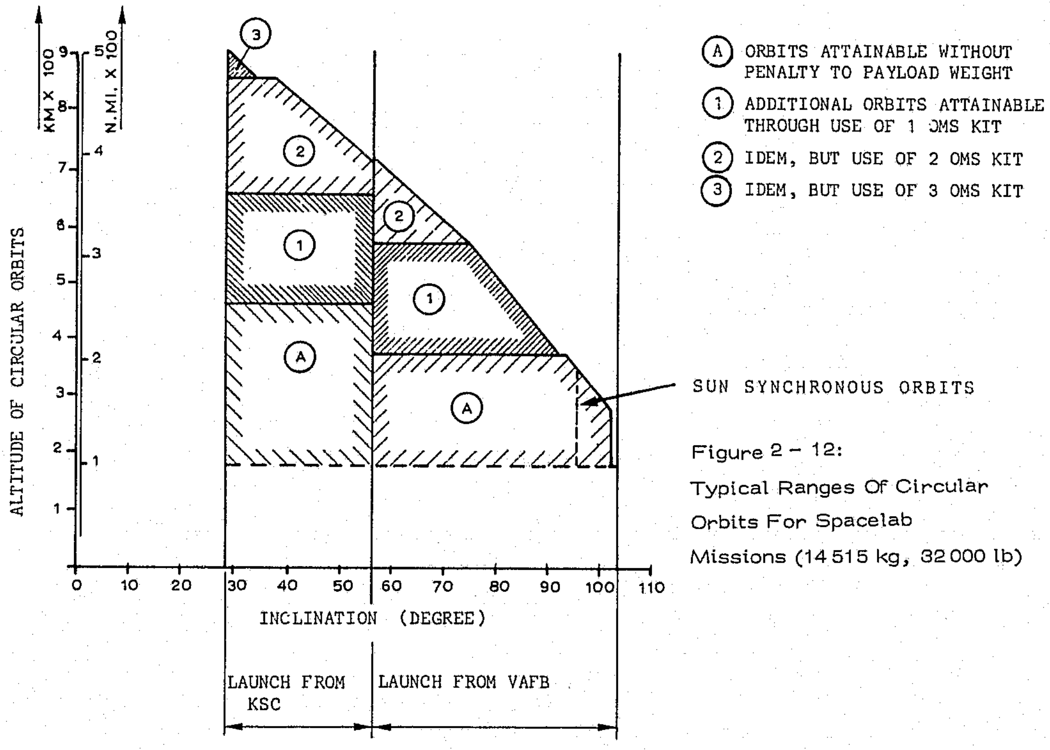


Figure 2 – 12: Typical Ranges Of Circular Orbits For Spacelab Missions (14 515 kg, 32 000 lb)

Figure 2-12 represents the capabilities of the Space Shuttle for typical sets of operational requirements In this figure, a RCS propellant consumption of 1406 kg (3100 lb) is assumed. It should be noted that the suborbital disposal of the External Tank presents limitations on some discrete inclinations between 56 and 70 degrees for launches from VAFB). Missions in this inclination region will have to be individually planned, because the performance shown is the maximum expected and trajectory changes to accommodate safe External Tank disposal will probably only degrade performance.

Figure 2-12 is derived from performance curves of the Space Shuttle for launches from KSC and VAFB (Figures 2-13, 2-14). The curves present the cargo weight (para. 2.2.5) to be placed into circular orbits as a function of orbital altitude, for various inclinations and number of OMS-kits. The weight of the OMS propellant in the integral OMS-tankage and OMS-kits necessary to obtain the indicated orbits has already been taken into account in establishing the performance curves of Figure 2-12 and 2-13 and, therefore, need not to be subtracted from the cargo weight given in these figures.

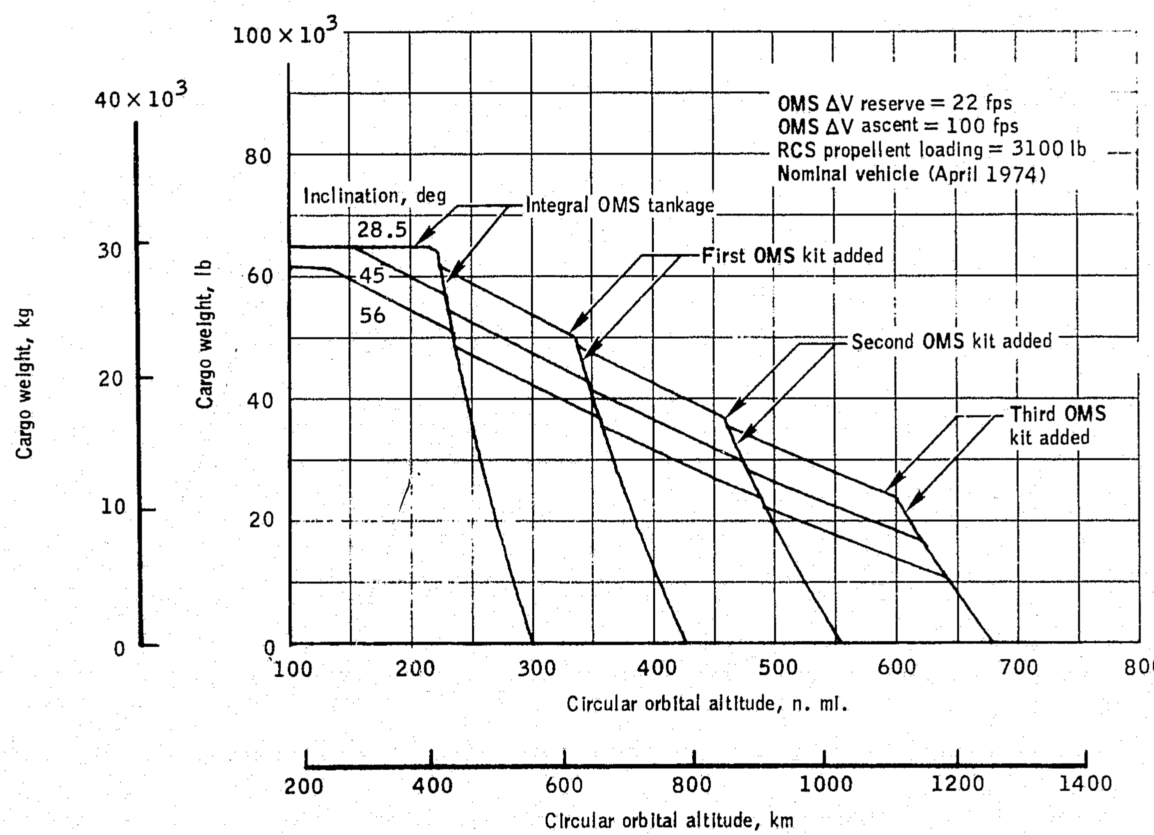


Figure 2-13: Cargo weight versus circular orbital altitude - KSC launch (Reference, Vo., XIV, Rev. D, Fig. 3.3)

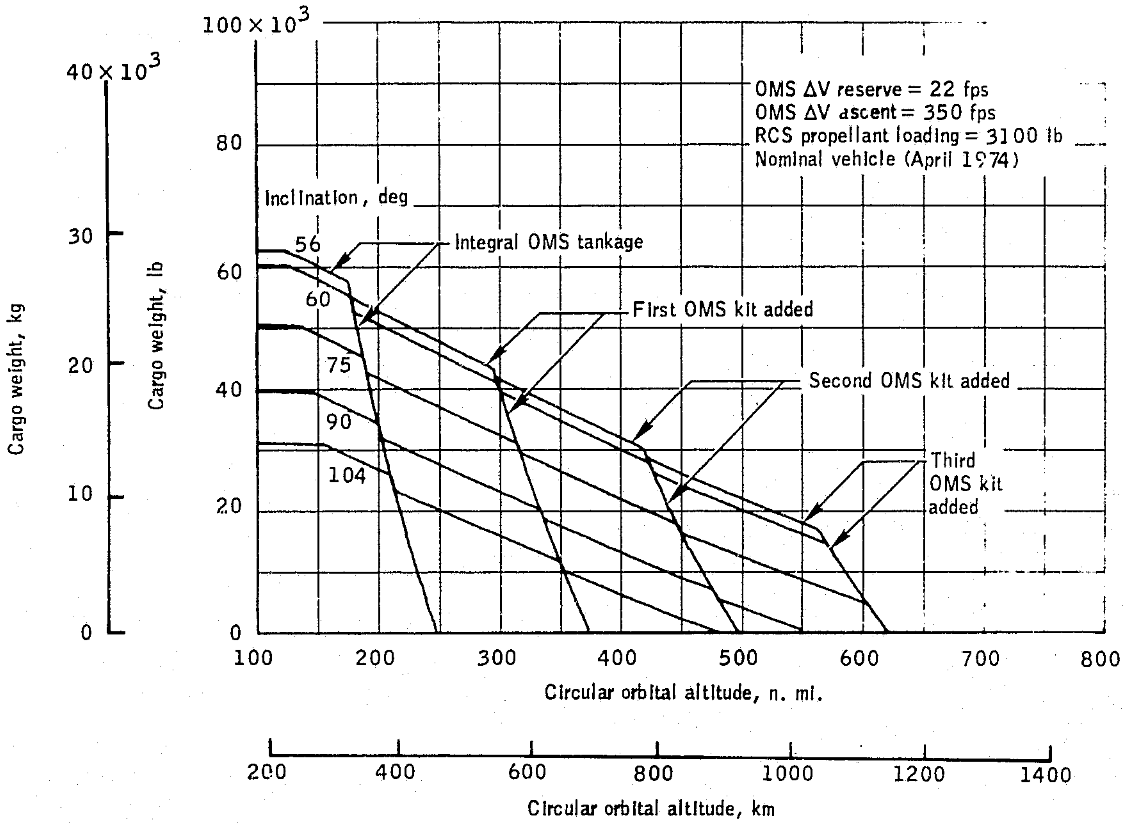


Figure 2-14: Cargo Weight Versus Circular Orbital Altitude - VAFB Launch (Reference, Vol. XIV, Rev. D, Fig. 3.4)

The Space Shuttle has also the capability to place Spacelab into elliptical orbits. This capability depends significantly on the de-orbit mode. Orbits with maximum excentricity can be obtained in a direct de-orbit mode, i.e. a procedure where the de-orbit maneuver is initiated at apogee. An alternative de-orbit mode (indirect de-orbit mode) is to return to a low altitude orbit prior to re-entry. The maximum achievable heights of apogee are shown in Table 2 - 4 for various inclinations and the two described de-orbit modes. This table is based on a height of perigee of 185 km (100 nautical miles) and a Spacelab weight, including payload, of 14 515 kg (32 000 lb). For the direct de-orbit mode there exist operational limitations such as the relationship of the landing site to the location of the de-orbit maneuver or constraints due to thermal protection system capabilities. In Table 2 - 4 an ideal relationship between the landing site and the location of the de-orbit maneuver and no constraint due to the Orbiter thermal protection system are assumed. The data concerning the indirect de-orbit mode are based on an 185 km (100 nautical miles) circular orbit prior to re-entry. The indirect de-orbit mode can always be flown. The exact capability of the Space Shuttle to obtain elliptical orbits will have to be assessed on an individual basis and will - in general - be between the figures for the two de-orbit modes, quoted in Table 2-4.

The Shuttle System has the capability to place the Orbiter into sun synchronous orbits (see Figure 2-12) which have nodal precession rates exactly matching the earth's angular motion around the sun.

It has already been pointed out that the data for the direct de-orbit modes given in Table 2-4 are based on an ideal location of the perigee with respect to the landing site. Other locations of the perigee and control of the location of perigee are possible, but these cases will have to be calculated on an individual basis.

In principle the Shuttle System is capable to cover the whole range of possible angles of right ascension of ascending nodes. Mission requiring specific angles of right ascension of ascending node have to be evaluated on an individual basis.

Table 2 - 4 Typical Eccentric Orbits
(Spacelab including its payload: 14 515 kg, 32 000 lb)

Inclination	direct de-orbit		indirect de-orbit	
	apogee in km (n.mi.)	number of OMS-kits	apogee in km (n.mi.)	number of OMS-kits
28.5°	2 500 (1 350)	3	1 150 (620)	3
55°	2 050 (1 100)	2	950 (510)	2
104°	280 (150)	0	550 (300)	0

Perigee 185 km (100 n.mi.)

2.3.3 Orbital Position Determination

The orbital injection errors for the Shuttle System are presented in Table 2-5.

Table 2 - 5 Orbiter expected 3 Sigma Insertion Accuracies (Reference: PR CBD 500781 R 1)

Launch Site	Position km (n.mi.)				Velocity m/sec (ft/sec)			
	Down-range $\pm x$	Cross-range $+ y$	Altitude $\pm z$	Radius Vector r	Down-range $\pm \dot{x}$	Cross-range $\pm \dot{y}$	Altitude $\pm \dot{z}$	Velocity Vector v
KSC	15.3 (8.3)	2.0 (1.1)	0.9 (0.5)	15.5 (8.4)	2.9 (9.4)	6.9 (22.5)	4.2 (13.7)	7.6 (25.0)
VAFB	20.2 (10.9)	2.8 (1.5)	0.9 (0.5)	20.4 (11.0)	3.1 (10.2)	6.9 (22.5)	4.5 (14.7)	8.8 (28.8)

Knowledge of the orbital position of the Orbiter at any time is dependent on the elapsed time since the last tracking pass and also the tracking system used during the last pass.

The on-orbit navigation accuracies, using the Spaceflight Tracking and Data Network (STDN) and the Tracking Data Relay Satellite (TDRS) system are given in Table 2 - 6 . These accuracies are being re-evaluated, based on current onboard software model definition. For each system, the estimated errors of the position and velocity of the Orbiter are given at the end of the last tracking pass and for one revolution later. The navigation accuracies using the STDN are based on at least two subsequent two-minute tracking passes above a 5° elevation angle and separated by approximately one revolution. The TDRS navigation accuracies are based on two tracking passes from a single TDRS.

Table 2 - 6: Expected On-Orbit Navigation Accuracies (3 Sigma) for 100 Nautical Miles (185 km) Orbital Altitude
(Reference: Vol. XIV, Rev. D.: Table 3.1)

Navigation System	Position, Feet (Meters)				Velocity, Feet/Sec (Meters/Sec)			
	Altitude	Down-track	Cross-track	Root Sum Square	Altitude	Down-track	Cross-track	Root Sum Sqare
STDN								
After last tracking pass	440 (130)	370 (110)	430 (130)	730 (222)	3,9 (1,2)	0,5 (0,15)	2,0 (0,6)	4,4 (1,3)
After one revolution	470 (150)	850 (260)	430 (130)	1 030 (315)	4,3 (1,3)	0,5 (0,15)	2,0 (0,6)	4,8 (1,4)
TDRS								
After last tracking pass	300 (90)	1 400 (430)	1 520 (460)	2 070 (630)	1,6 (0,5)	0,35 (0,11)	0,5 (0,15)	1,7 (0,5)
After one revolution	300 (90)	2 010 (610)	1 520 (460)	2 400 (740)	2,4 (0,7)	0,3 (0,1)	0,5 (0,15)	2,5 (0,7)

The time required to perform an orbit correction maneuver, to determine the new orbit and the position and velocity of the Orbiter with a certain accuracy is in the order of 15 to 30 minutes. Constraints related with these activities have to be assessed on an individual basis.

2.4 Attitude Control

Orbiter pointing and attitude control are performed by the Reaction Control Subsystem (RCS) using either primary or vernier thrusters. Basic RCS data and the arrangement of thrusters and tanks are given in Table 2-15.

The primary thrusters are mainly used for rotational maneuvers while the main function of the vernier thrusters is to maintain attitudes.

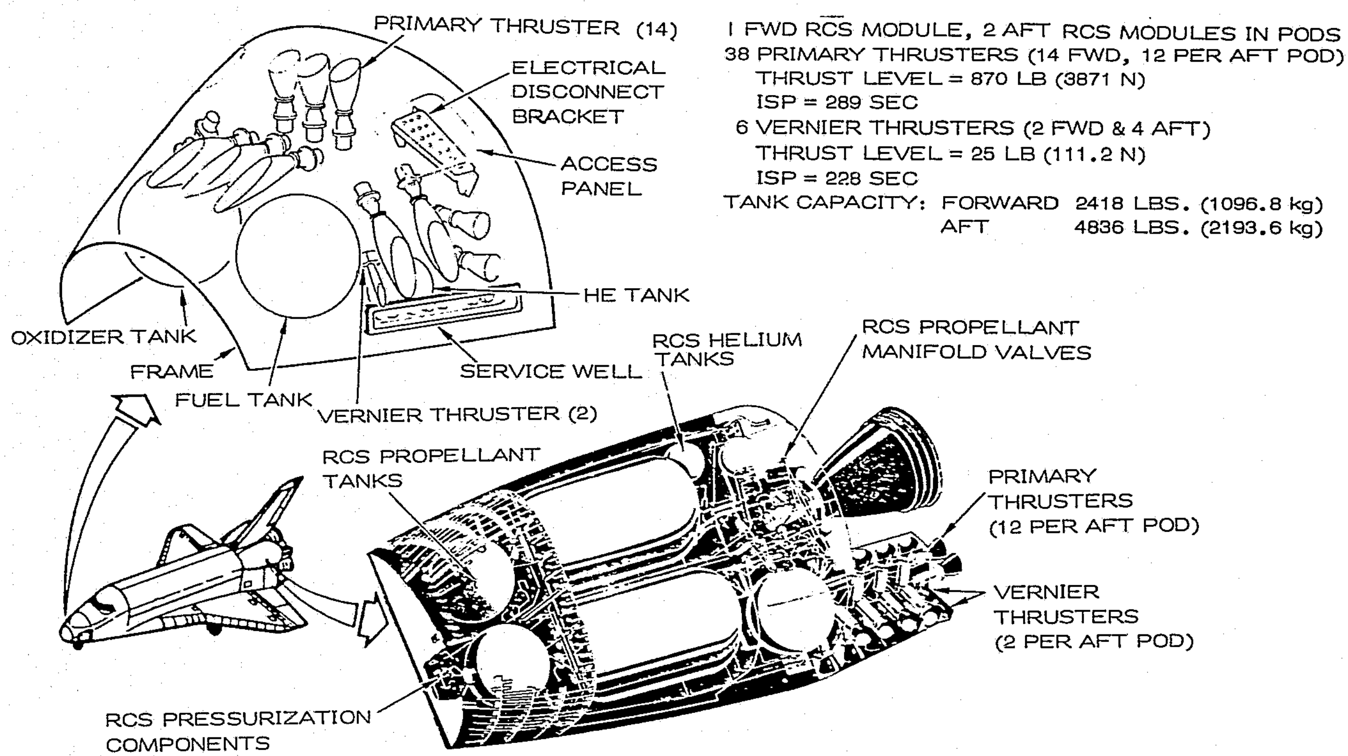


Figure 2 - 15: Orbiter Reaction Control Subsystem (RCS)
(Tank and Thruster Arrangement)
(Reference, Vol. XIV, Rev. D, Figure 3.2)

2.4.1 Propellant Consumption

2.4.1.1 Propellant Available

When the RCS propellant tanks are fully loaded there is approximately 4000 lbs (1814 kg) of propellant available for on-orbit operation of Spacelab and its payload. A typical RCS propellant budget is given in Table 2 – 7.

For missions with extremely large RCS propellant requirements, an additional 2000 lbs (907 kg) of RCS propellant can be obtained from the integral OMS tankage. Missions requiring the use of this RCS/OMS interconnect require special planning.

The weight of RCS propellant required for attitude control and maneuvering of Spacelab and its payload on-orbit will be charged to the launch weight of the Spacelab payload, but not to the landing Spacelab payload weight (Section 3).

2.4.1.2 Vernier Thrusters

The vernier RCS propellant usage for various orbital altitudes and Orbiter orientation modes is presented in Table 2 – 8 for a deadband of $\pm$ 0.1 degree per axis. These consumption rates include both aerodynamic and gravity gradient torques. The effects of attitude deadband on vernier RCS propellant usage for Orbiter pointing in a 100 n.m. (185.2 km) circular orbit are illustrated in Figure 2 – 16. For deadbands greater than 0.1 degrees per axis, the majority of the propellant is utilized for countering the aerodynamic and gravity gradient disturbances. Therefore, the propellant consumption for a large deadband, e.g. 5 degree, can be expected to be not significantly different from figures given in Table 2 – 8. A detailed mission analysis is required for an accurate assessment of the required propellant for such a large deadband. As the deadband is decreased below the $\pm$ 0.1 deg/axis, increased limit cycle frequency becomes the dominant effect on propellant usage. The values given for deadbands less than 0.1 degree are based on a perfect sensor, but also includes the aerodynamic and gravity gradient disturbances. Disturbing torques due to venting are not included in the data and could be significant.

2.4.1.3 Primary Thrusters

The propellant consumption of the RCS system for attitude maneuvers, for example pointing from one inertial direction to another, are described below for the primary thrusters. Table 2 – 9 gives the propellant usage for attitude maneuvers at rotation rates from 0.25 to 1.0 deg/sec using the primary thrusters. The maneuvers are performed by sequential rotation at the given rotation rate around the roll-, pitch- and yaw-axes of the Orbiter (Figure 2 – 2).

Table 2 – 7: Typical RCS Propellant Budget

(Reference, Vol. XIV, Rev. D, Table 3.8)

	lb	kg
Total RCS Loadable	7,391	3,353
Unavailable (Includes Residuals Plus Tank Loading Tolerance)	– 806	366
	6,585	2,987
Required for Insertion	– 228	103
	6,357	2,883
Required for Orbital Adjustment	– 899	408
	5,458	2,476
Required for Entry	– 1,164	528
	4,294	1,948
On-Orbit Dispersions & Contingencies	– 301	136
Available for Payload Support	3,993	1,811

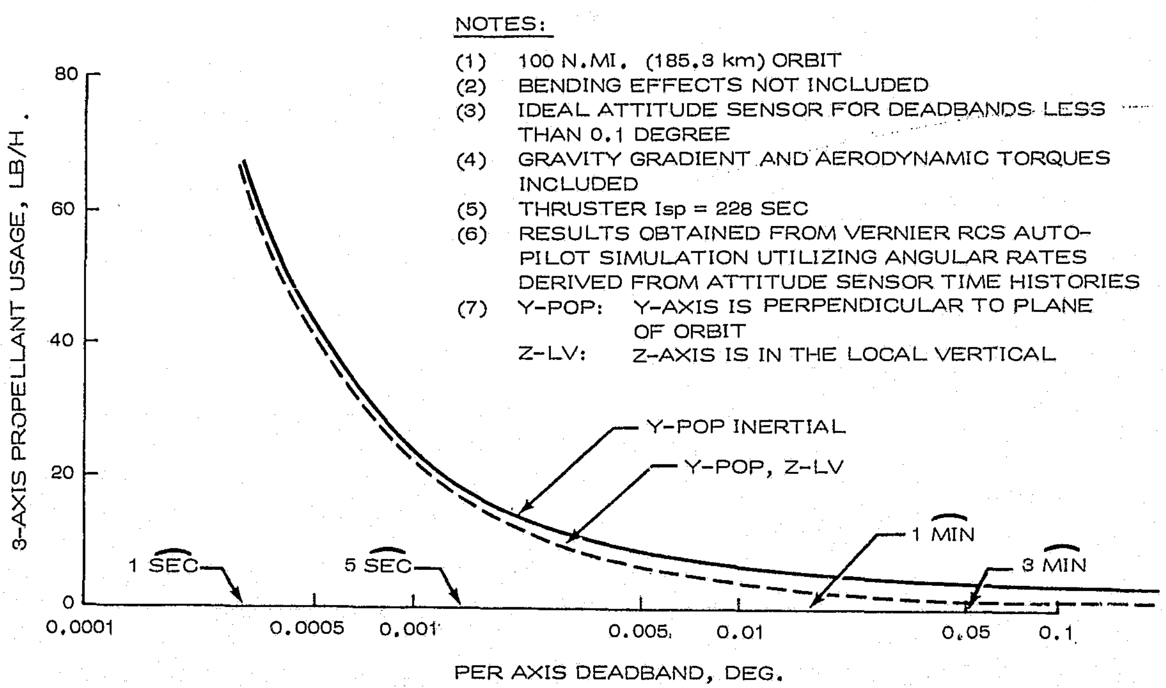


Figure 2 – 16: Effect of Attitude Deadband on Vernier Thruster Propellant for Typical Local Vertical and Inertial Orientation

(Reference, Vol. XIV, Rev. D, Fig. 3 – 14)

Table 2-8: Typical Propellant Usage of Vernier Thrusters for Various Altitudes and Orientation

(Reference, Vol. XIV, Rev. D, Table 3.4)

Propellant Usage[a], lb/h (kg/hour)

Orientation[b]	100 N MI. (185.2 km) Orbit		200 N MI. (370.4 km) Orbit		500 N MI. (926 km) Orbit	
Y-POP[c], Z-LV	0.7	(0.32)	0.6	(0.27)	0.4	(0.18)
Y-POP, Inertial	3.6	(1.63)	3.0	(1.36)	2.4	(1.09)
Z-POP, Inertial	11.9	(5.40)	4.5	(2.04)	3.7	(1.68)
X-POP, Inertial	10.2	(4.63)	2.4	(1.09)	2.0	(0.91)

Notes:

a) Utilizing baselined vernier thruster configuration (six 25 lb thrusters; I_{sp} = 228 sec, thruster minimum on time = 40 m sec.)

Effects of angular rate estimation with 20 arc sec IMU angle quantization and 10 arc sec IMU noise included.

Typical mass properties of Orbiter including a weight of Spacleab plus its payload of 32 000 lb (14 515 kg).

Gravity gradient and aerodynamic torques included, atmospheric density based on a 100 N MI (185.2 km) orbit.

Per axis deadband $\pm$ 0.1 deg.

Neglects venting effects

b) X, Y, Z axes are standard airplane axes with origin at center of mass, X axis forward and Y axis out right wing (Figure 2 - 2).

c) Perpendicular to orbit plane (POP)

d) Local vertical (LV)

Table 2 – 9: Effect of Orbital Altitude on RCS Vernier Porpellant Usage for Various Orbiter Orientations
(Per axis deadband of $\pm$ 0.1 deg)

For definition of Xo, Yo, Zo see Figure 2 – 1.

Pop: perpendicular to orbit plane
(Reference: Vol. XIV, Rev. C; Table 3.4)

Orientation	Propellant Usage lbs/Orbit (kg/Orbit)		
	100 NMI (185 KM)	200 NMI (370 KM)	500 NMI (926 KM)
Yo – Pop and Zo – Local Vertical	0.7 (0.318)	0.6 (0.272)	0.6 (0.272)
Yo – Pop Inertial	4.2 (1.920)	3.9 (1.770)	3.6 (1.630)
Zo – Pop Inertial	13.6 (7.160)	5.4 (2.450)	4.6 (2.090)
Xo – Pop Inertial	13.7 (6.210)	1.0 (0.453)	0.8 (0.363)

2.4.1.4 Translational Maneuvers

As has been pointed out in para. 2.3.1, the primary thrusters of the RCS can also be used for translational maneuvers required for Orbiter/External Tank separation and on orbit translational maneuvers such as rendezvous braking, docking, etc. The propellant consumed for the translational maneuvers is a function of the thruster select logic, cross-coupling compensation, Orbiter weight, incremental velocity, maneuver direction, and maneuver axis. Typical RCS propellant usages for translational maneuvers are presented in Table 2 – 10 as a function of maneuver axis.

2.4.1.5 Rendezvous

The Orbiter has the capability to rendezvous with orbiting payloads that are either cooperative (i.e. responding to signals emitted by Orbiter) or passive. In most cases it will use a multi-orbit and multi-impulse maneuver sequence associated with a parking orbit rendezvous mode, but it is also capable of performing a rendezvous and retrieval in one revolution. The rendezvous limits for cooperative and passive payloads are given in Table 2 – 11. RCS propellant estimates for rendezvous and payload retrieval are typically 1580 lb (717 kg) for a rendezvous and 360 (163 kg) for a payload retrieval maneuver, these figures refer to an Orbiter with a weight of 200 000 lb (90 718 kg).

2.4.1.6 Propellant Usage due to Attitude Constraints

There exist constraints on certain Orbiter attitudes due to limitations of the thermal design of the Orbiter. These constraints depend upon

- the position of the sun with respect to the orbital plane
- orbit altitude and inclination
- required heat rejection by the Orbiter

Details of the thermal by critical Orbiter attitudes are TBD.

Table 2 – 10: Typical RCS Propellant Usage for Orbiter Translational Maneuvers
(Reference, Vol. XIV, Rev. D, Table 3.6)

Maneuver Axis/Direction	Propellant Usage lb/fps (kg/mps)
+ X	27.4 (40.79)
– X	26.7 (40.12)
+ Y	44.5 (66.20)
– Y	44.6 (66.41)
+ Z	21.0 (31.29)
– Z	37.1 (55.17)

- Includes compensation for rotational and translational cross-coupling.
- X, Y, and Z are defined in Figure 2 – 2.
- Includes impingement effects on – Z jets.
- Utilizing baselined RCS primary thruster configuration (minimum on-time = 40 ms).
- Weight of Orbiter is 202 000 lb (91 627 kg) including a weight of Spacelab mass properties plus its payload of 32 000 lb (14 515 kg).
- Attitude hold (± 0.5 deg/axis) maintained during maneuvers.

Table 2 - 11: Rendezvous Limits

(Reference, Vol. XIV, Rev. D, Table 3-11)

Parameter	Type of Orbiting Payload	
	Cooperative (1)	Passive (2)
Range Limit	300 nmi to 100 ft (560 km) (30 m)	10.3 NMI to 100 ft (19 km) (30 m)
Range Rate Limit	TBD	TBD
LOS Angle Limit	$\pm$ 40° (Function of Range)	$\pm$ 40° (Function of Range)
LOS Angle Rate Limit		
(1) Acquisition	$\pm$ 4 mr/sec	$\pm$ 4 mr/sec
(2) Tracking	$\pm$ 5 deg/sec	$\pm$ 5 deg/sec

(1) Requires transponder on orbiting payload compatible with Orbiter radar.

(2) Orbiting payload has an average radar cross section of one square meter.

(3) LOS means: line of sight

Thermally critical attitudes of the Orbiter can only be maintained during limited periods of time and must be followed by durations of thermal conditioning. During the periods (typically 3 hours) of thermal conditioning the Orbiter will rotate at approximately 5 revolutions per hour about the Xo-axis with the orientation of the Xo-axis perpendicular (or within $\pm$ 20 degr.) to the Earth-Sun-Line, or be oriented as preferred thermal attitudes. On-orbit thermal conditioning for up to 12 hours is required prior to re-entry for missions where the temperatures of the Orbiter are outside certain design limits. The propellant usage for thermal conditioning can be derived from Table 2 - 8 and 2 - 9. A typical propellant requirement for a three-hour period of thermal control is 7.2 kg (16 lb) assuming that a roll rate of 0.25 deg/sec will be established and stopped about the Xo-axis. Propellant for pitch and yaw attitude control during thermal conditioning is assumed to be approximately the 1.63 kg/hour (3.6 lb/hour) value shown in Table 2-6 for X-POP inertial attitude hold. It may be assumed that the propellant required to maintain the roll rate is negligible.

2.4.2 Attitude Control Performance

2.4.2.1 Principle of Orbiter Pointing

The Orbiter contains a structural reference, referred to as Navigation Base. For Orbiter pointing this Navigation Base is related to an inertial reference which is derived in the Inertial Measurement Unit (IMU). The IMU contains gyros whose accuracy can be up-dated by star trackers. The Navigation Base and the IMU with its star trackers are located in the forward end of the Orbiter. The Orbiter Guidance, Navigation and Control System has the capability of pointing any vector defined in the Orbiter Navigation Base Axis System at any desired inertial, earth fixed or orbiting target or in the direction of the local vertical. Pointing accuracy for inertial or earth referenced directions is within a $\pm$ 0.5 degree (3-sigma) half cone angle. Pointing accuracy of continuous pointing is dependent upon thermal attitude constraints mentioned in Section 2.4.1.6 and upon the drift of the IMU.

The pointing accuracy specified above when utilizing the Orbiter IMU for Spacelab payload pointing does not include orientation alignment uncertainty between the Orbiter Navigation Base and, for example, a Spacelab payload. This alignment uncertainty can be greater than 2 degrees. In order to minimize the effect of this uncertainty the Orbiter Guidance, Navigation and Control System is capable of accepting compatible attitude information from a Spacelab payload supplied and Spacelab mounted sensor of comparable accuracy to the Orbiter IMU. Details of the interfaces between the Spacelab payload supplied sensor and the Orbiter are TBD. Utilizing this information, the Orbiter Guidance, Navigation and Control System is capable of pointing a vector defined in the sensor-fixed reference axis system at any direction defined above to within the same pointing accuracy. Duration of pointing will depend upon the drift characteristics of the Spacelab payload sensor.

2.4.2.2 Pointing Stability

For Spacelab payload pointing utilizing the Vernier Thrusters, the Orbiter Flight Control System provides a stability (deadband) of ± 0.1 deg/axis and a stability rate (maximum limit cycle rate) of ± 0.01 deg/sec/axis when no Vernier Thrusters have failed. When using the primary thrusters, the Orbiter Flight Control System is capable of providing a stability of ± 0.1 deg/axis and a stability rate of ± 0.1 deg/sec/axis. For pointing and/or stability requirements beyond the capability of the Orbiter, the Orbiter is capable to accept compatible commands from a Spacelab payload supplied and Spacelab mounted stabilization and control system.

2.4.2.3 Pointing Accuracy

The Orbiter capability to point a vector defined in the navigation base axes utilizing the Orbiter IMU for attitude information is summarized in Table 2 - 12 and described below:

a) IMU Inertial Attitude Hold: The error in inertial pointing utilizing the Orbiter IMU includes

- errors due to the deadband (± 0.1 deg/axis) of the Flight Control System
- errors due to the IMU alignment uncertainty of ± 0.133 deg/axis (3 sigma)
- read-out errors of the IMU (± 0.073 deg/axis, 3 sigma)
- drift rate of IMU (± 0.105 deg/hour/axis, 3 sigma)

Based upon these values, a vector defined in the Orbiter navigation base axes may be maintained to an inertial pointing accuracy of ± 0.5 deg for durations up to 1.0 hour, subsequent to which IMU realignment is required. Active IMU realignment can require interruption of attitude hold for durations up to 15 minutes and the Orbiter may require maneuvering to acquire the necessary stars. It it possible to realign the IMU during the sunlit part of the orbit, but this is a function of the stars available for the Orbiter star tracker(s) to acquire. Pointing duration can be extended beyond one hour by IMU inflight calibration (i.e. IMU realignment without interruption of attitude hold provided the necessary stars are within the field of view of the Orbiter star trackers).

For the second case shown in Table 2 – 12 (augmented inertial) the Orbiter star trackers are continuously tracking a suitable star pair which permits frequent updates of the IMU reference. For this case the attitude error due to drift is essentially eliminated; thus, the vector defined in the navigation base axes may be maintained to within $\pm 0.44^{\circ}$ of the desired direction for an indefinitely long period of time (determined by other factors such as propellant consumption, thermal conditioning and heat rejection requirements, etc.).

b) Earth Target and Local Vertical Pointing:

The error of local vertical and earth surface fixed target pointing includes the inertial pointing error sources described above, as well as additional errors due to Orbiter position uncertainty and Flight Control System tracking capability. Orbiter position accuracy is a function of orbital geometry with respect to ground tracking stations or TDRS tracking satellites and varies with orbital altitude and time. Based upon a 3-sigma navigation uncertainty of 2070 ft utilizing TDRS tracking with the Orbiter in a 100 n.mi. circular orbit (Table 2 – 6), continuous earth-surface-fixed-target pointing can be maintained for durations up to 0.5 hours after IMU realignment (Table 2 – 12). Continuous local vertical pointing to the required accuracy can be maintained for durations up to one hour after IMU realignment (Table 2 – 12). Pointing duration can possibly be extended beyond this time by IMU in-flight calibration and/or passive IMU realignment. The effects of navigation errors on earth target pointing as a function of orbital altitude and viewing angle are presented in Table 2 – 13.

c) Orbital Object Pointing: TBD

Table 2 – 12: Pointing Accuracy (Half-Cone Angle) Utilizing Orbiter IMU (Reference: Vol. XIV, Ref. D, Table 3.2)

Type of Pointing	Half-Cone Angle Pointing Accuracy (3-Sigma)	Pointing Accuracy Degradation Rate (3-Sigma)	Duration Between IMU Alignments
Inertial	± 0.5 deg	0.105 deg/hr/axis	1.0 hours
Augmented Inertial	± 0.44 deg	0	N/A
Earth-Surface-Fixed Target*	± 0.5 deg	0.105 deg/hr/axis	0.5 hours
Orbital Object	TBD	TBD	TBD
Local Vertical*	± 0.5 deg	0.105 deg/hr/axis	1 hour

*Tracking with TDRS, 100 n mi. (185 km) circular orbit.

Table 2 - 13: Pointing Errors for Earth Targets (one Sigma Values)
(Reference: Vol. XIV, Rev. D, Table 3.3)

	Orbiter Altitude		
	100 N MI. (185.2 km)	200 N MI. (370.4 km)	300 N MI. (555.6 km)
	DEG		
A) Local Vertical Pointing:			
o STDN	0.16	0.16	0.16
o TDRS	0.16	0.16	0.16
B) Earth Target Pointing:			
o Looking Vertical			
- STDN	0.18	0.16	0.16
- TDRS	0.28	0.20	0.18
o Looking 30° Off Vertical			
- STDN	0.20	0.17	0.16
- TDRS	0.29	0.20	0.18

2.4.2.4 Attitude Disturbance by Spin-Up and Release of Payloads

The angular momentum perturbation imparted to the Orbiter by a released Spacelab payload is a function of the payload mass and moment of inertia about the axis of rotation and the angular or spin-up velocity of the payload. The Orbiter should not be required to maintain a $\pm$ 0.5 degree pointing accuracy during the payload spin-up and release period because the induced perturbations on the Orbiter during this period could be large. Spin-up and separation system requirements and deployment mechanism design requirements need to be studied in detail, because of potential impact on Orbiter.

2.4.3 Passive Attitude Control

The Orbiter can also operate in either a free drift or (possibly, depending upon the magnitude and direction of disturbances resulting from crew motion and venting) a passive gravity gradient stabilized mode to satisfy acceleration levels below 10^{-4} g. A passively stable gravity gradient drift mode ($\pm$Xo-axis along local vertical) would only experience thermal constraints on attitude hold duration for angles between orbital plane and Earth-Sunline equal to or greater than 60 degrees. Star trackers of the Orbiter can passively keep the IMU platform aligned to within 1 degree as long as the field of view of the star trackers is kept on or above the local horizontal and suitable star pairs are available. A gravity gradient attitude of a Xo - downward along the local vertical and $\pm$ Zo perpendicular to the orbital plane would, therefore, have either no or occasional thermal constraints, and be compatible with the star tracker field of vieweld of constraint for passive platform alignment.

Translational accelerations due to the atmospheric drag acting on the Orbiter while in a free drift mode are given in Section 5. Drift mode translational acceleration level time histories could, however, also be expected to be affected by other mission dependent variables which include venting forces, disturbances from crew movement, orbit altitude, Orbiter orientation, and attitude control changes due to communication requirements. Experiment timeline and crew timeline constraints also need to be known before total meaningful attitude hold duration capabilities and requirements can be specified.

2.5 Crew

2.5.1 Crew Tasks and Crew Size

The Orbiter crew consists of the commander and pilot who are always required to operate and manage the Orbiter. Additional crewmen who may be required to conduct Orbiter/Spacelab payload operations are a mission specialist and one or more payload specialists. The duties of the crew are described in Section 6.

The crew size will be a function of the mission complexity and duration, but the maximum crew is seven persons: command, pilot, mission specialist and 4 payload specialists.

For Spacelab flights for a continuous 24 hour operation a total crew of 4 is required: commander and pilot to monitor and control Orbiter and Spacelab subsystems in alternating shifts, mission specialist and a payload specialist to serve as Spacelab crew for experiment operation in alternating shifts.

It is foreseen, that for each crew-member a sleep cycle of 8 hours is followed by an awake cycle of 16 hours. 8 1/2 to 10 1/2 hours of productive work can be expected within 16 hours awake time. Crew cycles will be arranged such, that an overlap for all crew-members of approximately 8 hours will be achieved. This will give convenient time each day for the total crew for briefings, flight plan updates, checklist reviews etc. See Figure 2 - 17.

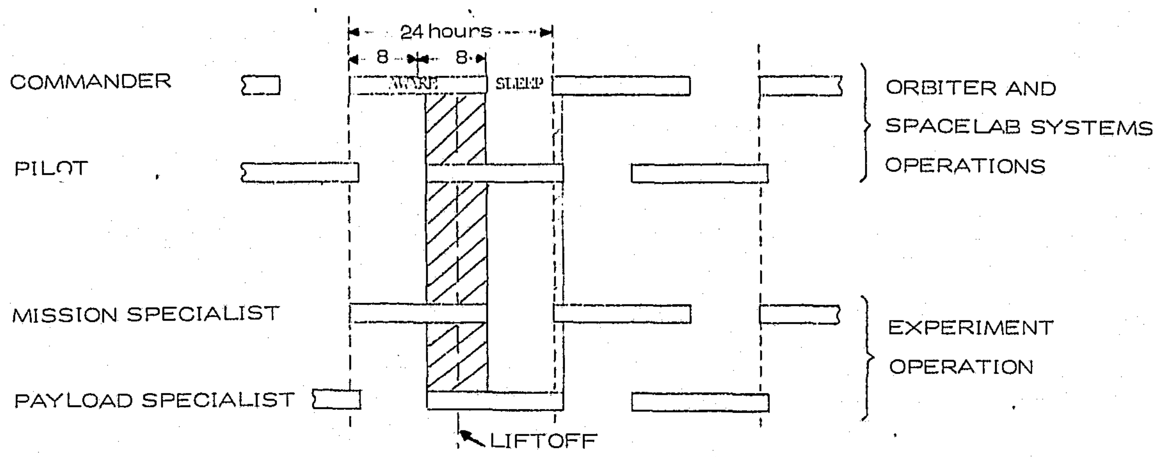


Figure 2 - 17: Crew Timeline for Continuous 24-Hours-Operation, Minimum Total Crew of 4

2.5.2 Crew Compartment and Accommodation

The orbiter crew compartment consists of a two-level cabin, the flight deck and the mid-deck as shown in Figure 2-18 through 2 - 21.

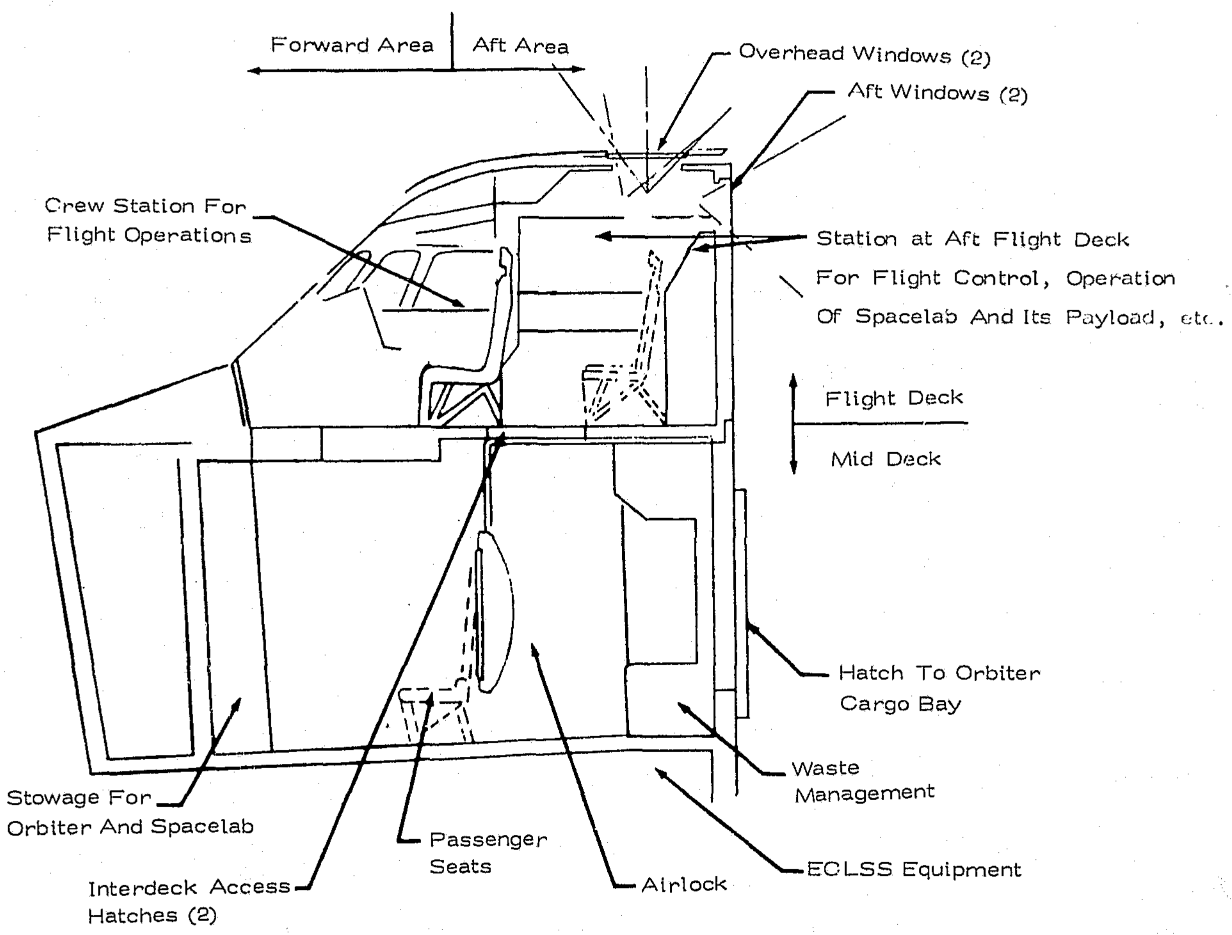


Figure 2-18: Crew Compartment Concept

The forward area of the flight deck is dedicated primarily to Orbiter operations during ascent and re-entry with displays, controls and seats for the Commander and Pilot.

The aft area of the flight deck contains two seats for a mission and a payload specialist during ascent and re-entry. These seats are removed for on-orbit operations. The aft flight deck contains controls and displays for Orbiter systems operations, Spacelab subsystem operation and Spacelab experiment operation.

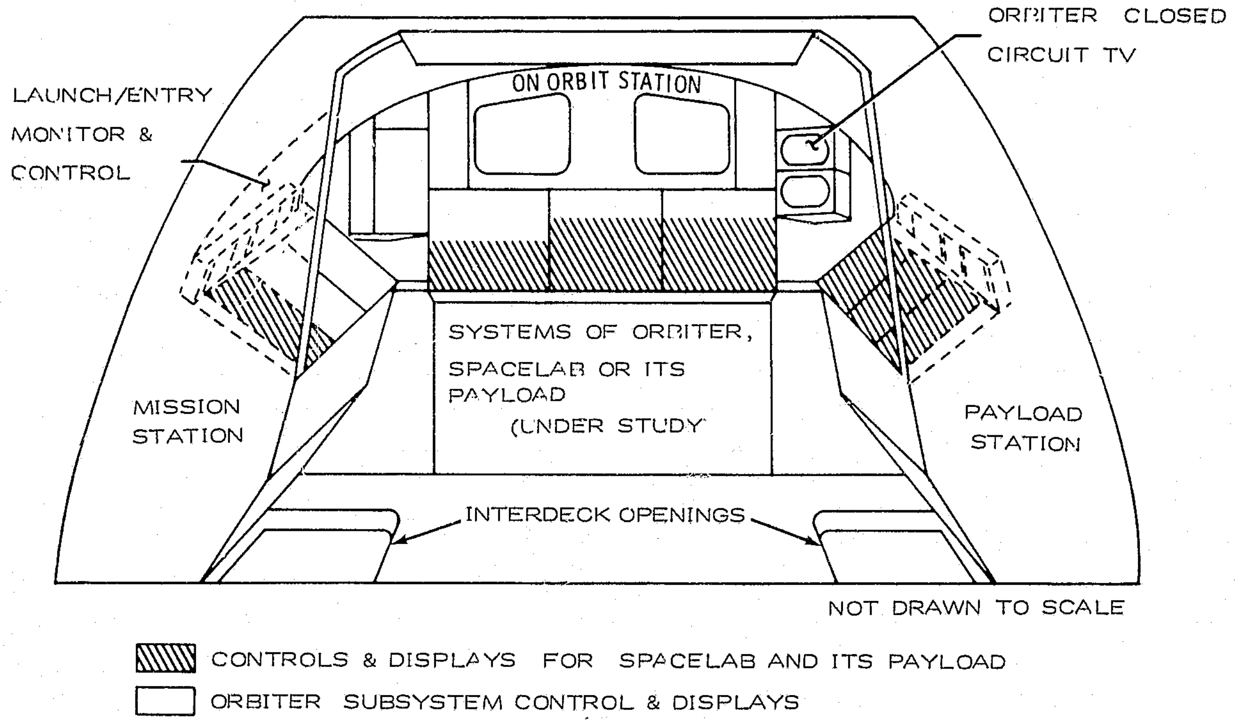


Figure 2-19: Crew Station at Aft Flight deck, view looking aft (conceptual layout)

The mid-deck area includes sleep stations, galley, waste management and loose equipment stowage provisions.

It is presently foreseen, that the Spacelab subsystem will be operated from the mission station, using Orbiter and Spacelab control and display equipment to access the system.

The Payload station and part of the on-orbit station are dedicated to experiment operations. Standard 19" openings will accommodate Spacelab and experimenter provided control and display equipment in support of experiment operations.

In the present conceptual design of the aft flight deck the panel surface area, volumes and panel depth allocated for Spacelab and its payload are 2.04 m^2 (22 ft^2), 0.99 m^3 (35 ft^3), and 0.508 m (20 inches), respectively. The heat rejection capability and power available to Spacelab and payload equipment at the aft flight deck is limited. The nominal heat rejection capability at the aft flight deck is between 0.35 kW and 0.75 kW depending on the heat rejection required in Spacelab. If 0.35 kW (0.75 kW) is available at the aft flight deck, 8.5 kW (8.1 kW) heat can be rejected in Spacelab. Peak heat rejection at the aft flight deck is 1 kW for 15 minutes every 3 hours. The nominal and peak power available at the aft flight deck is 0.75 and 1 kW respectively. The allocation of all the resources (panel surface area, volume, heat rejection and power) between Spacelab subsystems and payload at the aft flight deck is mission dependent and is currently TBD.

In the presently preferred concept for connecting Spacelab with the Orbiter the airlock indicated in Figures 2 – 18 through 2 – 21 will be removed from the mid-deck and placed into the Orbiter cargo bay to permit simultaneous operation of Spacelab and extra vehicular activities (EVA).

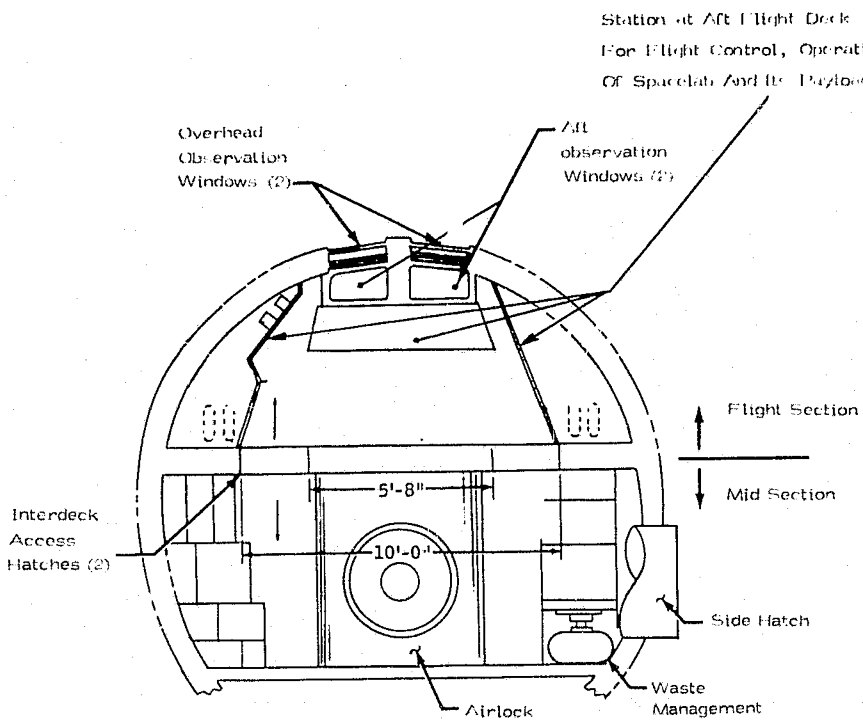


Figure 2–20: Crew Compartment Concept–Looking Aft

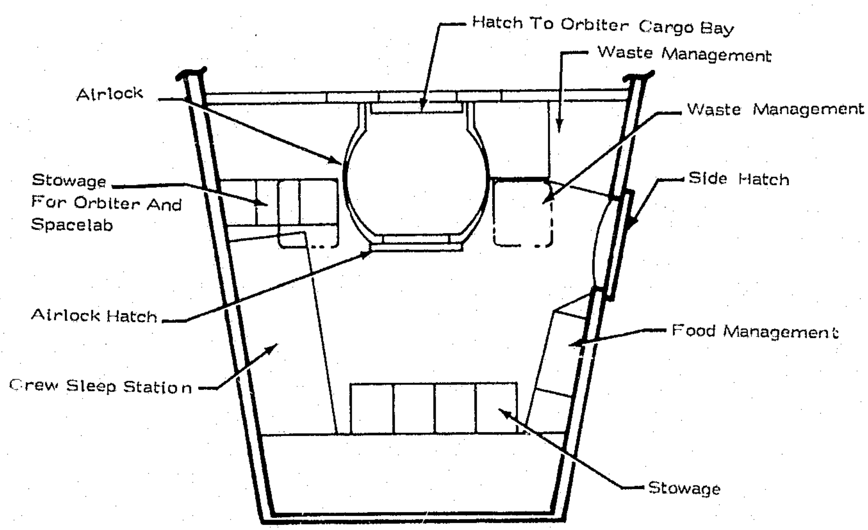


Figure 2–21: Crew Compartment Concept Mid-Deck

A total volume of approximately 150 cubic feet (4.24 m^3) will be provided in the crew compartment for loose equipment stowage of which approximately 95 percent will be on the mid-deck. Loose equipment includes those items which are not permanently mounted in the cabin. The allocation of containers for loose equipment of Spacelab is mission dependent. Any excess stowage capacity available above the Orbiter requirements may be utilized for stowage of loose equipment of Spacelab and/or its payload.

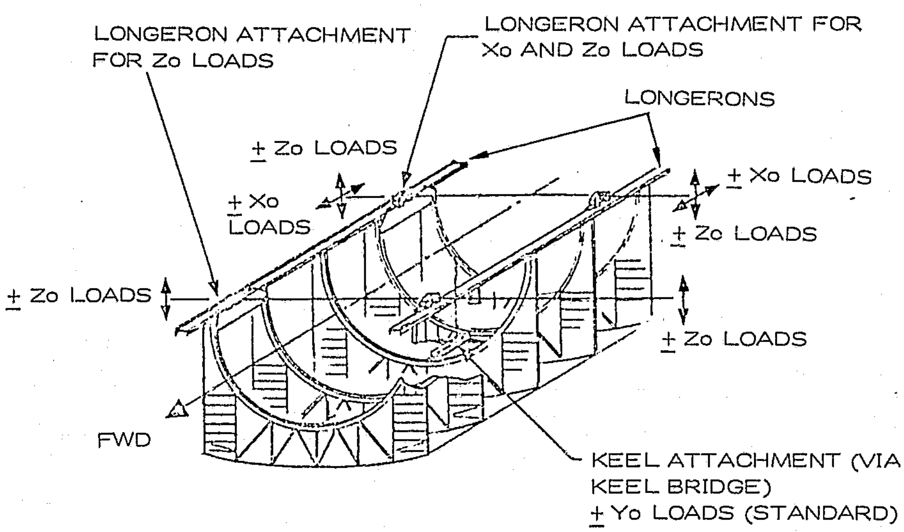
Expendables, such as oxygen, food, LiOH-canisters for CO_2 removal etc. to support the crew in orbit are supplied by the Orbiter (para 2.6.1.3).

The Orbiter provides - at no weight penalty to the Spacelab payload - for 28 mandays of expendables plus expendables for rescue operations for four men and for a duration of 4 days. In addition volume is provided for expendables for up to 42 mandays. Weight and volume above the outlined provisions will be charged to the Spacelab payload. Table 2-16 gives a survey of the items and services charged to the Spacelab payload.

2.6 Orbiter Support to Spacelab

2.6.1 Structural Support

The Orbiter provides structural attachment points along the Orbiter cargo bay. The Spacelab module and pallets are attached to the Orbiter via these attachments points. There are three sets of attachment points, one at the bottom (keel) of the Orbiter cargo bay and two on longerons on both sides of the Orbiter cargo bay at Zo = 414 inches. The keel attachment points are designed to take loads in the Yo direction only, while the attachment points on the longerons are able to take primarily loads in the Xo and Zo-directions. Figure 2-22 illustrates the attachment system. Table 2 - 14 lists the Xo-location of attachment points available to Spacelab. Table 2 - 15 lists additional attachment points which will be available on a longeron when the Remote Manipulator System is not installed on that same longeron.



NOTE: THE TWO TYPES OF LONGERON ATTACHMENTS ARE FUNCTIONALLY INTERCHANGEABLE.

Figure 2 – 22: Orbiter Attachment System

Figure 2 – 14: Attachment Points – Longeron and Keel

Xo Location – Inches (mm)

Attach Point No.	Inches	(mm)	Attach Point No.	Inches	(mm)
155	612.73	(15563.4)	208	821.20	(20858.5)
156	616.67	(15663.3)	209	825.13	(20958.4)
157B	619.00	(15722.6)	210B	829.07	(21058.3)
158	624.53	(15863.1)	211	833.00	(21158.2)
159	628.47	(15963.1)	212	836.93	(21258.1)
160	632.40	(16063.0)	213	840.87	(21358.0)
164B	649.00	(16484.6)	214	844.80	(21457.9)
165B	652.07	(16562.5)	215D	848.73	(21557.8)
166	656.00	(16662.4)	216D	852.67	(21657.7)
167	659.93	(16762.3)	217	856.60	(21757.6)
178B	703.20	(17861.3)	221LB	872.33	(22157.3)
179	707.13	(17961.2)	222	876.27	(22257.2)
180	711.07	(18061.1)	223	880.20	(22357.1)
181B	715.00	(18161.0)	224	884.13	(22457.0)
182B	718.93	(18260.9)	225	888.07	(22556.9)
183	722.87	(18360.8)	226B	892.00	(22656.8)
184	726.80	(18460.7)	227	895.93	(22756.7)
185	730.73	(18560.6)	228D	899.87	(22856.6)
186D	734.67	(18660.5)	229K	903.80	(22956.5)
187K	738.60	(18760.4)	230K	907.73	(23056.4)
188	742.53	(18860.3)	231K	911.67	(23156.3)
192L	758.27	(19260.0)	234L	923.47	(23456.0)
193	762.20	(19359.9)	235L	927.40	(23556.0)
194	766.13	(19459.8)	236	931.33	(23655.9)
195	770.07	(19559.7)	237B	935.27	(23755.8)
196B	774.00	(19659.6)	238	939.20	(23855.7)
197B	777.93	(19759.5)	239	943.13	(23955.6)
198D	781.87	(19859.4)	240	947.07	(24055.5)
199D	785.80	(19959.3)	241B	951.00	(24155.4)
200	789.73	(20059.2)	242B	954.93	(24255.3)
201	793.67	(20159.1)	243	958.87	(24355.2)
202	797.60	(20259.0)	244D	962.80	(24455.1)
203L	801.53	(20358.9)	245K	966.73	(24555.0)
207	817.27	(20758.6)			
246D	970.67	(24654.9)	286	1128.00	(28651.2)
247L	974.60	(24754.8)	287	1131.93	(28751.1)
251B	990.33	(25154.5)	289L	1135.87	(28851.0)
252	994.27	(25254.4)	290L	1139.80	(28950.9)
253	998.20	(25354.3)	292K	1151.60	(29250.6)
254	1002.13	(25454.2)	293K	1155.53	(29350.5)
255	1006.07	(25554.1)	294K	1159.47	(29450.5)
256B	1010.00	(25654.0)	295K	1163.40	(29550.4)
257	1013.93	(25753.9)	296B	1167.33	(29650.3)
258	1017.87	(25853.8)	297	1171.27	(29750.2)
259	1021.80	(25953.7)	298	1175.20	(29850.1)
260	1025.73	(26053.6)	299	1179.13	(29950.0)
261D	1029.67	(26153.5)	300LB	1183.07	(30049.9)
262D	1033.60	(26253.4)	301B	(TBD)	
266L	1049.33	(26653.1)	304	1198.80	(30449.5)
267	1053.27	(26753.0)	305D	1202.73	(30549.4)
268	1057.20	(26852.9)	306D	1206.67	(30649.3)
269B	1061.13	(26952.8)	307	1210.60	(30749.2)
270	1065.07	(27052.7)	308	1214.53	(30849.1)
271	1069.00	(27152.6)	309B	1218.47	(30949.1)
272	1072.93	(27252.5)	310B	1222.40	(31049.0)
273B	1076.87	(27352.4)	311	1226.33	(31148.9)
274	1080.80	(27452.3)	312	1230.27	(31248.8)
275L	1084.73	(27552.2)	313LB	1234.20	(31348.7)
276L	1088.67	(27652.1)	314L	1238.13	(31448.6)
277L	1092.60	(27752.0)	315L	1242.00	(31548.5)
279K	1100.47	(27951.9)	316L	1246.00	(31648.4)
280K	1104.40	(28051.8)	322L	1269.60	(32247.8)
281	1108.33	(28151.7)	323L	1273.53	(32347.7)
282	1112.27	(28251.6)	324LB	1277.47	(32447.6)
283	1116.20	(28351.5)	325L	1281.40	(32547.6)
284B	1120.13	(28451.4)			
285B	1124.07	(28551.3)			

NOTES: All attach points normally available are included in this list.
Attach points designated 'L' (e.g. number 192L) are available only at the longerons.
Attach points designated 'K' (e.g. number 295K) are available only at the keel.
Potential orbiter structural interference exists at locations designated 'P' (stability brackets) and 'D' (door drive support points).

Table 2 - 15 Additional Longeron Attachment Points
Xo Locations - Inches (mm)

Condition	Attachment Point No.	Inches	(mm)
Available when Manipulator not Installed	168	663.86	(16 862.2)*
	169	667.80	(16 962.1)*
	170	671.73	(17 062.0)
	171	675.67	(17 162.0)
	172	679.60	(17 261.9)
	173	683.53	(17 361.8)
	293 L	1155.53	(29 350.5)
	294 L	1159.47	(29 450.4)*
	295 L	1163.40	(29 550.4)*

*These points are not recommended for use

2.6.1.2 Electrical Power and Energy

Hydrogen/oxygen fuel cells provide the DC electrical energy for the Orbiter and Spacelab. The required fuel is stored in tank sets, referred to as energy kits, each energy kit providing approximately 840 kwh. The Orbiter baseline provides only 50 kwh of electrical energy for Spacelab use; the weight of one additional energy kit is included in the Spacelab design so that 890 kwh are available to Spacelab and its payload. Volume for three additional energy kits will be provided outside the dynamic envelope (Figure 2 - 4) of the Orbiter cargo bay. Further energy kits may be added, but they must be located within the dynamic envelope and, therefore, result also in a volume penalty for the Spacelab payload. The dry and wet weight of additional energy kits will be charged to the landing and launch weight of the Spacelab payload, respectively (para 3.). The weight of the fuel has to be duly accounted for in mission planning and in the assessment of the center of gravity.

Although additional energy kits may be used to increase the electrical energy available to the Spacelab payload, it is pointed out that the use of electrical power must be consistent with the available heat rejection capability (para. 2.6.1.3 c).

2.6.1.3 Environment Control and Life Support

The Orbiter ECLS Subsystem provides for the environment to support a shirtsleeve operation within the pressurized cabin of the Orbiter during all mission phases. The ECLS Subsystem will perform the function of:

a) Atmospheric Revitalization

b) Food, Water, and Waste Management Service

c) Active Thermal Control

d) Fire Suppression

The Orbiter provided expendables are outlined at the end para 2.6.2 and in Table 2-16.

a) Atmospheric Revitalization Subsystem (ARS)

The ARS furnishes a shirtsleeve environment at sea level pressure by controlling CO_2, humidity, odor, pressure, oxygen/nitrogen and temperature for the Orbiter cabin and cabin located equipment, and compensates for atmosphere leakage from the Orbiter cabin.

b) Food, Water and Waste Management Subsystem (FWW)

The food management section consists of a galley area for food preparation, food and equipment storage, hot and cold water dispensers, and waste storage.

The water management section stores, distributes and disposes of excess water generated by the fuel cells (para 2.6.1.1); excess water is dumped overboard in a non-propulsive fashion, at predetermined times, depending on mission constraints. Waste water is collected and stored.

The waste management section accumulates solid waste and collects, transfers, and stores liquid wastes. Urine and condensate are collected, separated from air, and stored for return to earth in three waste tanks of 75 kg (165 lbs) capacity each.

c) Active Thermal Control Subsystem (ATCS)

The heat generated by Spacelab and its payload is dissipated in Spacelab supplied coolant loops, transferred to the coolant loops of ATCS via a heat exchanger and finally transferred to space. The ATCS will provide a nominal on-orbit heat rejection of 8.5 kw for Spacelab and its payload with the doors of the Orbiter cargo bay open. This level of heat rejection capability is the maximum the Orbiter can supply. It is achieved by supplementing the basic Orbiter ATCS (6.3 kw capability) with a heat rejection kit which is included in the basic Spacelab weight, i.e. the increased heat rejection capability is not weight chargeable to the Spacelab payload.

d) Fire Suppression

The Orbiter provides portable fire extinguishers containing Freon 1301. Spacelab equipment located in the Orbiter cabin must be compatible with the use of Freon 1301.

2.6.1.4 EVA and Rescue Accommodation

The Orbiter provides the capability for Extra-Vehicular Activity (EVA) and rescue. The equipment and expendables required to support three, two-man EVA operations is supplied by the Orbiter. Two of these three operations may be utilized by Spacelab for either planned or unscheduled EVA operations, the third operation is for rescue. Additional EVA operations in support of Spacelab and/or its payload may be provided with the expendables being provided as items which will be charged to the weight of the Spacelab payload.

The prime mode of rescue is EVA. The Orbiter provides the equipment to support the rescue operation for 4 persons; 2 persons use EVA equipment for rescue and the 2 other persons use kits of a specially designed personal rescue system. The Orbiter provides expendables for 4 persons for 4 days as contingency for rescue operations (end of para 2.6.2 and Table 2-16). For any persons above four, the Spacelab payload will be charged for the weight of required kits of the personnel rescue system and expendables to support these persons.

2.6.1.5 Avionics

The Orbiter avionics provides for:

a) Receiving, transmission and distribution of voice
b) Transmission of operational telemetry
c) Receiving and transmission of Spacelab data (including payload data)
d) Transmission of commands from the ground or Orbiter to Spacelab CDMS subsystem
e) Furnishing Guidance, Navigation and Control data to Spacelab or its payload
f) Transmission and distribution of television signals
g) Tracking of active and passive targets
h) Transmission and reception of EVA data and voice

The Orbiter avionics provides also the interface between the Orbiter and:

a) Tracking and Data Relay Satellite (TDRS)
b) Space Tracking and Data Network (STDN)
c) Spacelab
d) EVA crewmen

e) Other space vehicles

f) Landing site facilities of the Orbiter

Details of the Orbiter avionics are discussed in connection with the Spacelab CDMS subsystem (para. 4).

2.6.1.6 Remote Manipulator System

The Remote Manipulator System and its installations in the Orbiter are shown in Figure 2 - 23 and 2 - 24. The Orbiter provides one manipulator 50 ft (15 240 m) in length on the left side of the Orbiter. In orbit the manipulator is capable of removing and installing a 15 ft (4.572 m) diameter, 60 ft (18.288 m) long, 65.000 lb (29.510 kg) object. The Remote Manipulator System does not infringe on the dynamic envelope to Spacelab and its payload (Figure 2 - 4).

The manipulator provides a light (tentative data: 150 Watts with a field of view cone no less than 60 degrees) for illumination and a TV camera (TBD) for remote viewing. Their locations on the manipulator are TBD.

If not required for a particular mission the Remote Manipulator System may be removed to provide additional payload weight capability, provided compensations made for the effect on the Orbiter center of gravity.

A second manipulator arm (Figure 2 - 23) can be installed if required. The weight of the second manipulator is chargeable to the payload of Spacelab.

The capability is provided to operate two Remote Manipulator Systems in serial-only (non simultaneous) operations. However, it is possible to hold or lock one manipulator arm while operating the other one.

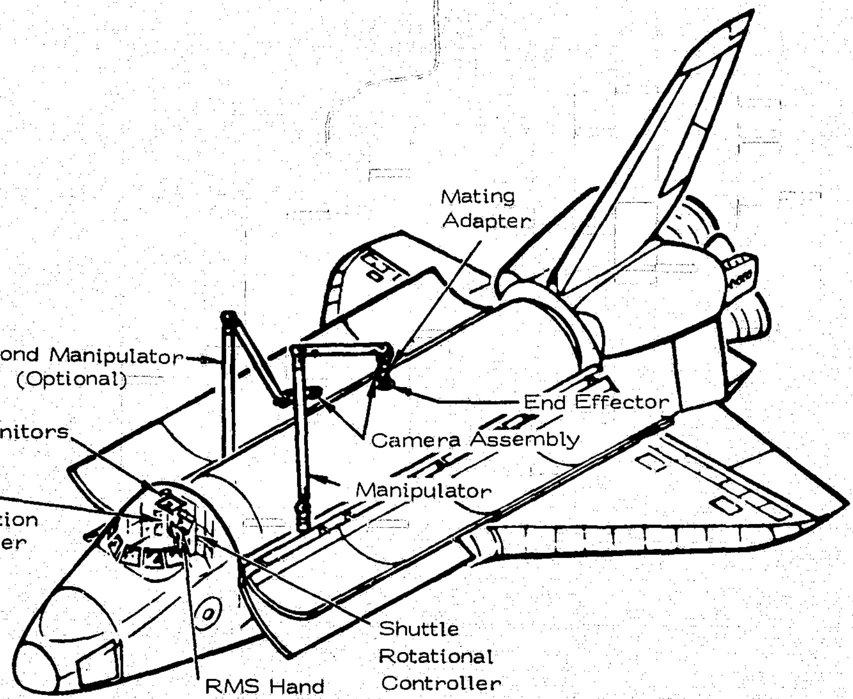


Figure 2 - 23: Orbiter Remote Manipulator System
(Reference, Vol. XIV, Rev. D, Figure 8.3)

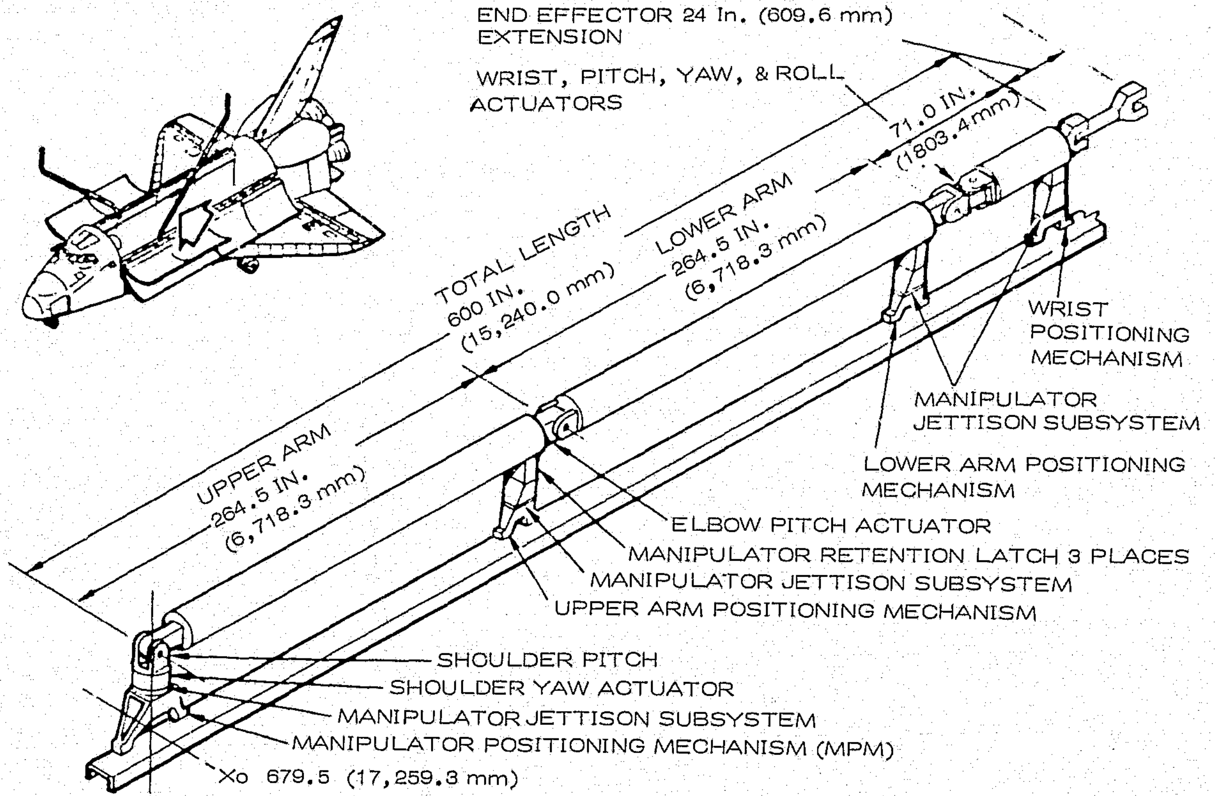


Figure 2 – 24: Manipulator Arm of the Remote Manipulator System Reference, Vol. XIV, Rev. D, Figure 8.1)

The RMS will be used only in zero g. The RMS reach and viewing capabilities are illustrated in Figure 2-25. However, the information presented in this figure does not account for the geometrical constraints imposed by Spacelab and its payload.

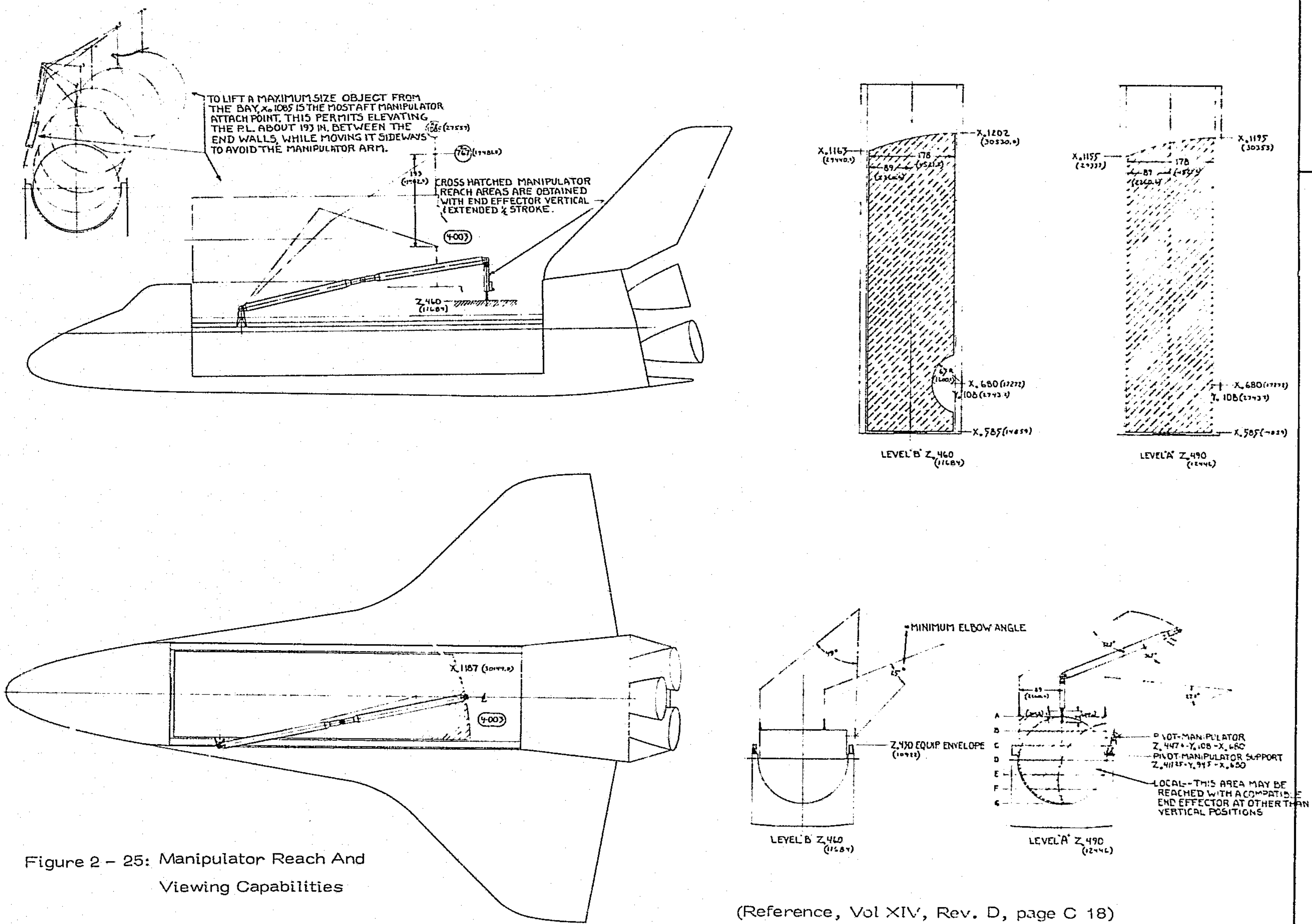


Figure 2 - 25: Manipulator Reach And Viewing Capabilities

(Reference, Vol XIV, Rev. D, page C 18)

2.6.2 Summary of Orbiter Services available to Spacelab Payloads

Services to the Spacelab payload in excess of those provided for in the Orbiter and Spacelab baselines are available at the discretion of the user(s). The weight, volume, etc. of the items required to render those additional services will be charged to the Spacelab payload and appropriate care must be taken to ensure that all items are correctly included in mission planning. Stringent control will be required to ensure compatibility with weights, center of gravity, volumeric and other constraints imposed by the Orbiter.

The services available to the Spacelab payload in excess of those provided for in the Orbiter baseline have been briefly outlined in the previous sections and are summarized in Table 2 – 16. Weights, center of gravity location and volume are in the process of being placed under formal control. The first column of Table 2 – 16 describes the services/items which may be charged to the Spacelab payload if required by the user; the second column describes the Orbiter baseline provided services/items which are not charged to the Spacelab payload. The third column indicates in general terms the items/services which are charged to the Spacelab payload. The fourth and fifth column presents the weight charged to the Spacelab payload weight at launch and landing, respectively. Figures for volume and center of gravity of Spacelab payload charged items are given in column 6 through 9. The last column refers to the sections providing more detailed information.

Items/Services	Orbiter provided/charged		Chargeable to Spacelab Payload to launch weight (lb)	to landing weight (lb)	to volume (ft³)	Center of Gravity (inches) Xo	Yo	Zo	Reference Section
• Crew	4 men	above 4 (maximum 7 total)	170 / men		possible locations: rescue only	490 490 490 490 520 520	48 28 8 - 12 - 43 - 63	349 349 34 349 351 351	2.5.1
• Crew Related									
Crew expendables (oxygen, food, LiOH, hygiene etc.)	28 MD[1)] supplies + 16 MD contingency	above 28 MD	10 MD		--	T B D			2.6.1.3
Seat & Restraints	4 men	above 4 (maximum 7 total)	54 / men		possible locations: for rescue only	494 494 494 494 524 524	48 28 8 -12 -43 -63	340 340 340 340 342 342	2.5.2
Storage Containers	42 MD storage	above 42 MD	TBD		TBD	435 479	-10 5	370 315	2.5.2
Stowage Packaging			30 % of each stowed item						2.5.2
Crew Equipment (helmet, garnet, food tray etc.)	for 4 men	above 4 (maximum 7 total)	45 / men		TBD				2.5.2
Waste Management	Waste water of 28 MD	Waste water above 28 MD	0	TBD	--				2.6.1.3.b
	Waste water storage for 42 MD + 16 MD contingency	above 42 MD	TBD		TBD				
• EVA & Rescue									
EVA Hardware	2 men capability with expendables for 3 EVA's (1 EVA per man is reserved for rescue)	Hardware above a 2 men capacity (incl. one charge of expendables)	203 / hardware set		TBD				2.6.1.4
		Expendables above Orbiter provision	20 / EVA		TBD				
Personal Rescue System	2 men capacity (this is sufficient for 4 men because 2 men use EVA for rescue)	One for each crew man above 4	43.5 / men		TBD	T B D			2.6.1.4
Contingency of Expendables for Rescue Operations	4 men for 4 days	above 4 men for 4 days	7 / MD		TBD				2.6.1.4
• Structural Support		Spacelab payload dedicated support	51 / attachment point						2.6.1.1
• Orbiter Services Spacelab Atmosphere Revitalization	TBD	TBD	TBD		TBD				4.3
• Heat Rejection	Nominal 8.5 kW heat rejection during orbital operations - kit increasing heat rejection from 6.3kW to 8.5kW charged to Spacelab		-		-				2.6.1.3.c
• Orbiter Cabin Leakage	7 days plus 4 days for rescue contingency	beyond 7 days flight duration	7 / day	-	TBD				2.6.1.3.d
• Electrical Energy (for Spacelab Subsystems and Payload)	890 kWh, i.e. 50 kWh Orbiter provided plus 1 energy kit (840 kWh) charged to Spacelab	additional energy kits of 840 kWh each, first 3 add'l kits - above 3 kits, 873 lbs fuel per kit.	1632/kit TBD	759/kit + TBD unusable and unused fuel TBD					2.6.1.2
• Remote Manipulator System	One manipulator on left side of the Orbiter	second manipulator on right side of the Orbiter	867		-	935.8	84.0	436.1	2.6.1.6
• Propulsion									
Reaction Control System (RCS)	Tankage capacity	Propellant for on-orbit attitude control:	3993 max., limited by existing tankage capacity	0	-				2.4.1
• Orbit Maneuvering System (OMS)	Integral Tank	One OMS kit	15 508	3 135	defined in Figures 2-9 to 2-11	1246.5 1243.1	4.4 6.2	377.9 343.7	
		Two OMS kits	28 841	4 095		1256.4 1244.9	- 2.0 -3.0	368.9 365.9	
		Three OMS kits	42 264	5 145		1244.4 1243.0	1.6 2.1	389.7 390.7	
• Avionics wide band communication	Single steerable high-gain antenna	Switching electronics and second antenna to improve coverage	290			590	- 90	480	TBD
Recording	-	MSS-PCM recorder kit	30						TBD

Note[1)]: MD = Man Day

Table 2-16: Orbiter Services Available to Spacelab Payloads

3. SPACELAB SYSTEM

This section describes some overall physical characteristics of the Spacelab system such as the possible flight configurations and their physical accommodation capabilities (mass, volume, c.g.-constraints). Also given is an overview of the module and pallet structure, including the igloo and crew transfer tunnel. A detailed description of module and pallet is given in Section 4, which also includes information about the electrical,data managementand environmental control capabilities of the Spacelab system. The crew habitability provisions are presented here in detail, and an overview of utility services and routing is given. Then the extended mission capabilities are discussed and summary tables of Spacelab payload accommodation capabilities are presented.

3.1 Hardware Content of Basic Spacelab

Table 3 - 1 shows a list of hardware content of major basic Spacelab elements, from which different Spacelab configurations may be built up. These items are essential parts of Spacelab which must always be flown in a particular configuration and which are included in the mass budget of the basic Spacelab (Table 3 - 4). The Spacelab Program provides, in addition to this basic (i.e. mission independent) hardware, a spectrum of mission dependent equipment (Table 3 - 5), which can be selected and flown according to the requirements of a particular mission.However, because of possible operational constraints some mission dependent equipment might not always be removed from Spacelab when not needed.

3.2 Spacelab Flight Configurations and Physical Accommodation Capabilities

3.2.1 Possible Flight Configurations

The modular elements of Spacelab introduced in Section 1 and listed in Tables 3 - 1 and 3 - 5 in more detail can be arranged in various flight configurations to suit the needs of specific mission/payload requirements and to meet Orbiter constraints.

Seven basic flight configurations are presented in detail with respect to their physical accommodation capabilities (Table 3 - 4). While other configurations are basically possible, only these seven configurations are under configuration control with respect to the mechanical interfaces to the Orbiter. In addition, of these seven basic configurations,four are formal baseline design configurations and as such are under formal configuration control within the Spacelab project. The formal baseline design configurations are

- Short module/3 pallets
- Long module
- 15 m pallet
- 9 m (independently suspended) pallet

The hardware of the Spacelab project, however, allows all seven basic and other possible flight configurations to be implemented by combination/deletion/addition of appropriate hardware elements.

Table 3 - 1 : Mission Independent Hardware Content of Major Spacelab Elements

Mode	Short Module	Long Module	Pallet
Major Equipment			
Mission Independent Spacelab Equipment Located In Orbiter Aft Flight Deck			
Integrated AFD Panel (R 7 Panel)	1	1	1
Data Display & Keyboard	1	1	1
Subsystem RAU	1	1	1
Interconnecting Station	1	1	1
Power Distibution Box (AFD)	1	1	1
Emergency Intercom	1	1	1
Mission Independent Spacelab Module/Pallet Subsystem Equipment			
Structure			
Primary Structure/Module			
2.7 m Cylinder Assembly	1	2	–
Forward Cone Assembly	1	1	–
Aft Cone Assembly	1	1	–
Fitting Assembly	1	1	as req'd
Secondary Structure/Module			
2.7 m Overhead Support Structure	1	2	–
2.7 m Floor Support Structure	1	2	–
1.1 m Floor Assembly	1	2	–
1.6 m Floor Assembly	1	2	–
Floor End Extension	2	2	–
Subsystem Subfloor	1	1	–
Aft Viewport	1		–
Double Rack Assembly for Subsystems	2	2	–
Pallet Structure			
2.9 m pallet segments	–	–	5
Structure/Per Pallet Segment			
Inner-Panels	–	–	24
Outer-Panels	–	–	24
EPDS			
EPDS Monitor and Control Panel	1	1	–
Power Control Box	1	1	1 2)
Emergency Box	1	1	1 2)
Subsystem Power Distribution Box	1	1	1 2)
Experiment Power Distribution Box	1	3	1 per pallet segment
Subsystem Inverter 400 Hz	1	1	1 2)
Harness (Signal / Power)	x	x	x
Lights	9	15	–

Mode	Short Module	Long Module	Pallet
Major Equipment			
ECS			
Atmospere Storage and Control System (ASCS)			
N_2 Tank Assembly	1	1	–
Oxygen Supply Assembly	1	1	–
Nitrogen Supply Assembly	1	1	–
Atmosph. Pressure Control Assembly	1	1	–
Atm. Supply Control Assembly	1	1	–
Airlock Supply Assembly	1	1	–
Cabin Pressure Relief Assembly	1	1	–
Small Experiment Vent Assembly (Forward Cone)	1	1	–
Atmosphere Revitalization System (ARS)			
Display and Control Panel	1	1	–
Cabin Fan Assembly	1	1	–
Condensing Heat Exchanger	1	1	–
CO_2 and Temperatur Control Assembly	1	1	–
Water Separator Assembly	1	1	–
Overboard Dumping Assembly	1	1	–
Distribution System - Cabin Loop	1	1	–
Avionics Fan Assembly	1	1	–
Avionics Heat Exchanger	1	1	–
Cabin Sensor Assembly (Temp./Press.)	1	1	–
Distrib. Syst./Avionics Loop	1	1	–
Fire/Smoke Detection Assembly	1	1	–
Active Thermal Control			
Water Loop Plumbing	x	x	x
Water Pump Package	1	1	–
Subsystem Cold Plates	9	9	9 2)
Freon Loop Plumbing	x 1)	x 1)	x
Freon Pump Package	1 1)	1 1)	1
Interloop Heat Exchanger	1 1)	1 1)	–
Passive Thermal Control			
High Performance Insulation	x	x	–
Thermal Coating	x	x	x

1) Module plus Pallet Configurations Only
2) located in Igloo in Pallet Only Mode
x) Hardware not easily quotable in discrete Units

Table 3 – 1 : Mission Independent Hardware Content of Major Spacelab Elements (cont'd)

Mode	Short Module	Long Module	Pallet
Major Equipment			
<u>Mission Independent Spacelab Module/ Pallet Subsystem Equipment (cont.)</u>			
<u>CDMS</u>			
Data Display Unit (CRT & Keyboard)	1	1	–
Subsystem Computer	1	1	1 2)
Back – up Computer	1	1	1 2)
Subsystem I/O Unit	1	1	1 2)
Data Display Unit & Keyboard	1	1	–
Mass Memory	1	1	1 2)
Subsystem Data Bus	x	x	x
Experiment Data Bus	x	x	x
Subsystem RAU	5	5	3 2)
Master Intercom Station	1	1	–
Remote Intercom Station	1	4	–
Video Monitor	1	[illegible]	–
RAAB	1	1	1 2)
<u>Crew And Habitability</u>			
Tools and Maintenance Equipment Set	1	1	–
Ceiling Rails	6	12	–
Console Vertical Rails	4	4	–
Console Horizontal Rails	2	2	–
Rack Foot Restraints	2	2	–
Portable Foot Restraints	2	2	–
Work Bench	1	1	–
Work Bench Containers	4	4	–
EVA Mobility Aid	x	x	x
Fire Extinguisher	2	2	–
Portable O_2 System	4	4	–

Mode	Short Module	Long Module	Pallet
Major Equipment			
<u>Utility Routing</u>			
Forward Utility Lines	x 3)	x 3)	x 3)
Module-to-Pallet Utility Bridge and Harnesses	1 1)	1 1)	–
Pallet-to-Pallet Utility Bridge and Harnesses	–	–	2
<u>Igloo Structure</u>	–	–	1
Structure			
Outer Shell	–	–	x
Coverplate	–	–	x
Inner Platform	–	–	x
<u>Transfer Tunnel</u> 3)			
Length/Number of Segments	TBD	TBD	–

1) Module Plus Pallet Configurations Only 2) Located in Igloo in Pallet Only Mode
3) Orbiter Provided x) Hardware not Easily Quotable in Discrete Units

3.2.2 Basic Flight Configurations

3.2.2.1 Long Module Configuration

The long module configuration (without pallet), which is also one of the baseline design configurations, is shown in Figure 3 - 2. It consists of the core segment and the experiment segment and provides the largest pressurized volume for Spacelab payloads. It is accessible from the Orbiter cabin through the transfer tunnel (not shown here). Utility services are routed from the Orbiter to the forward end cone feedthrough provisions and from there into the module interior. Basic Spacelab dimensions are shown (in mm), as well as the Orbiter stations of the module attach fittings. Orbiter stations Xo > 660 " have to be kept clear of Spacelab and Spacelab payload equipment. The volume is reserved for the Orbiter air-lock and tunnel adapter.

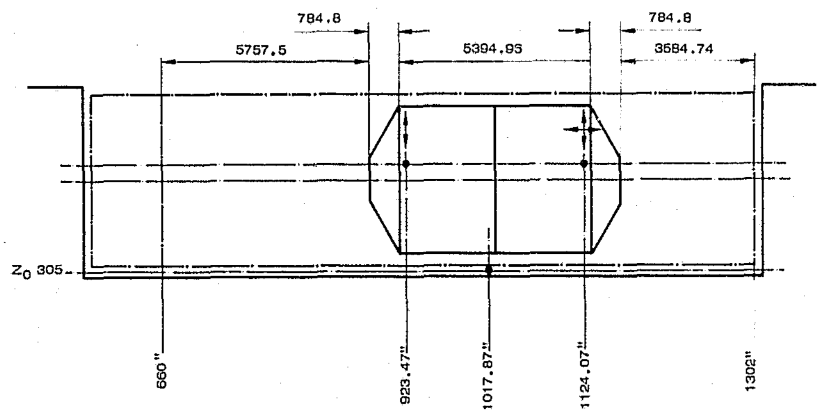


Figure 3 - 1: Long Module

3.2.2.2 Long Module plus One-Pallet Configuration

Figure 3 - 2 depicts the long module/one pallet configuration. This configuration provides both pressurized volume for payloaas and pallet mounting area for experiments requiring exposure to space environment. Utility services to the pallet are routed through the Module-to-Pallet utility bridge as described in para 3.7.2.

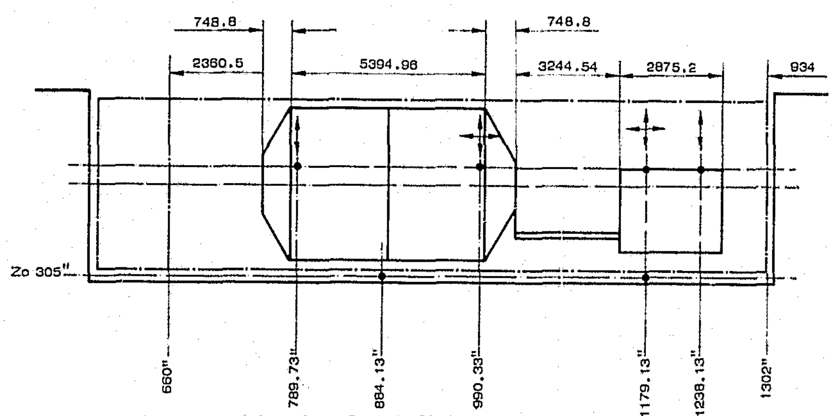


Figure 3 - 2: Long Module Plus One Pallet

3.2.2.3 Long Module plus Two-Pallet Train Configuration

This configuration (Figure 3 -3) increases the pallet mounting area by connecting two pallets to form a pallet train. Utility routing is the same as for the long module plus one-pallet-configuration.

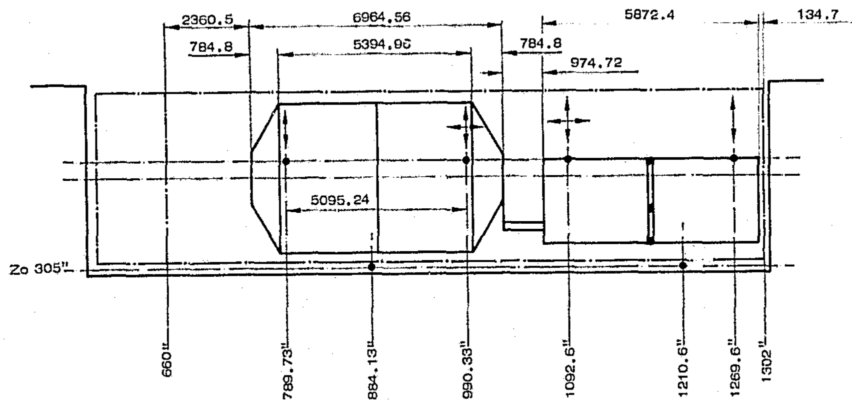


Figure 3 - 3 Long Module Plus Two Pallet Train

3.2.2.4 Short Module plus Two-Pallet Train Configuration

A short module may be used in place of the long module to provide the configuration shown in Figure 3 - 4.

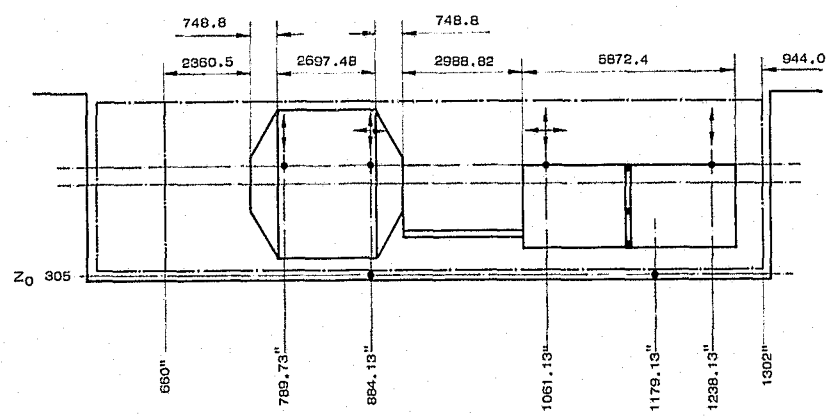


Figure 3 - 4 Short Module Plus Two Pallet Train

3.2.2.5 Short Module plus Three-Pallet Train Configuration

This configuration is one of the four baseline design configurations. It offers the largest pallet area which may be used in a module/pallet configuration, as shown in Figure 3 - 5. The three pallet segments are rigidly attached to form a single pallet train.
A module-to-pallet utility bridge is not used in this configuration because of the small gap between module and pallet.

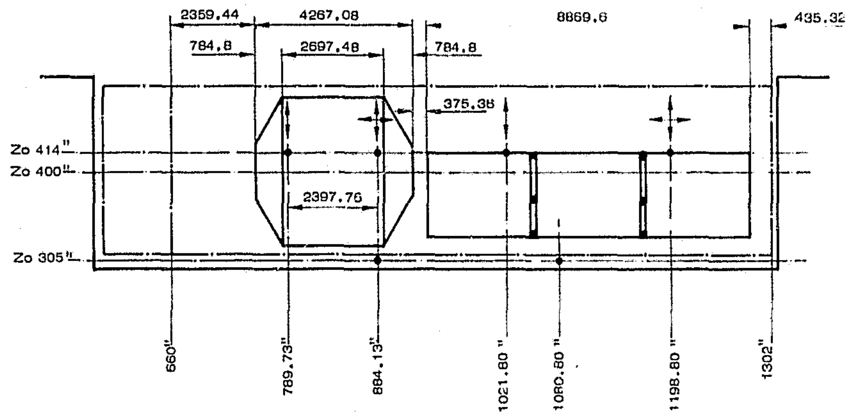


Figure 3 - 5: Short Module Plus Three-Pallet Train

3.2.2.6 Pallet-Only-Configuration/15 Meter Pallet

This configuration, which is also a baseline design configuration, provides the longest possible experiment platform for Spacelab payloads requiring exposure to the space environment. The configuration shown in Figure 3 - 6 consists of two independently suspended pallet trains separated by a dynamic clearance gap. One pallet train consists of three and the other consists of two structurally connected pallet segments.

The "Igloo" of the forward pallet provides a controlled pressurized environment for certain Spacelab subsystems normally located in the core segment of the module (see para 3.5.3). Utility services are routed directly from the Orbiter to the igloo/first pallet segment.

For the accommodation of payload structures, it must be ensured that such structures do not act as a rigid connection between the two pallet trains.

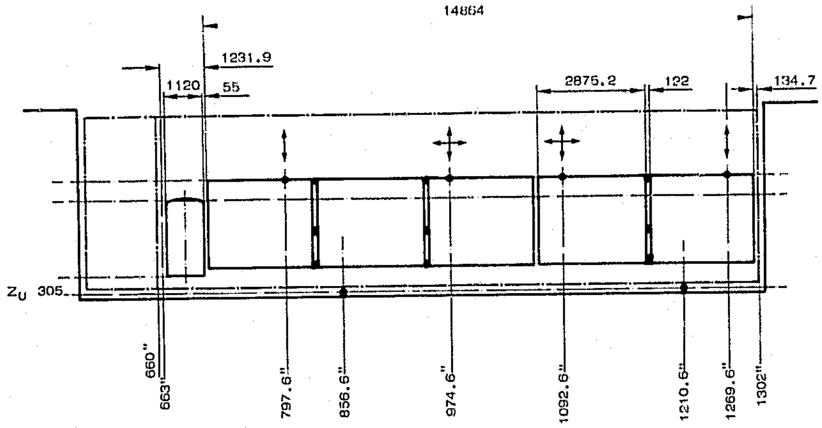


Figure 3 – 6: 15 Meter Pallet

3.2.2.7 Pallet-Only Configuration/9 Meter Independently Suspended Pallet

As shown in Figure 3 – 7, this configuration, which is also a baseline design configuration, consists of three independently suspended pallet segments. This configuration is especially suited for payloads not requiring a full 15 meter pallet for mounting (e.g. telescopes, antennas, etc.). The pallet segments are placed along the length of the cargo bay and interconnected by inter-pallet utility bridges, as described in para. 3.7.3. The "Igloo" at the forward pallet provides a controlled pressurized environment for certain Spacelab subsystem equipment normally located in the core segment of the module (see para 3.5.3). Utility services are routed directly from the Orbiter to the Igloo/first pallet.

For the accommodation of experiment structures it must be ensured that such structures do not act as a rigid connection between the pallet segments.

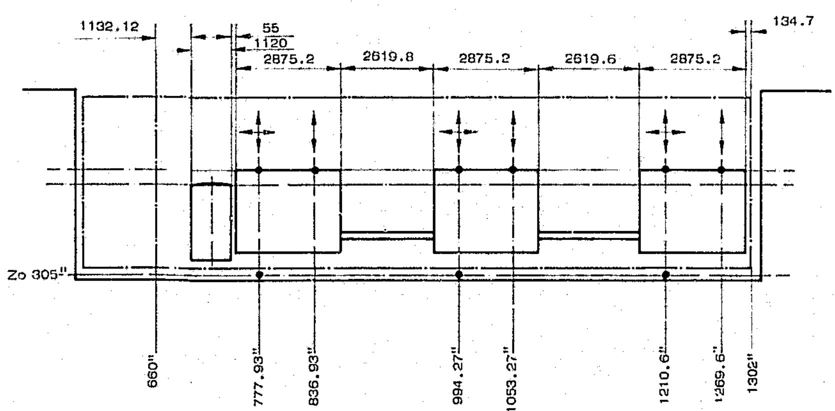
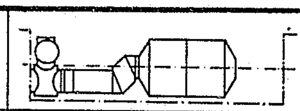
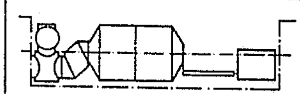
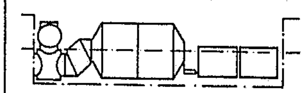
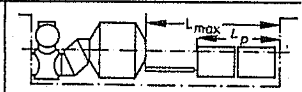
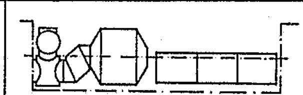
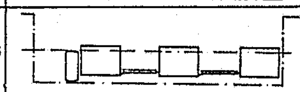
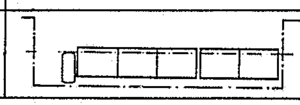


Figure 3 – 7: 9-Meter Independently Suspended Pallet

Table 3-2: Volume and Area Available to Spacelab Payloads for the Seven Basic Spacelab Configurations

CONFIGURATION		MODULE		PALLET		
		Volume m^3	Panel Mounting Area (m^2) 1)	Surface Mounting Area m^2	Length L_p(m)	Length L_{max}(m)
1		22.2	17.14	—	—	—
2		22.2	17.14	17.1	3.94	7.11
3		22.2	17.14	34.2	6.00	6.98
4		7.6	6.43	34.2	6.92	9.77
5		7.6	6.43	51.3	9.30	9.25
6		—	—	51.3	—	—
7		—	—	85.5	15.00	—

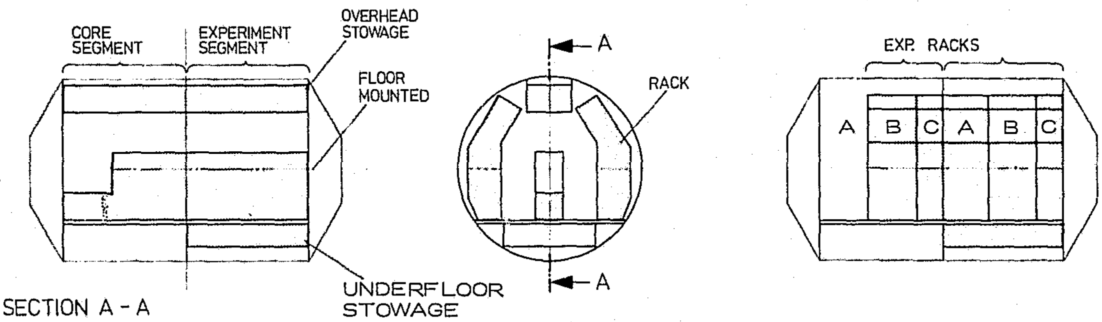
1) total available in the Experiments Racks for Payloads and Mission Dependent Equipment

3.2.3 Volume and Mounting Area Available to Spacelab Payloads

The overall volume and mounting area available for payload equipment is shown in Table 3-2 for the seven basic flight configurations. The values shown for the module volume are the maximum volumes available when all the available mission dependent racks and ceiling storage containers are used and when reasonable allowances are made for unrestricted crew movement and working conditions.

For the pallets, the areas shown represent the maximum available mounting surface. The maximum length indicates the space available for payload lengths including payload overhang.
Provisions available in the Orbiter aft flight deck are mentioned in para 2.5.2.
A more detailed breakdown of the available volume is shown in Table 3 - 3. For the racks the quoted values are for the available volume inside the racks. The overhead volume comprises the volume inside the mission dependent storage containers (of which 8 are provided in the baseline).

The center aisle volume is the maximum envelope available for payload equipment mounted on the floor without impacting crew habitability and safety requirements. The underfloor volume is available for payload use only in the experiment segment. If the racks are not used the resulting available volume is the envelope occupied by the rack structure. Some space inside the aft end cone may be available for experiment equipment; e.g. equipment mounted on the center floor may protrude into the end cone.

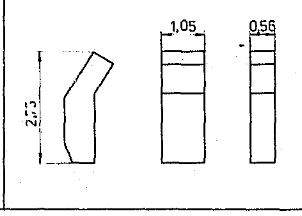
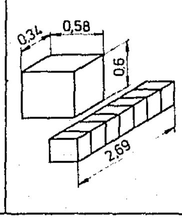
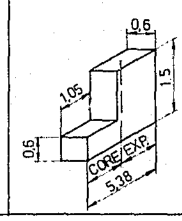
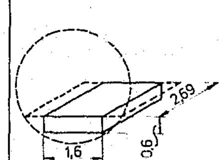


Editorial note: A subfloor for experiments in the underfloor area is not part of the current baseline.

Figure 3 - 6 Module Payload Volume Allocation

Figure 3-6 shows the approximate volume available for payload equipment in experiment racks, overhead and underfloor area and in the middle aisle. Table 3-3 depicts the gross volume available in those areas.

Table 3 - 3 Volume Breakdown (Total Volume)

	Volume inside Racks m^3 left side A	B	C	right side A	B	C	Ceiling (m^3)	Center Aisle (m^3)	Subfloor (m^3)	Total (m^3)
Core Segment	–	1.75	0.9	–	1.75	0.9	0.8	1.5	–	7.6
Experiment Segment	1.75	1.75	0.9	1.75	1.75	0.9	0.8	2.42	2.58	14.6
										

The usable net volume is reduced due to the structural elements and utilities of the racks (see also para. 3.3.2 and 4.1.2).

Due to the accessibility required around the aft viewport and the accommodation of various Spacelab equipment such as oxygen masks and fire extinguisher, no specific volume for experiments can be identified in the aft end cone, other than the aforementioned option to utilize parts of this volume for floor mounted experiment equipment.

3.2.4 Mass Available for Spacelab Payloads

Because of the variety of possible Spacelab configurations and the variety of mission dependent Spacelab equipment which can be flown according to specific mission requirements,a wide range of payload mass capabilities exists. The actual mass available to payloads for any given configuration of Spacelab and Orbiter hardware will be generally limited by the launch/landing mission capabilities of the Shuttle and the specific load carrying capabilities of Spacelab. The information which follows will enable the user to determine, for a wide range of configurations and constraints, the mass available for his equipment to meet his specific mission needs.

Editorial note: The actual mass data which is presented contains contingency provisions which are currently under discussion between ESA and NASA.

3.2.4.1 Spacelab Mass

Hardware and consumables which are included in the Spacelab mass budget comprise the following major equipment categories (described in more detail in 3.2.4.3):

- Basic Spacelab subsystem equipment
- Mission Dependent Spacelab subsystem equipment
- transfer tunnel
- Basic Orbiter Support Equipment (OSE)

3.2.4.2 Spacelab Payload Mass

Hardware and consumables which are chargeable to the Spacelab payload mass budget comprise the following mission dependent categories:

- all experiment hardware including any user-provided special experiment displays, controls, service panels, etc., located in Spacelab and/or the Orbiter
- instrument pointing system(s) and related equipment
- experiment consumables
- crew members in excess of four
- any additional mission dependent Orbiter support equipment such as
  - EVA equipment above the Orbiter baseline
  - RCS tankage and propellant for special Spacelab payload pointing requirements
  - OMS kits and propellant required by the Spacelab payload
  - additional electrical energy kits (in excess of the basic 50 + 840 kwh)
  - second Orbiter remote manipulator system
  - second Orbiter TDRSS antenna
  - extra hardware and consumables required for missions longer than seven days.

3.2.4.3 Overall Mass Breakdown

Table 3 - 4 shows the overall mass breakdown for 7 basic configurations of Spacelab. For 4 of these configurations (the baseline design configurations), a more detailed mass budget has been established which defines Spacelab development hardware control masses, nominal masses for payloads and payload reserve masses. The complete Spacelab mass control program that is applied to four basic configurations does not formally apply to other configurations and, in particular, there are no specification masses for payloads for the other 3 configurations (or for other possible configurations). These configurations and their associated mass details, which are derived from the representative configurations, are listed so that the user can derive an available payload mass for as wide a range as possible of mission configurations. An explanation follows for each column of information in Table 3- 4 :

Column 1 Mission Independent Spacelab Subsystem Control Mass
These control masses relate to the basic Spacelab equipment which will be flown on every mission for each particular configuration. This consists of equipment that is either essential for the operation of Spacelab or that cannot easily be removed from Spacelab between missions because of operational constraints.

Column 2 Transfer Tunnel Mass (Mission Independent)
The transfer tunnel is flown in all module configurations to provide access to and egress from Spacelab. The tunnel is modular and different lengths are used depending on configuration and center of gravity constraints. The two different weights indicated in the table are for mass control purposes only. The final tunnel hardware will allow several different tunnel lengths - each with its associated different mass properties. This may enable the user to change tunnel lengths to meet center of gravity constraints.

Column 3 Orbiter Support Equipment (Mission Independent)
This category comprises Orbiter supplied equipment which is necessary for Spacelab operation and which is considered as part of the Orbiter payload from a mass chargeability viewpoint. The following items comprise this category:

- Mass of Orbiter heat rejection subsystem components not included in the Orbiter baseline design (i.e. increase from 6.3 kw to 8.5 kw heat rejection capability).
- Tankage for fuel cell consumables for Spacelab power in excess of 50 kwh (i.e. for the first kit of 840 kwh capability).
- Consumables for Spacelab power in excess of 50 kwh (i.e. for first kit of 840 kwh).
- Orbiter payload attachment fittings in excess of four.
- Adapter and other associated hardware for attaching the EVA airlock to the tunnel and Orbiter cargo bay.

The difference between the launch and landing masses is accounted for by that part of the first power kit consumables which is not landed (i.e. overboard water dump).

Column 4 Total Mission Independent Equipment Mass
This represents the sum of the masses in columns 1, 2 and 3.

Column 5 Mission Dependent Equipment Mass
Mission dependent equipment comprises Spacelab hardware which is not an integral part of Spacelab but which can be added to the basic Spacelab to meet the requirements of a particular mission. Table 3 – 5 lists this equipment together with the mass of each individual equipment item. Also listed is the total number of each item which is delivered as part of the Spacelab program baseline. It should be noted that this baseline delivery number is not necessarily the maximum number of each item which can be physically accommodated by each Spacelab configuration. The last five columns of Table 3 – 5 indicate how the mission dependent units are typically distributed between the different elements of Spacelab and the Orbiter Aft Flight Deck. Again it should be noted that, while for some items such as top airlock there is only one possible location per Spacelab element, for items such as RAU's and cold plates the total number of possible locations greatly exceeds the total number of items supplied. For detailed information on the location restrictions of mission dependent equipment reference must be made to the appropriate subsystem description section.

For any particular Spacelab configuration and for particular mission requirements there is a very large number of possible different combinations of mission dependent equipment that can be flown. Each combination will have a different total mass. In order to have a reference for establishing the resultant payload mass capability for all of these possible conditions, it is necessary to establish a mission dependent equipment control mass for each Spacelab configuration. This control mass, indicated in column 5 of Table 3 – 4 for the seven basic configurations, is approximately equal to the mass of the total amount of delivered mission dependent support equipment that can be used with each configuration.

Editorial note: Because of possible operational constraints,some mission dependent equipment might not always be removed from Spacelab when not needed.

Column 6 Minimum Mass Available to Spacelab Payloads
If all the available mission dependent equipment is flown for a particular mission the resultant Spacelab hardware mass will be a maximum. The difference between this mass and the launch and landing weight Shuttle payload capabilities (which are described in Section 2) represents a minimum value available for the Spacelab payload mass but, as explained later, further constraints will, in practice, result in modifications to the actual available Spacelab payload mass. In column 6 the calculated masses assume that the total mission dependent equipment is represented by its control mass (see column 5) for each of the four representative configurations.

Column 7 Maximum Mass Available to Spacelab Payload

If no mission dependent equipment is flown for a particular mission, the resultant Spacelab hardware mass will be a minimum. The difference between this mass and the launch and landing weight Shuttle payload capabilities represents a maximum value available for the Spacelab payload mass but, as explained later, further constraints will, in practice, result in modifications to the actual available Spacelab payload mass.

Column 8 Spacelab Payload Carrying Capability

This column lists the overall load carrying capability for Spacelab payload equipment which results from the structural limitations of each configuration (when all the mission dependent equipment is carried). The load carrying capabilities of the module and pallet structures have further localized constraints (capabilities of racks, hardpoints etc.) and these are described in Section 4 . The values listed in column 8 are preliminary and subject to further refinement. Later editions of the Handbook will also include details of any extra load carrying capability which can be made available for payload equipment if the maximum amount of mission dependent equipment is not flown.

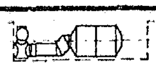
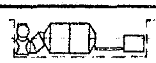
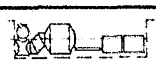
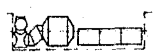
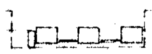
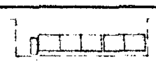
Column 9 Nominal Payload Mass

Because of the wide range of Spacelab total hardware masses which result from the flexibility in the use of mission dependent equipment, it is necessary to establish a reference condition on which to base the nominal payload mass. This reference condition is defined when each of the four representative configurations includes an amount of mission dependent equipment which has a mass equal to 50 % of the mission dependent equipment mass of column 5. The nominal payload mass is then as listed in column 9 and is the total mass of payload equipment allowable for each configuration and applies to the landed mass only.

If extra mission dependent equipment is carried, the resultant mass delta (i.e. the difference between its mass and the 50 % value masses quoted above) must be subtracted from the nominal payload mass. Conversely, if less mission dependent equipment is carried, the resultant mass delta can be added to the guaranteed payload mass.

Column 10 Payload Growth Reserve Mass

The masses listed in column 10 are currently foreseen as margins for Spacelab payload contingency and growth. These tentative reserve masses are subject to regular review by the Spacelab program and are listed here for information only.

Configuration	1 Mission Independent Spacelab Subsystem Control Mass(kg)	2 Transfer Tunnel Mass (kg)	3 Mission Independent Orbiter Support Equipment Mass(kg)		4 Total Mission Independent Equipment Mass (kg) (1+2+3)		5 Mission Dependent Spacelab Subsystem Control Mass (kg)
			Launch	Landing	Launch	Landing	
	Landing 4715	499	1269	918	6503	6132	2094
	5430	346	1473	1102	7249	6878	TBD
	6025	346	1473	1102	7844	7473	
	4995	346	1473	1102	6814	6443	
	5589	346	1473	1102	7408	7037	1806
	2910	–	1254	883	4164	3793	736
	3933	–	1060	699	4993	4632	818

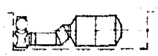
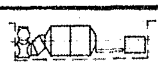
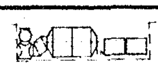
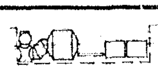
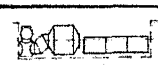
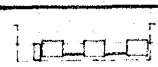
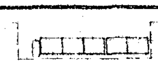
Configuration	6 Min Mass Available for Payloads when total Miss Dep.Eq. is used (kg)		7 Max Mass Available for Payloads when no Miss. Dep.Eq. is used (kg)		8 Payload Load Carrying Capability (kg)	9 Nominal Payload Mass (kg) (Landing)	10 Payload Growth Reserve Mass(kg)
	Launch 29 484 –	Landing 14 515 –	Launch 29 484 –	Landing 14 515 –			
	20887	6289	26922	8383	5500	5500	1100
	TBD	TBD	22235	7637	8600	TBD	TBD
			21640	7042	10,500		
			22670	8072	7200		
	20270	5672	22076	7478	7200	5500	1100
	24584	9986	25320	10722	9100	9100	1320
	23673	9065	24491	9883	10,000	8000	1499

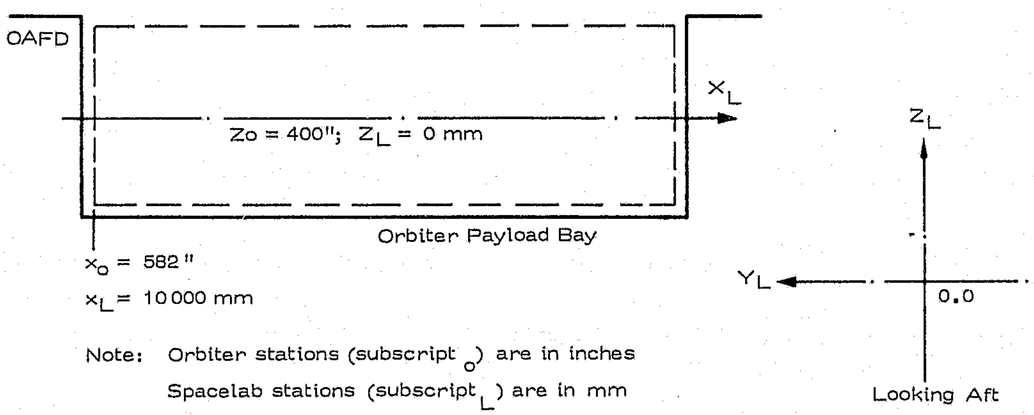
Table 3 – 4 : Overall Mass Breakdown

Table 3-5 : Mission Dependent Spacelab Subsystem Equipment

Item / System	Unit Weight (kg)	Unit Power Consumption (W)	Unit Size (mm)	Numbers of Units Delivered by S/L Progr. (Baseline)	Number of Units Foreseen Per				
					Core Segment	Experiment Segment	Pallet Segment	OAFD	Igloo *3)
Single Rack *1)	36.1	-	2753 x 564 x 760	4	2	2	-	-	-
Double Rack *1)	57.3	-	2753 x 1052 x 760	6	2	4	-	-	-
Pallet Hard-points	1.0	-	TBD x 105 x 65	120	-	-	24	-	-
Experiment Heat Exchanger	10.8	-	TBD	1	1	-	1	-	-
High Vacuum Venting Facility *7)	50.	-	TBD	1		-	-	-	-
Cold Plate (with freon)	6.3	-	750x500x6	8	* 8)	-	8	-	-
Cold Plate Stand-Off	0.9	-	TBD	8	-	-	8	-	-
Thermal Capacitor	12.5	-	750 x 500 x 28	4	* 8)	-	4	-	-
Freon Line System	*2)	-	-	*2)	-	-	-	-	-
Pallet Thermal Cover Set	19.8	-	TBD	5	-	-	1	-	-
Exp. Inverter 400 Hz	28.00	100 + 0.175 P DC *6)	200 x 394 x 473	1	1	-	-	-	1
Exp. Switch Panel	3.7	-	177x482x150	10	4	6	-	-	-
Power Harness	*2)	0.02 P * 6)	-	*2)	-	-	-	-	-
Exp. Computer	30.4	315 DC	1 ATR long	1	1	-	-	-	1
Exp. I/O-Unit	29.5	160 DC	193.5 x 391 x 550	1	1	-	-	-	1
Exp. RAU *4)	8.9	16 DC *11)	180x170x 408	8	4	6	4	-	-
Keyboard	4.5	12 AC	182 x 442 x TBD	1	1	-	-	-	-
Data Display Unit and Symbol Generator	24.25	250 AC	314 x 442 x 520	1	1	-	-	-	-
High Rate Digital Recorder (HRDR)	43.00	367 AC * 9)	537 x 442 x 152.6	1	-	-	-	1	-
High Rate Multi-plexer (HRM)	25 approx.	TBD	TBD	1	1	-	-	-	1
Stowage Container Ceiling	6.1	-	600 x 580 x 340	8	4	4	-	-	-
Rails Console Vertical	1.0	-	Cross Section 19x33.5	10	4	6	-	-	-
Portable Foot Restraint	1.7	-	33.5 x 228 x 670	1	-	1	-	-	-
Rack Foot Restraint	1.1	-	33.5 x TBD x 1000	8	3	5	-	-	-
Pallet - PSA Foot Restraint	5.5	-	TBD	1	-	-	1	-	-
Top Airlock	129.5	53 DC *10)	1000 Ø, 1000 long	1	1	1	-	-	-
Aft Airlock	153.9	56 DC *10)	1000 Ø, 1500 long	1	1	1	-	-	-
High Quality Window/Viewport Assy	62.8	TBD	1000 Ø	1	1	1	-	-	-
Film Vault Assy (3 Units)	78.2x3	-	451 x 645 x 1200	1	1	-	-	-	-
Intercom Remote Station	1.3		133 x 240 x 139	3	1	2	-	-	-

Table 3- 5 Mission Dependent Spacelab Subsystem Equipment
(Footnotes and Comments)

* 1):	Rack weight includes ducts and insulation and takes into account that floor panels otherwise installed will be removed.
* 2):	Mission configuration dependent
* 3):	Applies to pallet-only-mode
* 4):	Figures given are the maximum number of accommodable RAUś
* 5):	Adjusted for removal of close-out plates.
* 6):	P = Power drawn (W)
*7):	The high vacuum vent facility is presently under definition and not yet baseline
*8):	The location of cold plates is outside the module. The example given in this table refers to a cold plate thermal capacitor combination which may be required to accommodate heat peak loads. These cold plates are not available to experiments for mounting of payload equipment.
* 9)	Power given is for Playback mode; during recording mode it is 245W
* 10)	Excluding heaters (78 W DC) and lamps (25 W DC)
* 11)	Average continous power / RAU with no command 'on'; for 32 commands 'on' at a time add 6.9 W



Note: Orbiter stations (subscript $_o$) are in inches
Spacelab stations (subscript $_L$) are in mm

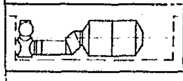
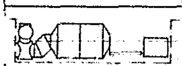
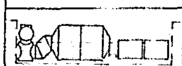
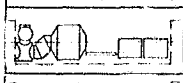
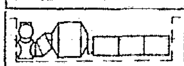
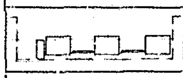
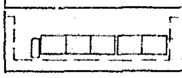
Figure 3 - 9 Spacelab System Coordinate System

3.2.5 Center of Gravity Constraints

As noted in Section 2, the Orbiter imposes stringent center of gravity location constraints on its payload. These constraints enable safe landing of the Orbiter but, to allow for an aborted launch, will also apply to the launch condition. Table 3 - 6 shows, for the 7 basic configurations, the center of gravity locations of the basic Spacelab (i.e. without mission dependent equipment and without payload). The c.g. locations are superimposed on the allowable c.g. location envelopes imposed by the Orbiter and the total Orbiter payload c.g. (i.e. the overall c.g. of the basic Spacelab plus mission dependent equipment plus Spacelab payload and consumables) must always fall inside these envelopes. This superimposition is presented for each of the three ref. axes in Figures 3 - 10, 3 - 11 and 3 - 12. The shaded areas of each envelope indicate the constraints on the launch condition when the overall Orbiter payload mass is greater than the allowable landing mass (e.g. when expendables, non-returnable satellites, etc. are carried as part of the payload). The Orbiter and Spacelab coordinate system is depicted in Figure 3 - 9.

It should be noted that the Spacelab c.g. locations shown in Figures 3 - 10, 3 - 11, 3 - 12, apply to specific locations of Spacelab in the Orbiter cargo bay. For the seven basic configurations the reference locations of the module and pallet are indicated in Figures 3 - 2 thru - 5. It should be noted that other configurations and locations are in principle possible allowing that, for any given mission, the configuration location may be optimized for several constraints in addition to the c.g. of the Spacelab and its payload (e.g. the combination of Spacelab with other Orbiter payload, retrieval and launch of satellites etc.).

Table 3 - 6 Center of Gravity for Mission Independent SPACELAB at Landing Condition

Configuration	Total Mission Independent Mass (kg)	Center of Gravity (m)			Moment of Inertia ($x10^3 kgm^2$)		
		X_L	Y_L	Z_L	Ixx	Iyy	Izz
	6132	19.137	.002	-.455	16.13	76.76	71.93
	6878	17.647	.021	-.503	20.03	108.42	104.55
	7473	18.144	.018	-.531	21.11	136.55	133.21
	6443	17.512	.024	-.627	18.76	113.13	110.50
	7037	17.769	-.006	-.545	18.37	115.78	112.59
	3793	19.232	.089	-.813	13.19	95.19	94.81
	4632	19.215	.059	-.850	14.05	115.18	115.30

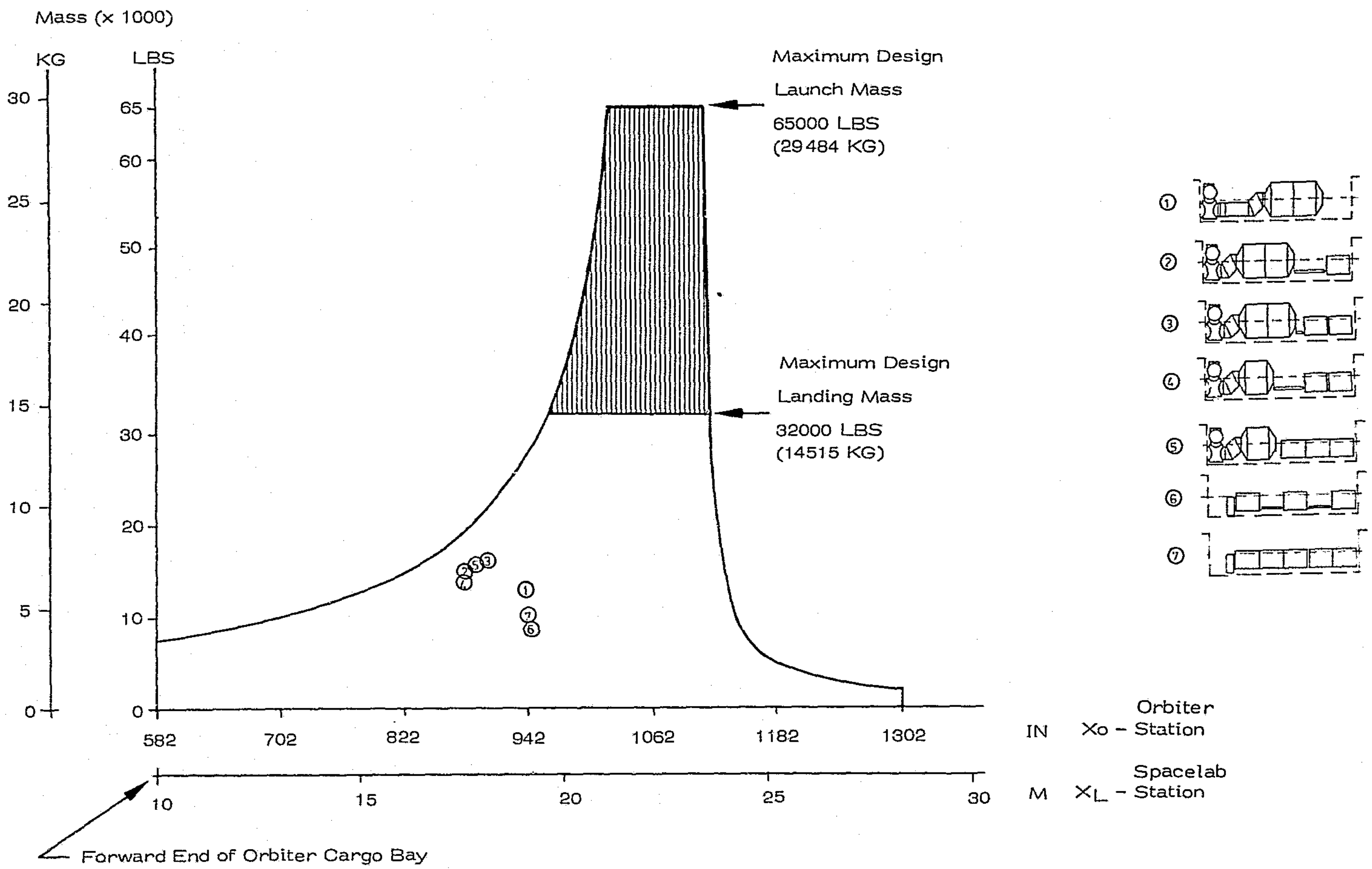


Figure 3-10 X-Axis CG Location of Basic Spacelab Configurations

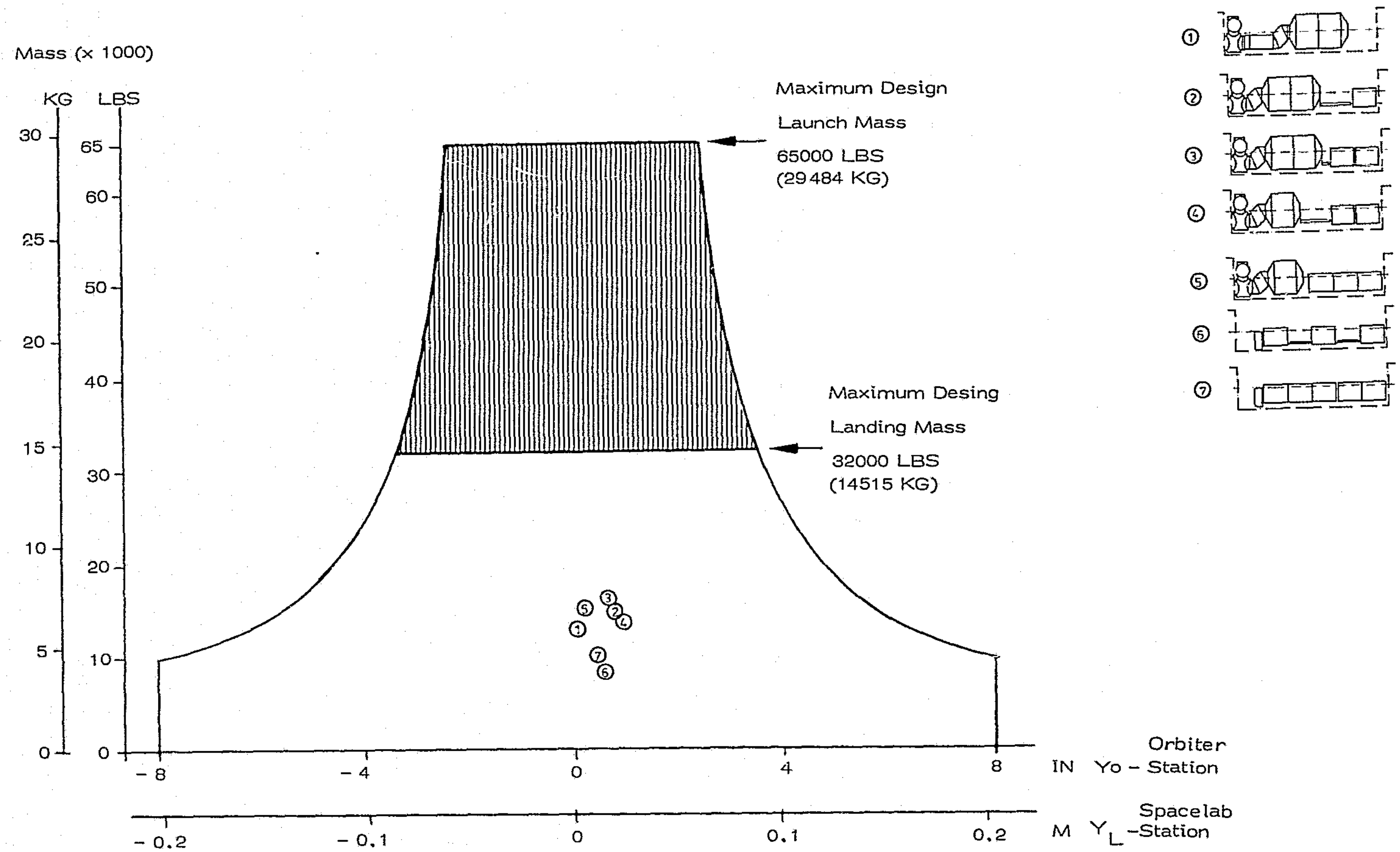


Figure 3 - 11 :Y-Axis CG Location of Basic Spacelab Configurations

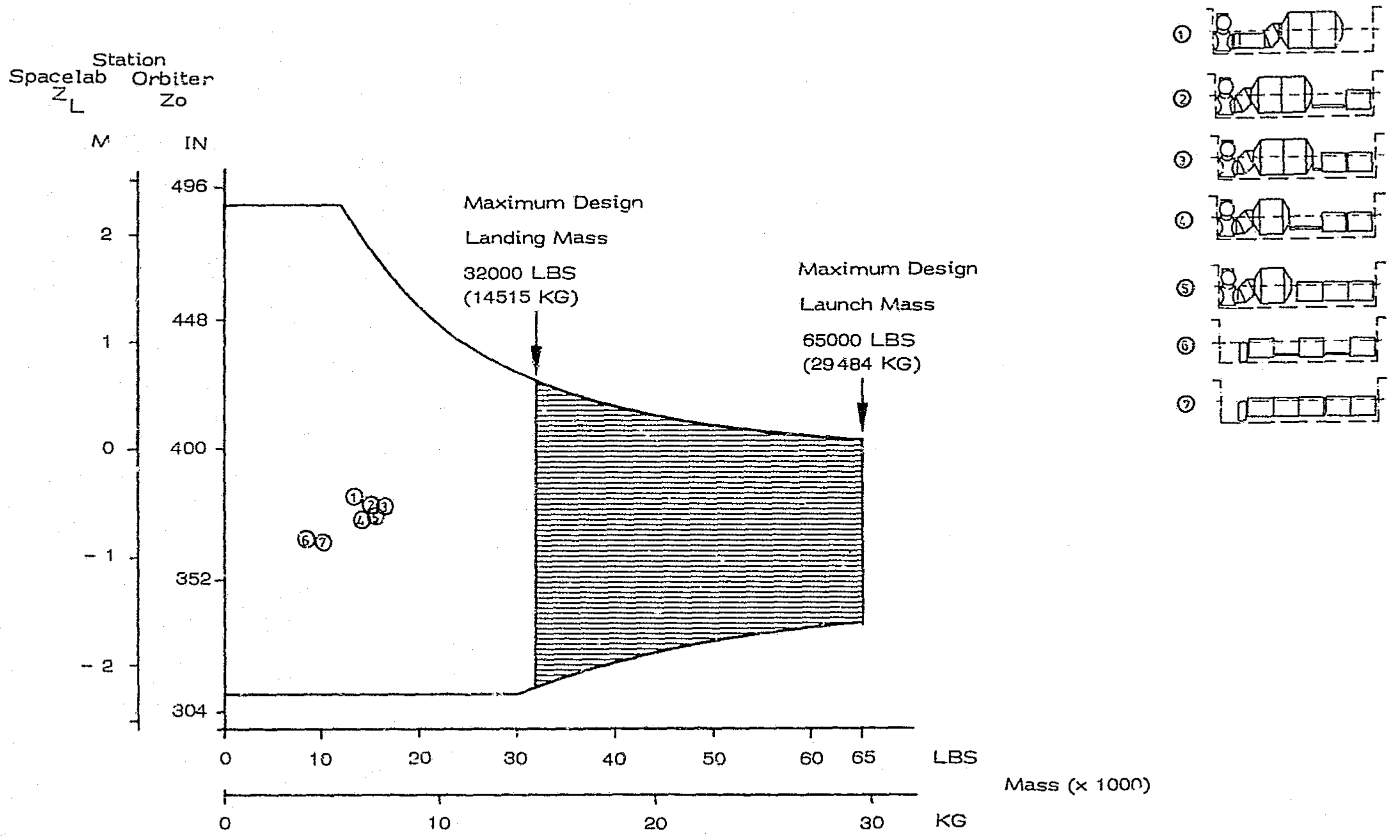
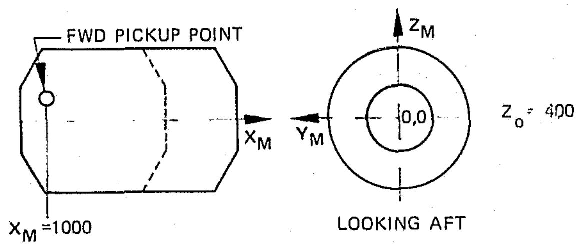


Figure 3 -12 : Z-Axis CG Location of Basic Spacelab Configurations

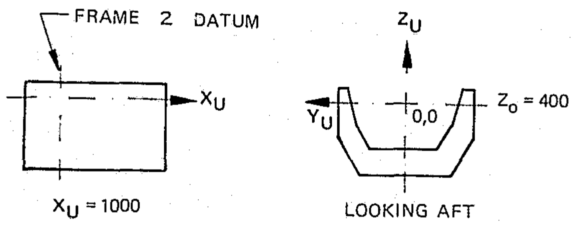
In order to establish the overall c.g. of a combined Spacelab plus Spacelab payload the masses and c.g. locations for the individual hardware and consumable items must be combined according to the requirements of each individual mission. Table 3-7 shows the masses and c.g. locations for the major items of mission dependent equipment. Items with masses less than about 10 kg (e.g. RAU's and foot restraints) have been omitted to keep the list to a reasonable size. These smaller items, however, must be considered in any final mission c.g. calculations and the relevant information is available in detail in a separate mass properties document. The Spacelab payload hardware and consumables supplied by the user will have a mass and c.g. location distribution which is dependent on their physical layout within the Spacelab and/or Orbiter. Also included in this category will be those further items (see 3.2.4.2) which are considered as part of the Spacelab payload mass. The mass and c.g. data for these items are described in Section 2 for the Orbiter related items, and in para 4.9 for the IPS.

The reference coordinate systems for the individual major Spacelab assemblies (module, pallet and igloo) are shown in Figure 3-13. For the tunnel and the IPS the reference coordinate systems are still TBD. The advantage of assembly related reference data lies in its convenience during planning and integration of these assemblies. It is obvious that computerized transformation to Spacelab and Orbiter coordinates, respectively, may be performed as a matter of routine when relative positions are defined.

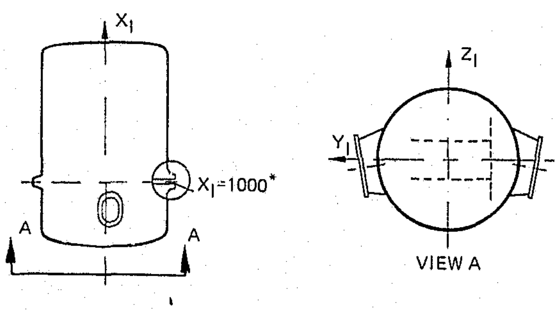
MODULE



PALLET



IGLOO



* X_I REFERENCE RELOCATED PER ECP 50354

UTILITY BRIDGE

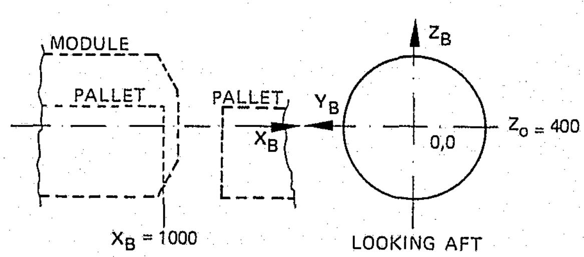


Figure 3 - 13 :Spacelab-Assembly C.G.Coordinate Systems

Table 3 – 7 : Center of Gravity Locations of Major Mission Dependent Equipment

ITEM	Location in Spacelab by Element	Center of Gravity Spacelab Assembly Coordinate System x_{cg}	y_{cg}	z_{cg}	Unit Weight (kg)
Single Rack	Module	3260	± 1270	184	See Table 3 – 5
Single Rack	Module	5936	± 1270	184	
Double Rack	Module	2452	± 1270	184	
Double Rack	Module	4068	± 1270	184	
Double Rack	Module	5128	± 1270	184	
Experiment Heat Exchanger	Module	3090	405	– 1770	
Thermal Capacitor Cold Plate Assembly	Module	405	– 1090	– 1090	
400 Hz Inverter	Module	2765	– 1240	– 820	
400 Hz Inverter	Igloo				
Experiment Computer	Module	1080	1450	– 370	
Experiment Computer	Igloo				
Experiment I/O Unit	Module	1080	1450	– 630	
Experiment I/O Unit	Igloo				
DDU / Keyboard	Module	1370	– 1420	– 60	
High Rate Digital Recorder	Module	2840	– 800	1140	
High Rate Digital Recorder	+ AFD	9517	0	869	
High Rate Multiplexer	Module	TBD	TBD	TBD	
Ceiling Stowage Containers	Module	1097	0	1438	
Ceiling Stowage Containers	Module	1467	0	1438	
Ceiling Stowage Containers	Module	1837	0	1438	
Ceiling Stowage Containers	Module	2207	0	1438	
Ceiling Stowage Containers	Module	2577	0	1438	
Ceiling Stowage Containers	Module	2947	0	1438	
Ceiling Stowage Containers	Module	3317	0	1438	
Ceiling Stowage Containers	Module	3787	0	1438	
Ceiling Stowage Containers	Module	4157	0	1438	
Ceiling Stowage Containers	Module	4527	0	1438	
Ceiling Stowage Containers	Module	4897	0	1438	
Ceiling Stowage Containers	Module	5267	0	1438	
Ceiling Stowage Containers	Module	5637	0	1438	
Ceiling Stowage Containers	Module	6007	0	1438	
Top Airlock	Module	2231	10	1640	
Top Airlock	Module	4924	10	1640	
Aft Airlock	Module	4151	25	– 50	
Aft Airlock	Module	6844	25	– 50	
High Quality Window Assy	Module	2097	0	2100	
High Quality Window Assy	Module	4786	0	2100	
Removable Top Cover	Module	2206	0	2050	
Removable Top Cover	Module	4899	0	2050	
Aft Cone End Plate	Short Module	4359	0	0	
Aft Cone End Plate	Long Module	7061	0	0	
Film Vault	Module	–	–	–	

\+ Orbiter Aft Flight Deck (AFD) located equipment uses Spacelab System Coordinate System

3.3 Module

3.3.1 Overall Configuration

The pressurized module consists of a combination of either one or two 4060 mm diameter cylindrical segments (see Figure 3-14) of 2694 mm length, each segment being equipped with a flange ring of 1300 mm internal diameter on the top (CPSE opening) to provide accommodation for the following mission dependent items:

- top airlock or
- optical window/viewport assembly

or

- high vacuum vent facility (not yet baseline but under definition)

When not used for any of the above items, the hole is closed with a coverplate.

Note: Planned and/or contingency access constraints during ground operations (late access in vertical position) may not allow the use of top airlock or high vacuum vent facility in the CPSE opening of the core segment.

The module end closures are conical sections of equal cone angle. The forward end cone is truncated at the diameter required to interface with the crew transfer tunnel which connects to the Orbiter. The aft end cone is truncated to provide an opening for the aft airlock or for the high vacuum vent facility. The module exterior is covered with high-performance insulation. EVA mobility aids are also located at the exterior.

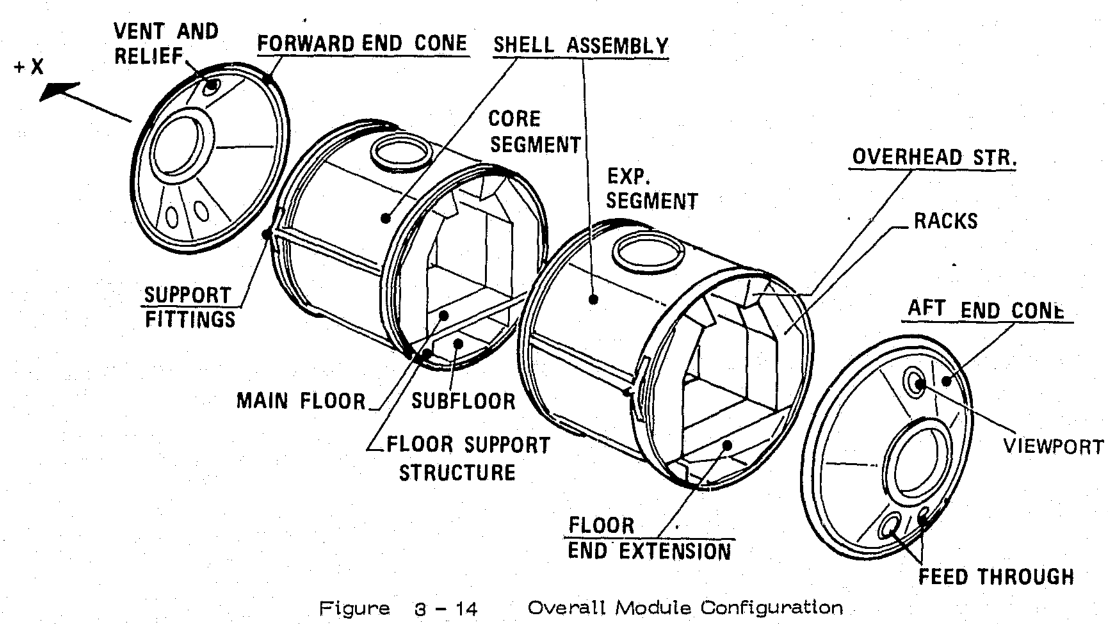


Figure 3 - 14 Overall Module Configuration

3.3.2 Accommodation Capability

All module flight configurations contain the same basic internal arrangement of subsystem equipment, the main difference being the volume available for experiment installation.

Subsystem equipment is located forward, primarily in the core segment, and experiment equipment is located primarily aft in the experiment segment. However, about 60 % of the volume in the core module segment is also available for experiments.

The floor is designed to carry the racks with their equipment and consists of segments which may be interconnected at the integration site and during transport. These segments allow the adaptability of the secondary structure to both module sizes. The floor itself consists of a load-carrying beam structure and is covered by panels on the main walking surface providing also for noise attenuation. The floor also contains openings equipped with debris traps to allow cabin air return flow. Except for the center floor plates, the panels are hinged to allow underfloor access in orbit and on the ground, as can be seen in Figure 3 - 15. Major features shown are the floor with the equipment rack assemblies pre-integrated. If experiment racks are replaced by stand-alone experiment equipment, the same attachment points as those for racks have to be used.

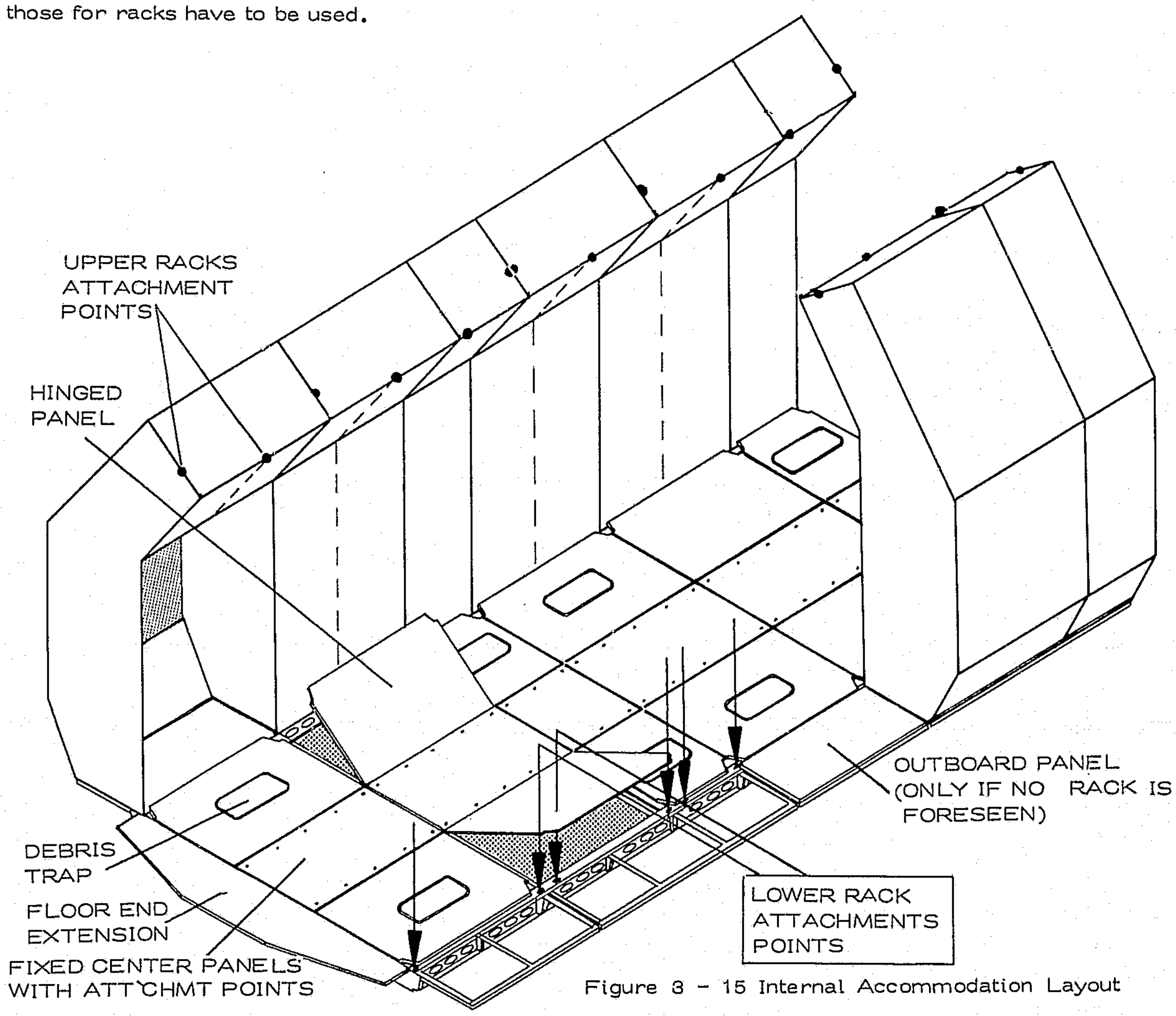


Figure 3 - 15 Internal Accommodation Layout

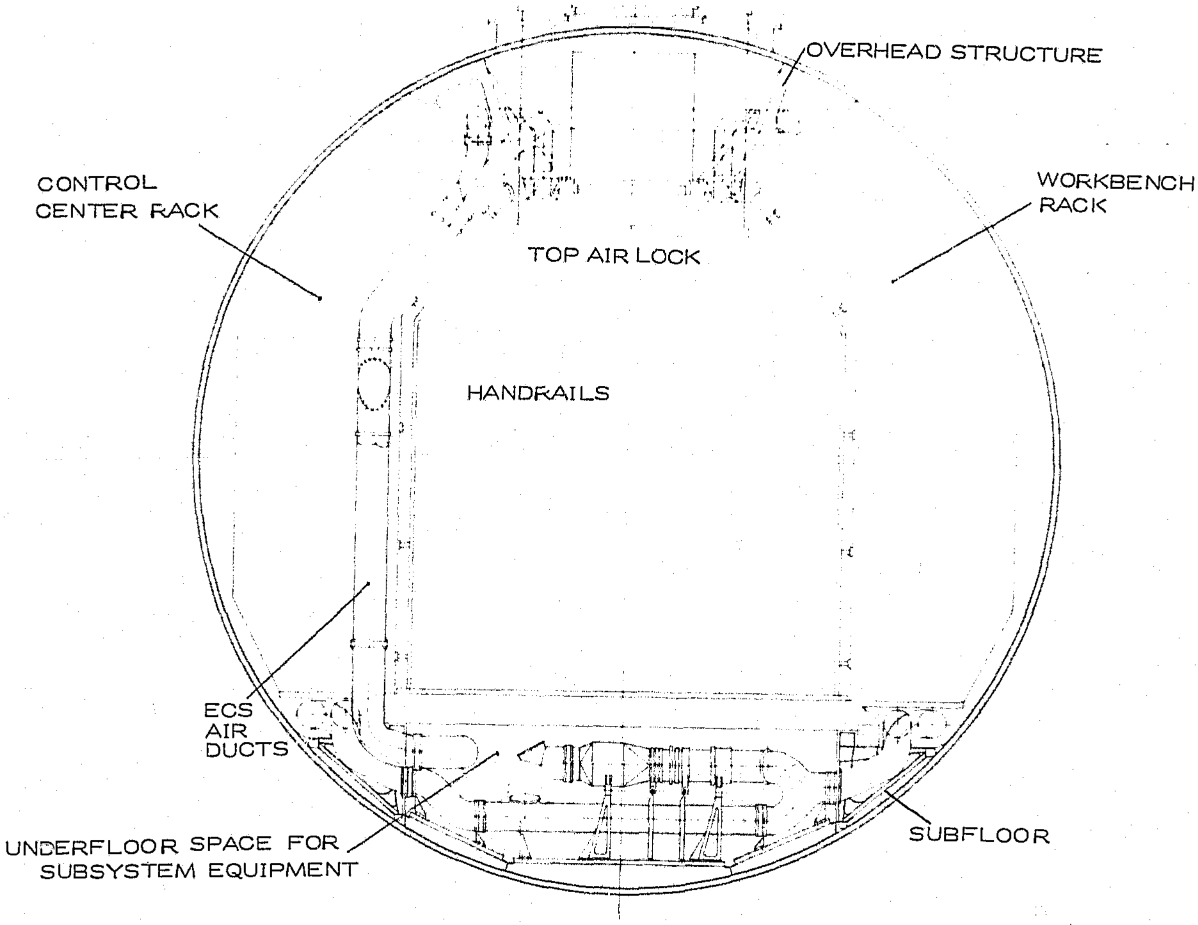


Figure 3-16 Core Segment Frontal View
(Forward End Cone Removed)

Figure 3-16 shows a frontal view of the core segment. Figures 3-17 and 3-18 show a longitudinal section through the module and illustrate the subsystem arrangements, top airlock, the air ducts and the underfloor subsystem equipment.

The underfloor subsystem equipment is mounted on a 2.7 m subfloor attached to a primary structure.

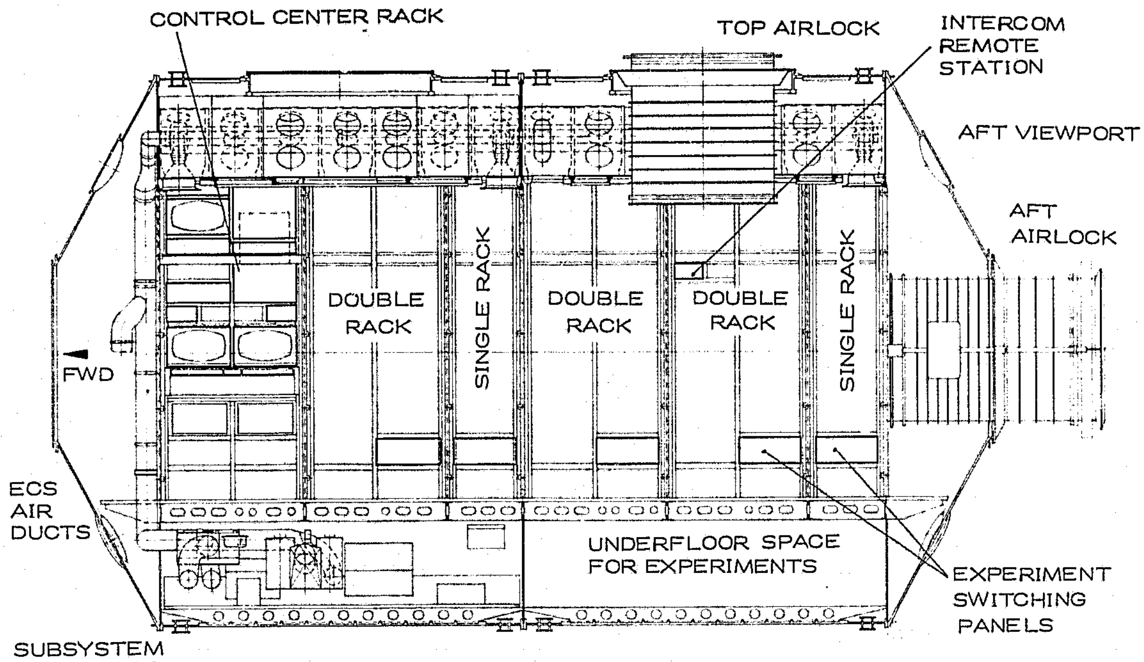


Figure 3- 17 Sectional View Starboardside

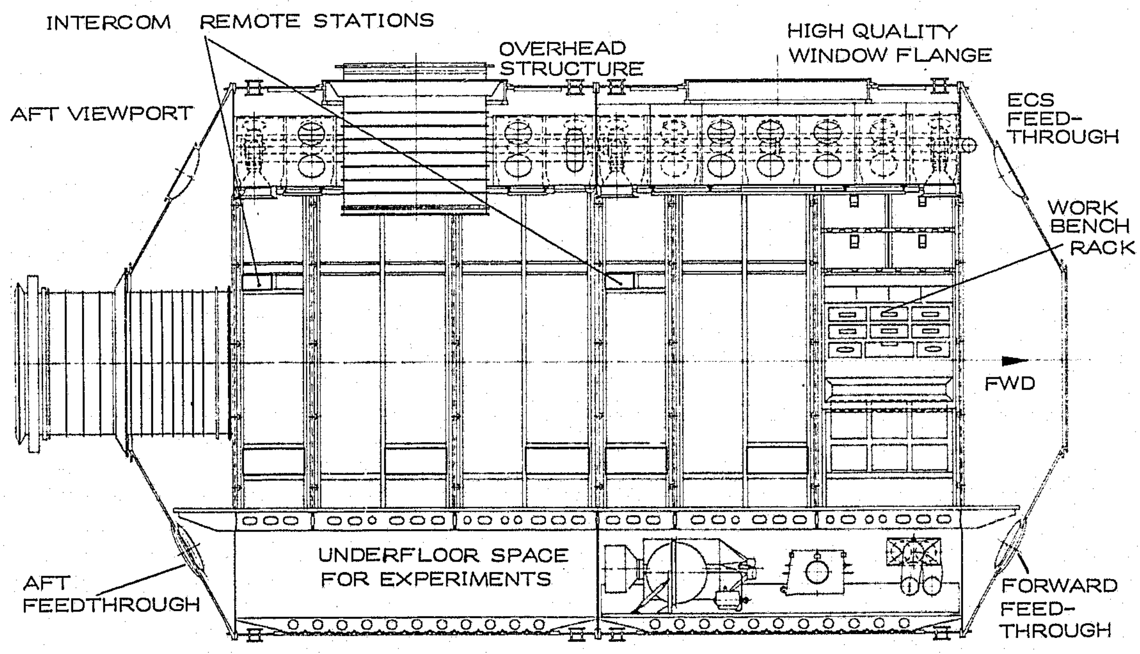


Figure 3- 18 Sectional View-Portside

The racks are standard 19 inch racks to accommodate standard as well as non-standard laboratory equipment. The total number of experiment racks is two double and two single in the core segment and four double and two single in the experiment segment.

Figure 3 - 19 depicts a typical cut away section of the core segment, showing the Control Center Rack in the forward part and the space available for experiments in the aft part.

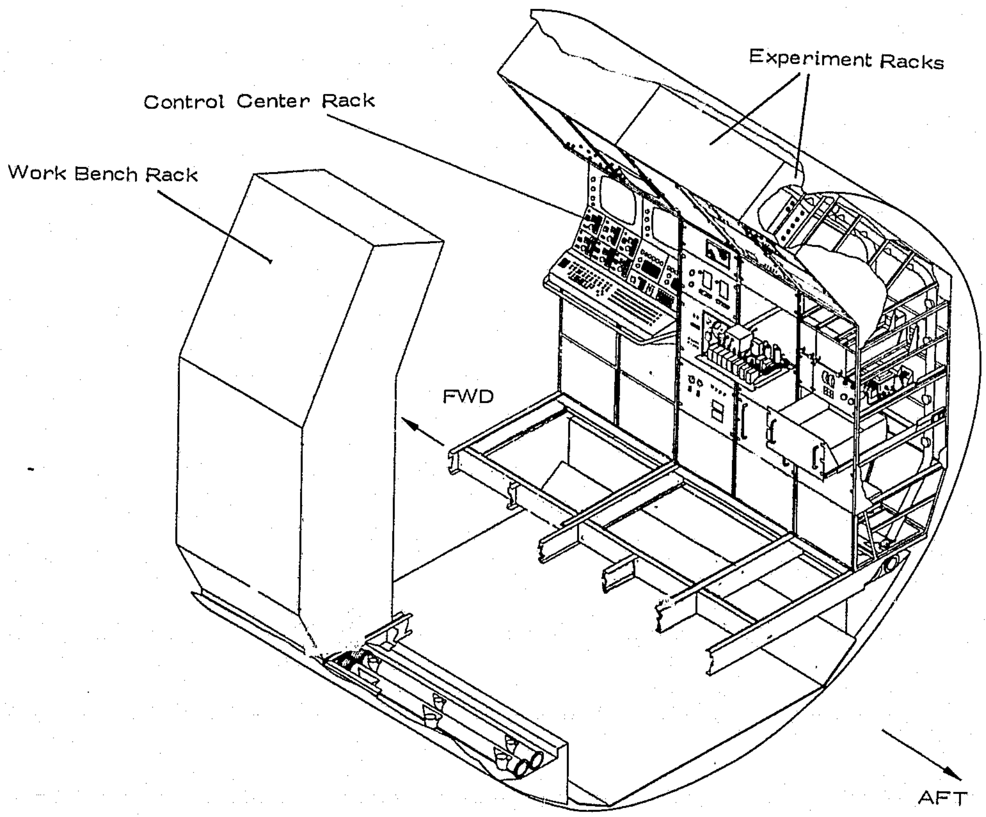


Figure 3- 19 Core Segment Cut-Away View-Starboard

Payload equipment (with or without racks) will normally be integrated with the floor structure when this is removed from the module. The complete floor/payload assembly will then be integrated with the module.

There is only a single interface plane between the subsystem rack assembly and experiment racks for electrical and avionics cooling loop connections after roll-in and before roll-out of the floor. The roll-in, roll-out concept for loading and unloading rack assemblies is shown in Figure 3- 20.

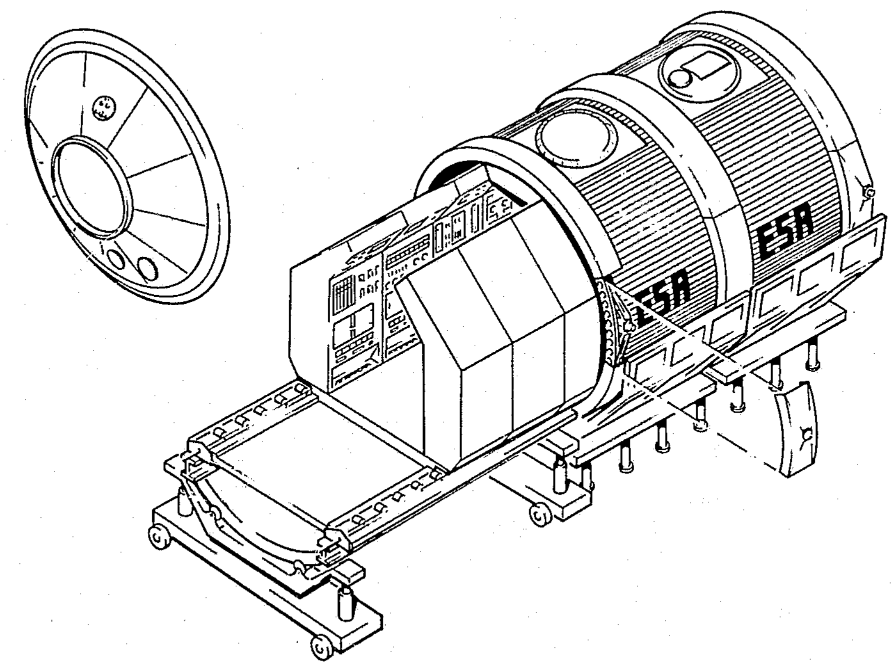


Figure 3-20: Loading/Unloading Concept of Rack Assembly

3.4 Crew Station and Habitability

The Spacelab design provides a 1-g longitudinal floor arrangement (Figure 3-21). The interior, however, is sized and shaped to allow optimum 0-g crew task performance. Foot restraints, handholds, and mobility aids are provided throughout the Spacelab to permit the crewmen to perform all physical tasks safely and efficiently in the most favorable body position. EVA mobility aids are located on the exterior for EVA operations.

The nominal illumination level in the module is 200 - 300 lumen/meter^2 but increases to 400 - 600 lumen/meter^2 at the work bench. Reduction of illumination level for handling sensitive films, reading faint images on a CRT, etc. can be provided by selectively turning out individual lights.

Spacelab acoustic noise is limited to a level compatible with NC 50. The cabin air temperature level is controllable between 18 and 27° C. The air flow rate is between 5 and 12 meters/minute and is directionally controllable. Tools are provided for spares replacement and contingency maintenance (there is no scheduled maintenance except replacement of LiOH containers). Storage bags, restraining straps and writing instruments are also provided.

3.4.1 Utility Work Bench

A work bench is provided in the core segment. This is intended to support work activities that are general in nature and not associated with a unique experiment.

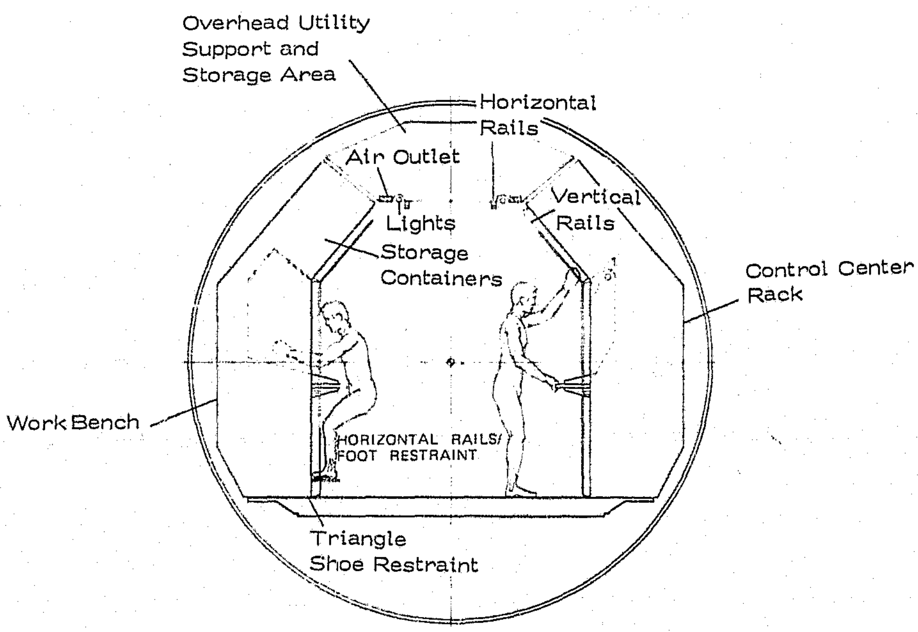
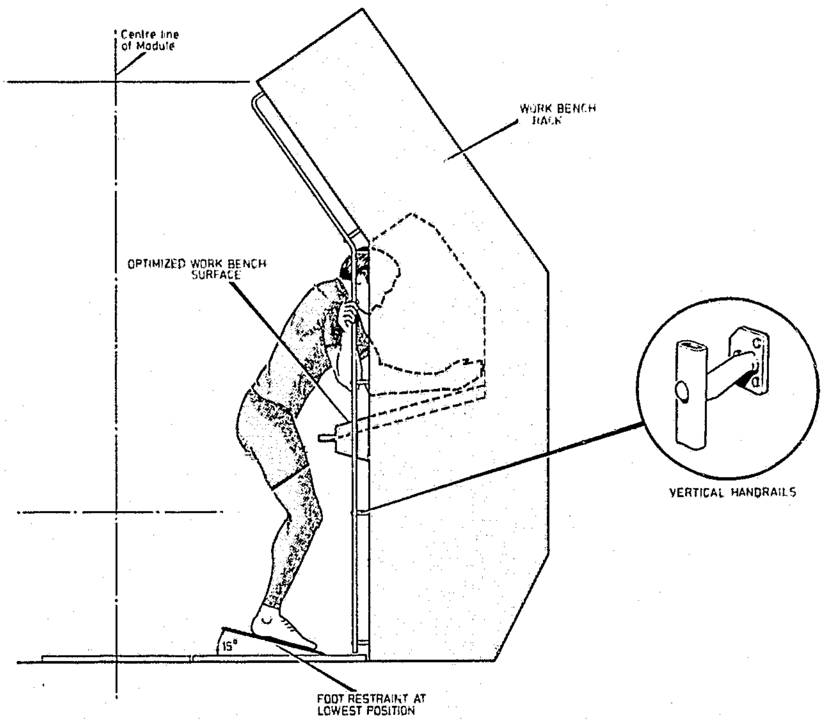


Figure 3- 21 Primary Crew Working Area (looking forward)

The design of the work bench provides optimum visual and reach capabilities for the operating crewman (Figure 3-22).



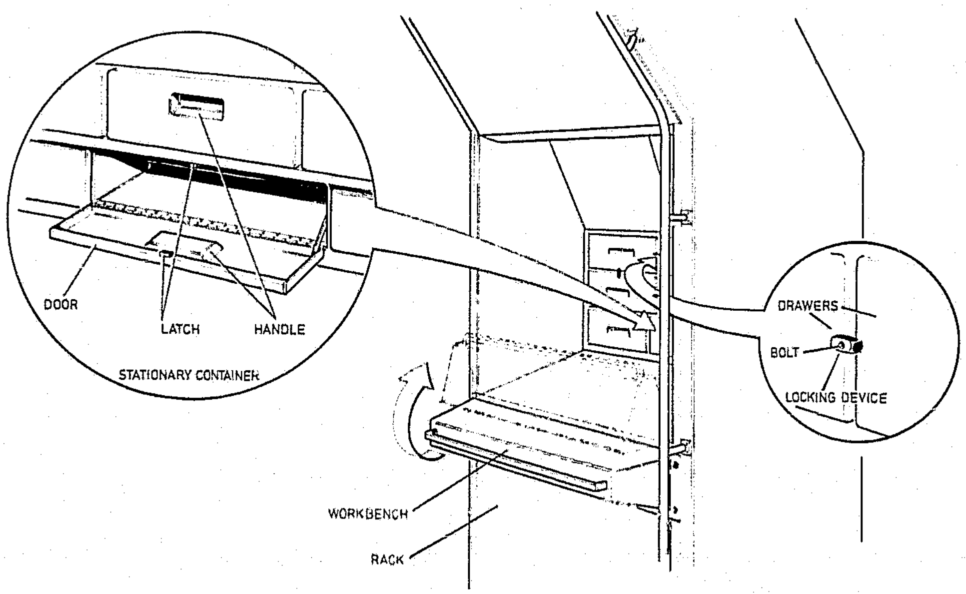


Figure 3- 22 Work Bench

Associated with the work bench are storage facilities such as utility drawers, file cabinets and tissue dispensers.

The work bench has lighting provisions installed in a recessed area above the primary working surface.

One electrical outlet, 28 VDC-100W, is available to support experiment activities. Other equipments associated with the work bench are:

- Tissues/wipes available for housekeeping tasks
- Writing instruments consisting of pens, pencils, markers, penlight writing paper, straight edges, etc.
- A stowage pouch offers individual compartments for items allowing easy identification and access for removal of single items without interfering with the stowed configuration of the remaining equipment

3.4.2 Stowage Container

The stowage containers provide storage space for experiment hardware, spare parts, consumables and other loose equipment. Four stowage containers are provided in the work bench and their allocation between Spacelab and experiment usage is not yet finalized, but it is expected that two containers will be available for payload use on most missions. 8 stowage containers are provided in the module ceiling and are exclusively for use by the experimenter (Figure 3- 23).

It is noted that a space for accommodation of up to 7 ceiling stowage containers per each module segment is available if neither the airlock nor the high quality window are used. It is in principle possible to accommodate up to 14 containers in a long module configuration but only 8 are delivered as Spacelab deliverable hardware.

The use of the high quality window reduces by 3 or more the number of possible stowage containers, depending on the space required for payload activities; the use of the top airlock reduces it by 6.

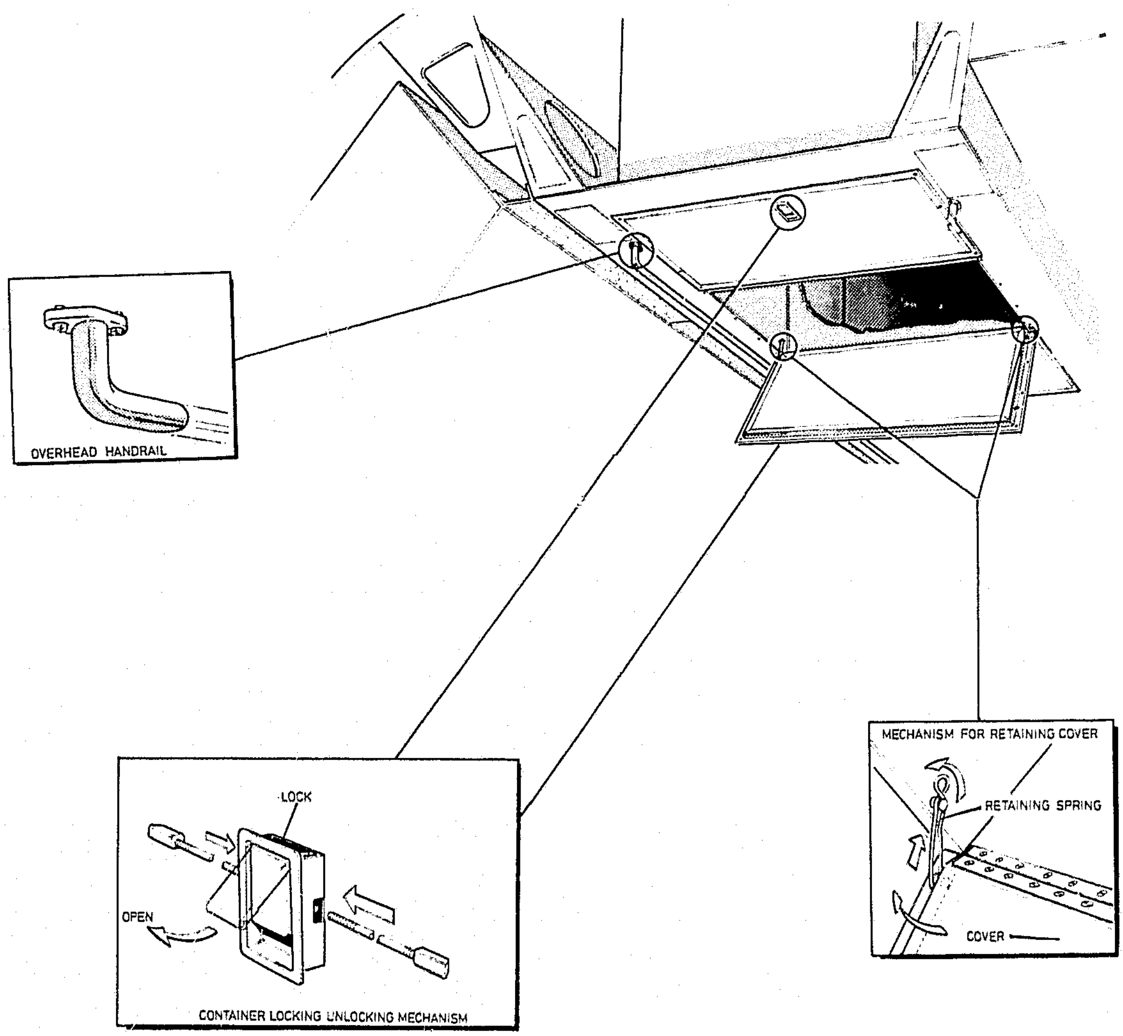


Figure 3 – 23 Typical Ceiling Stowage Container

Figure 3- 24 shows a work bench container. A grid pattern of mounting holes is provided inside the container for the attachment of internal restraints. During on-orbit access the door of the container can be locked in the open position. The characteristics are given in the following Table 3-8:

Table 3- 8 Characteristics of Stowage Containers

	Size (mm) A	B	C	Volume (m^3)	Loading capability (kg)
Work Bench Container	338	446	496	0.075	22.4
Ceiling Container	340	580	600	0.12	36

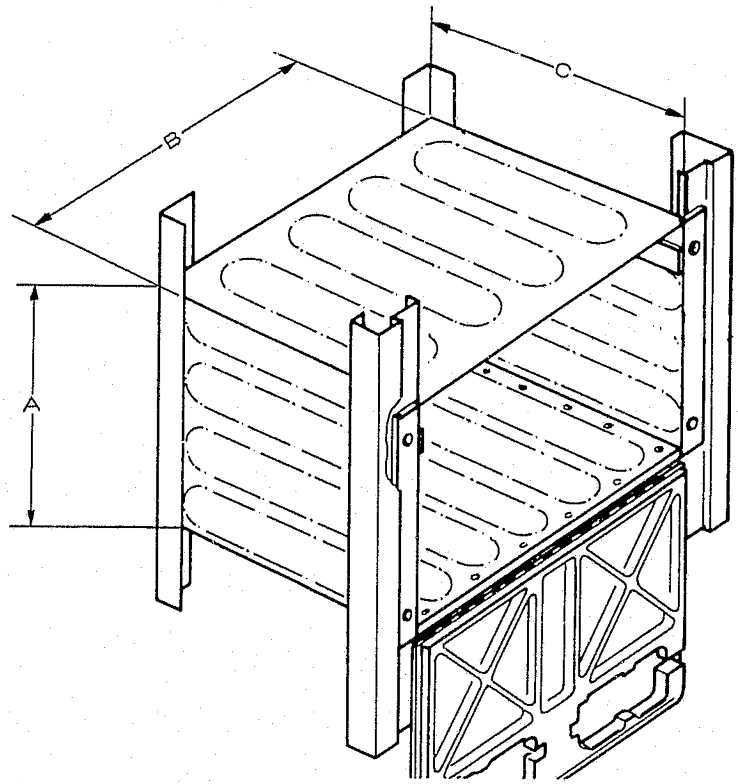


Figure 3- 24 Typical Workbench Stowage Container

3.4.3 Standard Equipment

A range of standard support equipment is located in the stowage containers at the work bench. The tools and maintenance equipment are also used to support Spacelab activities but will be available for experiment use.

3.4.3.1 Tool and Maintenance Assembly

A tool and maintenance assembly is provided, details will be subject to the outcome of an ongoing tool study: e.g.

- standard tools (off-the shelf tools)
- special tools (uniquely designed tools)
- maintenance equipment (tape, inspection mirror, etc.)
- utility box (to stow the tool and maintenance equipment)

3.4.3.2 Trash Disposal Bag

The trash disposal bag shown in Figure 3-25 provides for collection and stowage of trash. It is also used for temporary stowage and/or transportation of loose equipment.

Seven trash/disposal bags are provided folded and packaged together at launch in a stowage container. Snaps are provided in several locations in Spacelab to provide inflight hangers for the bags while in use. Full bags are stowed in TBD locations for return. The use of one bag per day for trash collection is assumed. The trash disposal bag will hold approximately 0.015 m^3 of trash.

The bags are available for containing dry trash as well as controlled liquid trash (such as moist towels or sponges) but no hazardous material can be placed into the bags. For potentially hazardous waste material the experimenter must provide the hardware and a special system to be used in dealing with such waste will be established.

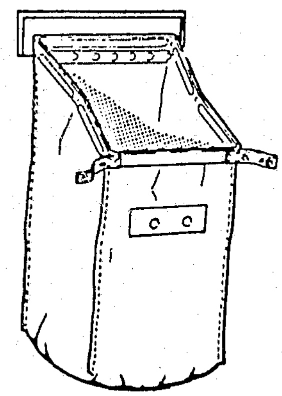


Figure 3 - 25 Trash Disposal Bag

For stowage of wet trash, waterproofed liners with simple twist tag closure will be provided.

3.4.3.3 On-Orbit Equipment Restraints and Stowage Provisions

On-Orbit restraints (utility straps and bungees) are provided which allow the crewman to temporarily restrain equipment at various locations throughout Spacelab. These utility straps and bungees provide a general capability for restraining objects in zero-g and are available for planned and unplanned restraint requirements. Attachment provisions for these on-orbit equipment restraints will be made available at TBD positions throughout Spacelab, subject to detail design.

3.4.4 Crew Restraints/Mobility Aids

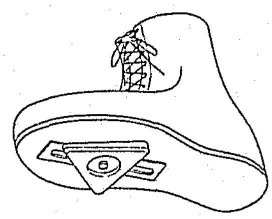
Several types of restraints and mobility aids are available for crew utilization: foot restraints, handholds, and locomotion aids. These restraints/mobility aids are situated throughout the vehicle for planned and unplanned tasks.

3.4.4.1 Foot Restraints

The foot restraint system in Spacelab and Orbiter will be compatible; the basic foot restraint system used for Spacelab is the Skylab-type triangle cleat shoe (see Figure 3-26). The triangle cleat at the shoe fits into a triangular isogrid structure. This structure can be mounted to the handrails which are available at each rack providing vertical positioning (see Figure 3-27).

In addition to the foot restraints at the consoles, portable foot restraints are provided to mount in a selected mounting pattern on the floor.

Figure 3 - 26 Triangle Shoe



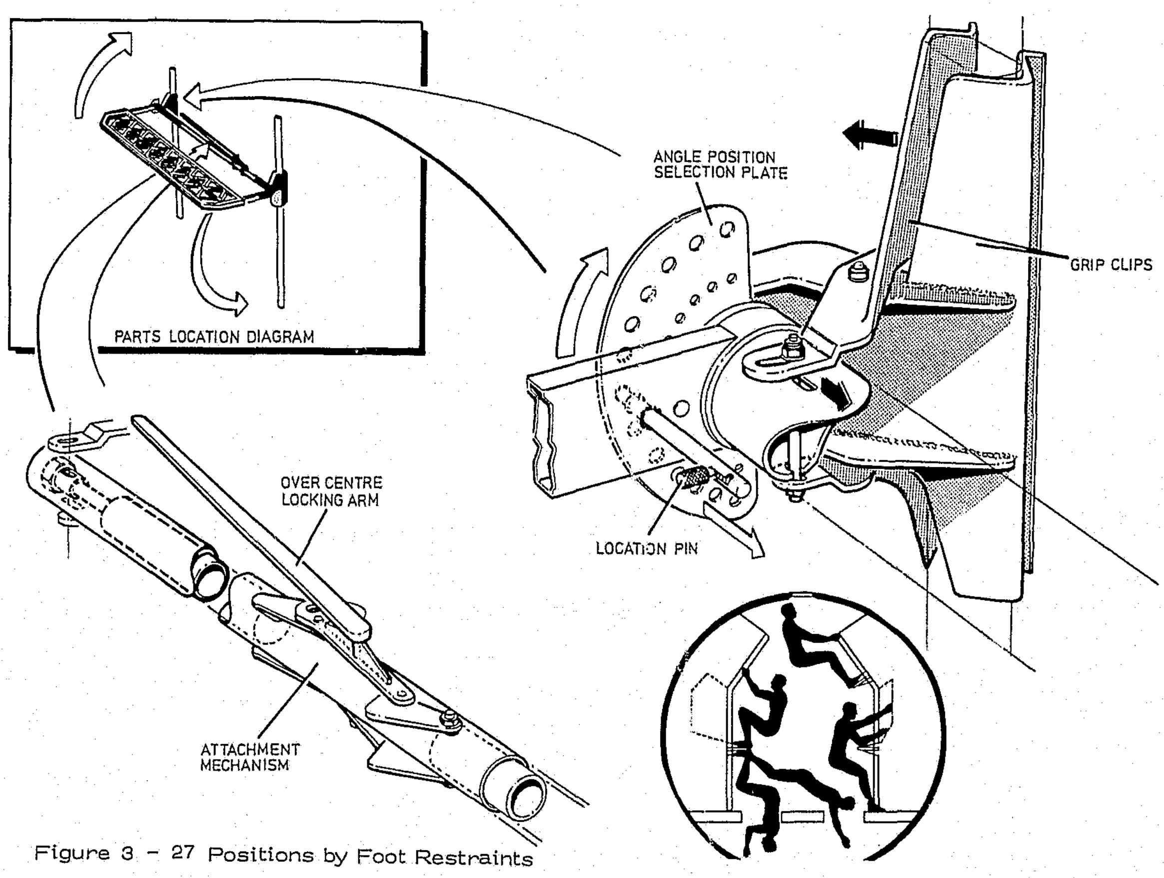


Figure 3 - 27 Positions by Foot Restraints

3.4.4.2 Locomotion Aids and Handholds

For crew activities inside the module, fixed handrails and handholds are provided throughout the habitable area. These devices aid crew member translation through the module and also provide a means of body stabilization while performing tasks in the immediate vicinity of the handrail or handhold.

The primary standard locations of locomotion aids and handholds are:

- Vertical handrails attached to the standard racks (Figure 3-28)
- Horizontal handrails attached along the overhead utilities/storage support structure (Figure 3 -29)

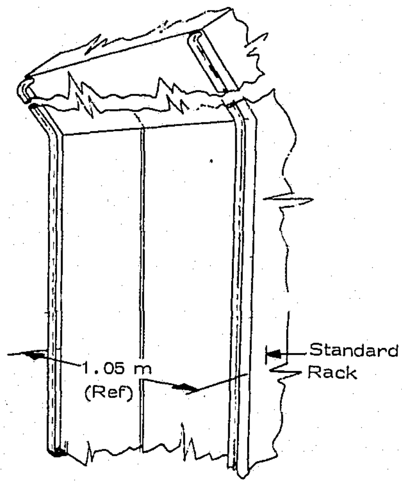
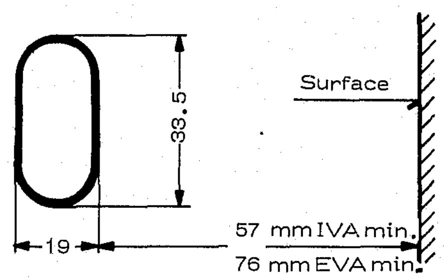


Figure 3 - 28 Vertical Handrails



Inside Module and EVA Handhold Cross Section

Note the increased spacing in the case of EVA

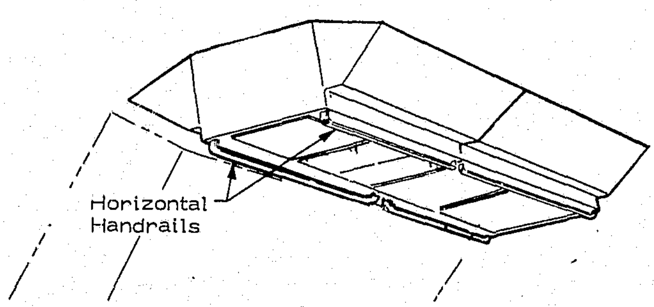


Figure 3 - 29 Horizontal Handrails

Other areas recognized for handrail placement are:

- At the inboard edge of console/work bench shelves
- On installed airlocks
- On interior structure around hatches and windows

3.4.4.3 EVA Restraint/Mobility Aids

EVA egress will be through the airlock in the Spacelab tunnel adapter, or through the docking module, depending upon the mission configuration. The size of the airlock and associated hatches limits the external dimensions of packages that can be transferred to payloads of 0.9 m diameter and 1.4 m length. The Orbiter provides the equipment and consumables required to support three two-man EVA operations. Two of these three operations may be utilized by the payload for either planned or unscheduled EVA operations. Additional EVA operations in support of payloads are possible with the expendables being provided as a payload mass chargeable item. EVA operations will utilize a self-contained life support system capable of supporting a six-hour EVA. At least three (3) hours of oxygen prebreathing is required. Post EVA operations take approximately 1.5 hours.

Most of the first two (2) hours of the three (3) hour prebreathing, scheduled to begin 3.5 hours before the start of an EVA, can be used to accomplish useful, non-EVA related activities by the EVA crewmen. The remaining 1.5 hours are used for EVA preparation.

Spacelab provisions for EVA have been made to allow the pressure suited crewman to perform EVA translation from the EVA hatch, up the end cone of the module, over the module, down the aft cone and along the pallet as shown in Figure 3-30.

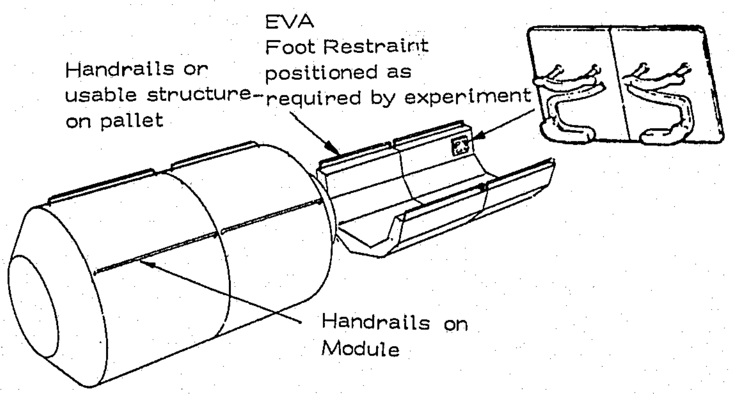


Figure 3 - 30 EVA Mobility Aids and Restraints

The details of the handhold restraints provided on the outer surface of the Spacelab module are as follows. Each end cone and the module cylinder provides EVA handrails. The cylindrical area has two sets of rails providing the crewman with the capability of translating on either the port or starboard side of the module. The handrails on the pallet are shown along the upper sill of the pallet. Provisions on the pallet are sufficiently flexible to allow for installation of handrails and EVA foot restraints as required for each mission. A standard man-mission model for EVA is shown in Figure 3-31 and Table 3-9 (Ref. doc. JSC-07700 Vol. XIV, Appendix D).

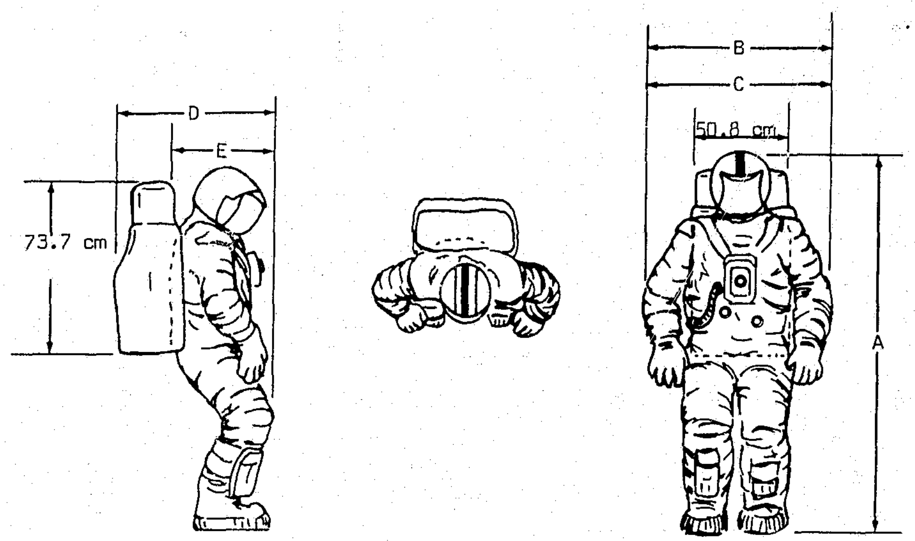


Figure 3-31 : Standard Man Dimensions for EVA

Table 3- 9 Standard Man Data for EVA

Dimensions (cm)	Percentile Man	
	5 %	95 %
A - Height	171.5	191.8
B - Maximum Breadth at Elbows (Arms Relaxed)	-	74.7
C - Maximum Breadth at Elbows (Arms at Side)	-	67.1
D - Maximum Depth with Portable Life Support Subsystem (PLSS) and Backup Oxygen (OPS)	66	72.1
E - Maximum Depth without PLSS	39.4	45.4
Mass (kg), with PLSS/OPS	TBD	TBD
Mass (kg), without PLSS/OPS	86.3	117.6

Weight benefits for payloads requiring no EVA are TBD

3.5 Pallet Segment

3.5.1 Basic Configuration

The pallet cross-section is U-shaped and of aeronautic-type construction. It provides hard points for mounting heavy experiments and a large panel surface area to accommodate lighter payload elements. Pallet segments are of 3 m nominal length and can be flown independently or interconnected. As many as three pallets can be interconnected to one pallet train and supported by one set of retention fittings.

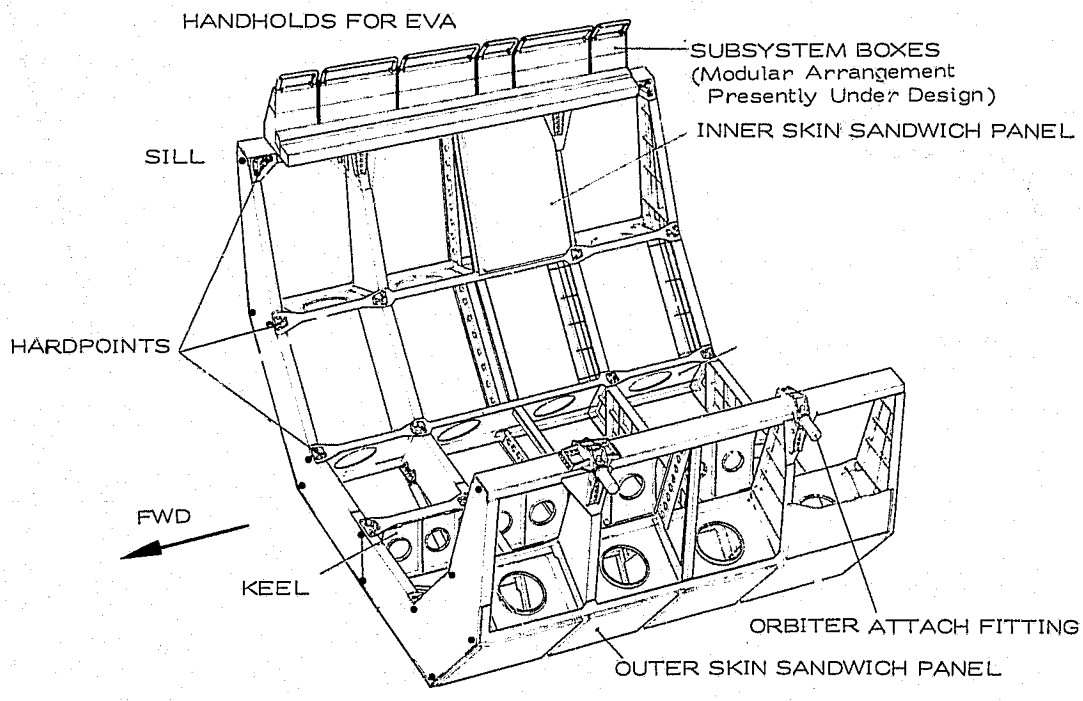


Figure 3 - 32 Pallet Segment

3.5.2 Accommodation Capabilities

Figure 3-32 shows a basic pallet segment with hardpoints and typical honeycomb sandwich skin panels. Each segment consists of the basic structure plus additional mission dependent and mission independent subsystem equipment which includes:

- subsystem and experiment electrical power busses
- experiment power distribution boxes
- subsystem and experiment data busses
- one subsystem and up to 4 experiment remote acquisition units (RAU)
- thermal insulation blankets
- cold plates and thermal capacitors
- plumbing

Details of cold plates/thermal capacitors and cold plate/thermal capacitor mounting are described in Section 4.3

A pallet-only configuration may consist of one to five standard pallet segments. Spacelab subsystem equipment, which in other flight modes is integrated within the module, is installed within the Igloo in the pallet-only mode, except operator interface equipment, such as CRT's, keyboards, TV monitor and SL control panels, which are not used in pallet-only modes since these are controlled from the AFD.

For reasons of thermal control and easy accessibility, pallet subsystem equipment will be mounted on the pallet sill, including subsystem cable harnesses. The arrangement will be modular and the location of the boxes along sill may be variable. Thus certain areas of the sill may be kept clear to avoid interference with payloads, e.g. with the field-of-view or pointing cone requirements of pallet-mounted telescopes.

3.5.3 Igloo

This Igloo is a one atmosphere pressurized cylinder as shown in Figure 3-33, used only for subsystem equipment. It is equipped with a removable bulkhead (Marman clamp) providing full access to the interior.

The weight of an equipped igloo is approximately 640 kg, the usable volume 2.2 m^3. The internal temperature is compatible with CAM equipment requirements and is achieved by active and passive thermal control devices .

The following is the list of the equipment (basic and mission dependent) which is foreseen to be mounted within the Igloo, however, some units might be pallet-mounted depending on the outcome of ongoing Igloo investigations:

- 3 computers
- 2 I/O units
- 1 mass memory
- 2 subsystem RAU's
- 1 experiment inverter (400 Hz)
- 1 subsystem inverter (400 Hz)
- 1 emergency box
- 1 power control box
- 1 subsystem power distribution box

- 1 Remote amplifier and advisory box (RAAB)
- 1 multiplexer
- 1 subsystem interconnecting station

The subsystem Igloo is mounted at the forward end of a pallet segment; as shown schematically in Figure 3-33. Final dimensions, however, may be subject to change.

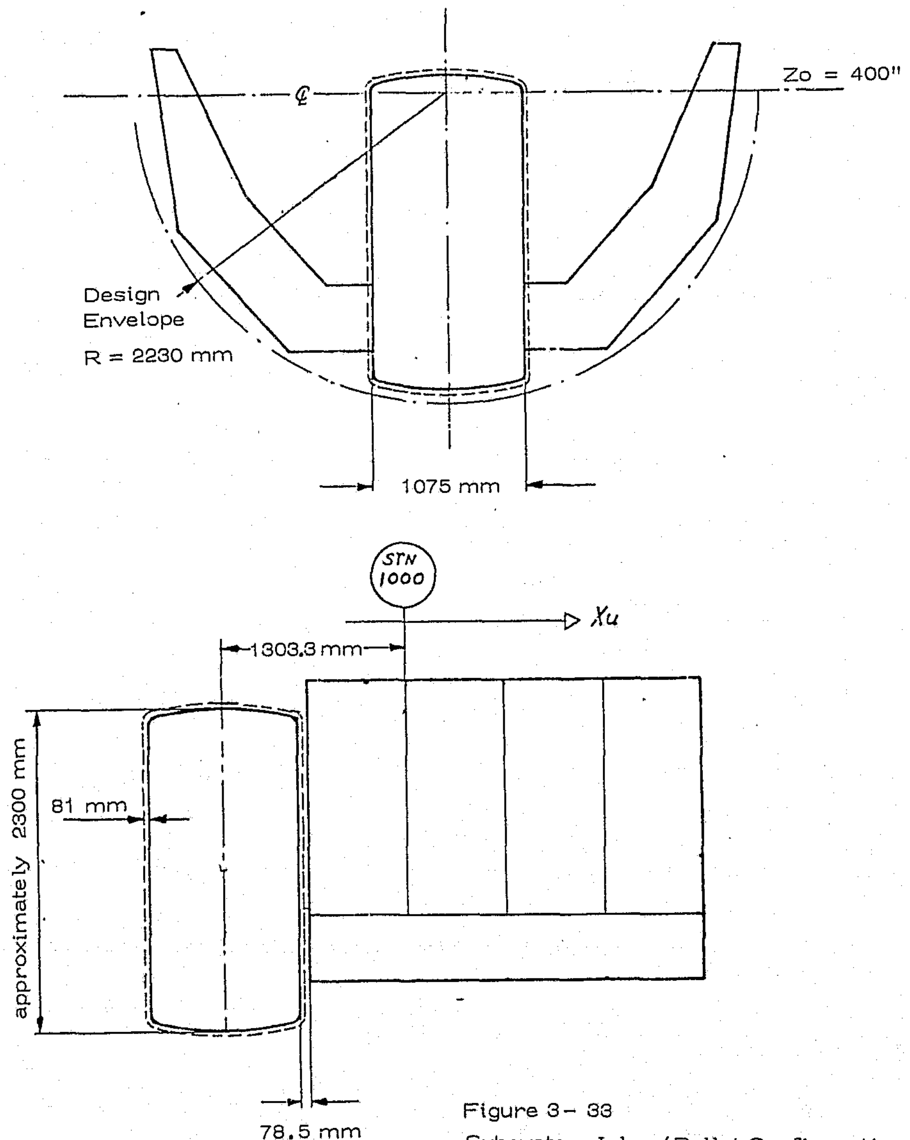


Figure 3-33
Subsystem Igloo / Pallet Configuration

3.6 Transfer Tunnel

Note: The transfer tunnel is NASA provided and information presented here is still preliminary.

The transfer tunnel will enable crew and equipment transfer between the Spacelab module and the Orbiter in a shirtsleeve environment. It is capable of functioning under orbital as well as ground operation conditions. It will have a minimum of about 1 m clear diameter, sufficient for handling a box of 0.56 x 0.56 x 1.27 m size and moving of a 1.95 m tall EVA man with a maximum elbow width of 0.75 m. The same internal atmosphere as in the Spacelab module is provided. Lighting is installed in the tunnel, as well as mobility aids for internal movements.

Figure 3 - 34 shows in a simplified form the mode of interfaces with the Orbiter and the Spacelab module. The tunnel adapter/airlock combination is provided by the Orbiter.

The tunnel consists of a S-shaped tunnel segment, a number of cylinder segments to accommodate different flight configuration locations and flexible elements for dynamic decoupling and tolerance compensation.

Equipment transfer during ground operations is described in Section 4.7, Mechanical Ground Support Equipment.

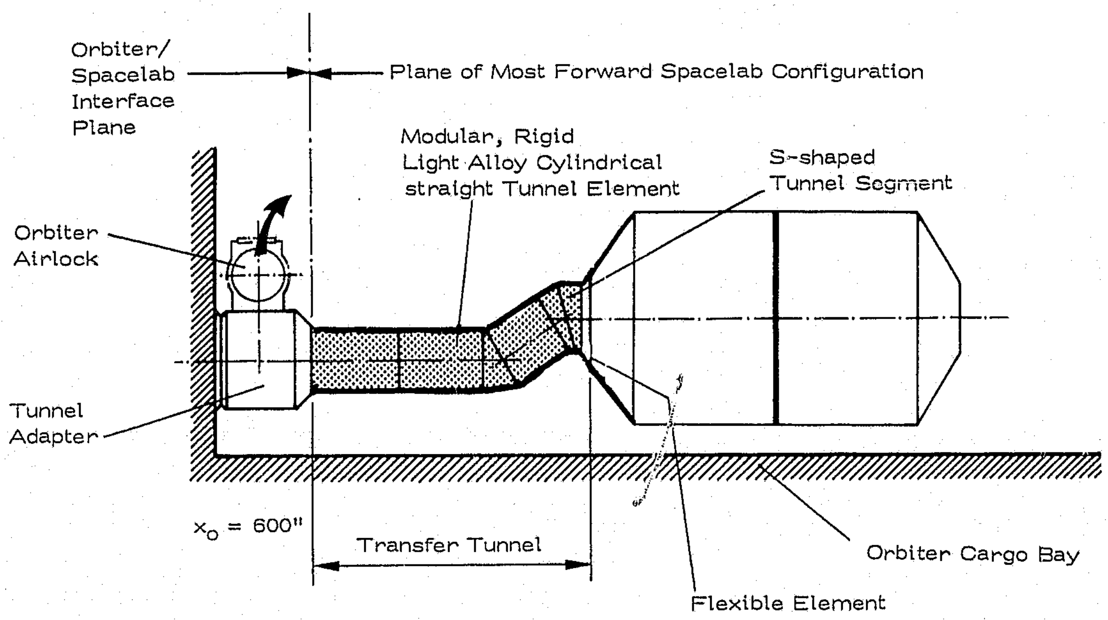


Figure 3 - 34 Transfer Tunnel

3.7 Utility Services

The routing of Spacelab utilities (signal, power and fluid lines) and utility interfaces, are shown schematically in Figure 3-35.

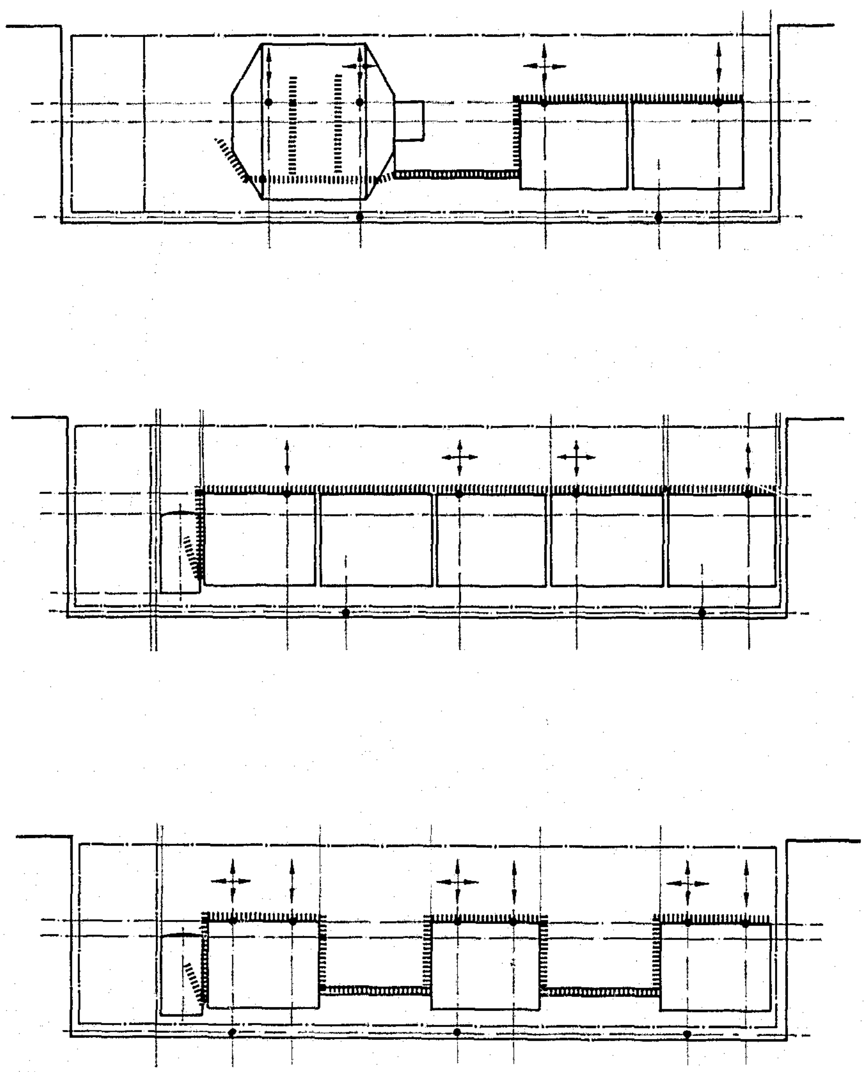


Figure 3 - 35 Schematic illustration of Spacelab utility routing

Principle utility routing and interfaces are as follows:

a) Module Configurations (Module-Only and Module-Pallet)
All utilities are routed from the Orbiter to feedthrough connectors and/or thermal control components (heat exchanger) on the forward end cone of the module. From there they are distributed inside the module and/or routed to feedthrough connectors in the aft end cone, except for freon lines which bypass the module interior and are routed externally to the pallet. From the aft end cone feedthrough connectors, utilities are routed to the pallet via the module-to-pallet utility bridge, except for the short module/3 pallet configuration which does not require the utility bridge (see Figure 3 - 5).

b) Pallet-Only Configurations
All utilities are routed from the Orbiter to feedthrough connectors in the Igloo and/or to thermal control components (heat exchanger) and interface plates on the pallet.
Utilities between pallet segments in the 9 m independently suspended pallet configuration are routed via interpallet utility bridges. No interpallet utility bridges are foreseen for utility routing between closely spaced pallet segments, e.g. in the 15 m pallet configuration or the module/pallet configurations.

3.7.1 Forward Utility Routing (Orbiter/Spacelab)

The utility lines and routing of utility lines from the Orbiter interface to the Spacelab interface are Orbiter provided. In addition to Spacelab power, signal and fluid lines, the following experiment dedicated lines are currently foreseen :

- 87 TSP (twisted shielded pairs), AWG 24
- 5 coax cables (3 x 75 ohm and 2 x 50 ohm)
- TBD fluid lines

These lines allow experiment equipment in the module and/or on the pallet to be connected with experimenter provided equipment in the Orbiter aft flight deck.

3.7.1.1 Module Only and Module/Pallet Modes

The experiment dedicated lines are routed to feedthrough connectors in the forward end cone (Figure 3-36). Inside the module, they are routed to connectors in a connector bracket in the core segment. This connector bracket, which is the interface to experiments, is located on the subfloor and accessible through the hinged panels of the mainfloor, as depicted in Figure 3-15.

3.7.1.2 Pallet-Only Modes

The experiment dedicated lines are routed to an electrical interface plate on the first pallet. Connector details are TBD.

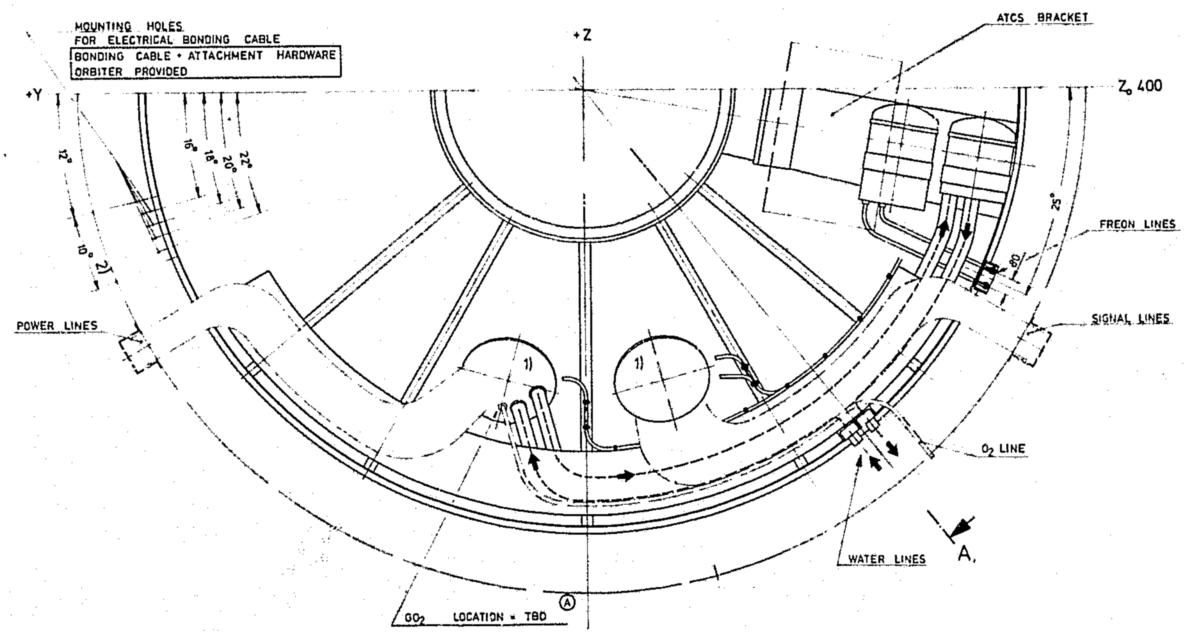


Figure 3 - 36 Forward Utility Routing

3.7.2 Module-to-Pallet Utility Routing

Spacelab subsystem and additional experiment provided utility lines are routed from feedthrough connectors in the aft end cone to the pallet via the module-to-pallet utility bridge (Figure 3 - 37).

Two feedthrough plates are provided in the aft end cone, one for Spacelab subsystem utilities and one for experiment provided utilities.

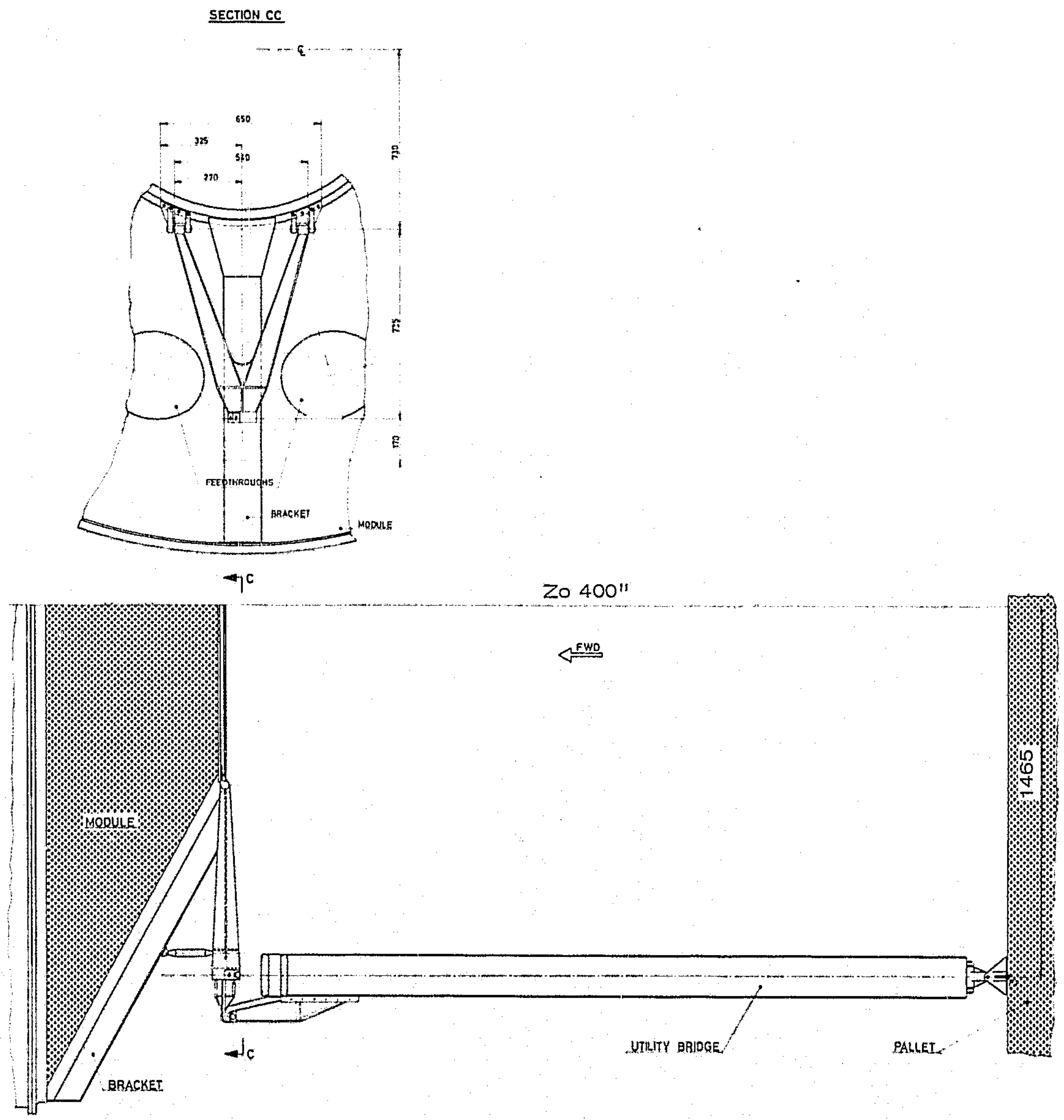


Figure 3- 37 Module-to-Pallet Utility Bridge

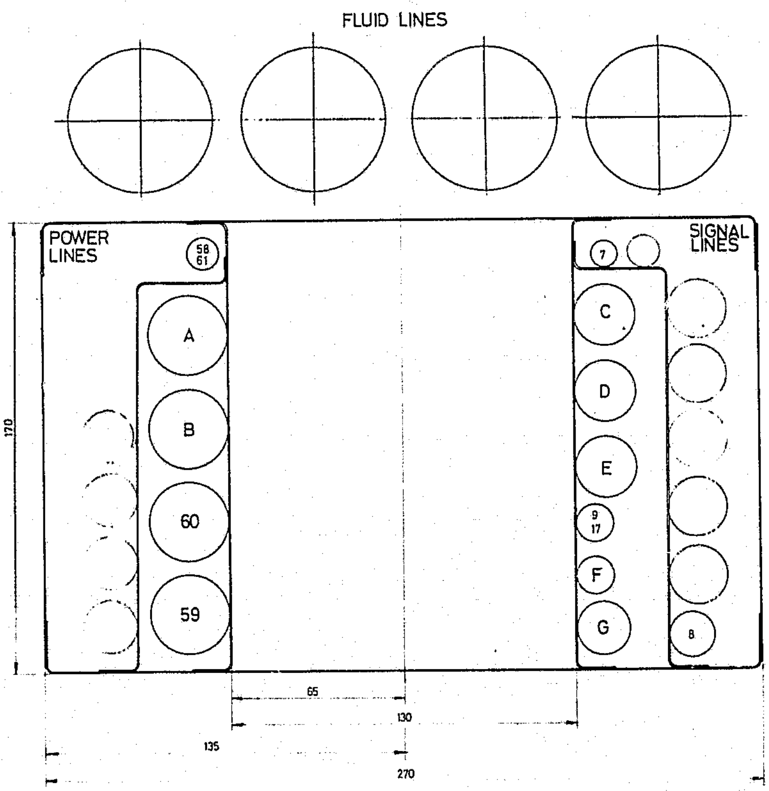
The experiment feedthrough plate (40 cm diameter) is a blank plate designed to accommodate at least the following lines:

100 TSP (twisted shielded pairs), AWG 24
100 SW (single wires), AWG 24
20 COAX cables
2 Power lines (5 kW), AWG 8
2 Fluid lines

Connectors and utility lines have to be experiment provided. The utility bridge is designed to accommodate the above listed lines in addition to the permanently installed subsystem lines.

A section through the utility bridge is shown in Figure 3 - 38, indicating the separate routing of power and signal lines in two parallel ducts to minimize EMI. The bridge is designed to allow the installation or replacement of the bridge as a unit or of each cable bundle and fluid individually. Utility bridges will be furnished in standard lengths for the applicable basic configurations.

On the pallet, Spacelab will offer provisions to mount interface connector plates for experiment provided utilities. Details are TBD.



SIGNAL LINES		
Nº	Ø	Class
7	10	II
8	17	III
9-17	14	IV

EXP. SIGNAL LINES		
Nº	Ø	Class
C	23	*
D	23	*
E	23	*
F	14	*
G	20	*

POWER LINES		
Nº	Ø	Class
59	30	I
60	30	I
58.61	12	I

EXP. POWER LINES		
Nº	Ø	Class
A	30	*
B	30	*

- FLUID LINES - Ø 55

* Assumptions on cable classification
- Cables are shielded (metallic barrier)
- No separation between cables in accord with SR-ER-0004 page 17 section 3.1.2.7.2

○ = Cable reserve

Figure 3-38 Cross Section and Cabling of Utility Bridge

3.7.3 Pallet-to-Pallet Utility Routing

Details of pallet-to-pallet utility routing are TBD. For closely spaced pallet and for pallet trains, utilities will be routed directly across pallet segments. For independently suspended pallet segments, utilities will be routed via pallet-to-pallet utility bridges. These have the same capabilities and are of similar design as the module-to-pallet utility bridge (Figure 3-39). Additional information on fluid line routing on pallets is given in Section 4.3.

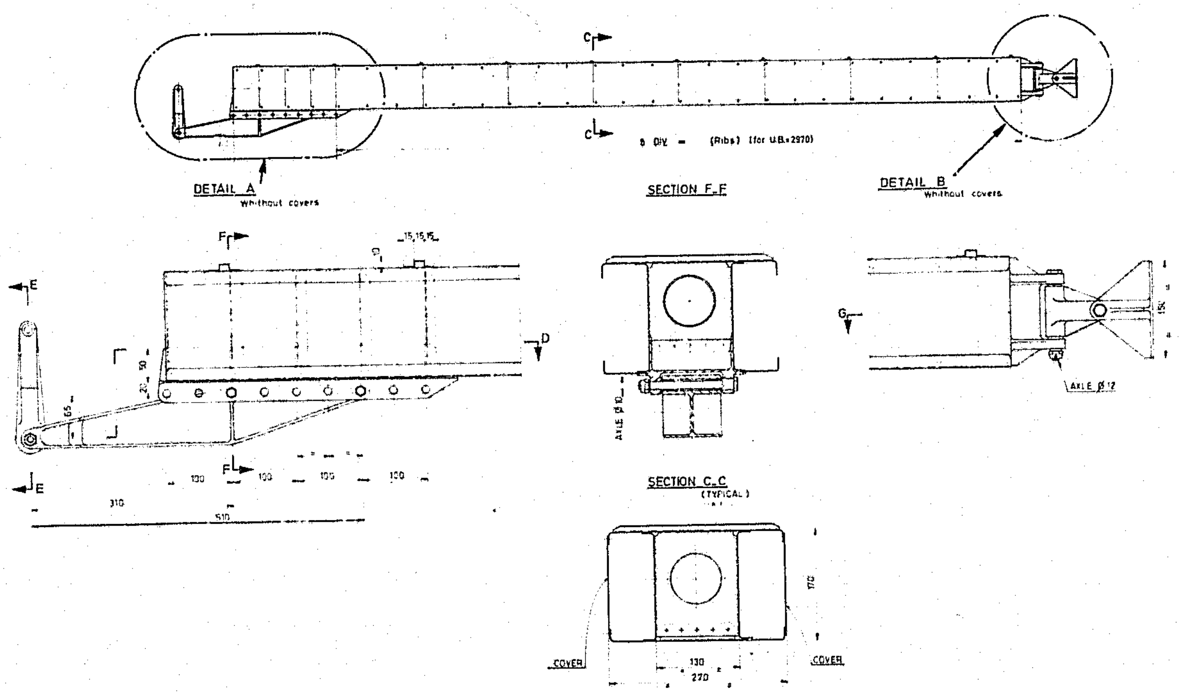


Figure 3-39 Inter-Pallet Utility Bridge

3.7.4 Routing of Experiment Provided Utilities Inside the Module

Experiment signal lines can be routed between experiment equipment inside the module (e.g. experiment racks, airlocks, center aisle, optical window, etc.), and the feedthrough connectors in the aft end cone and the connector bracket on the subfloor of the core segment. The cabling has to be experiment provided.

Routing of experiment signal lines between racks on the same and on opposite sides of the module will also be possible. However, additional experiment provided power lines can only be routed between racks on the same side of the module (e.g. from a power switching panel in one rack to experiment equipment in another one).

Also possible will be the routing of experiment cabling between experiment racks and airlocks, optical window, center aisle equipment, etc.

Detailed routing provisions offered by Spacelab for experiment provided cabling are under investigation and are TBD.

The routing of experiment fluid lines inside the module is TBD.

The routing of high power cables from the experiment feedthrough plate in the aft end cone is TBD.

3.7.5 Routing of Experiment Provided Utilities on the Pallet

The routing concept for experiment provided utilities and routing provisions offered by Spacelab are under investigation and are TBD. However, it should be assumed that experiment utilities will normally be routed on top of the inner pallet panels.

3.8 Extended Flight

The nominal on-orbit stay time of the current Orbiter/Spacelab design is seven days. Extended mission durations of up to 30 days can be achieved by the provision of additional hardware and consumables which by definition (para 3.2.4.2) are part of the Spacelab Payload Mass.

Spacelab Payload chargeables applicable for extended flights are summarized in the subsequent paragraphs.

3.8.1 Orbiter Related Items

- EPS kits for electrical energy
  - for Orbiter in excess of 7 days
  - for Spacelab and its payload in excess of 890 kWh
- Crew and crew support above Orbiter baseline
- N_2 for Orbiter cabin leakage
- RCS propellant and tankage in excess of Orbiter baseline.

3.8.2 Spacelab Related Items

- Additional nitrogen tanks above 12 day mission duration (24 kg for additional 12 days)
- Additional LiOH cartridges (4.5 kg every 0.75 days, based on 52 man-hours/day Spacelab occupation)

The Spacelab baseline includes the necessary provisions to accommodate the additional mission kits but does not include the kits themselves as deliverable items. With regard to condensate storage, periodic dumping of 47 kg of water appr. every 8 days is assumed.

For information only, tentative data on the weight impact of all mission duration dependent items are presented in Figure 3-40 for a total crew of four and seven, respectively, based on a power consumption of 12.5 kW for the Orbiter and 7 kW for Spacelab and Spacelab payload.

The most significant weight impact results from the additional energy kits. Energy kits 1 through 3 are located outside the dynamic envelope for Spacelab and its payload. All additional tanks, however, must be located within the dynamic envelope resulting in a significant volume penalty as well.

Thus the weight and volume requirements for additional energy kits will necessitate a Spacelab configuration optimization as a function of mission duration.

Due to the difference between "mass available for payloads" (column 6 and 7 of Table 3 – 4), i.e. the delta between Spacelab hardware weight and Orbiter payload launch and landing capabilities and the "nominal payload mass" (column 9 on Table 3 – 4) representing a reference condition based on Spacelab configurations including 50 % of mission dependent equipment, the weight of mission duration dependent items does not reduce the nominal payload mass immediately following mission day no. 7. For example, a long module configuration with a nominal payload mass of 5500 kg can be flown for 13 days. Only after the 13th day, the payload is reduced by approximately 295 kg/day. The corresponding mission durations for the other representative Spacelab configurations are:

Short module + 9 m pallet:	9 days with 5500 kg
15 m pallet:	12 days with 9000 kg
9 m pallet:	11 days with 9100 kg

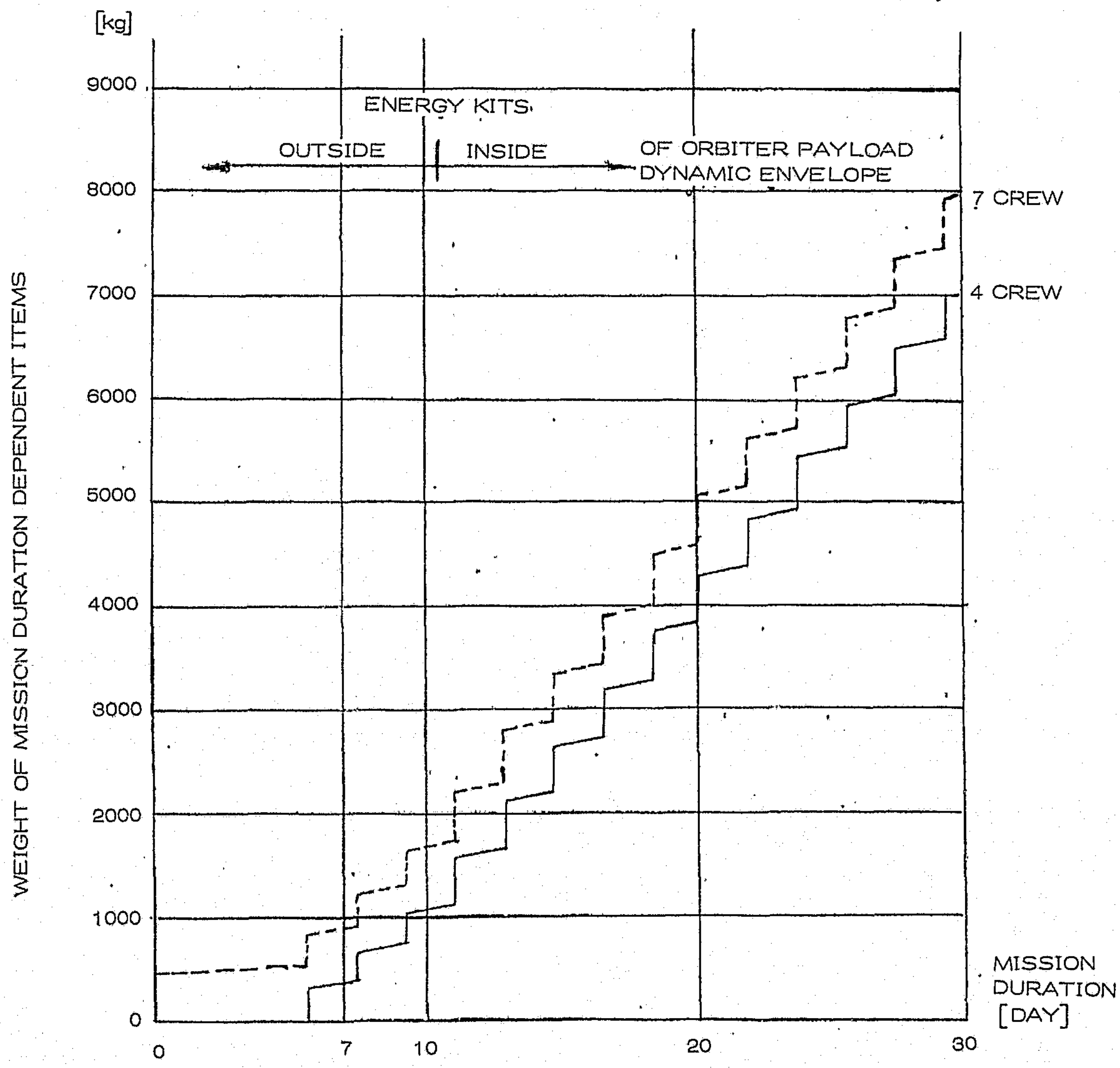
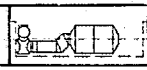
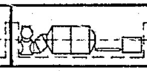
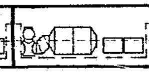
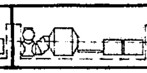
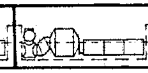
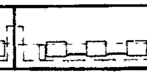
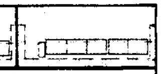


Figure 3-40: Weight Penalty of Mission Duration Dependent Items for Module Modes

3.9 Payload Accommodation Summary Tables

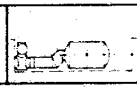
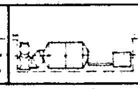
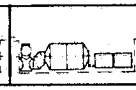
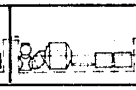
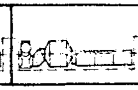
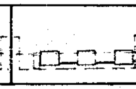
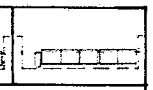
The following tables show a summary of the main payload accommodation features of Spacelab. The information is presented concisely to provide a quick reference source for data which is elaborated in more detail in other sections of the handbook.

Table 3 – 10 Major Spacelab Configuration Dimensional Aspects

Parameter	Configuration	Dim.							
Payload Accommodation – Dimensional Aspects									
Spacelab configuration	Location of most forward Orbiter attachment point utilized for reference configuration –Module (Xo) –Pallet (Xo)	in	923.47 –	789.73 1179.13	789.73 1092.6	769.73 1061.13	789.73 1021.80	– 777.93	– 797.6
Spacelab c.g. location	Location of c.g. of basic Spacelab without mission depend. subs. eqm. and without payload X_L Y_L Z_L	m	19.137 .002 –.455	17.647 .021 –.503	18.144 .018 –.531	17.512 .024 –.545	17.769 .006 –.545	19.232 .089 –.813	19.215 .059 –.850
Module inside dimensions	Total length available in X direction for payload – at sidewalls – at center aisle – at ceiling – under floor – inside endcones	m	4.28 5.38 4.76 2.69 1.47	4.28 5.38 4.76 2.69 1.47	4.28 5.38 4.76 2.69 1.47	1.61 2.69 2.38 – 1.47	1.61 2.69 2.38 – 1.47	–	–
Module inside volume	Total volume available for payload – at sidewalls – at center aisle – at ceiling – under floor – inside endcones	m^3	14.1 3.9 1.6 2.6 TBD	14.1 3.9 1.6 2.6 TBD	14.1 3.9 1.6 2.6 TBD	5.3 1.5 0.8 – TBD	5.3 1.5 0.8 – TBD	–	–
Pallet ☆ dimensions	Total length available for payloading – no overhang – with overhang	m	–	2.8752 7.0	5.8724 7.0	5.8724 9.8	8.8696 9.8	8.6256 17.0	14.854 17.0
mounting area	Total mounting area for payload projected on X-Y plane – no overhang – with overhang	m^2	–	11.5 27.7	23.0 27.7	23.0 38.7	34.5 38.7	34.5 67.2	57.5 67.2
	Total mounting area for payload on inner skin panels		–	17.1	34.2	34.2	51.3	51.3	85.5
volume	Total volume available for payloads – no overhang – with overhang	m^3	–	32.4 73.5	66.1 73.5	66.1 102.2	99.6 102.2	97.1 182.8	157.4 182.8
Orbiter aft flight deck	Total volume available for payload	m^3	TBD	TBD	TBD	TBD	TBD	TBD	TBD

☆ Overhang figures assume full use of available dynamic envelope inside Orbiter cargo bay taking into account a clearance of 1.219 m (4 ft) for EVA egress.

Table 3 - 11 Mass And Load Carrying Capabilities

Parameter \ Configuraration		Dim.							
Payload Accommodation - Mass and Load Capability Resources									
Spacelab mass	Mission independent Spacelab control mass	kg	4715	5430	6025	4995	5589	2910	3933
	Mission dependent sub-system equipment control mass		2094	TBD	TBD	TBD	1806	736	818
	Transfer tunnel mass		499	346	346	346	346	-	-
	Mission independent Orbiter support equipment mass		918	1102	1102	1102	1102	883	599
Nominal payload mass (1)		kg	5500	TBD	TBD	TBD	5500	9100	8000
Overall payload load carrying capability of the module (2)	at sidewalls	kg/m	580/side	→	→	580/side	→	N/A	N/A
	at ceiling		200			200			
	under floor		185			0			
	at center aisle		300			300			
	at forward endcone	kg	TBD	→	→	TBD	→		
	at aft endcone		TBD			TBD			
	Total module capability		5500			2200			
Overall payload load carrying capability of the pallet (3)	Total payload load carrying capability								
	- per pallet configuration			3110	5000	5000	5000	9100	10,000
	- per pallet train	kg	N/A	3110	5000	5000	5000	3110	5,000
	- per train with igloo			-	-	-	-	2880	5,000
Total payload carrying capability		kg	5500	8600	10,500	7200	7200	9100	10,000

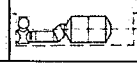
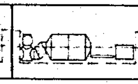
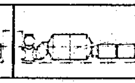
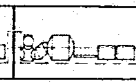
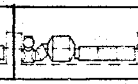
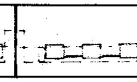
Editorial note: Further refinement of the detailed pallet and module structure will result in small variations of the local load carrying capabilities but the resultant final overall load carrying capabilities for the module and pallet will not be less than the values quoted in the table. Additional load carrying capability of TBD kg is available in the Orbiter AFD.

(1) applicable with 50 % of mission dependent subsystem equipment control mass

(2) combination of individual load carrying capabilities must never exceed total module capability

(3) refers to uniformly distributed loads

Table 3 - 12 Electrical Power And Energy Resources

Parameter \ Configuration		Dim.							
Payload Accommodation - Electrical Power and Energy Resources									
Electrical Energy	Energy available for payload and mission dependent equipment during flight from basic (890 kWh) Orbiter supply	kWh	420	TBD	TBD	TBD	369	588	588
Electrical power for payload at EPDS interface without operation of mission dependent subsystem equipment (max. values)	Pre launch phase max. cont. power peak power *)	kW kW	TBD	TBD	TBD	TBD	TBD	TBD	TBD
	Ascent phase max. cont. power peak power *)	kW kW							
Electrical power for payload at EPDS interface without operation of mission dependent subsystem equipment (max. values)	On Orbiter phase max. cont. power peak power *)	kW kW	3.70 7.7	3.25 TBD	3.24 TBD	3.40 TBD	3.40 7.4	5.00 9.5	5.00 9.5
	Descent phase max. cont. power peak power *)	kW kW	TBD	TBD	TBD	TBD	TBD	TBD	TBD
	Post landing phase average power peak power *)	kW kW							

*) 15 minutes duration for 3 hours intervals

1) Tentative values not subject to formal control

Editorial Note: Data given may be subject to change.

Table 3 – 13 Heat Rejection Capabilities And Module Atmosphere Aspects

Parameter / Configuration	Dim.	Module	Pallet
Module Atmosphere			IGLOO Atmosphere
nominal total pressure	bar	1.013 ± .013	max. GN_2 diff.press. 1.096 bar
partial oxygen pressure nominal	bar	.220 ± .017	min. GN_2 diff.press. .035 bar
partial CO_2 pressure nominal	bar	.0067	
cabin air temperature	°C	18 — 27	internal temperature 35°C max.
min. humidity (dew point)	°C	6	
max. rel. humidity	%	70	
max. allowable internal wall temp.	°C	45	
air velocity in habitable area	m/s	.1 – .2	
noise level	NC	50	
Heat Transport Capability Available to Payload and Subsystems			
during operational phase			
via cabin air loop	kW	1	–
via avionics cooling loop	kW	4.5	–
via experiment HX	kW	4	–
via pallet freon loop (6)	kW	–	8.5
total (5)	kW	8.5	8.5
prelaunch/postlanding, Orbiter GSE connected			
Orbiter powered down		same as operational phase	same as operational phase
Orbiter powered up	kW	1.5	1.5
ascent / descent	kW	1.5	1.5
Peak Heat rejection capability for Payload power peaks			
. preoperational phases	kW	TBD	TBD
minimum interval betw. peaks	minutes		
. during operational phase	kW	12.4	12.4
. minimum interval betw. peaks	minutes	165	165
. during postoperational phase	kW	TBD	TBD
. minimum interval betw. peaks	minutes		

(1) Prelaunch, ascent, on-orbit with cargo bay doors closed

(2) on-orbit with cargo bay doors open

(3) based upon 8.5 kW heat rejection of the Orbiter system

(4) on-orbit with cargo bay doors closed, reentry to landing.

(5) above numbers cannot be summed for total capability.

(6) includes igloo

Table 3 - 14 Command And Data Handling Resources

Payload Data Acquisition	
Housekeeping and Low Rate Scientific Data (to computer via RAU's)	
number of remote acquisition units (RAU's) of basic system	8
max number of RAU's (extension capability)	22
number of flexible inputs (analog or digital) per RAU	128
analog: resolution of analog/digital conversion	8 bit
discrete: number of inputs addressable as group	16
number of serial PCM inputs per RAU	4
clock rate	1 Mb/s
max number of words transferred per sample	32
word lengths	17 bit
max basic sampling rate	100 Hz
data rate of tarnsfer RAU / computer (incl. overhead)	1 Mb/s
Wide Band Scientific Data	
number of experiment channels of the High Rate Multiplexer (HRM)	16
min data rate of HRM input channels	64 kb/s
max data rate of HRM input channels	16 Mb/s
number of CCTV video input channels	1
number of 4.2 MHz analog channels	1
Data Transmission to Ground	
nominal data rate for housekeeping and low rate scientific data from subsystem and experiment computer	64 kb/s
max data rate for wide band scientific data (via TDRSS)	50 Mb/s
max data rate of High Rate Digital Recorder (HRDR) bridging TDRSS non-coverage periods	32 Mb/s
storage capability of HRDR	3.5×10^{10} bit
Payload Command Capability	
telecommand rate from ground via Orbiter	2 kb/s
number of on/off command outputs per RAU	64
number of serial PCM command channels per RAU	4
clock rate	1 Mb/s
max number of words per command	32
word length including parity bit	17 bit
Payload Data Processing and Displays	
payload data processing :	
word length	16 bit
speed (Gibson Mix)	350 kop/s
floating point arithmatic	32 (24+8) bit
mass memory	131 Mbit
payload data display: alphanumerical display screen	12 inch diagonal Tri-color

4. SPACELAB SUBSYSTEM CAPABILITIES/EXPERIMENT INTERFACES

4.1 Structure

4.1.1 Module Structure

The characteristic dimensions of the module structure are shown in Figure 4.1-1 and 4.1-2.

The module structure has mechanical provisions to accommodate subsystems, experiments and experiment related equipment by the following structural means:

- Floor
- Racks
- Stowage Containers

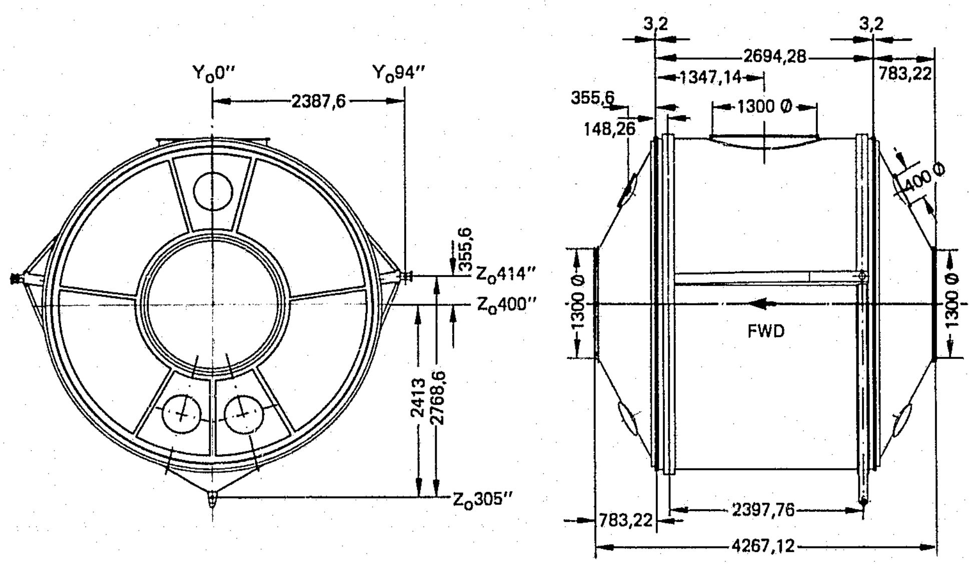


Figure 4.1-1 a: Short Module, Physical Dimensions

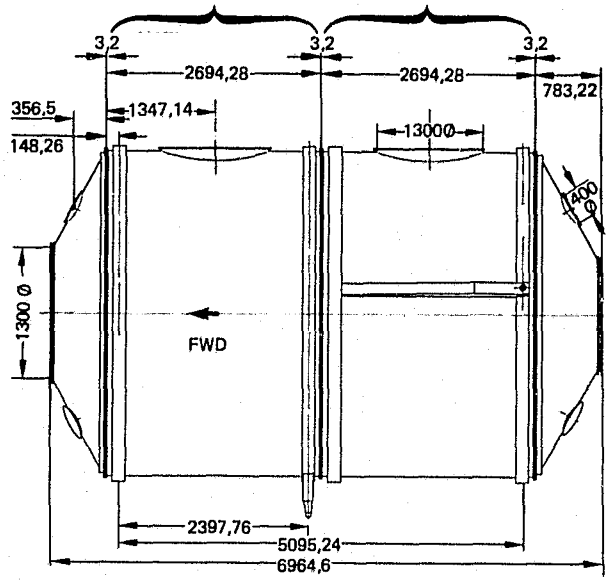


Figure 4.1-1 b: Long Module Physical Dimensions

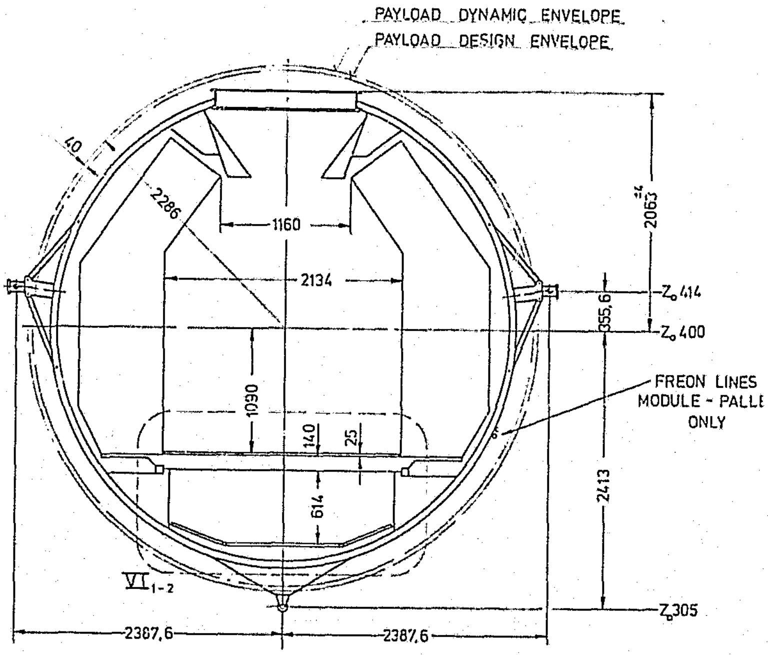
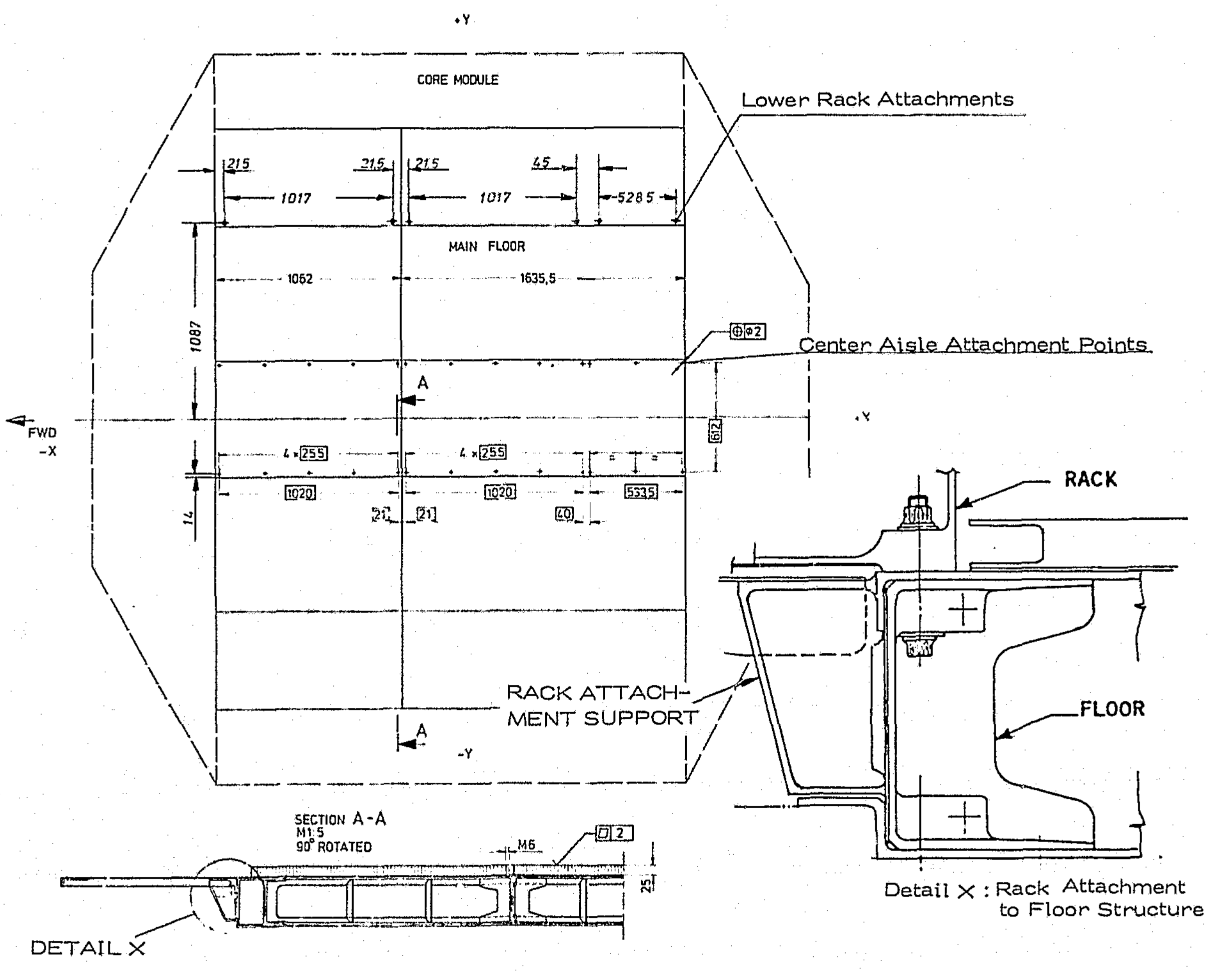


Figure 4.1-2: Characteristic Dimensions

4.1.1.1 Basic Structure-Floor

The floor in each module segment (core segment and experiment segment) consists of two individual sections, 1062 and 1635.5 mm long, respectively. Each section consists of a frame work of beams (Figure 4.1-3) providing support and mounting provisions for standard experiment racks and/or experiment equipment. Floor sections not occupied by racks or experiment equipment are covered with removable honeycomb sandwich panels. If required, the two floor sections can be bolted together to provide a single continuous floor during integration and flight. The floor is capable of carrying 300 kg/m length over its 0.61 m wide central portion; attachment points are provided, as shown in Figure 4.1-3.

In the case of center aisle mounted equipment, however, the C.G. must be no higher than 0.40 m above the floor level. The lower attachments for the racks (2 per rack) are located in the outer portion of the floor and shown in Figure 4.1-3.



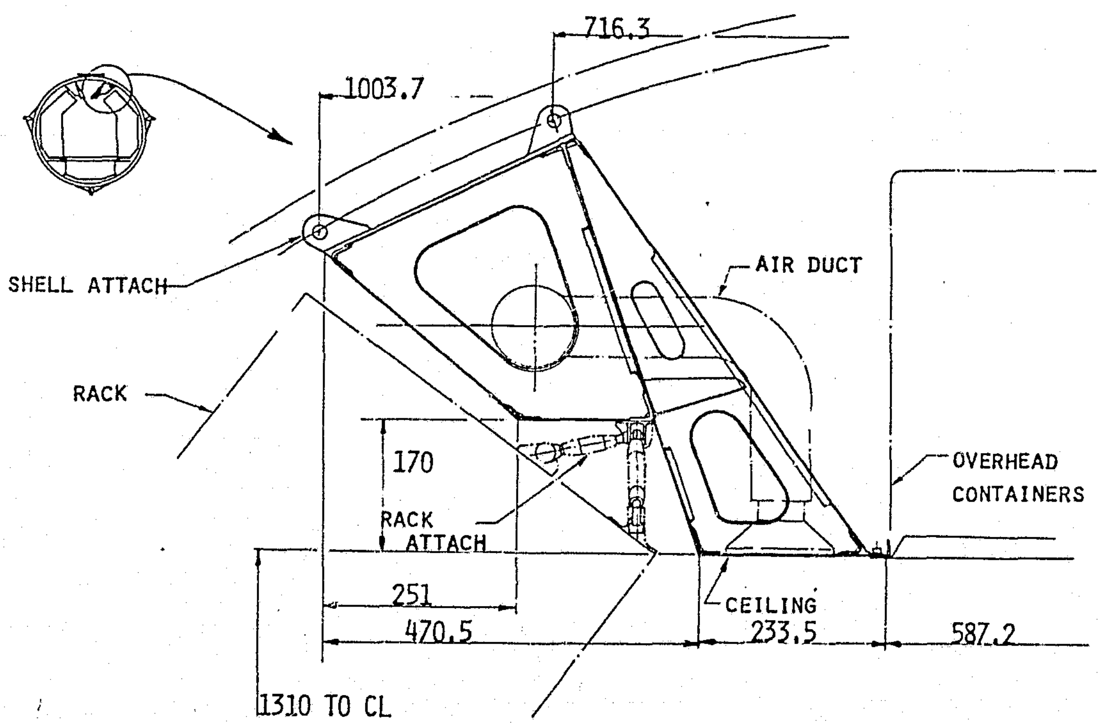
4.1-3: Floor Structure Interfaces

4.1.1.2 Overhead Structure

In the overhead structure lighting, air ducts and horizontal hand rails are installed. If top airlock installation is required , lighting fixtures and mounting brackets in the respective section, as shown in Figure 4.1-5, will be removed. The same is necessary if the optical window assembly or the high vacuum vent facility shall be installed.

The overhead structure, as shown in Figure 4.1-4, provides channels for the accommodation of stowage containers. The dimensions of the stowage containers are 350 x 580 x 600 mm; 8 containers are provided.

The upper attachment points for the racks are also located in the overhead structure as shown in Figure 4.1-4 thru 4.1-6. Further details of the racks attachment fittings are TBD.



4.1-4: Overhead Structure Interfaces

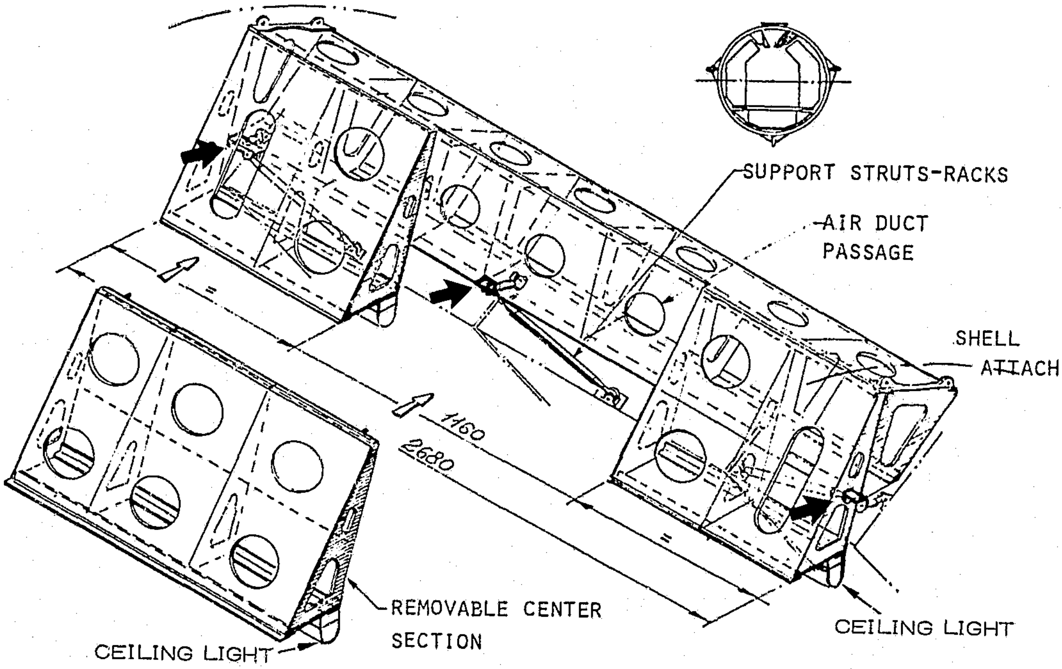


Figure 4.1-5: Rack Attachment Points at the Overhead Structure

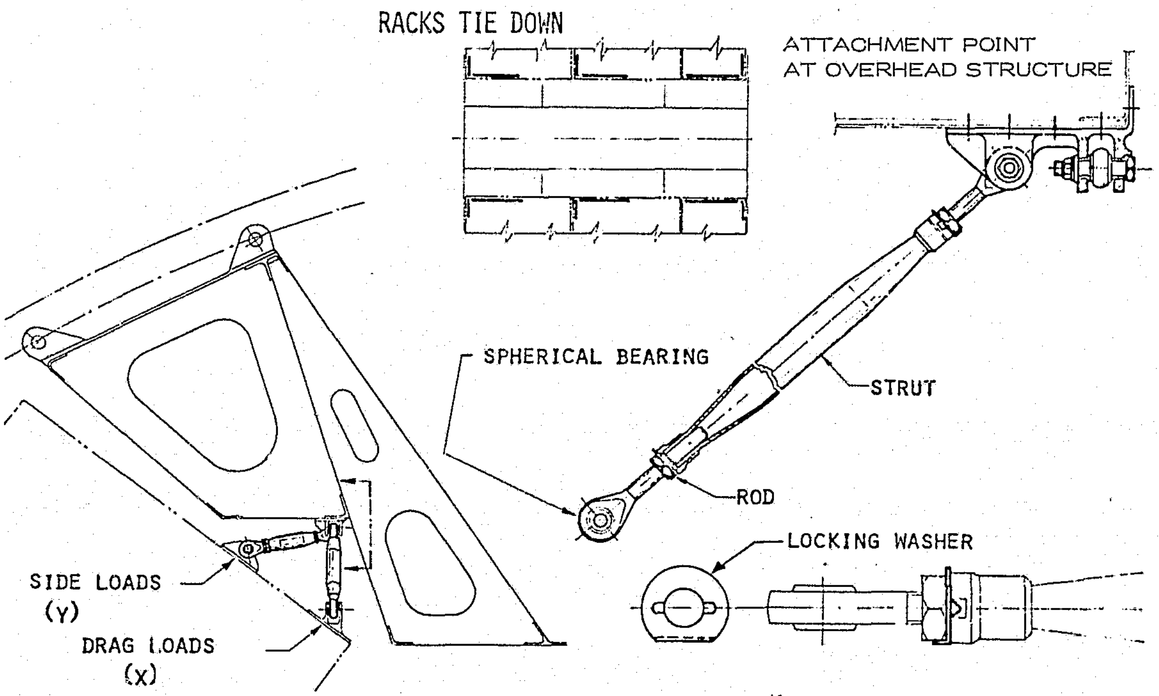


Figure 4.1-6: Rack Attachment - Details

4.1.2 Mission Dependent Structure – Racks

4.1.2.1 Standard Experiment Racks Description

Experiment racks are mission dependent Spacelab subsystem equipment and can be removed, if required. In this case, experiment equipment has to be mounted at the same attachment points of the racks on the floor and the overhead structure, as indicated in Figures 4.1-4 thru 4.1-6.

Location and arrangement of the racks inside the module are as indicated in Figure 3 - 15, 3 - 17 and 3 - 18. Two types of racks are available:

- single racks with an overall width of 563.48 mm

and

- double racks with an overall width of 1052 mm.

Both types of racks are 760 mm deep at their greatest depth and extend from the floor to the overhead structure (Figure 4.1-7).

The experiment racks are available for experiments, mission dependent equipment, experiment switching panels, RAU's and the intercom stations, as shown in Figure 3 - 18. It should be noted, however, that single and double racks are not interchangeable.

The double rack is designed to accommodate two side-by-side mounted 19 inch standard electronic equipment, while the single rack can accommodate a single row of standard 19 inch equipment. The hole pattern for attachment is in accordance with MIL-STD-189.

The width of a single rack, or half of a double rack, is also suitable for the accommodation of 1 1/2 size ATR equipment, according to ARINC 404 A, by means of user provided shelving.

A 255 mm high removable access panel is provided at the foot of each rack, where all wiring interface connections are routed through this area. Details are shown in Figure 4.1-10.

The rack is provided as a structural item with removable back panels, open at the front, a closed panel on top and a bottom panel with cooling system cutouts. A removable frame is also provided which, when installed, divides the double rack into two sections. This frame is of open truss type to facilitate direct cable routing between the left and right part of a double rack.

For experiments requiring the full width of the double rack (940 mm), the truss frame may be removed in the lower part of the rack.

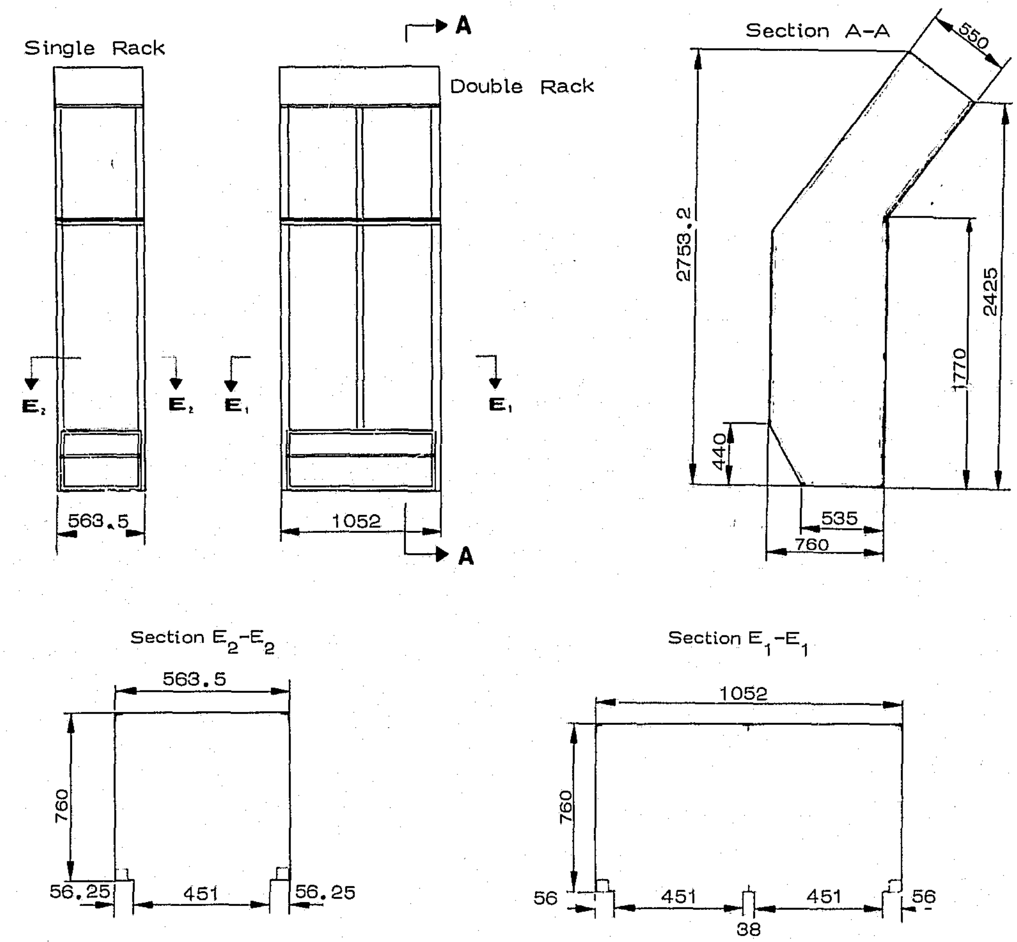


Figure 4.1-7: Standard Racks

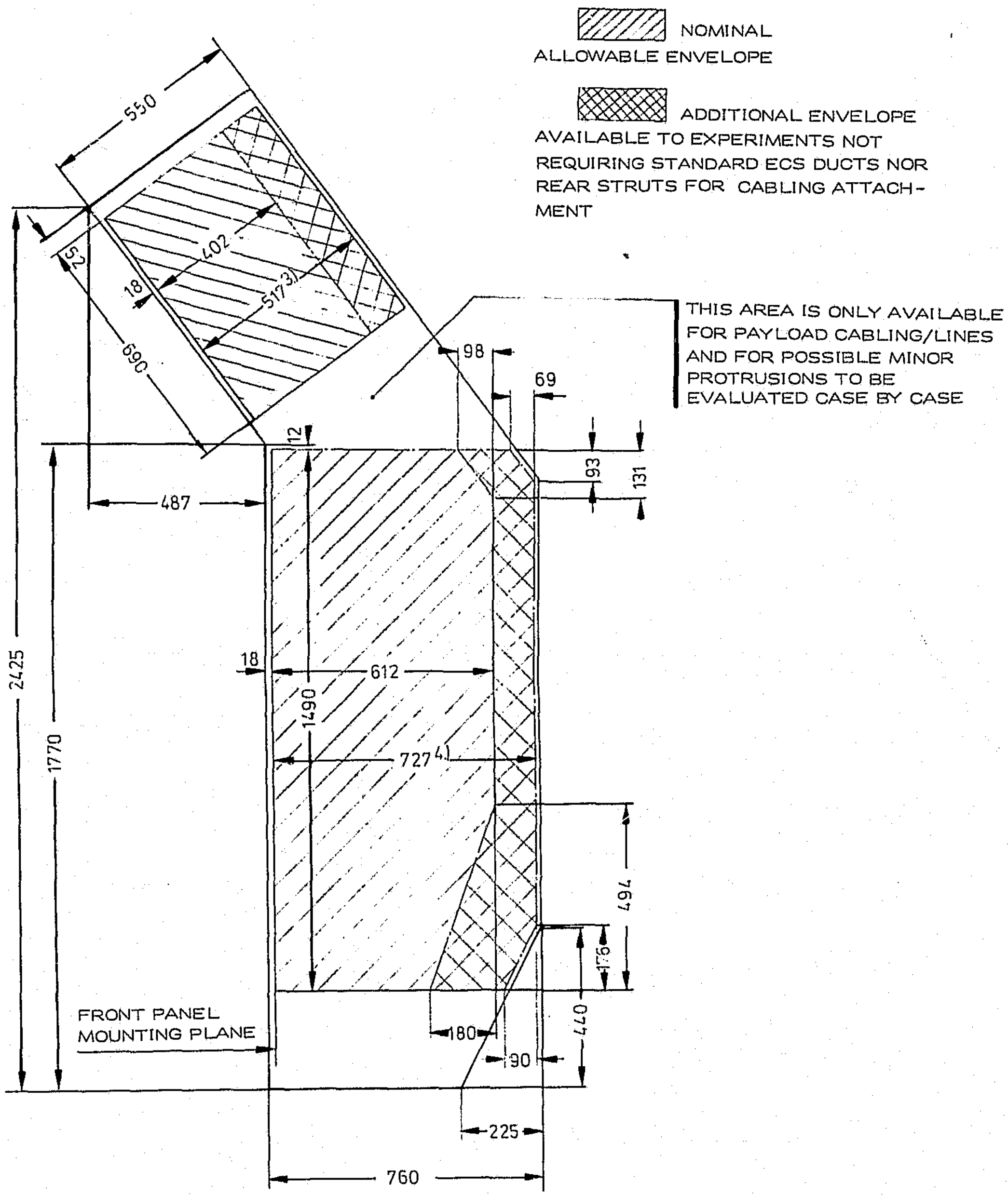
4.1.2.2 Standard Racks - Experiment Allowable Envelope

The nominal allowable envelope for experiment and mission dependent equipment mounted on the lower and upper (oblique) parts of the racks is shown in Figure 4.1-8. Minor protrusions of experiment equipment beyond the nominal allowable depths may be possible, if compatible with the experiment rear cabling and the spacing of the cabling support struts.

Experiments which require neither standard ECS air cooling ducts nor rear struts for cabling attachments, may utilize the entire internal depth allowed by the basic rack structure (as per Figure 4.1-8).

The height available for mounting of experiment (and Spacelab mission dependent) equipment is also shown in Figure 4.1-8.

Projection of experiment equipment in front of the panel mounting plane is allowed within limitations given by Crew Habitability constraints in para 7.2.5.



WITHOUT MIDDLE FRONTPOST:

3.) REDUCED UP TO 477

4.) REDUCED UP TO 687

WIDTH OF PAYLOAD ENVELOPE:

1. STANDARD SINGLE RACK: 451
2. STANDARD DOUBLE RACK: 2 x 451
   (WITHOUT MIDDLE FRAME: 940)

Figure 4.1-8: Experiment Allowable Envelope in Racks

4.1.2.3 Standard Racks Carrying Capability

The following maximum experiment masses, including Spacelab mission dependent equipment and experiment cabling, may be accommodated in the racks.

- Single rack 290 kg
- Double rack (overall) 580 kg
- Either side of a double rack (left or right) 290 kg

In addition the following limitations also apply:

- Max. mass of a single experiment box = TBD (120 kg tentative)
- Max. mass/height of the experiment box (single rack or half of a double rack) = TBD (200 kg/m tentative)

Mass and c.g. limitations for payloads supported only via the front panels are TBD. Other limitations for double racks with the center frame removed are TBD.
Payloads exceeding the quoted carrying capability are not necessarily excluded, but must be subject to a case by case analysis.

4.1.2.4 Payload Mounting Interfaces within Racks

Mounting of payload equipment may be done by means of front panels (according to MIL-STD-189) and via user provided internal shelving, runners, rails.

Rows of attachment holes for such internal structure are provided as shown in Figure 4.1-9. The same spacing between rows of holes (454 mm) holds also for single racks. Details of the hole spacing are TBD. Horizontal bracing platforms across the rack between experiment equipment may be necessary at one to three locations per rack, if the front panels of the experiment equipment are not suitable for the transfer of shear loads and if no additional shear panels can be installed. Each of these bracing platforms requires TBD (tentatively 40 mm) space between experiment equipment. In addition, all payloads that are not designed to accept secondary stresses due to the general deformation of the rack in operation, may require strain isolation fixtures.

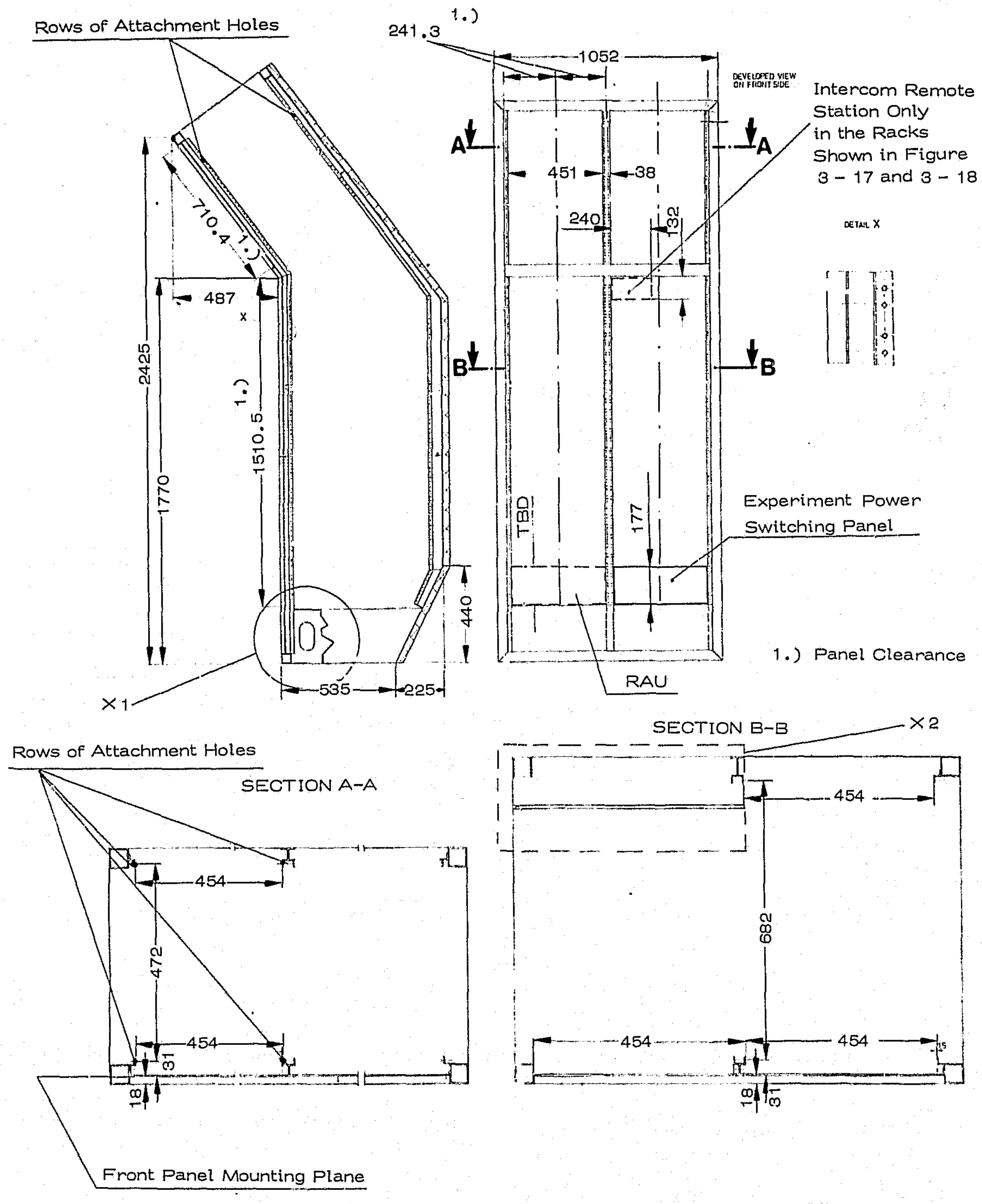


Figure 4.1-9: Double Rack

4.1.2.5 Payload Interfaces to ECS, EPDS, CDMS within Racks

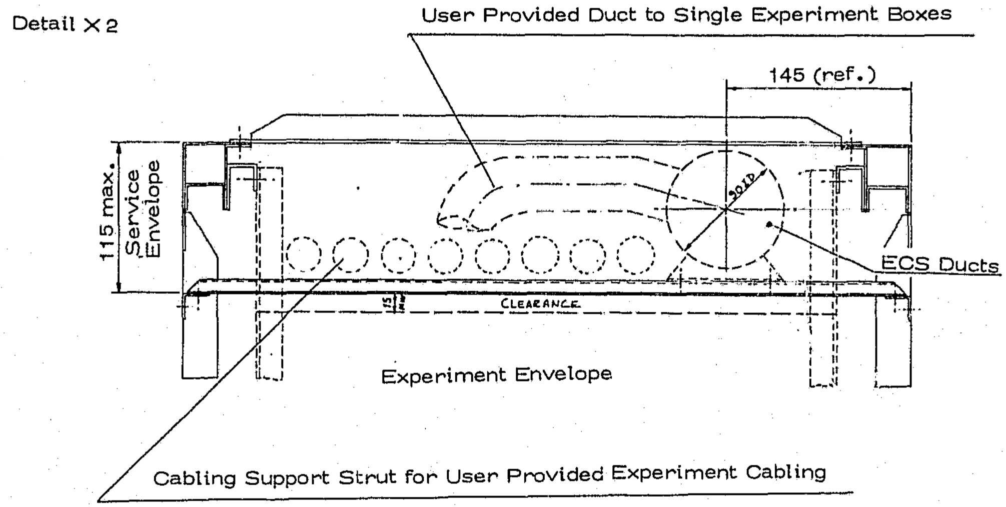
Standard ECS air suction ducts in the rear part of the racks provide orifice interfaces for the connection with single experiment equipment (Figure 4.1 – 10). The capabilities of the rack air cooling system are described in para 4.3. Mounting of payload in the racks shall permit complete closure of the front rack aperture. Payload front panels, although not leak tight, shall therefore be appropriately sealed to permit a satisfactory avionic cooling operation. (The equivalent orifice area allowable per unit front panel area is TBD.)

The interface with EPDS is located at the experiment power switching panel, one per rack is shown in Figure 4.1–9. Power outputs available at the power switching panel are detailed in para.4.2.3.2.

A standard set of horizontal struts is provided at the rear of the racks for mounting of experiment cabling, as shown in Figure 4.1 – 10. The number and vertical location of those struts may be varied within TBD limitations, according to payloads needs.

In the lower part of the rack, the connectors are located. For experiment cabling between racks, respective cables must be routed through these connectors and the cut-outs at the bottom of the rack, as shown in Figure 4.1 – 10.

The interface with CDMS is provided by means of one remote acquisition unit (RAU) per rack, as described in para 4.4.



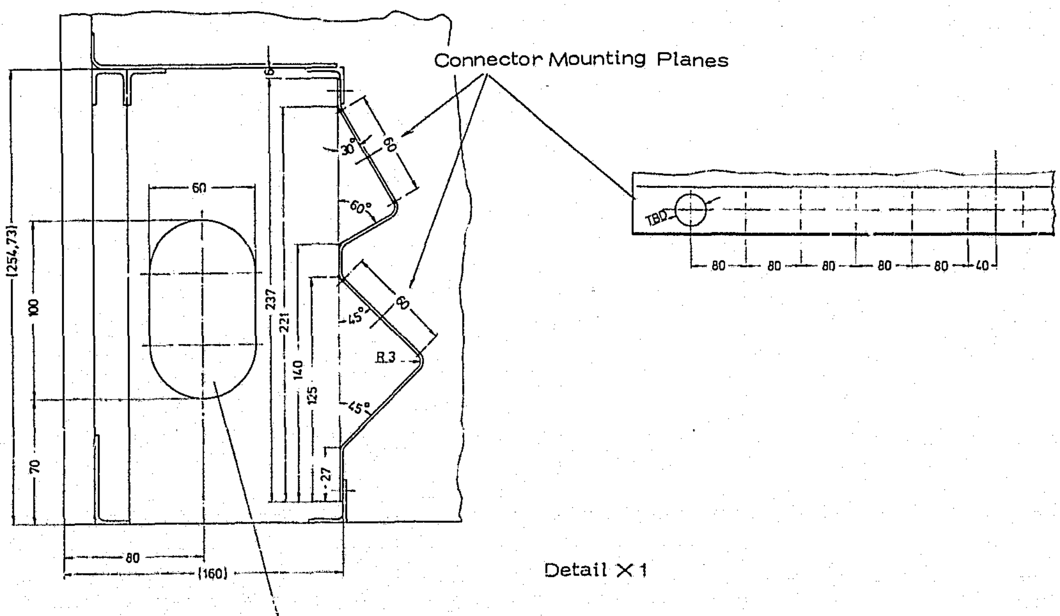


Figure 4.1-10: Detail X of Figure 4.1.-9

4.1.3 Pallet Structure

The pallet structure accommodates experiment equipment for direct exposure to space. The general structural configuration of each pallet segment is shown in Figure 4.1-11 and 4.1-12. Pallet segments can also be mounted as double and triple pallet assemblies as shown in Figure 4.1-13.

The pallet provides mounting support for the experiments either directly on the inner honeycomb skin panels, or - as mission dependent equipment - through specific hardpoints for better dispersion of concentrated loads. Within the dynamic envelope indicated in Figure 4.1-12, experiments, e.g. telescopes, may overhang at both ends of the pallet, the only geometric limitations being given by the Orbiter Cargo Bay dynamic envelope (Figure 2-4), the Igloo (Figure 3-6/3-9).

The surface of the pallet is painted white with Z 202, which has a high emissivity and low absorption to limit solar heating of exposed areas. Surface conductance is provided through aluminium panels.

The outer surface of the pallet is also coated with Z 202 white paint to avoid thermal coupling of the pallet segment and the Orbiter payload bay.

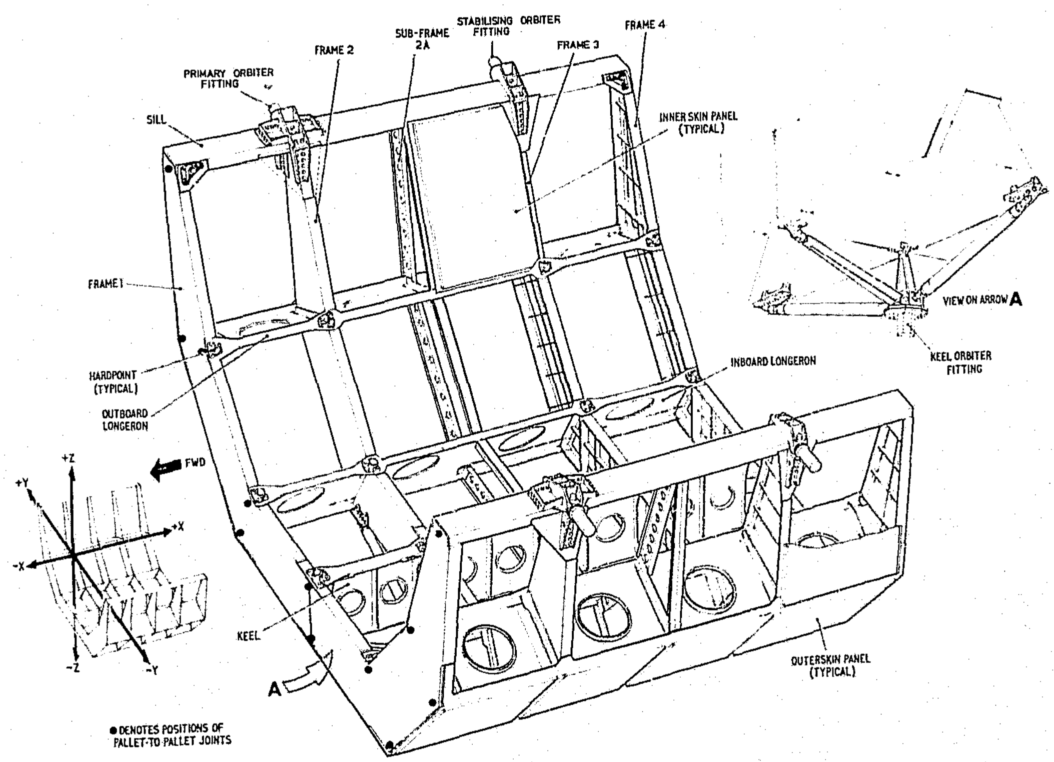


Figure 4.1-11: Pallet Structure

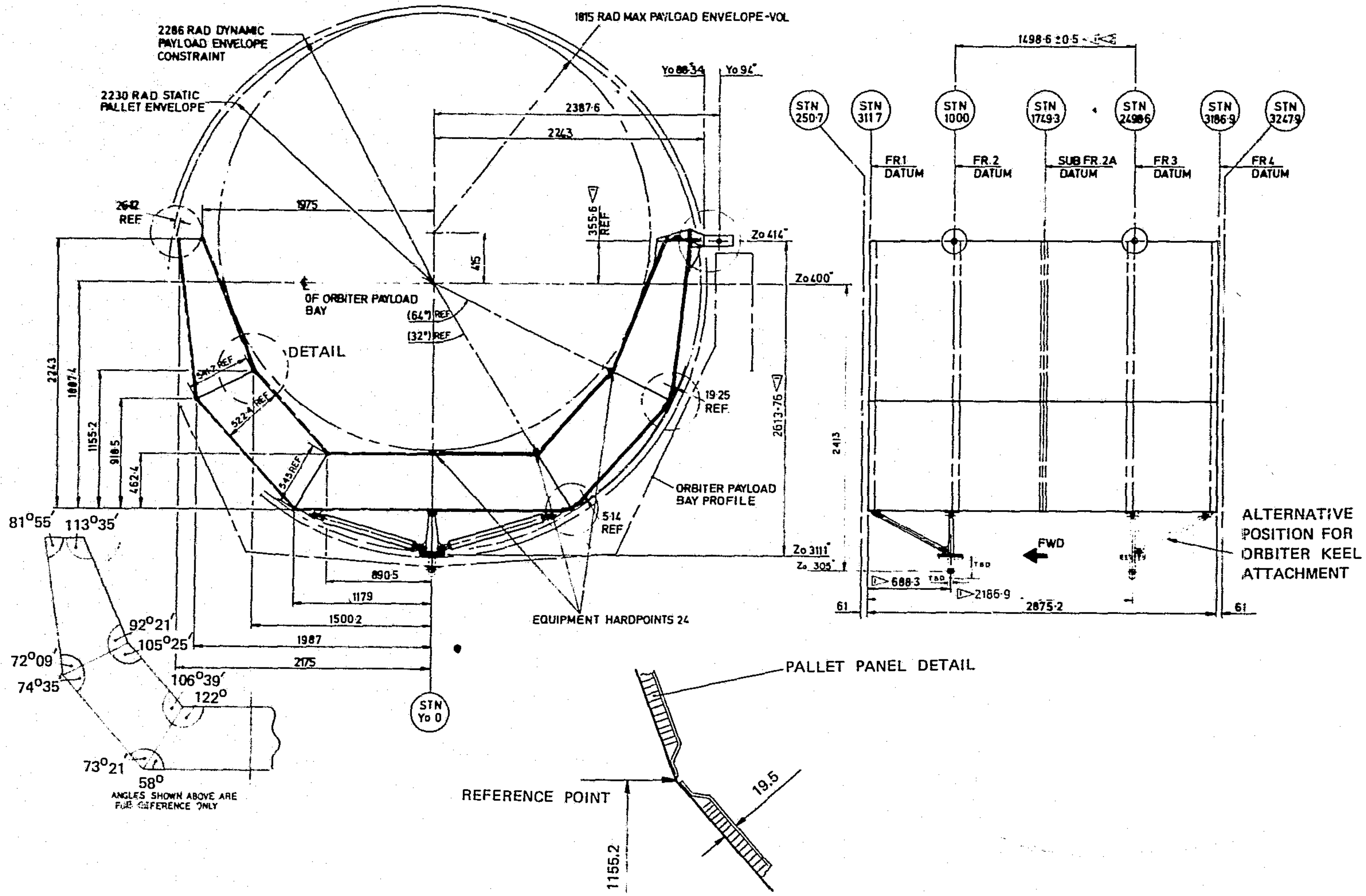


Figure 4.1-12 Pallet Structure Geometry

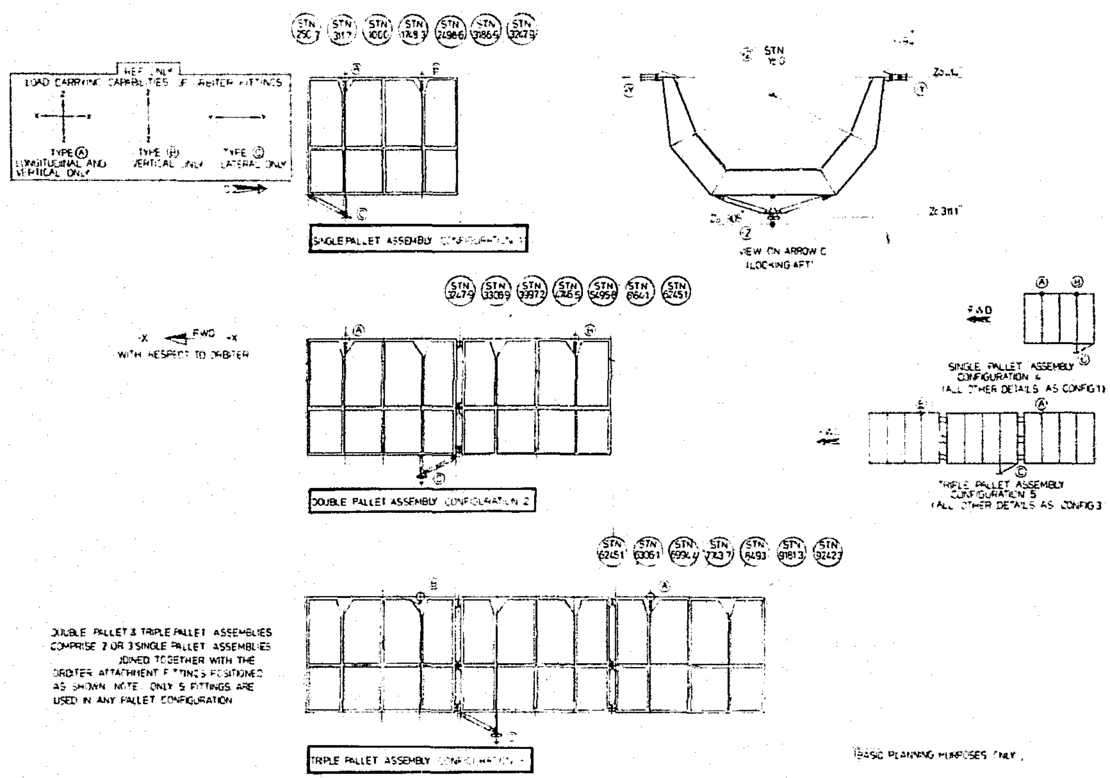


Figure 4.1-13: Pallet Configurations

4.1.3.1 Basic Structure

Inner Panels and Outer Skin

Experiments can be mounted on both the inner panels and the outer skin panels. Figure 4.1 - 15 indicates the geometry of the inner panels. Actual equipment mounting provisions in the honeycomb sandwich panels, e.g. inserts, are TBD.

The inner panels can support masses of 50 kg/m^2, whereas the outer skin panels can support 10 kg/m^2. All panels are removable for pre-integration purpose.

No mounting provisions for experiments are foreseen at the pallet fore and aft face areas, as e.g. for the subsystem Igloo

4.1.3.2 Mission Dependent Structure

Hardpoints

For equipment masses exceeding the floor panel carrying capability, 24 standard equipment hardpoints are foreseen on each pallet segment, located at the inter-section of the frames and longitudinal members of the inner surface (see Figures 4.1 - 14 and 4.1 - 15). The actual hardpoints are inserted as shown in Figure 4.1 - 14; each hardpoint providing a spherical nut with 12 mm diameter metric thread, bolted to the pallet structure.

The load carrying capability of the hardpoints is limited by the structure at the respective intersections.

Table 4.1 - 1 gives the load capabilities for each of the hardpoints on the pallet structure with the exception of the hardpoints on the pallet sill. The capabilities of each hardpoint are shown in the X, Y and Z directions both for flight and crash cases. Flight case loads are limit loads, whilst crash case loads are ultimate. On the figures, individual values are given for each hardpoint. In fact, each of the hardpoints along the length of the pallet at any radial station - e.g. nodes 1, 2, 3, 4 - has been designed to carry the worst loads on any of the hardpoints along that line, i.e. the maximum load at one specific radial station can be taken as applicable to all of the hardpoints along the respective line of the pallet.

These load limits apply irrespective of whether the pallet is flown in a single, double or triple configuration.

The load limits quoted at the hardpoints at the kinks of the pallet are the same for both sides of the pallet, i.e. port side and starboard.

Any experiment crash load, which falls within the flight case value, is acceptable even if it does not fall with the crash value. The reverse is not true.

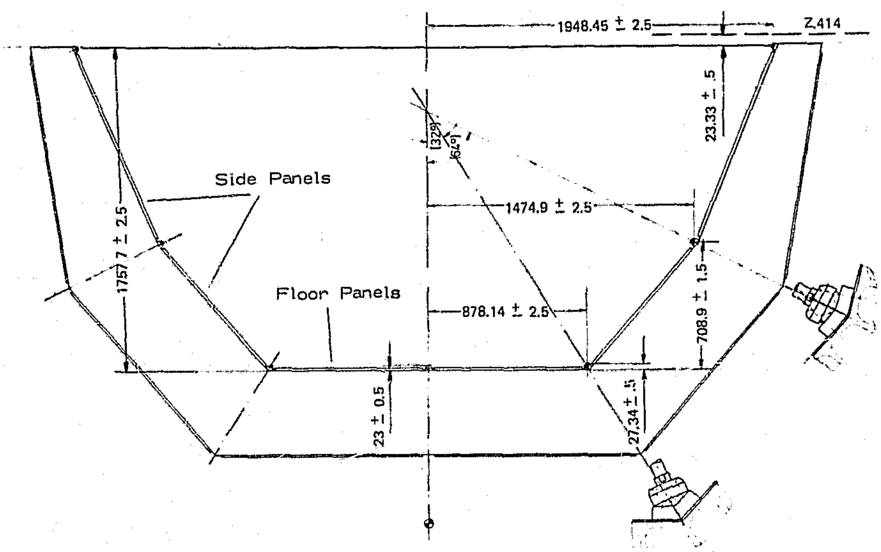


Figure 4.1 - 14: Cross Section through Pallet

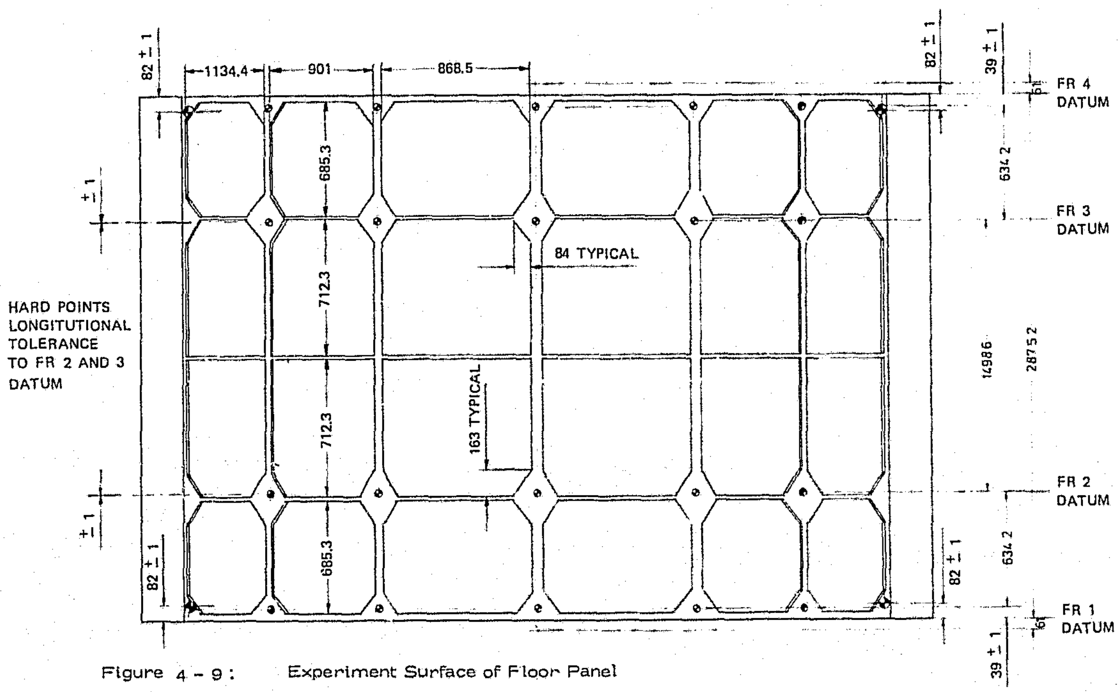


Figure 4 - 9 : Experiment Surface of Floor Panel

Figure 4.1 - 15: Experiment Surface of Floor Panel

Editorial note: The load limits given here are provisional and will be updated, as the pallet design progresses. They do, however, give a good indication of the capabilities of the hardpoints. It should be mentioned that the compliance of a payload with these hardpoint load limits does not, on its own, mean that payload falls within the design limits of the pallet, as described in the subsequent paragraph.

4.1.3.3 Physical Accommodation Capability

The physical accommodation capability of a single pallet segment is as follows:

- The overall payload carrying of a single pallet segment is about 1000 kg/m (uniformly distributed over the pallet). C.G. limitations are given by the envelope shown in Figure 2-1 thru 3-13.

- The overall payload carrying capability of a 2 and 3 pallet train is listed in Table 4.1-2 with the same c.g. limitations applied. Payloads exceeding, even considerably, the overall mass of Table 4.1-2 may still be accommodated, depending on the specific mass distribution and the specific hardpoints pattern utilization. Individual analyses are required for such payloads on a case-to-case basis.

  In addition, the pallet design provides the possibility to increase the payload mass carrying capacity up to 2000 kg/m by use of additional structural elements of TBD mass and volume. These elements, however, are not part of the deliverable items.

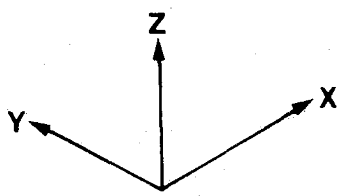
- A single pallet provides 33 m^3 volume above the floor.

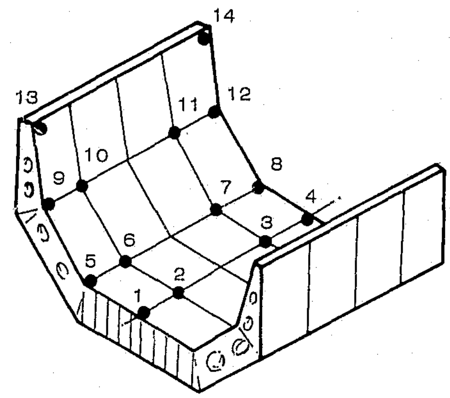
- The floor panel of a single pallet segment provides about 17.0 m^2 of mounting area, which is available for mounting payload equipment.

- The pallet structure has provisions for the insertion of hardpoints.

Up to five pallet segments can be combined as a pallet-only configuration, with a maximum of three pallets being rigidly interconnected as pallet train.

In addition to hardpoints, Figure 4.1-16 depicts examples of mounting heavy payload equipment on the pallet. The various platforms and support structure are shown for information only and are not provided by Spacelab.





Hardpoint Location

Table 4.1-1: Pallet Hardpoint Load Limitations ($N \times 10^{-3}$)

Nr.	Flight Cases (Limit Loads)			Crash Cases (Ultimate Loads)		
	X	Y	Z	X	Y	Z
1	+ 18.4 − 8.2	± 2.5	+ 19.0 − 14.7	− 37.4	± 8.8	− 37.3
2	± 14.0	± 5.3	+ 34.0 − 67.1	− 77.4	± 17.6	+ 144.4 − 40.8
3	± 58.0	± 6.7	+ 31.5 − 63.0	−133.0	± 22.2	+ 133.0 − 37.3
4	± 14.0	± 3.4	+ 34.0 − 63.0	− 62.5	+ 11.2	+ 144.4
5	+ 14.3 − 7.0	+ 7.5 − 5.3	+ 40.0 − 24.8	− 33.1	+ 11.2 − 16.8	− 88.0
6	+ 29.0 − 14.6	+ 12.3 − 10.5	+ 50.0 − 26.5	− 62.1	+ 22.4 − 25.3	+ 33.2 − 100.0
7	+ 29.0 − 14.6	+ 11.2 − 12.3	+ 40.0 − 25.0	− 62.1	+ 25.4 − 16.8	+ 35.3 − 87.5
8	+ 8.4 − 12.5	+ 6.0 − 7.2	+ 8.1 − 16.4	− 33.1	+ 16.8 − 4.5	− 33.2
9	+ 7.0 − 4.0	+ 3.9 − 2.2	+ 9.8 − 6.6	− 15.6	± 6.7	+ 22.9 − 22.2
10	+ 17.7 − 12.4	+ 5.6 − 4.2	+ 13.8 − 11.1	− 37.3	+ 6.7 − 7.9	+ 22.9 − 27.5
11	+ 17.7 − 12.4	+ 5.6 − 10.0	+ 10.0 − 13.6	− 37.3	+ 8.1 − 6.7	+ 30.8 − 22.2
12	+ 7.0 − 4.0	+ 3.3 − 2.9	+ 5.2 − 9.6	− 15.6	+ 6.7	+ 22.9
13	TBD	TBD	TBD	TBD	TBD	TBD
14	↓	↓	↓	↓	↓	↓

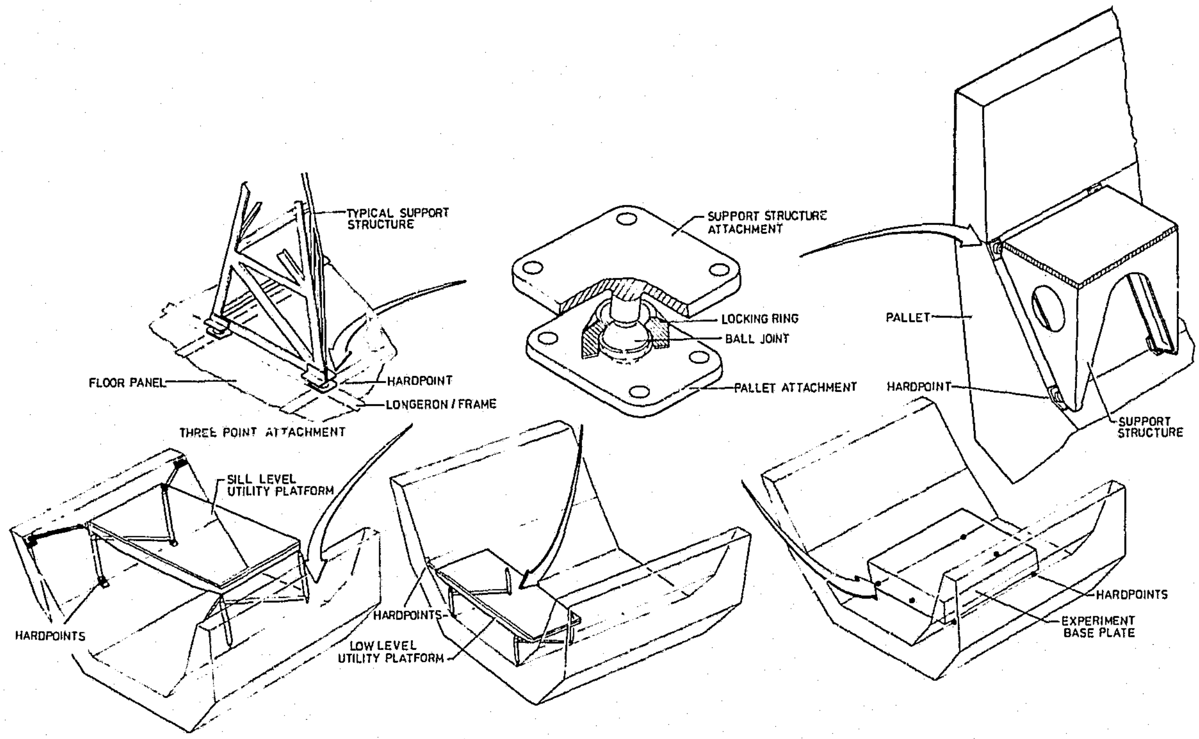
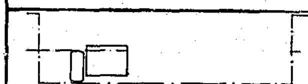
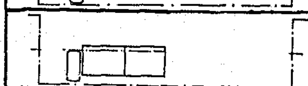
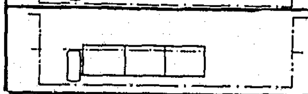


Figure 4.1-16: Typical Heavy Experiment Equipment Installation

Table 4.1-2: Payload Carrying Capability of Pallet Trains

Train Configuration	Payload carrying capability (kg)	
	without igloo	with igloo
	3118	2880
	5000	–
	5000	5000

4.1.3.4 Pallet Deflection

Pallet deflections from quasi-static inertia loads, Orbiter torsion loads and temperature effects may result in deflections in the order of 30 mm for pallet trains. In principle, secondary stresses on both pallet and attached experiments can be avoided by the use of statically determinate systems for the support structure.

It is important that experiment structures extending over more than one independently suspended pallet or pallet train shall not act as a rigid bridge connection, in order to avoid unnecessary secondary stresses both on the experiment and the pallet.

4.2 Electrical Power and Distribution Subsystem

4.2.1 General Description

The Electrical Power and Distribution Subsystem (EPDS) receives its primary power from a dedicated power source in the Orbiter. The primary power delivered from the Orbiter during orbital operations is 7 kW maximum continuous and 12 kW peak (for a maximum duration of 15 min. once every 3.0 hours) at a nominal voltage of 28 VDC. However, the supply of peak power is related to the Orbiter heat rejection capabilities, which are attitude dependent. The details are still TBD.

In case of failure of the dedicated power source, the degraded power available is 5 kW continuous and 8 kW peak from a back-up source shared with the Orbiter.

In addition, 750 W are available in the Orbiter AFD for Spacelab subsystems and experiments located in the AFD.

The energy available to Spacelab subsystems and experiments is 890 kWh.

The EPDS equipment is summarized in Table 4.2-1.

Table 4.2-1: EPDS Equipment

Basic Spacelab	Mission Dependent
1. Power Control Box	1. Experiment 400 Hz inverter
2. Subsystem 400 Hz inverter	2. Experiment power switching panels
3. Emergency Box	
4. Standard harness for power distribution	
5. Subsystem power distribution box	
6. Experiment power distribution boxes	
7. Orbiter AFD power distribution box	
8. Normal and emergency lighting	

The principle arrangement of the EPDS with respect to experiments is shown in block diagrams, Figure 4.2-1 and 4.2-2. The support provided to experiments is the same in principle for the module and pallet only configurations. Only the experiment power switching panels located in the experiment racks within the module are not available for pallet use. The power bus system for experiment power running through all module and pallet segments provides the wiring for:

- Primary DC (28 V nominal)
- 115/200 VAC at 400 Hz, 3 Ø
- Essential power for experiments

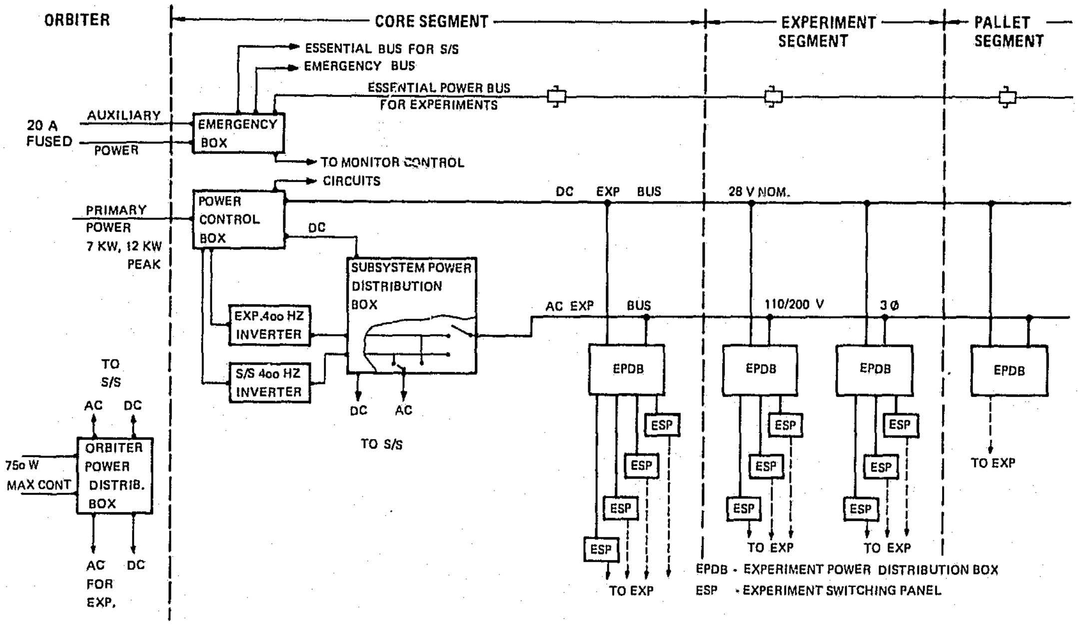


FIGURE 4.2-1 POWER DISTRIBUTION SCHEME, MODULE/PALLET CONFIGURATIONS

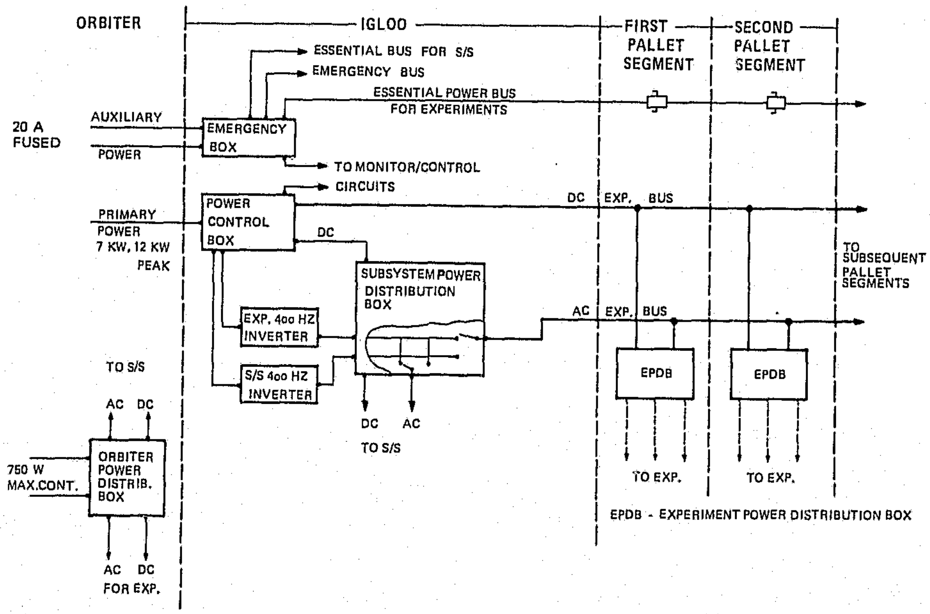


FIGURE 4.2-2 POWER DISTRIBUTION SCHEME, PALLET-ONLY MODE

The distribution of electrical power is generally separated between subsystems and experiments. In particular the subsystem inverter provides power to subsystem equipment including mission dependent equipment. The experiment inverter is totally dedicated to experiments.

However, as shown conceptually in Figure 4.2-1 and 4.2-2, it is possible to connect the AC experiment bus to the S/S inverters and, conversely, the S/S AC bus to the experiment inverter. This feature is intended to provide flexibility in case of inverter failure.

4.2.2 Power and Energy Available

Maximum continuous power is defined as power available with no time limitations except those due to the overall energy available.

4.2.2.1 Ground Operations

During the prelaunch and post landing phases power is provided to Spacelab either by Orbiter GSE or by the Orbiter power supply system itself. In the case of GSE support the power supplied to Spacelab is 1.0 kW maximum continuous and 1.5 kW peak (with the Orbiter subsystems powered up) and 7.0 kW maximum continuous and 12.0 kW peak (with the Orbiter subsystems powered down). For periods during which no GSE support is available (e.g. during transportation of the Space Shuttle to the launch pad) 1.0 kW max. cont. and 1.5 kW peak are available to Spacelab only at certain periods. The allocation of these amounts between Spacelab subsystems and payload is TBD.

4.2.2.2 Ascent and Descent Phase

The primary power available to Spacelab from the Orbiter during ascent and descent is 1.0 kW max. cont. and 1.5 kW peak. The peaks are limited to 2 min. total duration, per phase.

In the present operational concept Spacelab will be inactive during launch, ascent and descent (except for monitoring of emergency and warning and caution signals, and the activation of necessary safing commands. Both functions are hardwired into the Orbiter and controlled from the aft flight deck).

Editorial note: The provision of limited resources and services to experiments during ascent and descent is presently under investigation. The concept under consideration would result in a partially active Spacelab by providing power and cooling to experiments but no CDMS services. Power and cooling available to experiments would be that amount delivered by the Orbiter during these phases, minus the amount used by the partially active Spacelab.

4.2.2.3 On-Orbit Operational Phase

The primary power supplied to Spacelab on-orbit, with the Orbiter payload bay doors open and the Orbiter radiator kits activated, is 7.0 kW max. cont. and 12.0 kW peak (15 min. every 3 hours).

According to current Orbiter timelines it is expected to have full power and cooling available approximately one hour after lift-off, until about 8 hours before de-orbit.

The power available to experiments depends on the power consumption of the basic Spacelab subsystems and is also a function of the use of mission dependent equipment. As in the case of the weight budget, a maximum amount of power is available to the payload if no mission dependent equipment is used, and a minimum amount if a maximum arrangement of power consuming mission dependent equipment has been selected.

The power budget differs from the weight in one important aspect, however, in that actual power consumption during a mission is highly dependent on the operational usage of equipment during that mission. Within the maximum limits for power available to payloads, there will be a wide range of actual available power depending both on the selection of mission dependent equipment and the actual operational use of that equipment to meet specific payload requirements. To establish an accurate mission power budget an extensive timelining effort is required once basic experiment accommodation and functional requirements are fixed. In addition, the energy budget has to be considered. Table 4.2-2 summarizes the allocation of the Orbiter supplied power between Spacelab subsystem equipment and Spacelab payload, as explained below:

- Power Required by Basic Spacelab Subsystem Equipment

  The power required by the basic Spacelab S/S equipment is shown in column A of Table 4.2-2. Max. power of column A assumes all the basic Spacelab equipment switched on and operating at its maximum continuous power.

  The Minimum Power of column A includes power saving by selective use of module lighting, use of reduced power modes of certain subsystem equipment, etc., resulting in minimum power saving of 200 - 300 W. The average nominal power is the power assumed as a basis for calculating the energy consumption of the basic subsystem equipment, over a 7 day period.

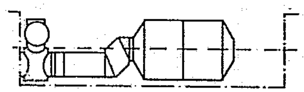
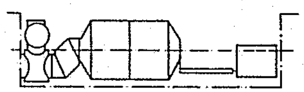
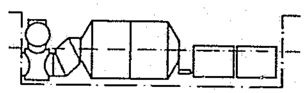
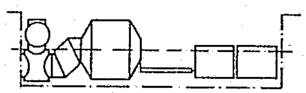
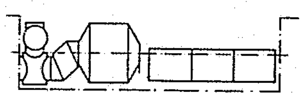
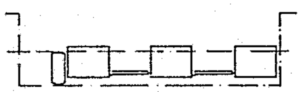
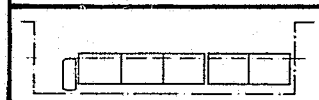
- Max. Continuous Power Available for Payload and Spacelab Mission Dependent Equipment

  Column B of Table 4.2-2 shows the maximum continuous power available for Payload and S/L mission dependent equipment. The minimum values shown assume maximum values of the power required by the basic Spacelab. The maximum values, still TBD, imply the use of power saving measures for Spacelab equipment mentioned above.

- Power Required by Mission Dependent Spacelab Equipment

  The power consumed by the mission dependent equipment will, as noted above, vary both from mission to mission and also at different times during each mission. It is not realistic to state here theoretical maximum and minimum values which might result from the worst case possible combination of mission dependent equipment since this may not be representative of any typical payload. For a particular mission a selection is made from the available mission dependent equipment listed in Table 3-5 and the resultant power consumption values can be estimated. In practice, different values will apply at different times during the mission dependent on the operational time-

Table 4.2-2: Spacelab Power and Energy Budget

	A Power required by basic Spacelab Equipment kW			B Power available for Payload and S/L Mission dependent Equipment – Max. continuous kW		C Peak Power available to Payload and S/L Mission dependent Equipment kW		D Energy available to Payload and S/L Mission dependent Equipment kWh
	Min.	Average Nominal	Max.	Min.	Max.	Min.	Max.	Nom.
	TBD	2.80	3.30	3.70	TBD	7.7		420
	TBD	TBD	3.75*	3.25*	TBD	TBD		TBD
	TBD	TBD	3.76*	3.24*	TBD	TBD		TBD
	TBD	TBD	3.60*	3.40*	TBD	TBD		TBD
	TBD	3.10	3.60	3.40	TBD	7.4		369
	TBD	1.80	2.00	5.00	TBD	9.5		588
	TBD	1.80	2.00	5.00	TBD	9.5		588

Note: * Tentative values not subject to formal control.

– Energy of column D includes energy used by Payload and S/L Mission Department in AFD.

– Electrical power available to payload in Orbiter AFD is available in addition to that of column B.

lines. The power consumption per unit listed in Table 3-5 are the maximum values for each unit. Some items also have reduced powers when operated in different modes e.g. tape recorders and airlock lighting.

The power dissipated inside the inverters will depend on the output power delivered and, as shown in Table 3-5, the fixed wattage stated must be added to the variable wattage, which is roughly proportional to the power output, in order to arrive at the total dissipation. The portion of power dissipated in the inverters due to power delivered to Spacelab mission dependent equipment is also chargeable to Spacelab mission dependent equipment power.

- Net Maximum Continuous Power Available to Payload

The net maximum continuous power available to Payload may be determined, for each particular flight phase, by subtracting the power required by Spacelab mission dependent equipment from the power of column B. It should be noted that the power loss in the experiment inverter due to AC power delivered to Payload is also payload chargeable, as is the DC power loss down stream of the experiment power distribution box.

In the limiting case where no mission dependent equipment is required by Payload, the available power is given directly in column B.

- Peak Power Available to Payload and Mission Dependent Equipment

Column C shows peak power available to payload and mission dependent equipment, and is formulated on the same principles as column B. The peak power availability is limited to a maximum duration of 15 min. to occur no more frequently than once in any 3.0 hour period. In addition peak power usage is limited by thermal control constraints of both the Orbiter and Spacelab.

- Energy Available for Payload and S/L Mission Dependent Equipment

The basic energy available from the Orbiter for all Spacelab missions will be 890 kWh.

The Energy available to Payload and Spacelab mission dependent equipment, for a nominal 7 days mission, is given in column D. Use of mission dependent equipment will reduce the Payload available energy according to the specific mission timelines.

This available energy (but not power) for each mission can be increased by the addition of extra Orbiter energy kits which provide an incremental increase of 840 kWh for each extra kit. The weight penalties associated with these extra kits are noted in para. 2.2.2.

Additional power will have to be experiment provided, along with the then necessary additional heat rejection provisions.

4.2.2.4 DC Power

The primary experiment DC bus runs through all module and pallet segments. It has the following characteristics:

Voltage:	28 VDC nominal
Voltage range:	28 VDC $\pm$ 4 VDC
Impedance:	TBD
Maximum power:	Harness layout for 11 kW

4.2.2.5 AC Power

The 400 Hz experiment power bus receives power from the mission dependent 400 Hz Experiment inverter, and has the following characteristics:

Voltage:	115/200 VAC $\pm$ 5 %
Outputs:	3 phases plus neutral
Frequency:	400 Hz $\pm$ 1 % Sine Wave
Harmonic distortion:	5 %
Maximum deviation between phases:	$\pm 2^{\circ}$
Power:	2.25 kVA max. cont. 3.375 kWA/2 s peak
Efficiency:	approx. 85 %
Power factor:	0.9 lead to 0.85 lag

The same characteristics apply to the Subsystem inverter. The 400 Hz inverters are designed to accept an external synchronization signal.

4.2.2.6 Experiment Essential Power

An independent redundant auxiliary DC power line from the Orbiter provides up to a maximum of 400 W emergency power to Spacelab and feeds the Spacelab emergency box. This 400 W is included in the 7 kW maximum continuous power delivered by the Orbiter, and is also available when only degraded power is delivered to Spacelab.

From the Spacelab emergency box one emergency bus, and two essential busses, one for subsystems and one for experiments, are routed through Spacelab.

The experiment essential power bus is powered up simultaneously with the emergency and S/S essential busses and fused for TBD Amps. The interface to experiments are one connector in the core segment and two connectors in the experiment segment of the module, and one connector on each segment. In the module these connectors are located on connector brackets under and accessible through the mainfloor. On the pallet the connector is located on a pallet connector bracket.

The essential power available to experiments is currently 25 W.

4.2.3 Power Activation and Distribution

Experiment equipment receives its power from the Experiment Power Distribution Boxes (EPDB´s) on the pallet and from the Experiment Power Switching Panels (ESP´s) in the module (or from EPDB´s in the module if the ESP´s are not used). Experiment equipment in the Orbiter AFD is supplied with power from the AFD power distribution box (which also supplies power to subsystem equipment).

4.2.3.1 EPDS Activation

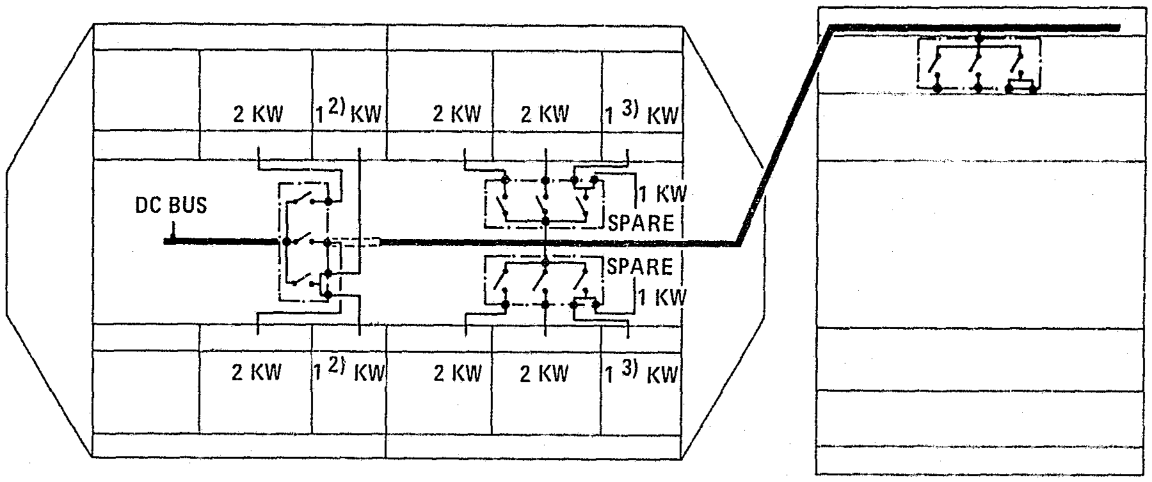
The EPDS is activated from the Spacelab Integrated AFD Control Panel (R 7 Panel) located in the Orbiter AFD. In addition Spacelab Remote Control also allows:

- the activation and control of EPDS functions via Orbiter MDM during ground and flight phases
- the activation and control, via Spacelab CDMS commands, of experiment EPDB´s.

Further details on Spacelab subsystem control are given in para. 4.4.5.

4.2.3.2 Experiment Power Distribution

Electrical power is distributed from the experiment power busses via Experiment Power Distribution Boxes (EPDB´s), one in the core segment, two in the experiment segment and one per pallet segment (Figure 4.2-3).



NOTES – 1) – A SIMILAR DISTRIBUTION SYSTEM APPLIES FOR THE AC POWER GIVING A MAXIMUM OF 2.25 KVA PER RACK

2) – UP TO 2 KW NOMINAL POWER IS AVAILABLE IF THE OPPOSITE SINGLE RACK IS NOT OPERATING

3) – UP TO 2 KW NOMINAL POWER IS AVAILABLE IF THE SPARE OUTPUT IS NOT USED

Figure 4.2-3: Experiment Power Distribution Box Location

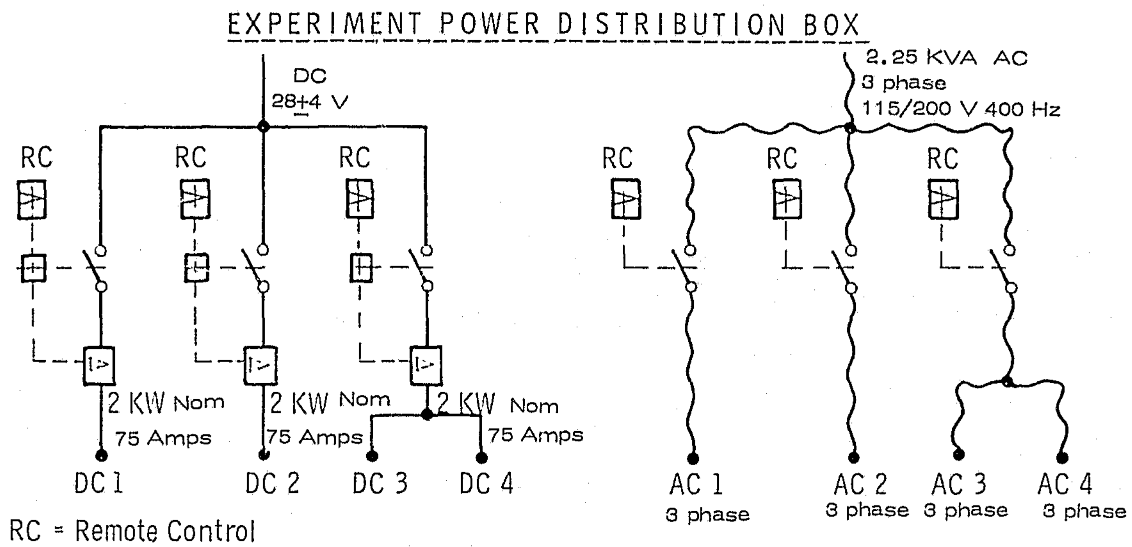
Within the module the busses are routed from the EPDB's to Experiment Power Switching Panels (ESP's) located in each rack (single or double rack). Each ESP provides connectors for rack internal access and one front outlet, as shown in Figure 4.2-4. Each output is protected against overload and switched manually from the front side of each panel.

Inside the module the experiments normally interface with the experiment power switching panels. Experiments not using the Spacelab provided experiment racks have to provide mounting provisions according to MIL-STD-189 for the ESP's. If the ESP's are not used the experimenter has to provide any necessary cabling, connectors etc. to interface with the EPDB outlets.

The location of the experiment power switching panel is shown in Figure 4.2-5 for a double rack.

On the pallet, experiments interface directly with the EPDB's (Figure 4.2-3 and 4.2-4), located on the pallet sill (Figure 3-32), one on each pallet segment.

Details on the power distribution boxes, experiment switching panels and the AFD power distribution boxes are given below:



- Via S/S RAU with one "on-off" command per circuit breaker EPDB.
- Via Orbiter MDM with one "on-off" command for all circuit breakers simultaneously.

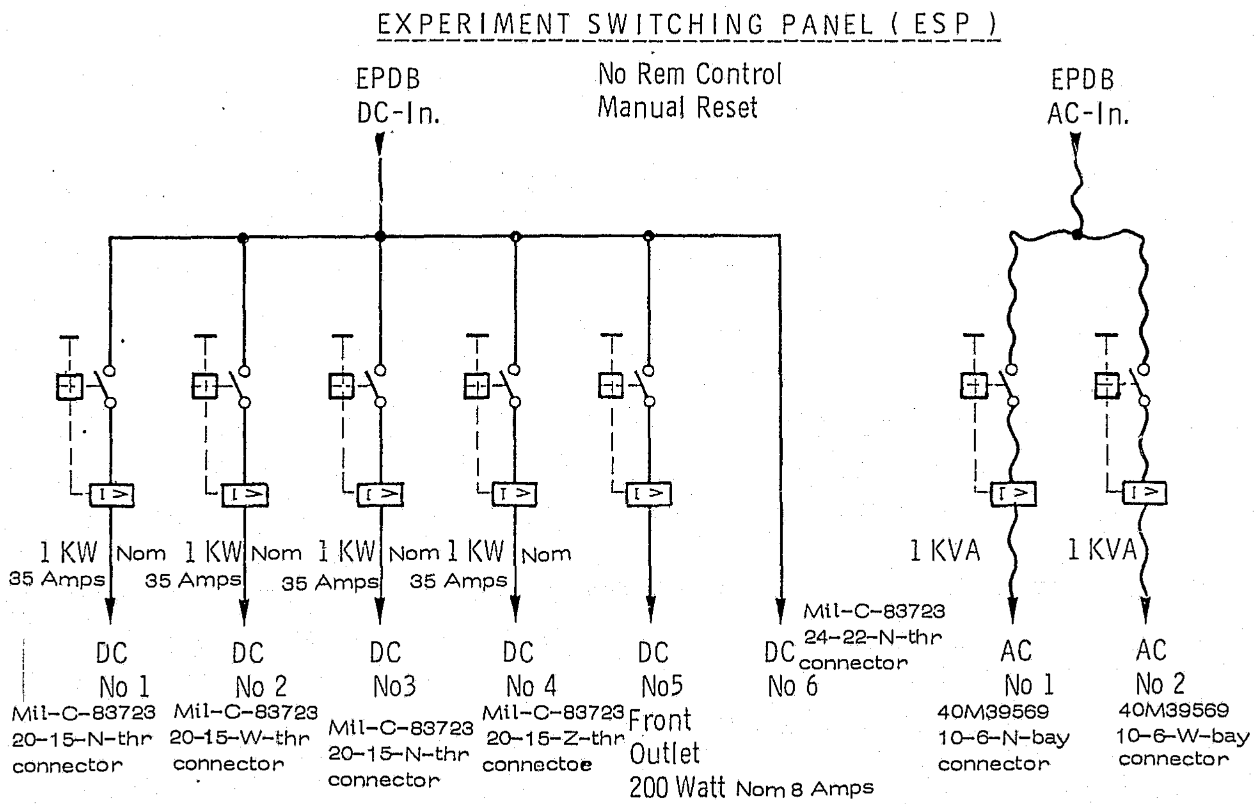
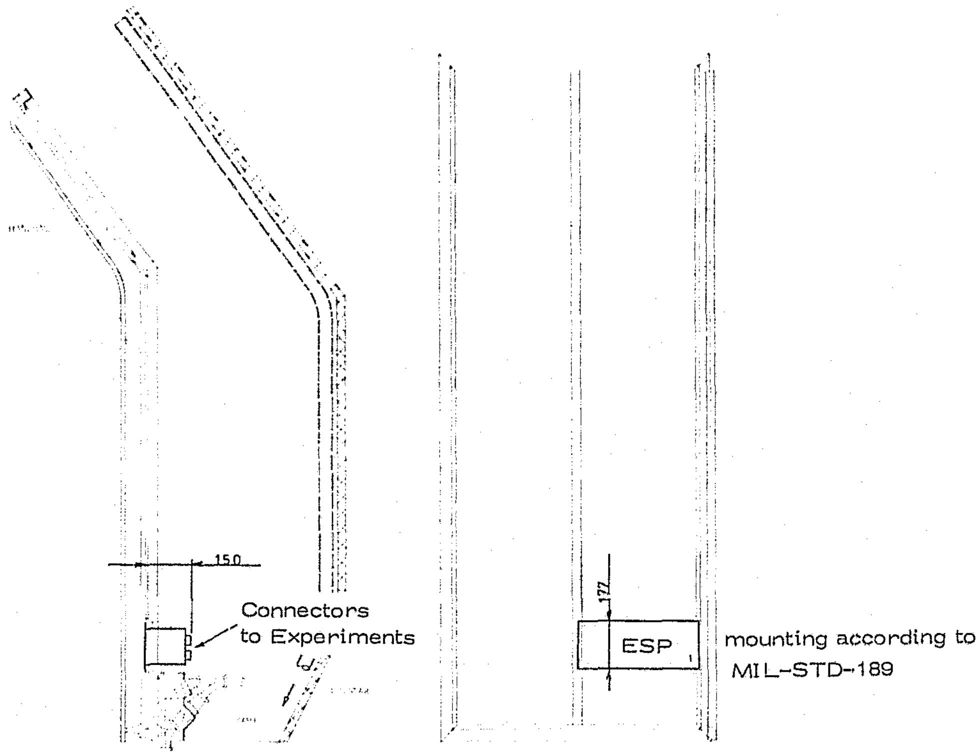


Figure 4.2-4: Experiment Power Distribution Box (EPDB) and Experiment Switching Panel (ESP)



- Similar location in a single rack
- For connector type see Fig. 4.2-4

Figure 4.2-5: Experiment Switching Panel (ESP) Location in a Double Rack

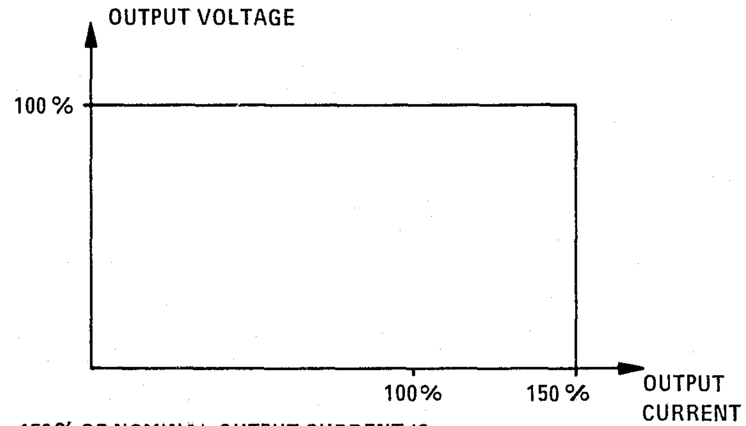
- Experiment Power Distribution Box

The Experiment Power Distribution Boxes supply the following outputs (see Figure 4.2-4):

- 4 DC outputs, 28 ± 4 V
  - 2 outputs, 75 A, 2 kW nominal per output
  - 2 outputs, 75 A, 2 kW nominal in total (max. 75 A per output)
- 4 AC outputs, 3 phase, 115/200 V, 400 Hz
  not fused, a current limiter in the 400 Hz inverter limits the total AC power available to 2.25 KVA (overload characteristics see Figure 4.2-6)
  - 2 outputs, 1 remote controlled switch per output
  - 2 outputs, 1 remote controlled switch for both outputs

The outputs of the experiment power distribution boxes may be remotely controlled:

- via S/S RAU with one on-off command per switch
- via Orbiter MDM with one on-off command per box driving all the switches within one box simultaneously.



150% OF NOMINAL OUTPUT CURRENT IS
SUSTAINED FOR A MINIMUM OF 30 SEC.

Figure 4.2 - 6 Overload Characteristics of the S/L 400 Hz Inverter

- Experiment Power Switching Panel

The Experiment Power Switching Panels provided with the experiment racks have the following outputs (see Figure 4.2-4):

for rack internal connection

- 5 DC outputs, 28 $\pm$ 4 V
  - 4 DC outputs, 35 A, 1 kW nominal per output selectively fused, manually controlled
  - 1 DC output, 75 A, 2 kW nominal, remote controlled via experiment power distribution box
- 2 AC outputs, 3 phase, 115/200 V, 400 Hz
  1 KVA per output, 3 A per phase selectively fused, manually controlled

for rack external access

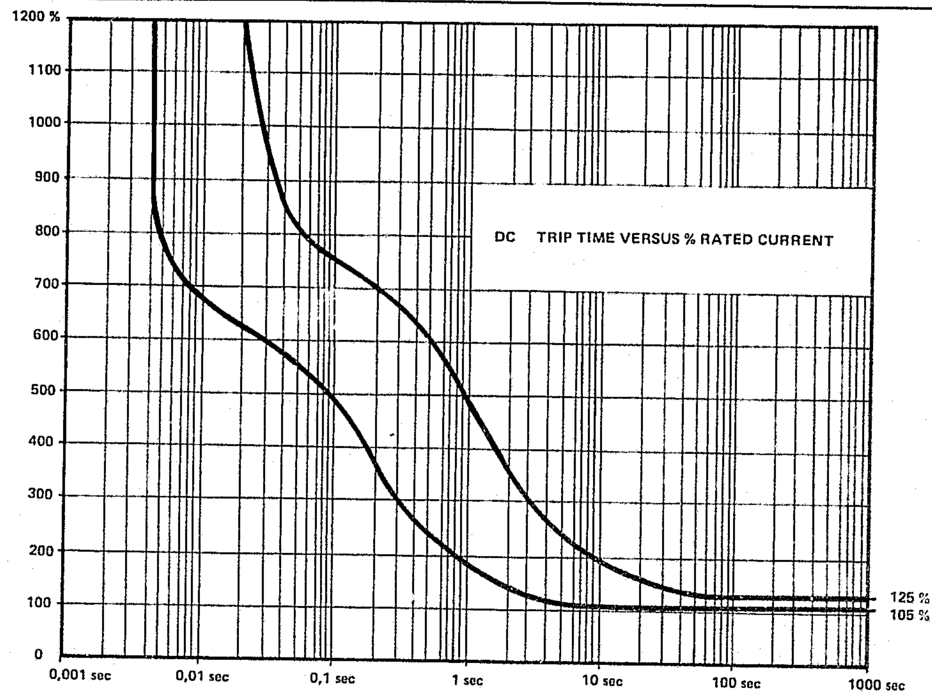
- 1 DC output, 28 $\pm$ 4 V
  8 A, 200 W nominal
  selectively fused, manually controlled

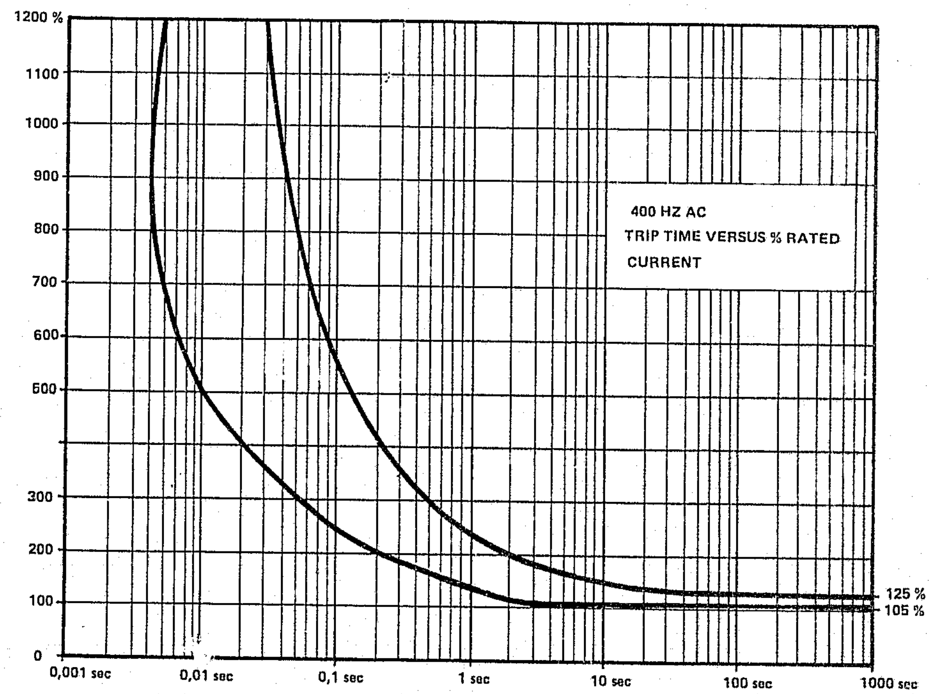
The tripping time characteristics are presented in Figure 4.2-7.

- Orbiter AFD Power Distribution Box

The AFD power distribution box supplies, in addition to outputs for subsystems, the following outputs for experiments:

- 2 DC outputs, 28 $\pm$ 4 V
  - 1 DC output, 250 W nominal
  - 1 DC output, 200 W nominal
- 1 AC output, 3-phase, 115/200 V, 400 Hz
  690 W in total, 230 W per phase.





All circuit breakers will take 100% of rated load.

They may trip between 105 % and 125 % and will always trip at 125 % of rated load.

Figure 4.2 – 7 DC and AC Trip Curves of the ESP

The Orbiter AFD power distribution box is fed directly by the Orbiter power distribution system. No switching is included in the power distribution box itself; therefore, the power output may be only switched on-off upstream of the box itself via the Orbiter switching system or downstream by switches included in the experiment hardware.

4.2.3.3 Power Integration Interface

For integration at different sites, the racks and pallets are provided with standard interfaces to ground support equipment. This allows assembly of experiment equipment into the racks and on the pallets and ground testing with simulated Spacelab power.

4.2.3.4 Grounding

Spacelab will be supplied with DC power from an Orbiter fuel cell via a two (2) conductor DC power bus which are connected to the Spacelab primary power system. A schematic diagram of the present grounding concept is shown in Figure 4.2-8.

The primary power system will be grounded to the Orbiter structure at a single point, defined as a single ground point or primary DC power single ground point (SGP) to be located near the fuel cell plant. Thus, when disconnected from the Orbiter DC supply, the power and the power return line of the Spacelab primary distribution system will be isolated from structure by at least 1 M (DC to 25 Hz).
The 400 Hz power return lines are grounded to the Spacelab structure at the 400 Hz inverter boxes.

In addition, Fig4.2-8 depicts present grounding concept of Spacelab subsystems. The case of each box is bonded to the structure. Power or signal ground is connected to chassis ground.

As a consequence for the signal flow between different units the use of:

- transformer coupling between signal input and output

or

- floating differential signal inputs

is necessary.

Experiment design requirements concerning grounding and isolation of experiments is given in Section 7.7 .

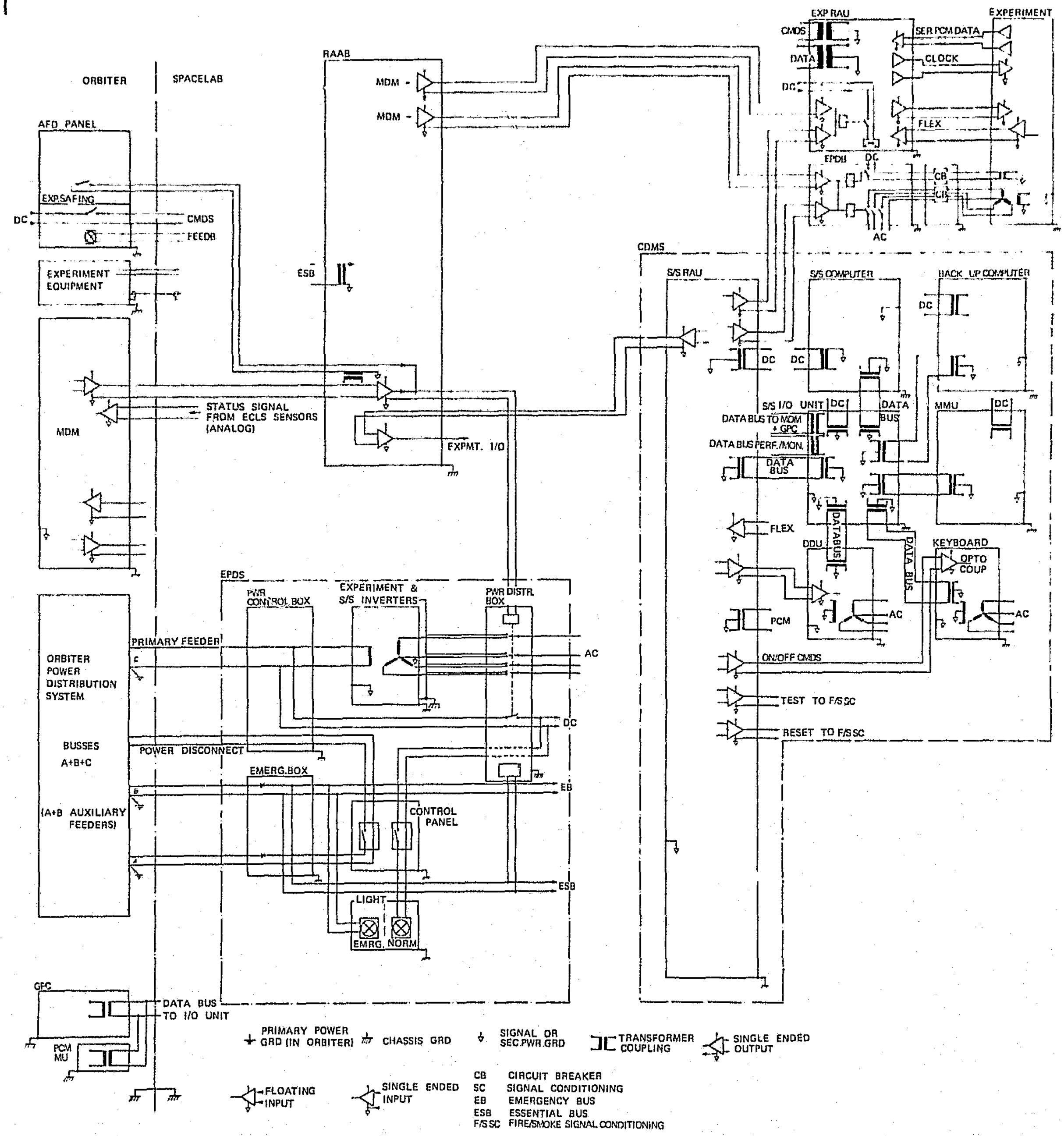


Figure 4.2 – 8 : Grounding Schematic Diagram

4.3 Environmental Control Subsystem

The environmental control subsystem (ECS) consists of the environmental control and life support subsystem (ECLS) and the thermal control subsystem (TCS). These subsystems comprise basic (mission independent), subsystem equipment which is configuration dependent, and mission dependent equipment which can be selected by the user according to his requirements.

The ECS provides the following services for Spacelab and its experiments:

- Pressurized environment inside the module
- Contamination control inside the module
- Module equipment cooling
- Pallet equipment cooling
- Passive thermal control
- Airlock repressurization
- Experiment venting

Table 4.3-1 lists the basic and the mission dependent ECS equipment and provisions.

Table 4.3-1: ECS Provisions and Equipment for Experiments

Basic Spacelab	Mission dependent
Atmosphere control inside the module	Cold plates (up to 8)
Contamination control inside the module	Thermal capacitors (up to 4)
Air cooling for experiment racks	Experiment dedicated heat exchanger inside the module
Airlock repressurization	Additional airlock repressurization (in excess of basic provisions)
Small experiment vent assembly	

The Spacelab ECS is designed to provide a shirt-sleeve 1 atmosphere environment for up to 4 crew-men, and provides cooling for equipment located in the pressurized module and on the pallet. In addition to the experiment support functions listed above, the ECS provides several emergency functions including portable oxygen supply, fire and smoke detection, fire extinguishing and protection for the module against overpressure and negative differential pressure. Figure 4.3-1 presents an overall ECS schematic, showing the major ECLS and TCS components and interfaces.

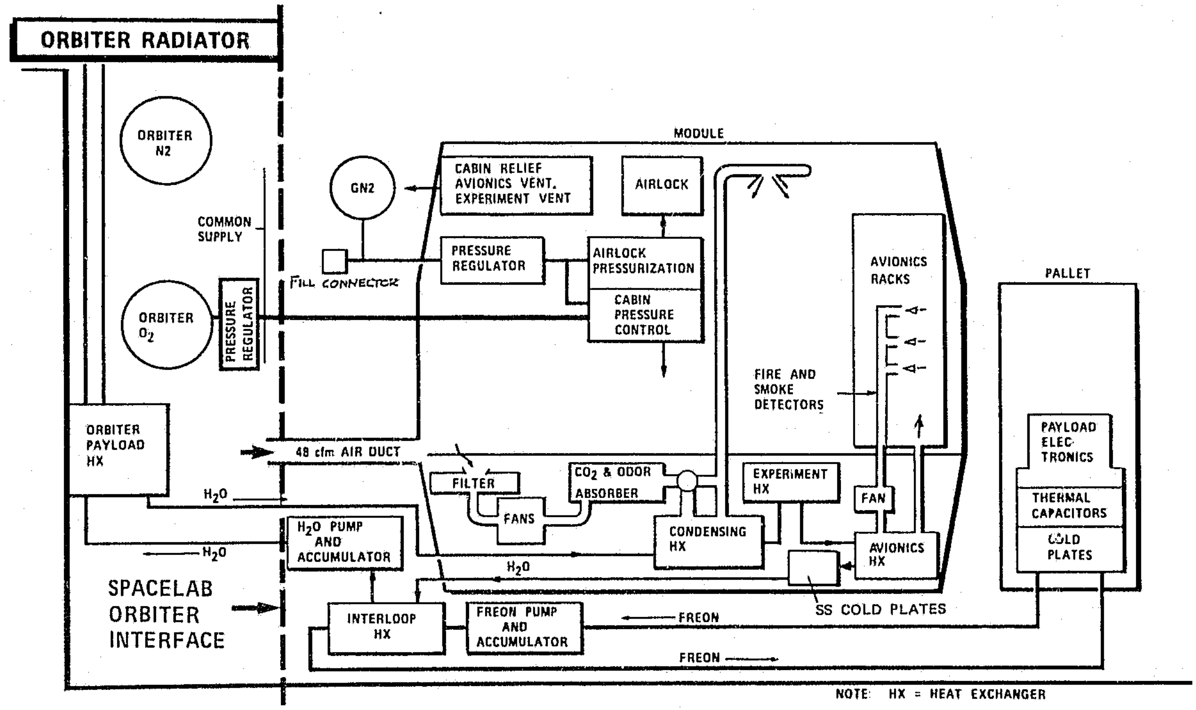


Figure 4.3-1: ECS Schematic

4.3.1 Environment Control and Life Support

The Spacelab Environmental Control and Life Support subsystem (ECLS) consists of the Atmosphere Storage and Control Section (ASCS) and the Atmosphere Revitalization Section (ARS). A shirt sleeve environment, compatible with the Shuttle Orbiter, is maintained in the pressurized module of the Spacelab. Gaseous oxygen supplied from the Orbiter and Spacelab supplied nitrogen provide the consumable gases for a 7 day mission, including airlock repressurization.

Carbon dioxide removal, and humidity and temperature control is effected by the cabin airloop, which rejects its heat to a waterloop. A separate avionics airloop is used to cool the rack mounted subsystem and experiment equipment.

A general overview of the ECLS subsystem is given in Figure 4.3-2, showing the arrangement of major ECLS components and the air ducting system inside the module.

A majority of the ECLS equipment is common with the Shuttle Orbiter.

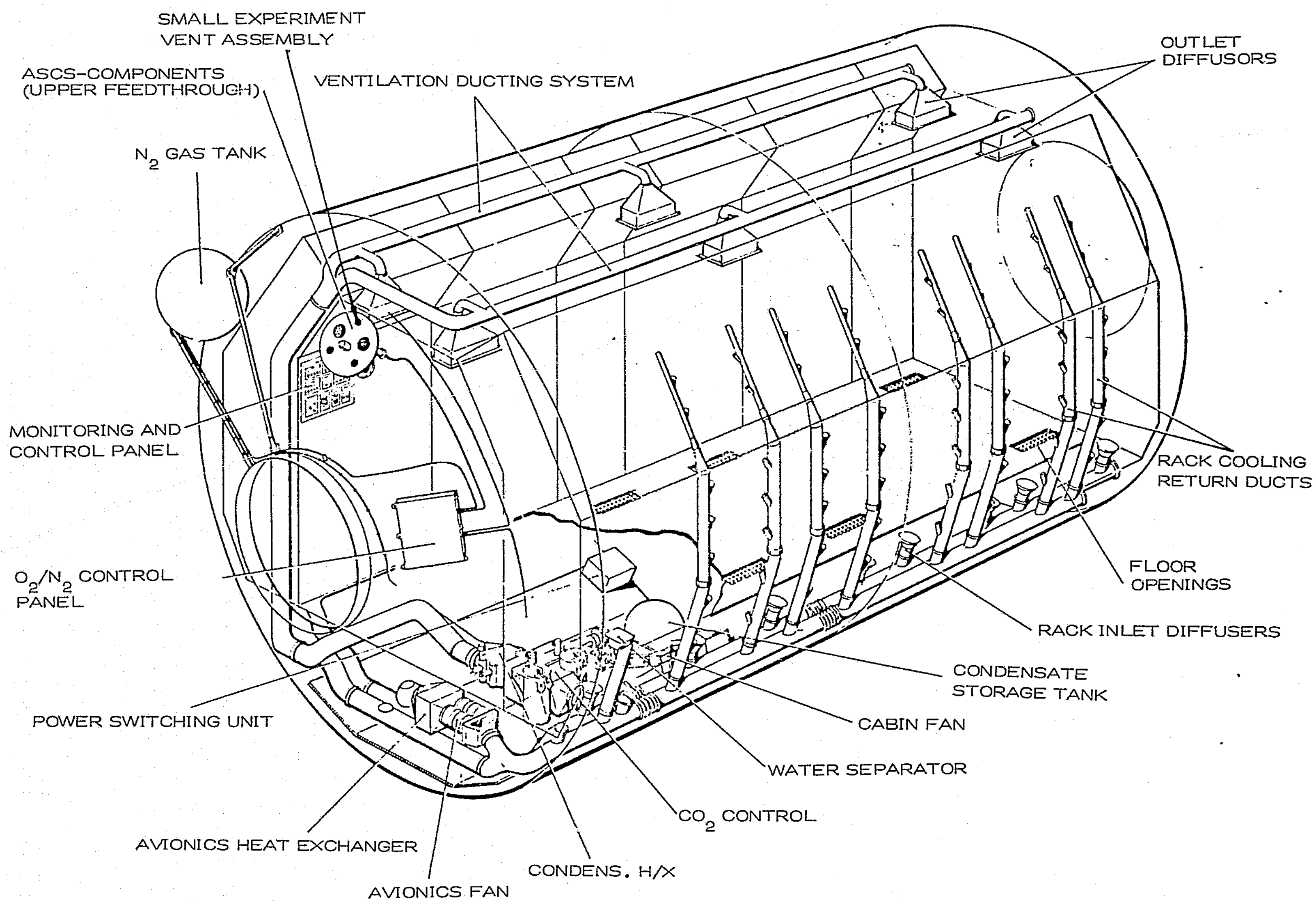


Figure 4.3-2: Environmental Control Life Support Subsystem

Table 4.3-2 lists some of the design characteristics of the ECLS subsystem.

The Spacelab ECLS is also to provide the protection of the pressurized module for the emergency cases of overpressure and negative differential pressure. Contamination monitoring, fire and smoke detection and fire suppression are being assessed and will be finalized in the near future.

Table 4.3-2 ECLS Design Characteristics

1. Atmosphere Composition	O_2/N_2 mixture Total pressure 1.013 $\pm$ 0.013 bar
2. Oxygen Partial Pressure	0.220 $\pm$ 0.017 bar
3. Cabin Air Temperature	Adjustable between 18° and 27° C Automatic control $\pm$ 1° C from the set point
4. Humidity Control	Maximum relative humidity 70 % Minimum water vapour pressure 0.0093 bar
5. Air Velocity in the Crew Area	5 to 12 m/minute
6. Metabolic Heat Loads Watts/Man	Max. 178 Nom. 164 Min. 118
7. Metabolic Oxygen Consumption g/man day	Max. 1000 Nom. 840 Min. 770
8. CO_2 Generation Rates g/man day	Max. 1180 Nom. 990 Min. 910
9. Mean Radiant Temperature	30° C Maximum
10. Max. Touch Temperature	45° C
11. Air Atmosphere Leakage	1.35 kg/day
12. CO_2 Control	Nom. 0.0066 bar or less Max. 0.01 bar
13. Particular Contamination Control	280 micron Filter
14. Airlock Repressurization	1.18 m^3, 7 times total for a 7 day mission

4.3.1.1 Atmosphere Storage and Control Section (ASCS)

The ASCS consists of one N_2 high pressure (207 bar) storage tank containing 17.6 kg of nitrogen and a high pressure regulator. O_2 is supplied via a line connected with the Orbiter O_2 supply system.

Cabin gas composition control is provided by a continuous regulation system similar to that in the Shuttle Orbiter.

The gases delivered from the high pressure gaseous storage are controlled by redundant supply assemblies.

The Spacelab structure is protected against excessive negative and positive differential pressures by an assembly containing redundant pressure relief valves.

4.3.1.2 Atmosphere Revitalization Section (ARS)

In the cabin loop (Figure 4.3-2), air is drawn from the cabin underfloor volume into a filter/debris trap assembly (280 micron filter) upstream of redundant cabin fans. Check valves prevent recirculation through the inactive fan. Downstream of the cabin fans lithium hydroxide canisters provide carbon dioxide control. They also contain a mixture of activated charcoal and $KMnO_4$ (Purafil) to control odor and trace contaminants in the cabin. Two canisters are provided in the air loop and operated simultaneously. The airflow through each canister is 20 kg/hr. The canisters used in Spacelab are identical to those used in the Orbiter.

Cabin air cooling and humidity control is provided downstream of the lithium hydroxide canisters by a condensing heat exchanger which interfaces with the water loop. Relative humidity is maintained within the required range for all predicted operating conditions including cabin temperature, condenser water supply temperature and cabin heat load variations.

Water is separated from the air by centrifugal effect and delivered into the condensate storage tank, and the dried air is then returned to the cabin.

Cabin ventilation is provided by air outlet diffusers located in each module segment. Air velocities within habitable areas of the cabin are kept within the required range of 5 to 12 m/min. as specified for crew comfort.

4.3.2 Thermal Control

The Spacelab Thermal Control Subsystem (TCS) transfers the heat generated by Spacelab subsystem and experiment equipment and by metabolic processes to the Orbiter heat rejection system.

The TCS can transfer 8.5 kW maximum continuously to the Orbiter. It can accommodate peak heat loads of up to 12.4 kW (i.e. 3.9 kW in addition to the 8.5 kW for 15 minutes every three hours).

The 8.5 kW transfer capability matches the Orbiter heat rejection capability offered to Spacelab and its payload. This is also consistent with the maximum continuous and peak power levels which the Orbiter provides to Spacelab, and the metabolic heat generated by the payload specialists working inside the module.

The 8.5 kW heat transfer capability is allocated on the Spacelab side to various cooling loops as described below. While certain design assumptions have been made to size the various cooling loops and their individual components, flexibility exists to change the distribution of heat loads over the various cooling loops to accommodate changing mission needs and experiment and subsystem equipment timelines.

The TCS consists of an Active Thermal Control Section (ATCS) and Passive Thermal Control Section (PTCS).

Schematic ATCS diagrams are shown in Figures 4.3-3, 4.3-4 and 4.3-5 for module-pallet, module-only and pallet-only configurations.(Not shown in Figure 4.3-3 and 4.3-4 are subsystem cold plates in the water loop.)

In module/pallet and module only configuration the heat generated by Spacelab and its payload inside the module is collected by a water cooling loop. A freon loop is used to collect heat generated on the pallet, and inside the igloo in pallet-only configurations. The freon loop interfaces with the water loop through the Spacelab interloop heat exchanger located outside the module on the forward end cone (together with the water and freon pump packages) in module-pallet configurations.

The heat is transferred to the Orbiter via the water loop in module configurations, and via the freon loop directly in pallet-only configurations. The water loop and freon loop, respectively, interface with the Orbiter payload heat exchanger.

The water loop contains several heat exchangers and cold plates in series (Figure 4.3-3 and 4.3-4). The condensing heat exchanger is used in the cabin air cooling loop that controls module air temperature. An experiment heat exchanger is provided to offer liquid cooling capability for experiments. The experiment heat exchanger can be installed either upstream or downstream of the avionics heat exchanger, depending on experiment temperature requirements. Then follows the avionics heat exchanger which is used in the avionics air cooling loop which cools rack mounted experiment equipment and some subsystem equipment.

Downstream of all three heat exchangers are subsystem dedicated cold plates for thermal control of subsystem equipment in the control center and workbench rack, i.e. the computers, I/O units, mass memory, multiplexer and 400 Hz inverters. Cold plate cooling for Spacelab subsystem equipment has been introduced

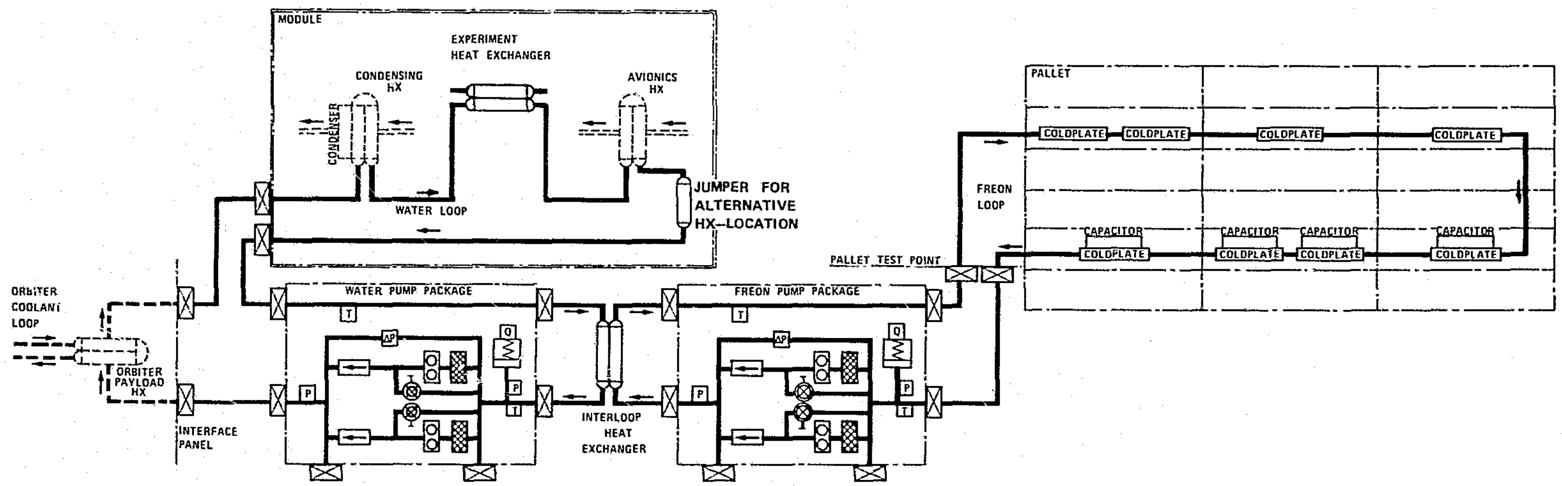


Figure 4.3-3: TCS Schematic Module-Pallet Configurations

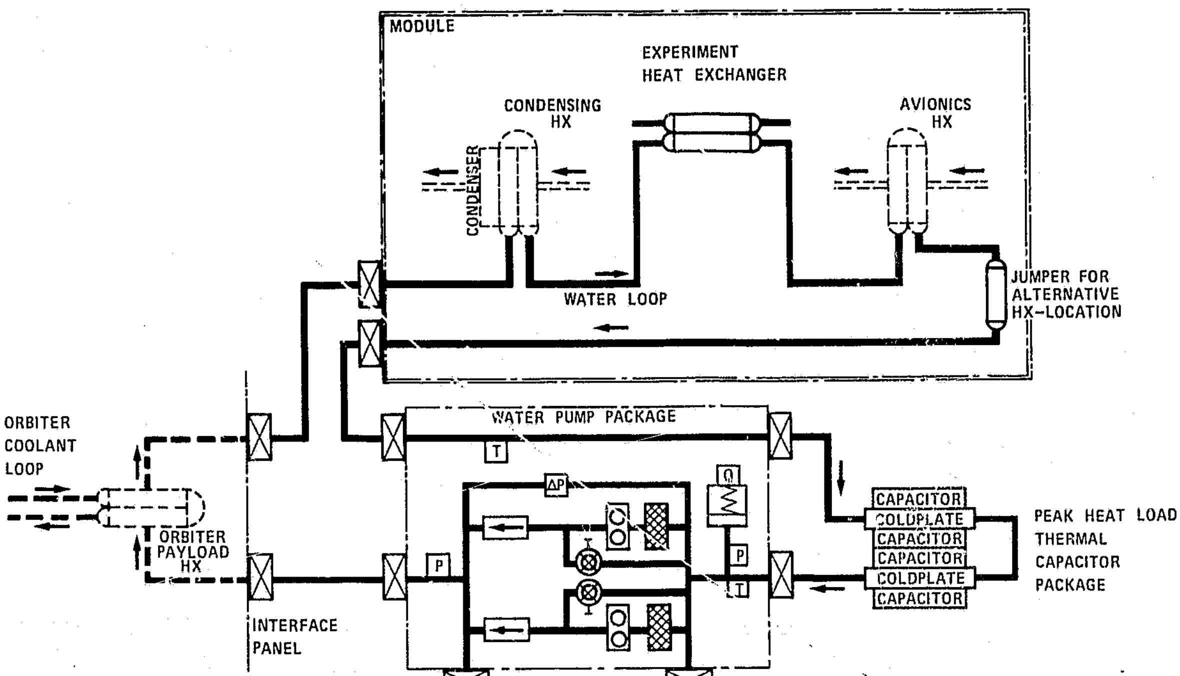


Figure 4.3-4: TCS Schematic Module-Only Configurations

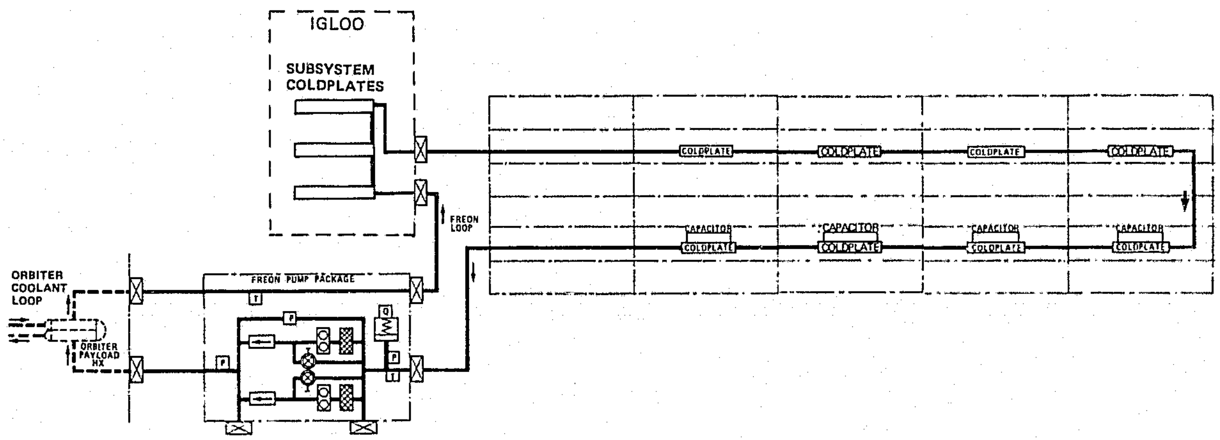


Figure 4.3-5: TCS Schematic Pallet-Only Configurations

to remove the subsystem induced heat load from the avionics air cooling loop. The subsystem cold plates are not available for experiment use in the module and should not be confused with the cold plates offered to experiments on the pallet.

To accommodate peak heat loads four thermal capacitors are provided. In the module only configuration these capacitors are installed on the outside of the forward end cone in a package consisting of the four capacitors and two cold plates. This package is not subdividable i.e. it is not possible to use only 3, 2 or 1 capacitor to save weight. For module/pallet and pallet-only configurations the thermal capacitors can be mounted on the experiment cold plates (one per cold plate) and any number between 1 and 4 may be used depending on mission requirements.

The freon loop in module-pallet configurations provides cooling for pallet mounted experiment equipment through the use of cold plates (Figure 4.3-5). In pallet-only configurations the loop also provides cold plate cooling for subsystem equipment located in the igloo. The igloo subsystem cold plates are separate and have to be distinguished from the cold plates on the pallet which are dedicated to experiments. Thermal control of subsystem equipment mounted on the pallet sill, e.g. RAU's and experiment power distribution boxes, is accomplished semi-passively with radiators and electrical heaters.

The PTCS employs high performance insulation, thermal covers and thermal coatings to protect the module/pallet, utility lines and other externally mounted Spacelab subsystem equipment and experiments from excessive temperature variations.

4.3.2.1 Cabin Air Loop

The primary purpose of the cabin air loop is to provide conditioned (temperature and humidity) air within established comfort criteria for the crew in the module. The cabin loop components and ducting are shown in Figure 4.3-2 and are described in Section 4.3.1. The nominal capacity of the cabin loop is 2.7 kW. This allows, in addition to the metabolic heat generated by the crew, transfer of an addition 1 kW of experiment generated heat to the water loop. This capability can be used to air-cool experiment equipment mounted in the center aisle, at the optical window/viewport assembly or at the high vacuum vent facility (which is not yet baseline).

Since the airflow in the cabin is determined by crew comfort requirements, experiments might have to provide their own fans to increase the airflow in localized areas (e.g. a center aisle rack). To avoid unnecessary airflow disturbance in the habitable area, outlet air from experiment equipment might have to be ducted into the subfloor area (it is currently under investigation to provide cutouts in the center aisle floor panels to allow ducting of outlet air into the subfloor area).

When using the capacity of the cabin air loop for experiment cooling account must be taken that a certain heat flow from rack mounted experiments into the module air can occur.

The cabin air temperature can be adjusted within the range 18° and 27° C, and will be controlled to within ± 1.1° C of the set point at full heat load.

4.3.2.2 Avionics Air Loop

4.3.2.2.1 Physical Layout and Ducting

The avionics air loop provides air cooling for rack mounted equipment and since subsystem equipment is mostly cold plate cooled, almost the entire capacity of the avionics loop is available to experiments.

The avionics loop components and ducting system is depicted in Figure 4.3-2. The avionics fan assembly establishes the airflow through the ducting system and the racks. (The flow is directed in such a way that air entering the racks is sucked out of the racks through return ducts.)

The fan assembly consists of two redundant fans and contains filters (280 micron) for particulate control. Downstream of the fan assembly the airlfow is ducted through the avionics heat exchanger, which interfaces with the water loop, and from there into supply ducts routed under the main floor on both sides of the module. It then enters the rack interior through short stubs and, after cooling rack equipment, it is sucked through the return ducts inside the racks into the return duct under the main floor, and back to the avionics fan.

This concept has the advantage that cool air enters the rack interior, thus minimizing the heat transfer to the cabin loop, and that the hot air is confined to the return ducts. There is one return duct in each single rack and one in each section of a double rack.

Cooling of rack mounted equipment is possible in two ways as shown schematically in Figure 4.3-6: (1) surface cooling for open equipment, and (2) ducted cooling for enclosed equipment.

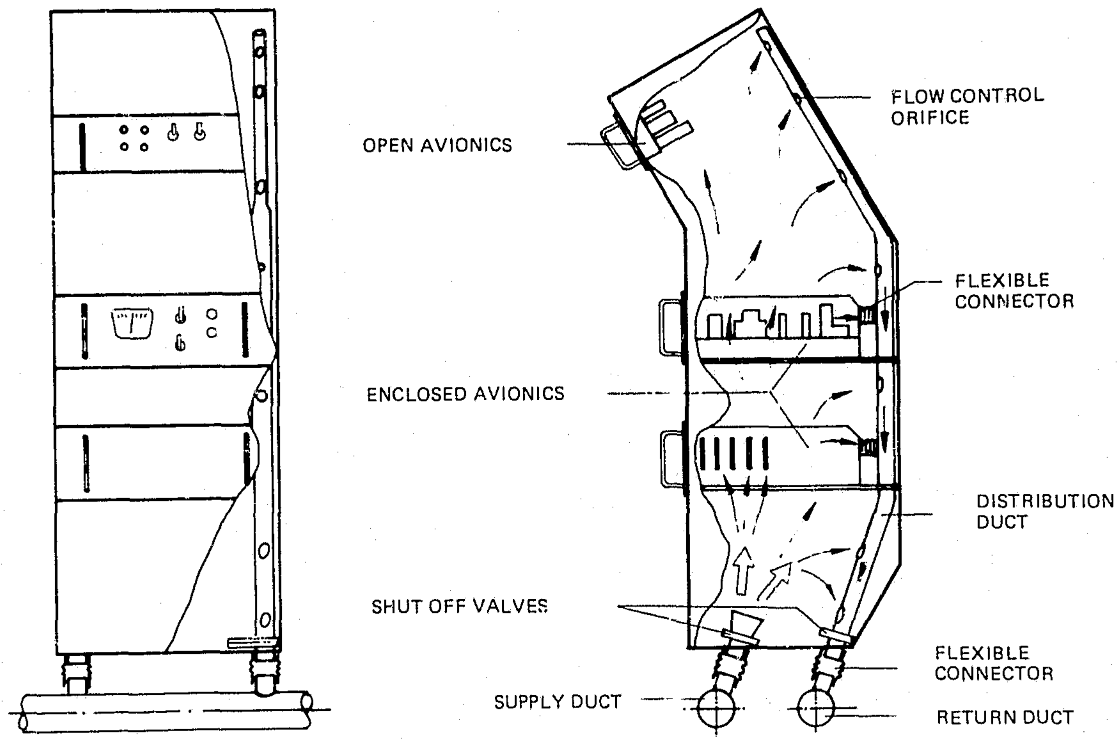


Figure 4.3-6: Rack Equipment Cooling Concepts

The normal mode of experiment cooling is ducted cooling. Air enters the experiment equipment through cutouts in the enclosure, cools the equipment, and is then sucked into the rack return duct. Experiment equipment is connected to the return duct in the back of the rack with flexible connections.

Figure 4.3-7 shows the layout of the return duct in a single experiment rack, as well as the number and the location of the inputs to the return duct. There are eight inputs to the return duct; the detailed interfaces with these inlets are under investigation.

Return duct inlets can be capped-off when not used and an adjustable orifice can be installed into each inlet to adjust the air flow/pressure drop in the connected experiment equipment.

For open experiment equipment the airflow inside the rack can be used for surface cooling. Air return and flow regulation is possible through unused return duct inlets.

In addition it is possible to shut off the airflow to an entire rack with manually operated shut off valves located in the rack supply and return ducts as shown in Figure 4.3-6. In the case of the double rack the entire rack has to be shut off from the airflow, i.e. the supply and return ducts in both halves of the rack are shut off together.

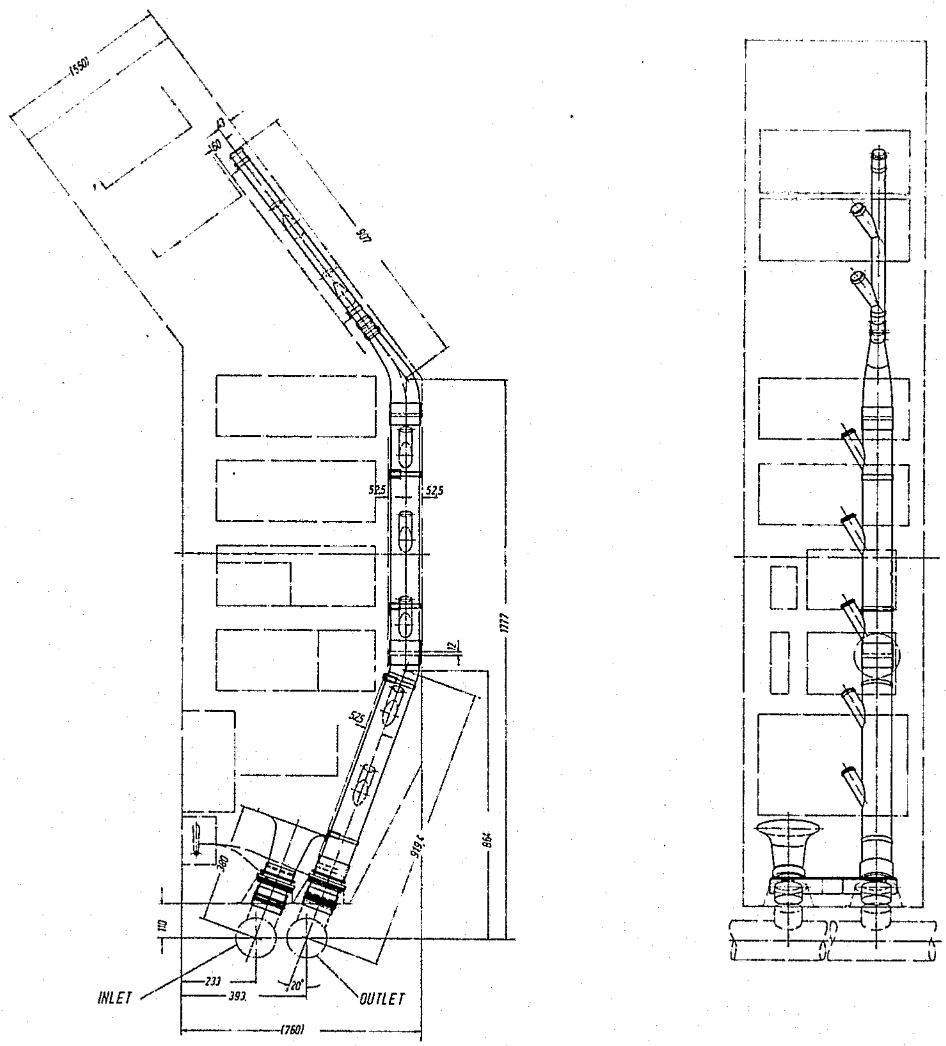


Figure 4.3-7: Air Duct System Single Experiment Rack

4.3.2.2.2 Cooling Capacity

The avionics loop nominal capacity is 4.5 kW of subsystem and experiment heat transferred to the water loop. However, the actual heat that can be collected is constrained by the 8.5 kW maximum continuous heat transfer to the Orbiter, and by the actual heat load distribution in all Spacelab cooling loops as mentioned before.

Table 4.3-3 illustrates this situation in more detail. Shown are four different heat load distributions, all resulting in 8.5 kW total heat to the Orbiter, for a module-only configuration and without using the experiment heat exchanger.

Selected are four different heat loads for the cabin loop (metabolic heat plus experiment heat) and fixed heat loads from subsystem equipment. Calculated are the remaining heat loads for the avionics loop (for subsystem and experiment equipment), the water temperature of various points in the water loop, and the air temperature in the avionics supply and return ducts (T_1 and T_2) and the avionics fan outlet (T_3). These numbers have been derived by analysis; they represent a certain design status and are, therefore, subject to confirmation.

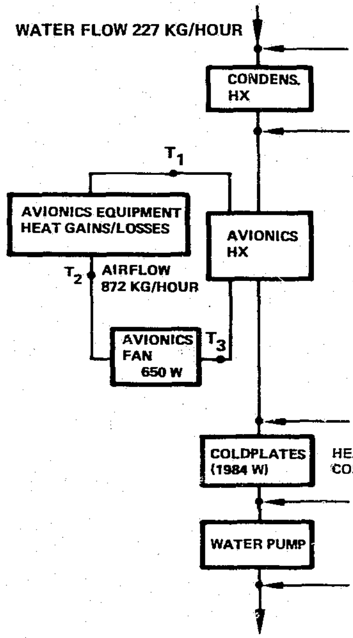
In the avionics loop about 1 kW is currently foreseen for air-cooled subsystem equipment. The approximate range of experiment heat that can realistically be accommodated is, therefore, in the range of about 1.7 - 4.2 kW, resulting in cool air return temperatures of about 37 - 41.5° C (under peak load conditions the return temperature can go up to about 50° C).

The avionics fan itself is sized for an airflow of 872 kg/hr (corresponding to an Arinc Standard heat load of 4 kW, i.e. 21.8 kg/hr per 100 W) and a pressure rise of 170 mbar between inlet and outlet.

The ducting layout and resulting airflow/pressure drop allow a maximum heat load on one side of the module of 3.13 kW (Arinc Standard). For proper operation of the avionics loop a 80 : 20 airflow upper ratio between the two sides of the module has to be maintained.

The rack return ducts are sized for a maximum heat load of 1.6 kW in a single rack and 3.13 kW in a double rack (distributed over two ducts). The actual heat load (Arinc Standard) that can be carried away depends, however, on the location of the rack within the module. The full 3.13 kW capability, i.e. the corresponding airflow in kg/hr, is only available in the double experiment rack next to the control center rack or work bench rack, respectively (position 2 or position 5 in Figure 4.3-8).

The maximum capability, i.e. the maximum possible airflow, in the last double rack and last single rack simultaneously, either on the left side or on the right side of the long module, is 2 kW and approx. 1 kW, respectively, (position 8 and 9 or possition 11 and 12 in Figure 4.3-8).



CASES	1.	2.	3.	4.
WATER SUPPLY TEMP.	7.3 $^{\circ}$C			
CABIN LOOP	3262 W	2782 W	1782 W	782 W
WATER TEMPERATURE	19.7 oC	17.8 $^{\circ}$C	14.0 $^{\circ}$C	10.3 $^{\circ}$C
AVIONICS EQUIPMENT	2689 W	3189 W	4189 W	5189 W
HEAT GAINS/LOSSES *	242 W	242 W	242 W	242 W
	2931 W	3431 W	4431 W	5431 W
AVIONICS FAN	650 W	650 W	650 W	650 W
AVIONICS HX	3581 W	4081 W	5081 W	6081 W
AIRLOOP TEMPERATURE T_1	24.9 $^{\circ}$C	23.7 $^{\circ}$C	21.4 $^{\circ}$C	19.0 $^{\circ}$C
AIRLOOP TEMPERATURE T_2	36.9 $^{\circ}$C	37.8 $^{\circ}$C	39.6 $^{\circ}$C	41.1 $^{\circ}$C
AIRLOOP TEMPERATURE T_3	39.6 $^{\circ}$C	40.5 $^{\circ}$C	42.3 $^{\circ}$C	44.1 $^{\circ}$C
WATER TEMPERATURE	33.3 $^{\circ}$C	33.3 $^{\circ}$C	33.3 $^{\circ}$C	33.3 $^{\circ}$C
HEAT LOAD FROM COLDPLATES TO WATER LOOP	1574 W			
WATER TEMPERATURE	39.2 $^{\circ}$C			
WATER PUMP	63 W			
WATER TEMPERATURE	39.4 $^{\circ}$C			
	8500 W	8500 W	8500 W	8500 W

* HEAT GAINS/LOSSES OF THE AVIONICS LOOP ARE AS FOLLOWS :

GAIN FROM COLDPLATED EQUIPMENT	+ 410 W
LOSS TO CABIN	– 125 W
LOSS TO UNDERFLOOR	– 43 W
TOTAL	+ 242 W

Note: For examplification only; data will be subject to further changes .

Table 4.3–3: Cooling Loop Heat Load Distribution Module-Only Configuration

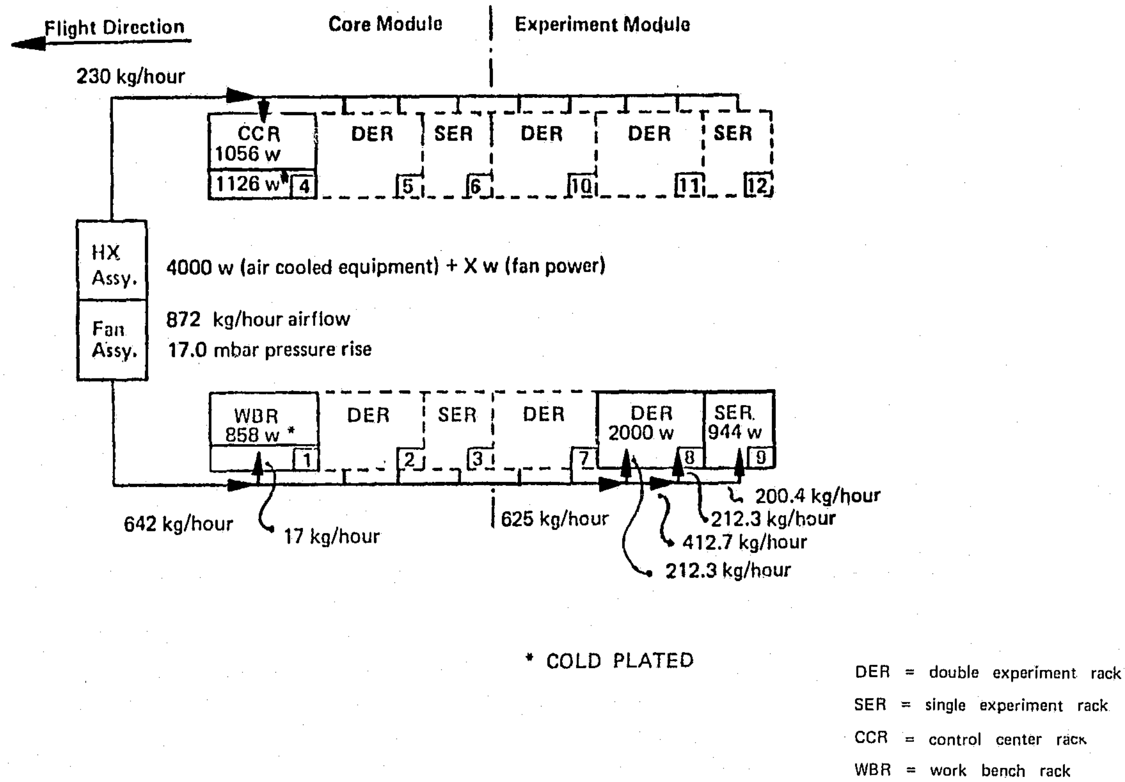


Figure 4.3-8: Avionics Loop Air Flow Distribution

Figure 4.3-8 illustrates a possible configuration of airflow distribution. It shows the split of the airflow between the left and the right side of the module and the further distribution between racks in various positions. The airflow into the control center rack has to be maintained continuously for subsystem cooling. The small airflow into the work bench rack is necessary for proper functioning of fire detection sensors but is not used for cooling. A large number of cooling flow distributions can be established because of the capability to shut off individual racks from the airflow.

4.3.2.3 Experiment Heat Exchanger

In addition to air cooling of experiment equipment in the cabin and avionics loop, it is possible to employ liquid cooling for experiments through the use of the experiment heat exchanger. This heat exchanger can be located either upstream or downstream of the avionics heat exchanger (Figure 4.3-4), and it is designed to transfer about 4 kW of experiment heat to the water loop. This heat, of course, cannot be transferred in addition to the nominal heat loads of the cabin and avionics loop, and the actual heat transfer capability is constrained by the total heat transfer of 8.5 kW to the Orbiter and the actual heat loads of the other cooling loops. Depending on the location of the experiment heat exchanger and of the heat loads in the cabin and avionics loop, the water inlet temperature to the heat exchanger can vary over TBD temperature range.

The experiment heat exchanger is located on the subfloor in the core segment of the module. The user has to provide all necessary plumbing, pumps, cooling liquid etc. necessary for cooling his equipment. The interface to experiments is currently at the coolant liquid inlet and outlet connectors of the heat exchanger. However, it is being considered to provide a more conveniently located and accessible interface, e.g. a connector bracket under the mainfloor. Routing provisions for experiment provided plumbing are also under investigation.

The normal working liquid for the experiment heat exchanger would be water, but any coolant liquid allowable in manned Spacecraft can be used.

The experiment heat exchanger is a mission dependent item and can be removed from the module when not required there. It should be noted, however, that there are no specific Spacelab provisions (e.g. attachment points, special plumbing, etc.) for using the experiment heat exchanger in any other location. Since the heat exchanger is identical to the interloop heat exchanger the use of the experiment heat exchanger in the pallet freon loop is in principle possible but would require extra user provided plumbing and attachments, and some analysis would have to be made of pressure drops in the freon loop, heat exchanger characteristics with freon on both sides, etc.

4.3.2.4 Cold Plates

Experiment equipment located on pallet segments in module-pallet and pallet-only configurations is provided with the possibility to use Spacelab provided cold plates for active thermal control. Passive thermal control measures for experiments on the pallet are the User's responsibility, except for some thermal blankets which are Spacelab provided.

The cold plates are part of the freon loop, which also provides coolant fluid to subsystem cold plates in the Igloo in pallet-only configurations (Figure 4.3-5).

The freon loop is designed to accommodate nominally up to eight experiment cold plates (in addition to subsystem cold plates). Cold plates are normally mounted to the inner pallet panels. The baseline Spacelab provides a total of eight (8) inner panels with inserts arranged in a standard hole pattern for cold plate mounting. The panels fit the 48^{o} sections of the pallet segments, and they have been selected to make it possible either to mount all 8 cold plates on one pallet segment or to distribute them over several pallet segments.

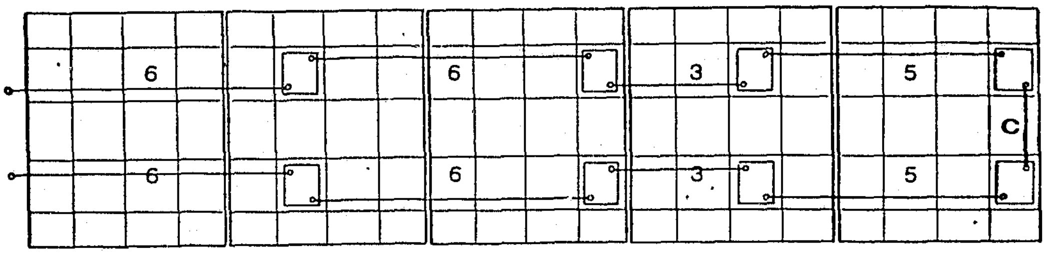
The Spacelab provides 8 cold plates for experiments together with a set of freon line plumbing that allows the implementation of various configurations of cold plates. The plumbing between cold plates consists of a range of stainless steel hard and flex lines which are connected to cold plate inlets and outlets with screw fittings. The set of plumbing allows a range of possible freon loop/cold plate configurations, of which three configurations are shown in Figure 4.3-9. Other configurations may be possible if additional (outside the present baseline) plumbing and/or cold plate attachments are provided and if the necessary analysis of pressure drops is performed. Alternative configurations with cold plates mounted directly onto experiments, for example, or with experimenter provided heat exchanger mounted into the freon loop would require further analysis of the technical and operational aspects involved.

In addition to cold plates, Spacelab also provides four (4) thermal capacitors to temporarily store peak heat loads exceeding the Orbiter nominal heat rejection capability of 8.5 kW. The thermal capacitors have to be mounted to experiment cold plates when needed.

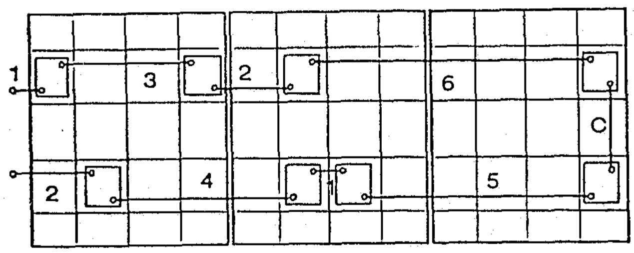
Cold plates and thermal capacitor physical dimensions are shown in Figure 4.3-10 and 4.3-11. Both elements provide the same standard hole pattern as the corresponding inner pallet panels. Experiment equipment is mounted to the pallet panel and cold plate (and thermal capacitor) with titanium bolts with thermal stand-offs between cold plate and pallet panel, to limit heat transfer between pallet and experiment. The mechanical load carrying capability of cold plates (and cold plate/ thermal capacitor assembly) is limited by the inner panel load carrying capability.

The cold plates are designed for a nominal heat rejection capability of 1 kW. Heat from experiments is transferred in the area around the mounting holes which are used to bolt experiment equipment to the cold plate and pallet panel. The design foresees a heat transfer of 13^{o} W/ ^{o}C per bolt with a conductance of 1 W/^{o}C. The use of thermal filler to establish optimum thermal conduct might be required. The actual heat rejection

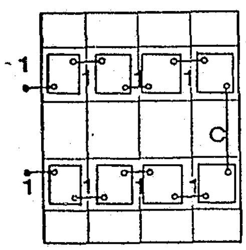
5 Pallets



3 Pallets



1 Pallet



Note : Numbers indicate length type of line

Figure 4.3-9: Baseline Cold Plate and Plumbing Configurations

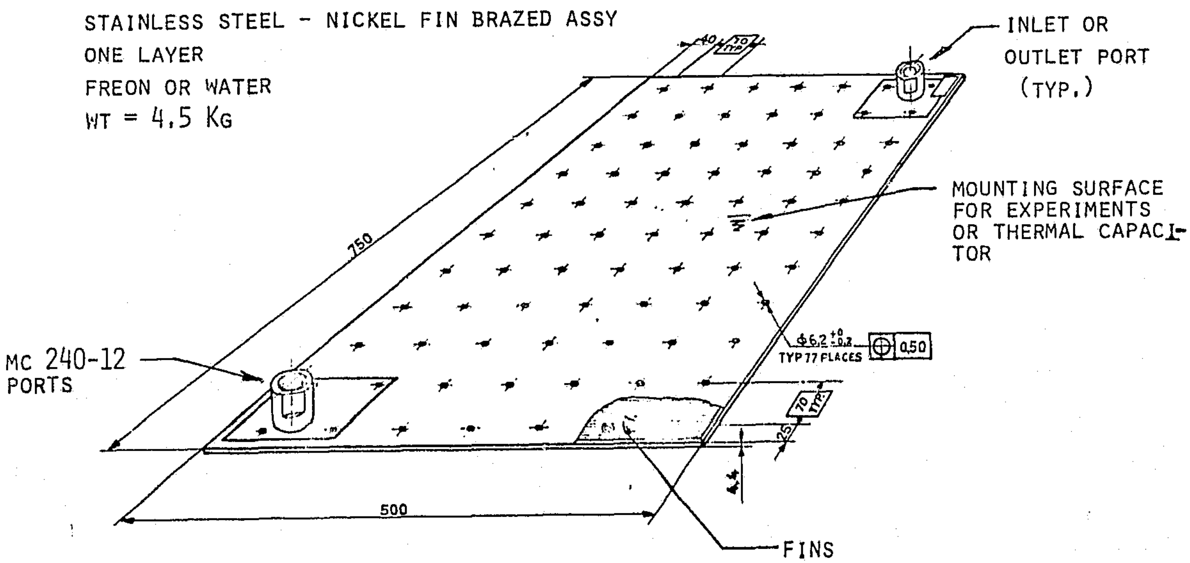


Figure 4.3-10: Cold Plate Assembly

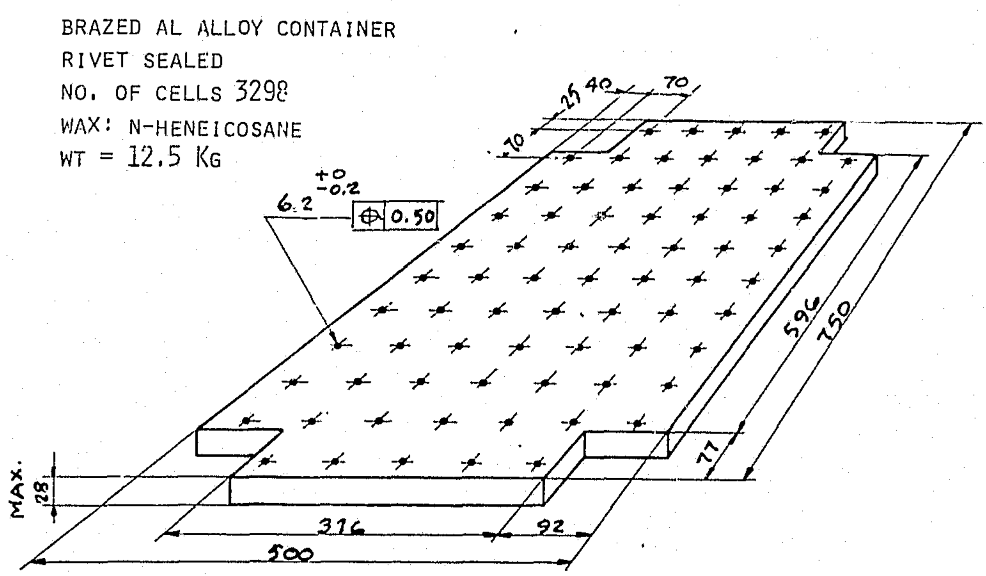


Figure 4.3-11: Thermal Capacitor Assembly

of the freon loop is again constrained by the Orbiter capability of 8.5 kW and the heat load of the other Spacelab cooling loops. In module-pallet modes, the experiment can expect to have about 4.85 kW maximum heat transfer available, and in pallet-only modes about 6.6 kW.

Since all cold plates are connected in series, additional constraints might be given with respect to the heat that can be transferred per cold plate due to the different cold plate inlet and outlet freon temperatures, according to the location of the cold plate in the freon loop.

For module-pallet modes the freon inlet temperature at the first cold plate will normally be not below 24^{o} C. The outlet temperature of the last cold plate can go up to about 43^{o} C at the full 8.5 kW heat transfer to the Orbiter.

In pallet-only modes the freon loop inlet temperature at the first pallet is about 10^{o} C. Outlet temperatures at the last cold plate can go up to about 43^{o} C. Detailed thermal analysis of the freon loop is ongoing and the results will be given in the next edition of the handbook.

4.3.2.5 Passive Thermal Control

The Spacelab passive thermal control employs insulations, surface coatings and thermal covers to protected the module, pallet segments, utility lines and other externally mounted subsystem equipment. Experiment passive thermal control is primarily the user's responsibility. However, Spacelab provides the possibility to attach thermal blankets around the circumference of each pallet panel, and also provides a set of tent like thermal blankets (details TBD) consisting of goldized Kapton layers, which can be used by experiments. Experiments need to be aware that the Orbiter payload bay temperature range between hot and cold case is approximately from -150^{o} C to $+80^{o}$ C, and that the pallet follows similar temperature excursions. For the pallet thermal design it was assumed that experiments are basically thermally decoupled from the pallet.

4.3.2.6 Orbiter Aft Flight Deck Thermal Control

The Orbiter provides air cooling for Spacelab subsystem and experiment equipment located in the Aft Flight Deckt (see also para. 2.5.2).

The minimum average cooling capability provided by the Orbiter is 0.35 kW during on-orbit operational phases. This is in addition to the 8.5 kW transferred by Spacelab to the Orbiter payload heat exchanger. The maximum AFD cooling capability is 0.75 kW average under the condition that the heat transfer in the Orbiter payload heat exchanger is limited to 7.85 kW. Peaks of 1 kW for 15 minutes every 3 hours can also be accommodated.

During prelaunch, launch descent and post landing phases 0.35 kW average heat rejection is available.

4.3.2.7 Summary of Thermal Control Capability

Table 4.3-4 summarizes the nominal heat load budgets for various cooling loops and Spacelab configurations.

Table 4.3-4: Spacelab Thermal Control Budget

Configuration	Basic Spacelab Subsystem	Mission Dependent Spacelab Subsystem		Experiment Equipment					
				Module					
		Long Module	Pallet	Cabin Loop	Avionics Loop	Exp.HX	Max. Available	Pallet	Total Available
max./nom. Module/ Pallet Peak	TBD	TBD	TBD	1 kW TBD	3 kW TBD	4 kW TBD	4 kW TBD	4.85 kW TBD	TBD
max./nom. Module-Only Peak				1 kW TBD	3 kW TBD	4 kW TBD	4 kW TBD	–	TBD
max./nom. Pallet-Only Peak				–	–	–	–	6.6 kW	TBD

It has to be noted that the maximum nominal values quoted are not additive, and are limited by the 8.5 kW heat transfer capability to the Orbiter and the actual heat load distribution over the various cooling loops. As pointed out in the preceeding sections, the inherent flexibility of the Spacelab thermal control subsystem allows changing of the heat load distributions according to mission needs and experiment and subsystem equipment timelining.

Detailed Spacelab thermal analysis is ongoing and results relevant to the user (e.g. heat load distribution inside experiment racks, temperature distribution in the freon loop, etc.) will be available in the next edition of the handbook.

4.3.3 ECS Capabilities During Ground Operations and Ascent and Descent

4.3.3.1 ECS Capabilities Prior to Installation of Spacelab into Orbiter

The ECS system is capable of providing all on-orbit conditions and on-orbit operational cooling capabilities for a complete Spacelab configuration during times when Spacelab GSE services are available (details on ground operational phases and timelines are TBD). This is done by connecting GSE services to the appropriate ECS GSE connections. The following GSE connections are available which allow full conditioning capability without operating flight pumps and fans:

a) Module cabin loop supply air duct connector, GSE duct is provided through the module forward end cone opening.

b) Module avionics loop supply and return air duct connectors in the module subfloor area. GSE ducts are provided through the module forward end cone opening.

c) Water supply and return connections for the Water Pump Assembly (water cooling loop for module heat exchangers).

d) Freon supply and return connections for the freon pump assembly (for pallet cold plates).

4.3.3.2 ECS Capabilities After Installation of Spacelab into the Orbiter (Prelaunch)

After installation of Spacelab into the Orbiter the Orbiter can provide limited cooling to Spacelab as long as the Orbiter GSE is connected and operating. The heat transfer from Spacelab is limited to 1.5 kW, allowing a partially active Spacelab. Under investigation is to provide part of this capability to experiments by running the cabin air loop, experiment heat exchanger, and/or the freon loop for pallet mounted experiments.

The capability exists, however, to increase the heat transfer capability on ground to several kW by connecting the Orbiter payload heat exchanger directly to special Orbiter GSE. This would require additional mission dependent plumbing in the Orbiter cargo bay which will be chargeable to the Spacelab mass budget (details are under investigation). Details on Orbiter ground operational timelines are TBD.

4.3.3.3 ECS Capabilities during Ascent and Descent

During ascent and descent phases the Orbiter provides 1.5 kW heat rejection capability to Spacelab (in addition to 0.35 kW in the AFD). This allows to partially activate Spacelab and to provide limited cooling capability to experiments. Under investigation is to provide cooling to experiments via the module cabin loop, experiment heat exchanger and/or the freon loop. The ascent phase extends to about one hour after lift-off, and the descent phase starts about 8 hours before landing.

4.3.3.4 ECS Capabilities Post Landing

Post landing ECS capabilities are the same as during the prelaunch phases after Orbiter GSE has been connected (including increased capabilities). Detailed Orbiter post landing timelines are TBD.

4.3.4 Small Experiment Vent Assembly

A small experiment venting assembly is located in the upper part of the ASCS feedthrough plate in the forward end cone shown in Figure 4„3-3. The assembly consists of a modified Skylab butterfly valve with an 11.4 cm^2 orifice area flow restrictor and quick disconnect. The flow rate is 0.18 kg/s max., at a 1.013 bar differential pressure and 21° C. Internal leakage is 3 SSCM, operating temperature is -54° to $+93^\circ$ C.

Experiment provided plumbing can be connected to the quick disconnect to vent experiment vacuum chambers and similar equipment in the module, e.g. in the experiment racks. Details of plumbing routing are under investigation.

To provide high pumping speeds for experiments inside the module, the development of a Spacelab provided High Vacuum Vent Facility is currently under consideration (para. 4.6.6).

4.4 Command and Data Management Subsystem

4.4.1 General

The Command and Data Management Subsystem (CDMS) provides a variety of services to Spacelab experiments and subsystems.

These services include:

- data acquisition
- data processing
- data formatting
- data transmission
- recording
- monitoring
- display
- command and control capability for experiments
- command and control capability for subsystems
- audio intercommunication
- caution and warning
- provisions for closed circuit television

The equipment provided by the CDMS to Spacelab experiments is listed in Table 4.4-1.

Table 4.4-1: CDMS Equipment for Experiments

Basic Spacelab	Mission Dependent
1. Exp. Data Bus	1. Experiment Computer
2. Back up Computer	2. Experiment I/O Unit
3. Mass Memory	3. Experiment RAU's (8 total) (22 may be accommodated)
4. Keyboard/CRT (2)	4. Keyboard/CRT (1)
5. Intercom	5. High Rate Multiplexer
	6. High Rate Digital Recorder

Figure 4.4-1 presents a functional block diagram of the CDMS. It shows the location of CDMS equipment for the module plus pallet mode and for the pallet-only mode, using the igloo.

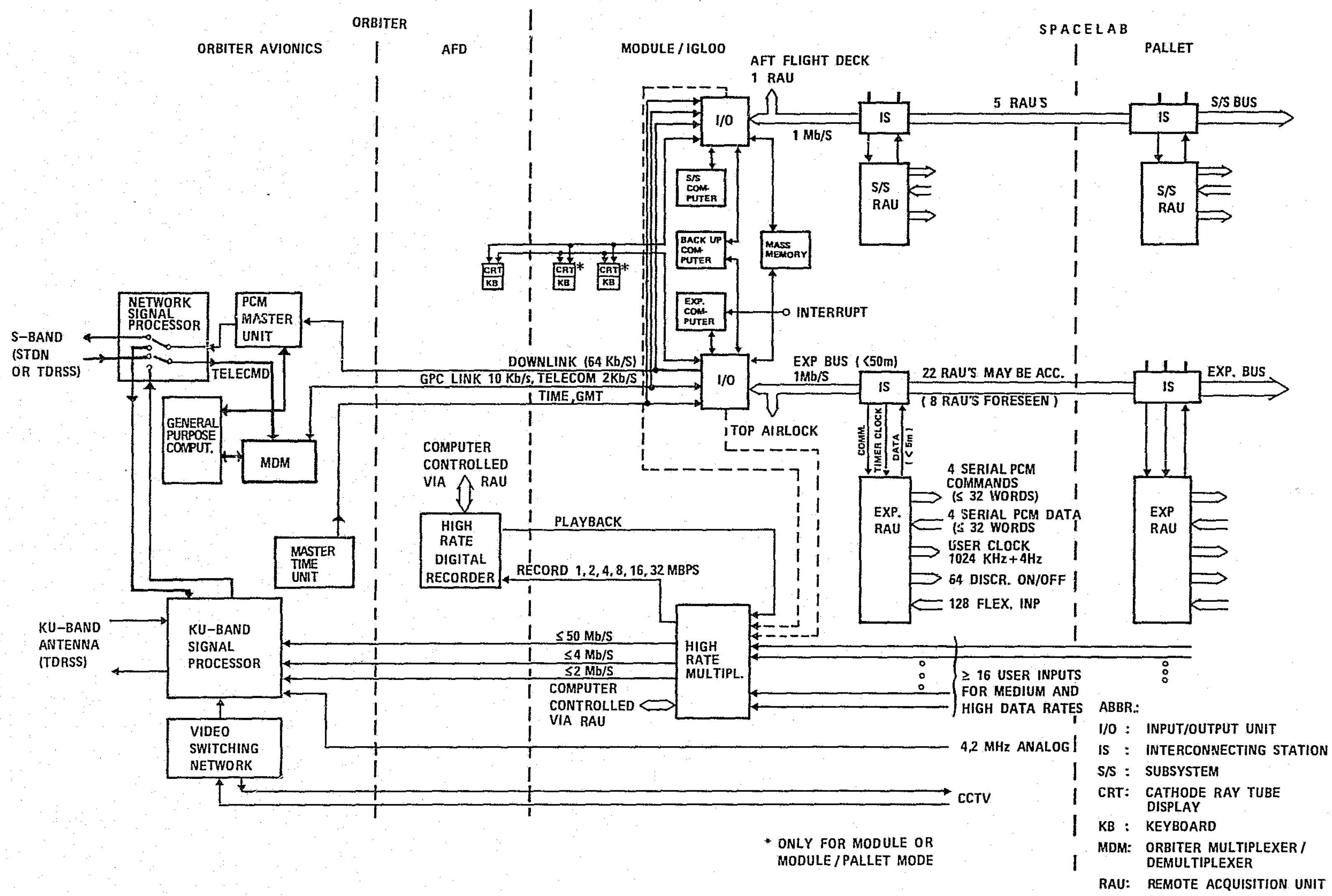


Figure 4.4 - 1 CDMS Block Diagram

Experiment outputs delivering housekeeping and low speed scientific data that need further on-board processing, are sampled by Remote Acquisition Units (RAU's) and transferred to the experiment computer via Interconnecting Stations (IS), the experiment data bus, and the Input/Output (I/O) unit.

On the same path, serial PCM and On/Off commands are transferred from the experiment computer, via the RAU's to the experiments.

The RAU User Time Clock delivers precision reference timing information.

Typical functions for on-board processing of scientific data by the experiment computer are quick look analysis, data compression, wave form analysis etc. Programs for control and processing of experiments exceeding the capability of subsystem and/or experiment computer can be loaded at execution time from the Mass Memory Unit (MMU).

A backup computer, which is primarily intended as backup for the subsystem computer, is also available to experiments in case of experiment computer failure. The backup computer is normally filled with subsystem programs. Before operating as experiment computer the core memory has to be loaded with the appropriate experiment software from the MMU.

The subsystem and experiment branches of the CDMS are identical and built up with the same components, (computer, I/O unit, data bus, and RAU modules) except the user time clock capability which is unique for experiments. However, it should be noted that there is no direct link between the subsystem and experiment branch.

The primary mode for experiment and subsystem commands and control is automatically via CDMS

These commands are initiated automatically through pre-programmed sequences in the computer (or stored in the MMU), or semi-automatically by interaction of the keyboard/CRT with the computer, or from the ground by telecommands through the Orbiter uplink (2 Kb/s).

Data, processed by the experiment or subsystem computer can be displayed on the CRT's having vector and graphic display capability.

Low bit-rate scientific and housekeeping data processed by the experiment computer can be transmitted by the Orbiter down link via STDN or TDRSS.

Medium and high rate scientific data can be accepted by the High Rate Multiplexer (HRM) for transmission to ground. The HRM is capable of multiplexing the data from up to 16 experiment input channels and can route the combined output data stream to the Orbiter Ku-Band down link via TDRSS. The mission phase dependent input channel allocation and input data rates are computer controlled. (The HRM is in the technical definition phase and not yet baseline.)

A variable speed High Rate Digital Recorder (HRDR) which is also integrated into the multiplexer system is provided by CDMS primarily to bridge TDRSS non-coverage periods in the Orbiter Ku-Band down link.

Spacelab provides the necessary electrical interfaces for experiment provided CCTV equipment to form an extension of the Orbiter CCTV. There is space for a TV monitor in the control center rack and an electrical interface for an EIA standard video camera.

The Spacelab provides a 4.2 MHz analog channel for use by the experiment, e.g. to accommodate non-EIA-standard TV signals.

CCTV and analog signals are transmitted to the ground through the same analog channel of the Ku-Band down link. TDRSS non-coverage times are not bridged by a recorder.

Duplex voice links for onboard or Orbiter ground communication are provided by the Intercom System.

Emergency, warning and caution conditions are detected and displayed by the Caution and Warning System (C & W).

4.4.2 Data Acquisition and Control

4.4.2.1 Remote Acquisition Units

4.4.2.1.1 Functional Concept

The RAU´s are the principal interfaces for the bidirectional link between experiments and the CDMS for acquisition of low bit rate digital data, analog data and for commands. The data exchange between RAU´s and I/O unit is performed via a bit serial bus with a 1 Mb/s clock rate. The data are encoded in a self clocking bi-phase code (Manchester II).

Data acquisition is initiated by the I/O unit and performed in a demand/response manner. Commands to the experiments are checked for validity in the RAU. The result is sent back to the I/O unit.

Each experiment RAU incorporates the following user interfaces:

Inputs: 128 flexible differential inputs for analog or discrete signals
4 serial PCM data channels with associated clocks, code NRZ-L + clock

Outputs: 64 On/Off command channels
4 serial PCM command channels with associated clock, code NRZ-L + clock
1 User Time Clock 1024 KHz
1 User Time Clock Update, 4 pulses/second synchronized with on-board GMT

A block diagram of the RAU is given in Figure 4.4-2.

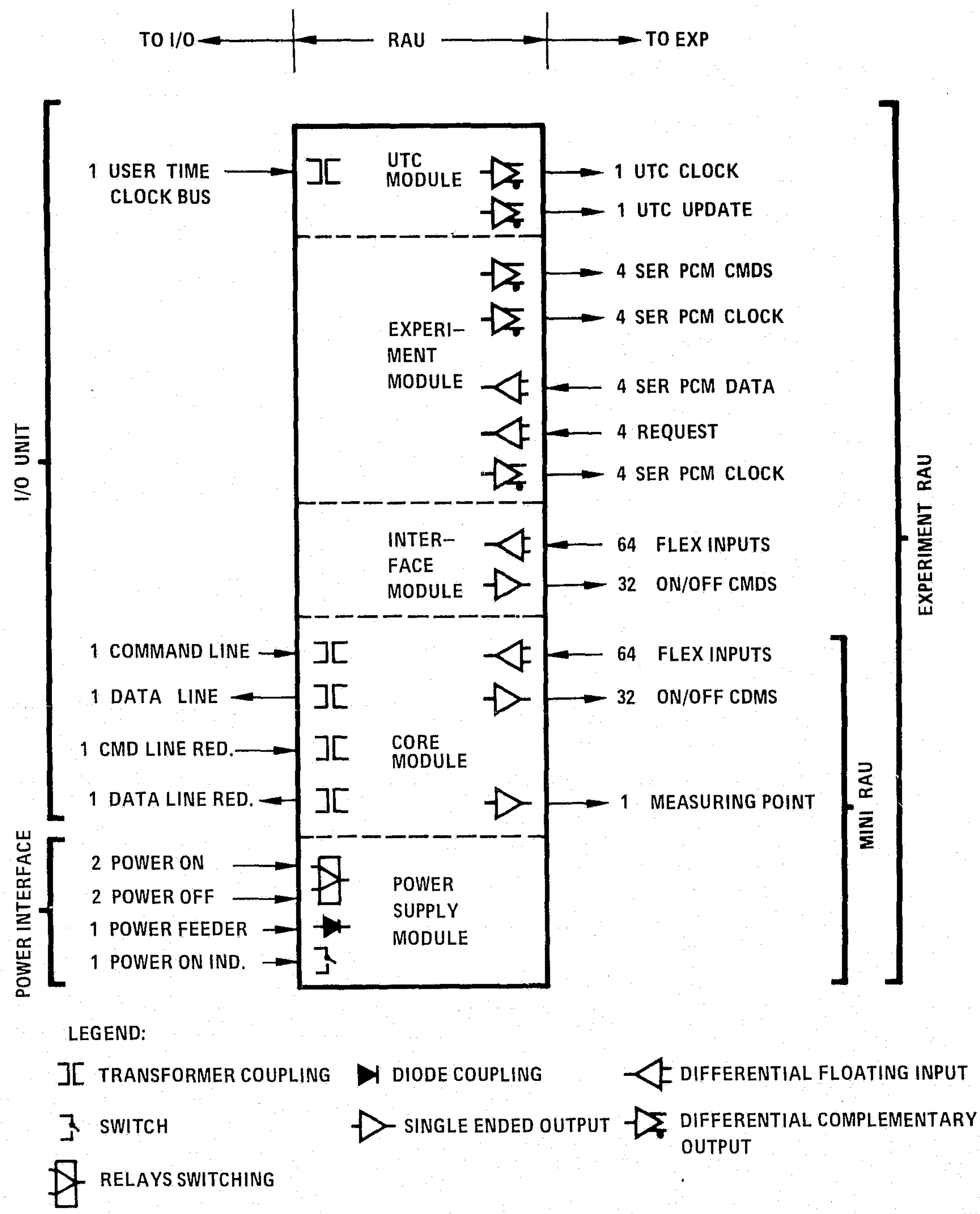


FIGURE 4.4–2 REMOTE ACQUISITION UNIT BLOCK DIAGRAM

The RAU data acquisition is based on a software controlled concept. The software for subsystem data acquisition and control is provided by Spacelab. The software for experiment data acquisition and control has to be provided by the experimenter in accordance with his requirements. Applicable portions of the Spacelab software can be used by the experimenter.

The RAU's will be scanned periodically with periods of 10 ms, 100 ms, or 1 s. Each scan cycle will be initiated and controlled by the General Measurement Loop which is part of the Spacelab computer software. The experimenters may design their own software to generate additional measurement cycles using the operating system task scheduler. This scheduler will accept priority levels and queue up experiment software requests for data and command transmission.

4.4.2.1.2 Physical Concept

Experiment RAU's can be connected to the experiment data bus at a number of interconnecting stations (IS) in the module and on each pallet. There are 2 interconnecting stations in the core segment, 3 in the experiment segment, and 2 on each pallet segment.

Each station accommodates two RAU's. The Spacelab baseline contains 8 experiment RAU's. The electrical characteristics of the experiment bus allow the accommodation of up to 22 RAU's although the computer allows the address of up to 32 RAU's.

In the Spacelab baseline the RAU's are located in the lower part of the experiment rack and on the pallet sill. However, the concept allows RAU's to be mounted in other rack and pallet locations as long as the cabling to the interconnecting station is below 5 meters.

In addition to the Spacelab provided locations the experimenter may use his own mounting provisions for RAU accommodation, e.g. if he uses his own racks and/or experiment equipment mounted directly to the module floor or to the pallet floor. In every case the environmental specifications for the RAU have to be met. The physical dimensions and connector locations of the RAU's, as depicted in Figure 4.4-3 are preliminary.

The Spacelab provides the facilities to accommodate an experiment RAU in the Top Airlock. This RAU is connected to the experiment I/O-unit via a dedicated branch of the experiment data bus which is routed through the module overhead structure.

To satisfy the various user requirements, a modular RAU concept was chosen. The smallest unit available to the user is the Mini RAU, consisting of the Power Supply Module and the Core Module (see Figure 4.4-2). There will be additional Modules such as the Interface Module (doubling the Core Module) the Experiment Module (providing serial PCM inputs and outputs) and the User Time Clock (UTC) Module. The functions of these modules are described in the following section.

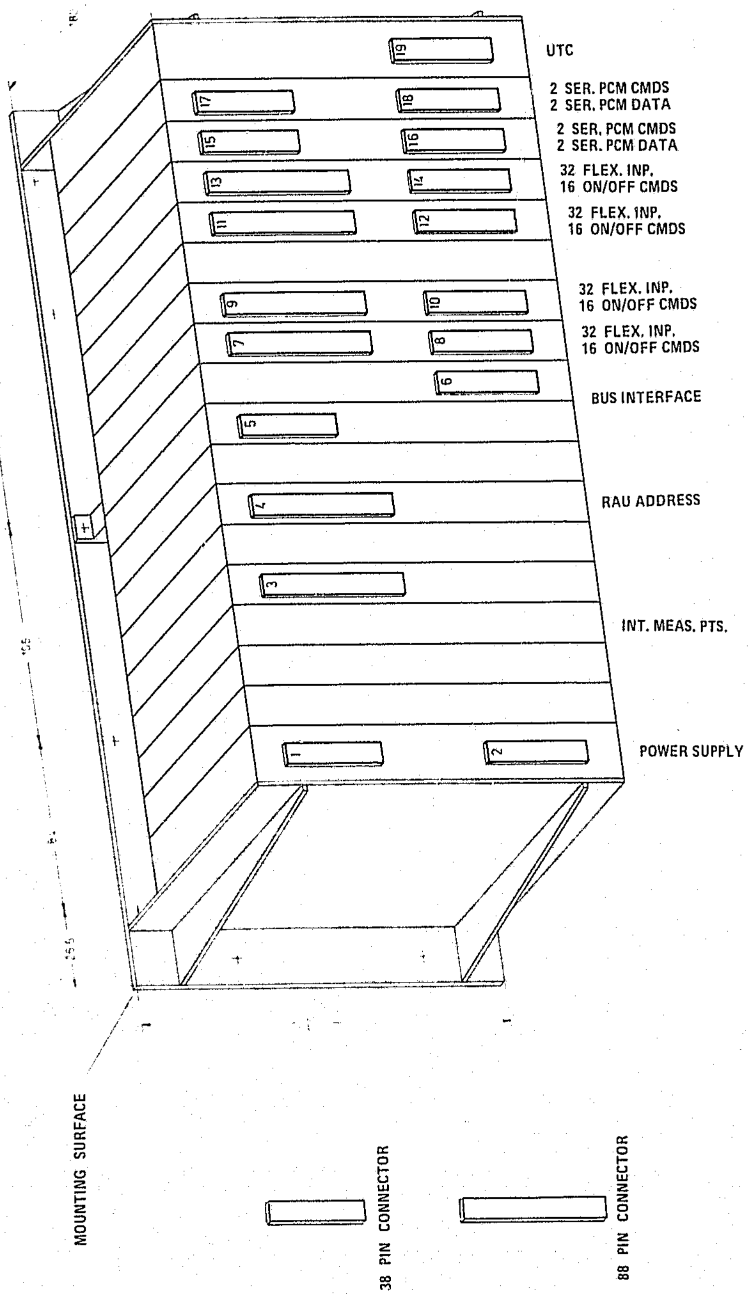


Figure 4.4 - 3 Physical RAU Dimensions

4.4.2.1.3 RAU/Experiment Links

The RAU/experiment links are depicted in Figure 4.4-2. The detailed electrical interface is presented in 4.4.2.1.6.

4.4.2.1.3.1 Data from RAU to Experiments

On/Off Commands

The RAU will provide 64 (32 + 32) On/Off commands as constant voltage levels to the experiments. These outputs may be used to set or reset experiment functions. Each On/Off command output has to be individually addressed by the computer software. After a software determined time (min 30 ms) the On/Off output will be reset automatically to save power. The load capability of these RAU outputs is designed to drive opto-couplers directly.

Serial PCM Commands

Four (4) RAU channels can deliver serial PCM commands to the experiments, in connection with a RAU provided 1 MHz clock frequency. The code is NRZ-L + clock. The maximum blocksize per software requested transaction will be 32 16-bit-words (plus parity bit). The time gap between each two transmitted words will be 3 µs. In addition to commands to control experiment functions the user may receive via this link additional software generated information such as GMT, ground data, Orbiter State Vector, etc.

User Time Clock Outputs

The experimenter can receive timing information from the User Time Clock (UTC) module. A 1024 KHz (duty cycle 0.5) and a 4 Hz update output (pulse width 10 µs ± 1µs) will be available. These clock signals are derived from the master oscillator in the Orbiter Master Time Unit and are therefore synchronized with GMT.

4.4.2.1.3.2 Data from Experiment to RAU

Flexible Inputs

The RAU provides 128 (64 + 64) flexible inputs. The electrically identical differential inputs can be programmed to accept either:

- discrete input signals, i.e. one bit of parallel digital data
- analog input signals which are digitized in the RAU

In the case of discrete data acquisition, groups of 16 inputs are addressable. Thus 16 bits in parallel are accepted and, after addition of one parity bit in the RAU, they will be serially transferred to the computer via the I/O unit. The number of 16 bit groups accepted during one scan cycle is software controlled and may vary from 1 to 8.

In the case of analog data acquisition, two adjacent input channels (single acquisition mode) or groups of 16 input channels (scanning acquisition mode) are addressable. The acquisition mode is software controlled.

The analog/digital conversion has a resolution of 8 bits, thus the conversion of signals in two adjacent input channels leads to a 16 bit word. This word, after addition of one parity bit, is sent via the serial data bus to the I/O unit. In the scanning acquisition mode up to 64 words can be transmitted to the I/O unit. The time for sequentially scanning the input channels is determined by the transmission time only (20 μs per word) because digitizing in the RAU is performed by two alternating fast ADC´s.

The analog/digital conversion in the RAU has the following characteristic:

Full scale voltage range:	- 5.12 V to + 5.08 V
Common mode voltage range:	± 6 V
Conversion gain:	1 / 40 mV
Encoding format :	8 bit, 7 bit amplitude + 1 bit sign
Accuracy (common mode ± 1 V):	± 31 mV

The flexible inputs may be wired to balanced or single ended sources. (Single ended wiring may result in degraded performance due to ground loops and noise pick up.)

Serial PCM Data

Four (4) RAU channels are available to transfer NRZ-L coded serial PCM data from experiments via RAU to the computer. Each channel consists of a data line, a clock line, and a request line. The RAU will accept from the experiment 17 bit words, including a user generated parity bit as long as the user provides a logic `one´ level on the request line. However, an internal timer in the RAU will restrict the number of serial words accepted to a maximum of 32. If the request line level changes from `one´ to `zero´ during the transmission of a word, all 17 bits of this word will be accepted by the RAU and transmitted to the computer. Each serial PCM data channel will provide the user with a 1 MHz clock signal to read out the data contained in the experiment buffers.

With appropriate software it is feasible to announce the request for serial PCM data by an On/Off command to the experiment.

4.4.2.1.4 I/O Unit/RAU Links

The experiment RAU´s are linked to the I/O Unit by the experiment bus. The data bus consists of a simplex "command line" which carries instructions and data from the IOU to the RAU´s and a simplex "data line" which carries responses from RAU's back to the I/O unit. Instructions and data are transferred at 1 Mb/s in 16 bit plus parity words in Manchester II (Bi-phase-level) code. Each word is preceded by a 3 µs non-valid Manchester synchronization signal.

A "clock bus" is also provided which distributes the MTU derived 1024 KHz clock and update pulses to RAU's for the user. The data bus and the RAU bus interface is dual redundant.

The subsystem bus connecting the subsystem RAU's to the subsystem I/O unit is similar to the experiment bus except that the "clock bus" is not provided.

4.4.2.1.4 Command Transfer

The serial PCM command transfer from the I/O unit to the RAU is depicted in Figure 4.4-4.

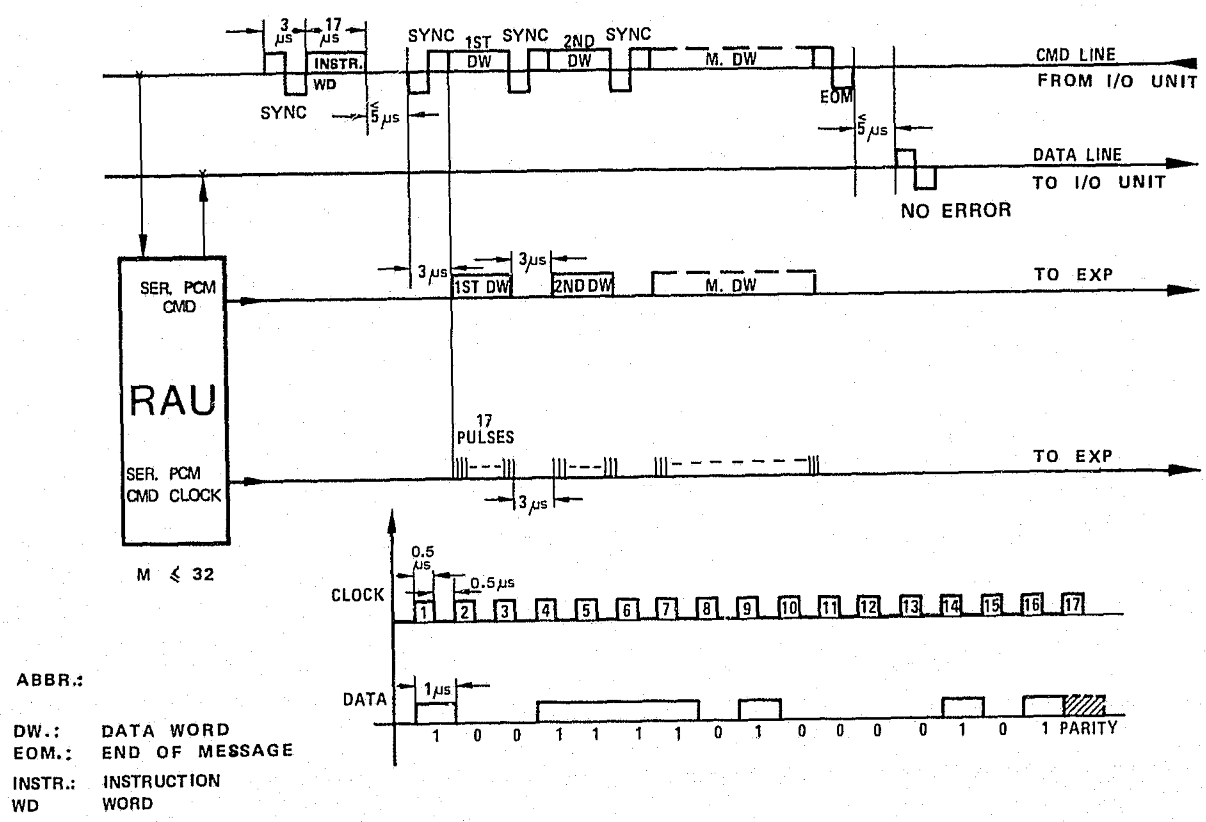


Figure 4.4-4: Serial PCM Command Transfer

The transfer will start with a sync pattern followed by an instruction word. This instruction word includes RAU address, RAU channel designation, and command type. The structure of this instruction word is sketched in Figure 4.4-5. After a short delay (<5µs), necessary for the internal set up of the RAU, the command information for the experiment is transmitted as 16 bit + parity data words blocked to a maximum of 32 words per transaction. Each word is preceded by a sync pattern. The end of the block is indicated by an End of Message signal.

The RAU will check the received data words by checking the Manchester code pattern and by checking the parity. It will send back a signal on the data line, if no errors are detected. In case of an error detected the RAU will shut down its output immediately. Otherwise the RAU will decode the received data words and transmit them to the experiment in real time.

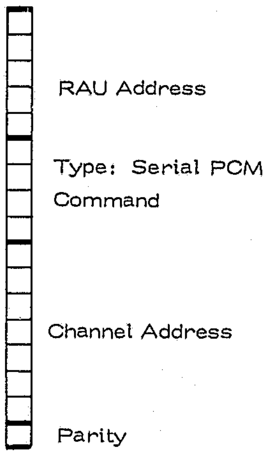


Figure 4.4-5: Instruction Word
Serial PCM
Command

The ON/OFF command transfer from the I/O unit to the RAU is depicted in Fig 4.4-6. The transfer will start with an instruction word preceded by a sync pattern. This instruction word includes RAU address, RAU channel designation, and command type information, including the level to which the ON/OFF output of the RAU has to be set (see Figure 4.4-6).

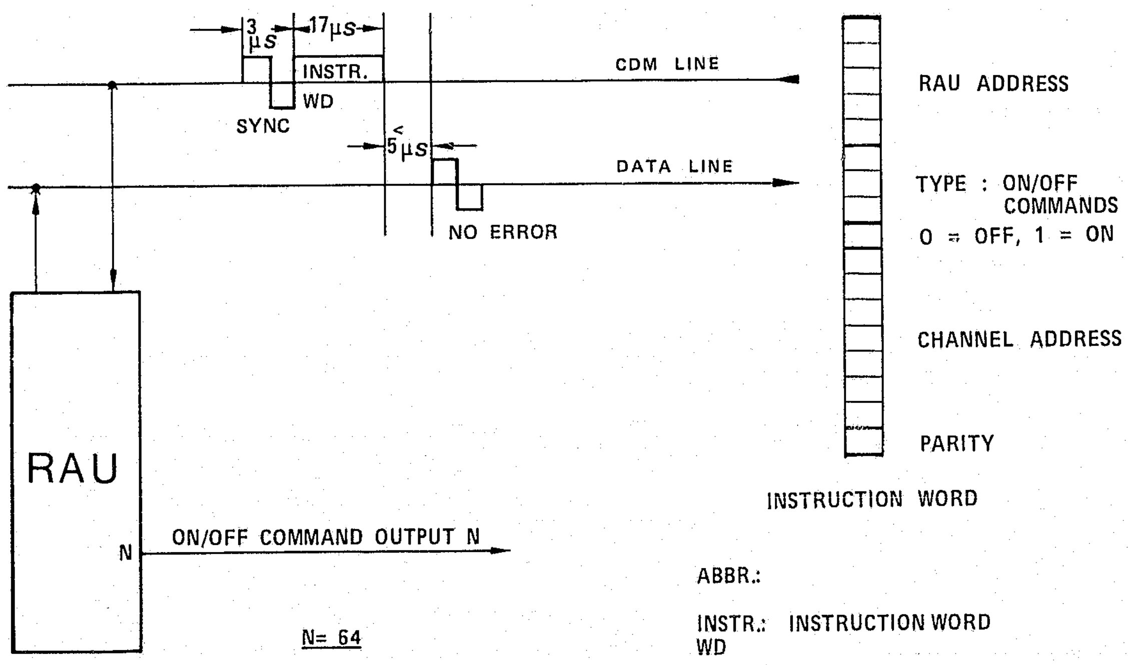


Figure 4.4 - 6: On/Off Command Transfer

The validity of the instruction word is checked as described above. If no errors are detected the RAU will send back a signal to the I/O unit on the data line.

4.4.2.1.4.2 Data Transfer

Serial PCM Inputs

The transfer of serial PCM data from an experiment to the I/O unit will be initiated by a software provided instruction word (see Figure 4.4-7). The structure of this instruction word is sketched in Figure 4.4-7. After receiving this signal the RAU will check the status of the request line of the addressed serial PCM data channel.

If the status of the request line is detected as 'zero' by the RAU this information will be transmitted to the I/O unit via the status word. In this case the computer system will repeat this procedure or stop the transfer as determined by software.

If the status of the request line is detected as 'one', the RAU starts to deliver 17 clock pulses for each word to the experiment as long as the request line level is 'one' and as long as the number of words is not greater than 32. The data shifted out by these clock pulses from the experiment are encoded and transmitted to the I/O unit by the RAU in real time. The RAU completes the transfer by the generation of an End of Message signal.

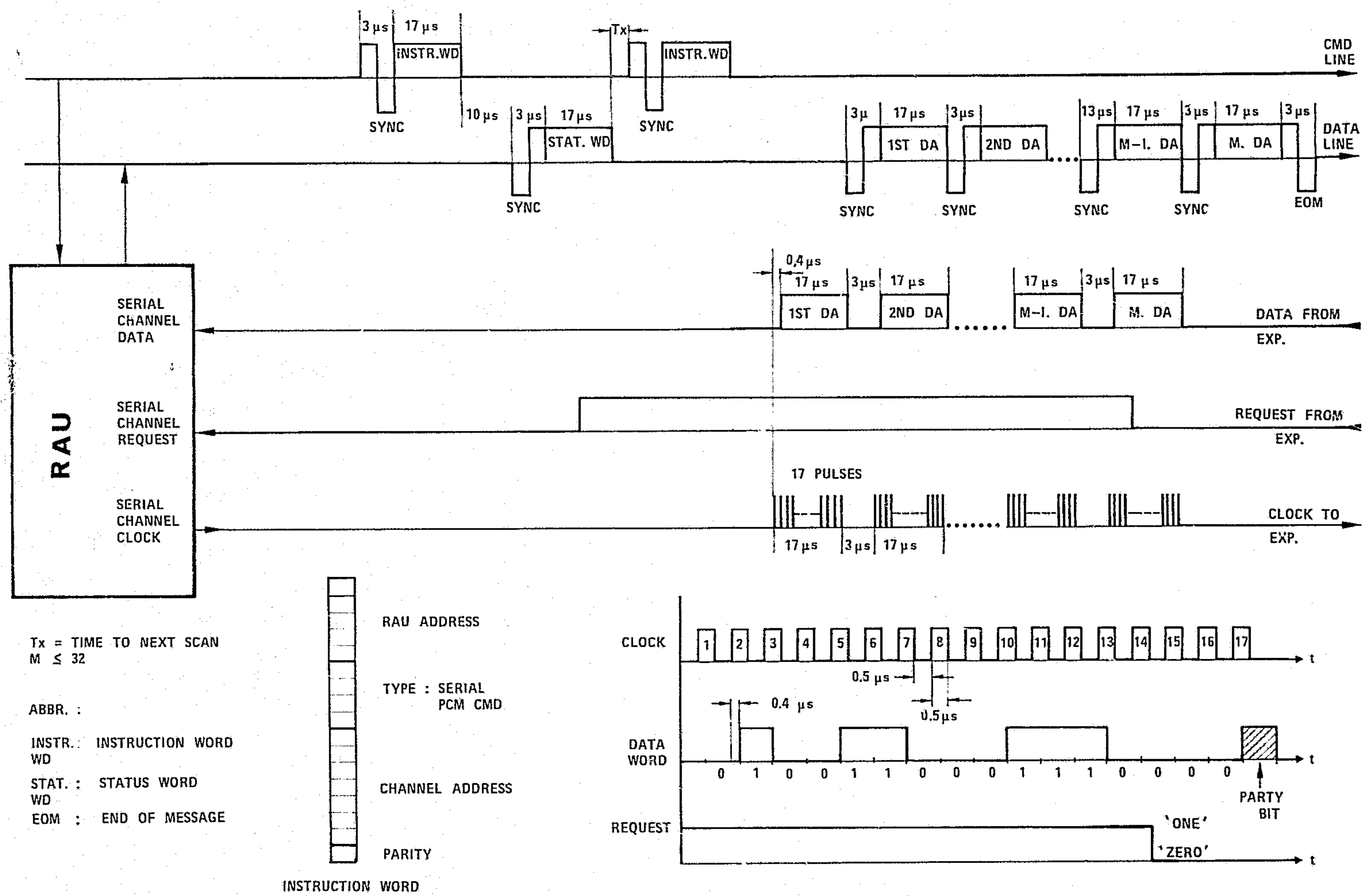


FIGURE 4.4-7 SERIAL PCM DATA TRANSFER

Flexible Inputs

Through the flexible inputs the acquisition of analog signals (analog mode) as well as parallel digital data (discrete mode) is performed.

The acquisition of analog data is depicted in Figure 4.4-8. The instruction word includes RAU address, RAU channel designation, and the information to sample analog signals. In addition the instruction word contains information on the sampling mode (see Figure 4.4-8). Two modes are possible:

- In the single mode only the addressed channel and the next following one will be sampled.
- In the scanning mode groups of 16 input channels ($N \times 16$) are sampled. The number of groups ($1 \leq N \leq 8$) is included in the instruction word.

After receiving the instruction word the RAU initializes the 8 bit analog/digital conversion. The digitized signals of two adjacent channels form a 16 bit word. The RAU adds a parity bit, encodes the word, and starts the transfer to the I/O unit less than 20µs after receiving the instruction word. Since the analog/digital conversion is performed with two fast ADC's operating alternatingly there is no additional time delay for the following data words. The transfer is completed by an End of Message signal.

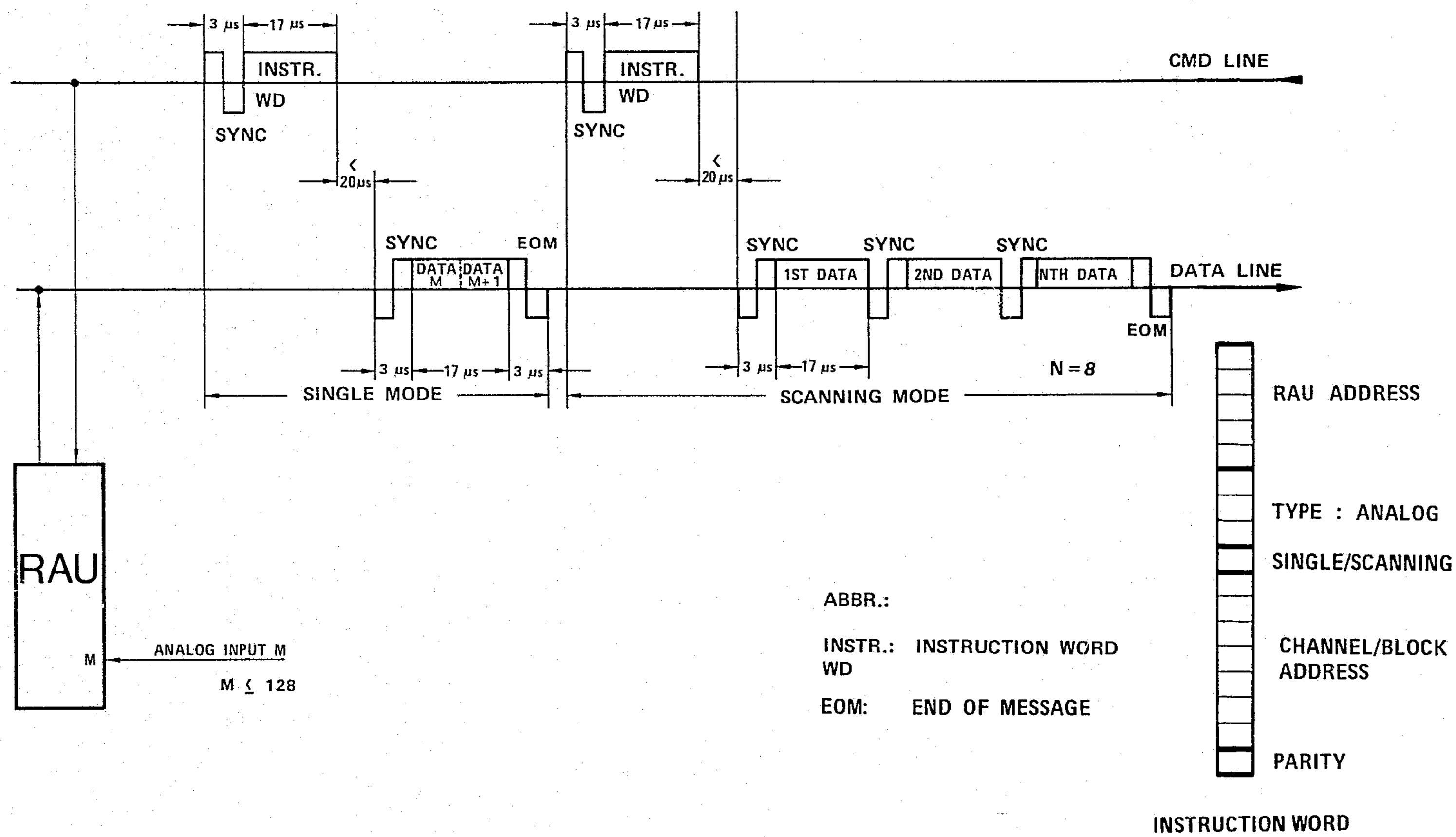


Figure 4.4 – 8: Data Transfer of Analog Signals via Flexible Inputs

The acquisition of discretes is depicted in Figure 4.4-9. The instruction word includes the information to sample groups of 16 discrete inputs. The number of groups (1 N 8) i.e. number of 16 bit parallel digital words, to be sampled is included in the instruction word. The RAU serializes, encodes, and transmits these words sequentially to the I/O unit. The transfer is completed by an End of Message signal.

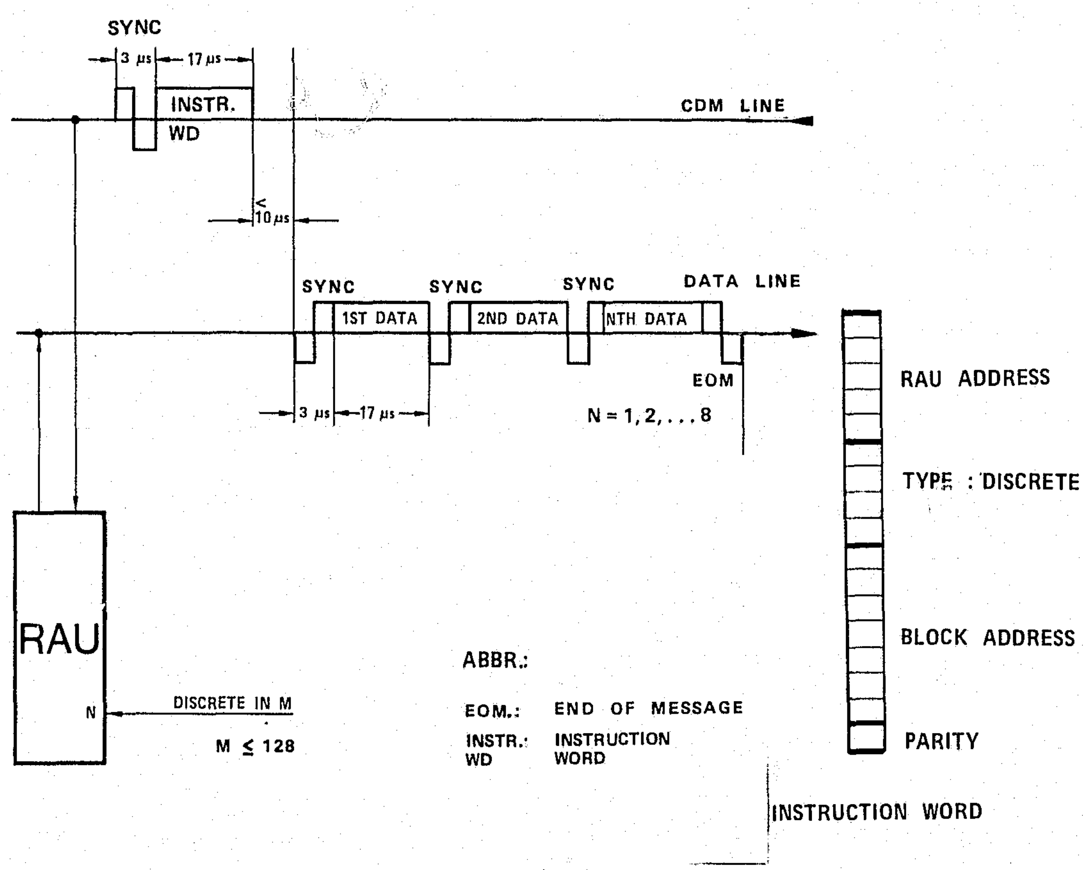


Figure 4.4 - 9: Data Transfer of Discretes via Flexible Inputs

4.4.2.1.5 Timing Information

The Master Time Unit (MTU) in the Orbiter generates and distributes a central "on board GMT". The GMT information can be delivered by software from the experiment computer through the serial PCM command RAU outputs. The experimenter may get time information such as: day, hour, minute and second. Time resolution will be less than 10 ms due to computer - RAU dialogue constraints. The long term drift rate will be 1×10^{-9}/day giving an accuracy better than 3 ms during a 7 day mission. The deviation of the onboard GMT from ground GMT will be logged on ground with an accuracy better than 1 ms. If the deviation is more than $\pm$ 10 ms, the Orbiter MTU will be readjusted externally.

The User Time Clock Module of the RAU delivers to the experimenter a User Time Clock for GMT update and time tagging of experiments.
The 1024 KHz pulses are derived from the master oscillator in the MTU. The characteristics are as follows:

Frequency drift rate Δ f/f:	10^{-9} per day
Rise and fall time:	100 ns
Short time resolution:	50 ns

The 1024 KHz pulses will be overlayed with synchronized 4 Hz update pulses and then transmitted via the User Time Clock bus to the RAU's. After decoding in the RAU the 1024 KHz pulses, with a pulse length of 500 ns, and the 4 Hz pulses, with a pulse length of 10 μ s, are available for experiments.

By utilizing the three time signals,(GMT, clock and clock update)it is expected that data may be tagged to an absolute accuracy (relative to the Orbiter MTU) of 1 millisecond and a relative accuracy (between experiments using the facility) of 10 microseconds. It must be emphasized that these are target figures.

Three time tagging schemes are visualized. Two apply to data acquired by RAU's and transmitted to the computer system, the third to data transferred directly to the wideband data acquisition system (High Rate Digital Recorder, High Rate Multiplexer, Orbiter Wide Band Communication System).

- When data are acquired directly from RAU flexible input channels, time tagging may be performed directly by software in the computer system. In this case the accuracy of the time may be not better than ten milliseconds with respect to the Orbiter MTU reference. If no time tagging is performed by Spacelab Software then it is possible to determine the acquisition time of directly acquired data transmitted to the ground via the Orbiter PCM Master Unit by reference to the PCMMU major and minor frame timing information.

- Data that are acquired within an experiment and transferred to the computer system via RAU serial PCM input channels may be time tagged in a two step process. The time of acquisition in the experiment within a 250 millisecond frame must be determined by experiment hardware based on the 1024 KHz and 4 Hz signals supplied by the RAU. A sixteen bit word indicative of this acquisition time must be interleaved with the data words when they are transferred to the computer system. Software is available in the computer system to calculate the GMT appropriate to the data and assemble the complete time tag accurate to 1 millisecond relative to the MTU reference. Using this technique the relative time tagging accuracy for events occurring in different experiments should be better than 10 µs.

- Data transferred directly to the wideband Data Acquisition System must be time tagged, if necessary, completely within the experiment. This may be achieved by requiring the computer system to output GMT as data words on an RAU serial PCM output channel.

  The RAU 1024 KHz and 4 Hz signals may then be used to interpolate between the GMT updates to obtain the same accuracies as attainable in the previous case.

Besides time tagging of experiment data another application of the GMT and UTC for experiments is the synchronization of commands.

The coincidence of a software generated On/Off command, with a 4 Hz pulse and a 1024 kHz pulse (to be performed with experiment hardware) defines a precise point of time. It enables to switch different experiments synchronized with a relative accuracy better than 1 µs.

4.4.2.1.6 Electrical Interface

This section described the electrical interface characteristics between the RAU and experiments.

4.4.2.1.6.1 Flexible Inputs

4.4.2.1.6.1.1 Analog Signals (Experiment to RAU)

Signal type:	balanced (unbalanced and single-ended signals are also acceptable, but will degrade RAU performance)
Full scale voltage range:	− 5.12 to + 5.08 V
Common mode voltage:	± 1.0 V maximum (± 6.0 V causes degraded performance, but acceptable)
Output impedance:	1 k Ω at DC to 500 Hz
Driving capability:	200 k Ω minimum in parallel with 2500 pF maximum
Fault voltage emission:	± 10 V maximum
Fault current:	± 10 mA maximum

4.4.2.1.6.1.2 Discrete Signals (Experiment to RAU)

Signal type:	single-ended or differential
Digital 'zero' signal:	0 ± 0.5 V
Digital 'one' signal:	+ 5.0 ± 1.0 V
Rise/Fall Time:	1 µs to 2 ms (measured between 10 and 90 percent of the peak signal values on a load of 50 k Ω in parallel with 2500 pF)
Output impedance:	1 k Ω maximum at DC in either logic status
Driving capability:	50 k Ω minimum in parallel with 2500 pF maximum
Fault voltage emission:	± 10 V maximum

4.4.2.1.6.2 Serial PCM DATA Channel

4.4.2.1.6.2.1 Request Line (Experiment to RAU)

Source

Signal type:	single-ended or differential
Digital 'zero' signal:	0 ± 0.5 V
Digital 'one' signal:	+ 5 ± 1.0 V
Rise/Fall Time:	less than 2.0 μs (measured between 10 and 90 percent of the peak signal values on a load of 50 kΩ in parallel with 2500 pF)
Output impedance:	1 kΩ maximum at DC in either logic status
Driving capability:	50 k Ω minimum in parallel with 2500 pF maximum
Fault voltage emission:	± 10 V maximum

Load

Input type:	differential, isolated
Digital 'zero':	+ 2.0 V to - 0.5 V
Digital 'one':	+ 2.5 V to + 6.0 V
Impedance:	500 kΩ minimum line-to-ground shunted with 200 pF maximum
Power-off-impedance	2 k Ω minimum (measured line-to-line with a differential signal of 5 V and a common-mode signal of 5 V)
Fault voltage emission:	± 15 V maximum line-to-ground as a result of any single-point failure
Fault current limitation:	± 15 m A maximum

4.4.2.1.6.2.2 Data Line (Experiment to RAU)

Source

Signal type:	complementary
Digital 'one' signal line-to-ground:	+ 2.5 to + 4.5 V (high state)
Return line-to-ground:	− 0.5 to + 0.5 V (low state)
Digital 'zero' signal line-to-ground:	− 0.5 to + 0.5 V
Return line-to-ground:	+ 2.5 to 4.5 V
Rise and fall time:	less than 150 ns on a load as shown in Figure 4.4−10 (measured between 10 and 90 percent of the peak signal values)
Impedance:	83 Ω maximum
Driving capability:	The experiment output shall be able to drive a load as shown in Figure 4.4−10
Fault voltage emission:	± 15 V maximum, line-to-ground as a result of any single-point failure
Fault current:	± 15 mA maximum
Distortion (overshoot, ringing):	0.25 Vp line-to-ground positive 0.15 Vp line-to-ground negative
Bit rate:	1 Mb/s ± 0.1 percent
Block size:	maximum 32 16 bit + parity bit words

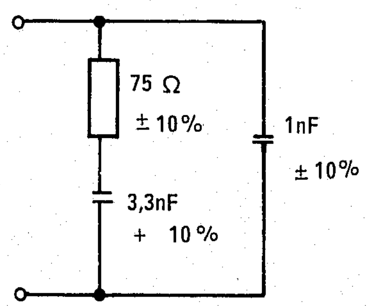


FIGURE 4.4−10 TEST LOAD FOR THE SERIAL CHANNELS

Load

Signal type:	differential
Digital 'one':	+ 2.0 V to + 6.0 V line-to-line
Digital 'zero':	− 2.0 V to − 6.0 V line-to-line
Threshold (line-to-line):	8 ± 0.5 V
Rise/ fall time :	less than 200 ns (measured between 10 and 90 percent of the peak signal values on a load impedance of 75Ω ± 10 percent)
Impedance:	75Ω ± 10 percent
Bit rate:	1 Mb/s ± 0.1 percent
Common mode rejection:	less than ± 10 V DC to 2 MHz line-to-ground

4.4.2.1.6.2.3 Clock Line (RAU to Experiment)

Same source and load characteristics as specified in 4.4.2.1.6.2.2

4.4.2.1.6.3 Serial PCM Command Channel

4.4.2.1.6.3.1 Command Line (RAU to Experiment)

Same source and load characteristics as specified in 4.4.2.1.6.2.2 .

4.4.2.1.6.3.2 Clock Line (RAU to Experiment)

Same source and load characteristics as specified in 4.4.2.1.6.2.2.

4.4.2.1.6.4 ON/OFF Commands

Source

Signal type:	signal-ended
Digital 'zero' signal:	0 ± 0.5 V
Digital 'one' signal:	+ 5 V ± 1.0 V
Rise/fall time:	less than 100 μs (measured between 10 and 90 percent of the peak signal values and on a load of 10 kΩ in parallel with 1000 pF)
Driving capability:	20 mA minimum
Pulse duration:	30 ms minimum
Fault voltage emission:	± 15 V maximum line-to-ground as a result of any single-point failure

Load

The experiment load shall be either an OP Amp or an OPTO Coupler with the following characteristics:

OP Amp

Input type:	Differential isolated
Digital 'zero' state:	+ 2.0 V or less
Digital 'one' state:	+ 2.5 V or greater
Impedance:	50 kΩ minimum line-to-ground shunted with 200 pF maximum
Power-off impedance:	2 kΩ minimum (measured between line-to-line with a differential signal of 5 V and a common mode signal of 5 V)
Fault voltage emission:	+ 10 V maximum line-to-ground
Fault current limitation:	+ 20 mA maximum
Common mode rejection:	$-10\ V \leq V_{CM} \leq +10\ V$ DC to 500 Hz line-to-ground on both signal terminals shall not activate receiver circuits

OPTO Coupler

Input type:	Isolated
Digital 'zero' state:	+ 1.5 V or less
Digital 'one' state:	+ 3.0 V or greater
Input current:	15 mA maximum at 5 V
Input capacity:	200 pF maximum line-to-line
Power-off impedance:	same as power on
Fault voltage emission:	± 10 V maximum line-to-ground as result of any single point failure
Fault current limitation:	± 20 mA maximum

4.4.2.1.6.5 User Time Clock Lines (RAU to Experiment)

Same source and load characteristic as specified in 4.4.2.1.6.2.2.

Clock

Frequency:	1024 kHz

Clock update

Frequency:	4 Hz
Pulse width:	10 μs ± 1 μs

4.4.2.2 Input/Output Unit

All communication between the computers and the rest of CDMS are handled by the Input/Output Units which control the transfer of external data into the computer memory and the transfer of data from the memory to all peripherals. A simplified block diagram of the I/O Unit is shown in Figure 4.4-11.

The I/O unit has seven interfaces with the rest of the CDMS and the Orbiter. These are:

Remote Acquisition Units	CDMS
Data Display Units and Keyboards	CDMS
Mass Memory	CDMS
High Rate Multiplexer (under discussion)	CDMS
MDM	Orbiter
PCM Master Unit	Orbiter
Master Time Unit (via RAAB)	Orbiter

Each interface is controlled by a 'coupler' which is attached to the non-redundant internal parallel bus of the I/O unit. Each coupler, except the 'time coupler' is dual redundant and communicates to the rest of the CDMS or Orbiter as appropriate via serial data buses. Only one coupler of a redundant pair is powered at any time.

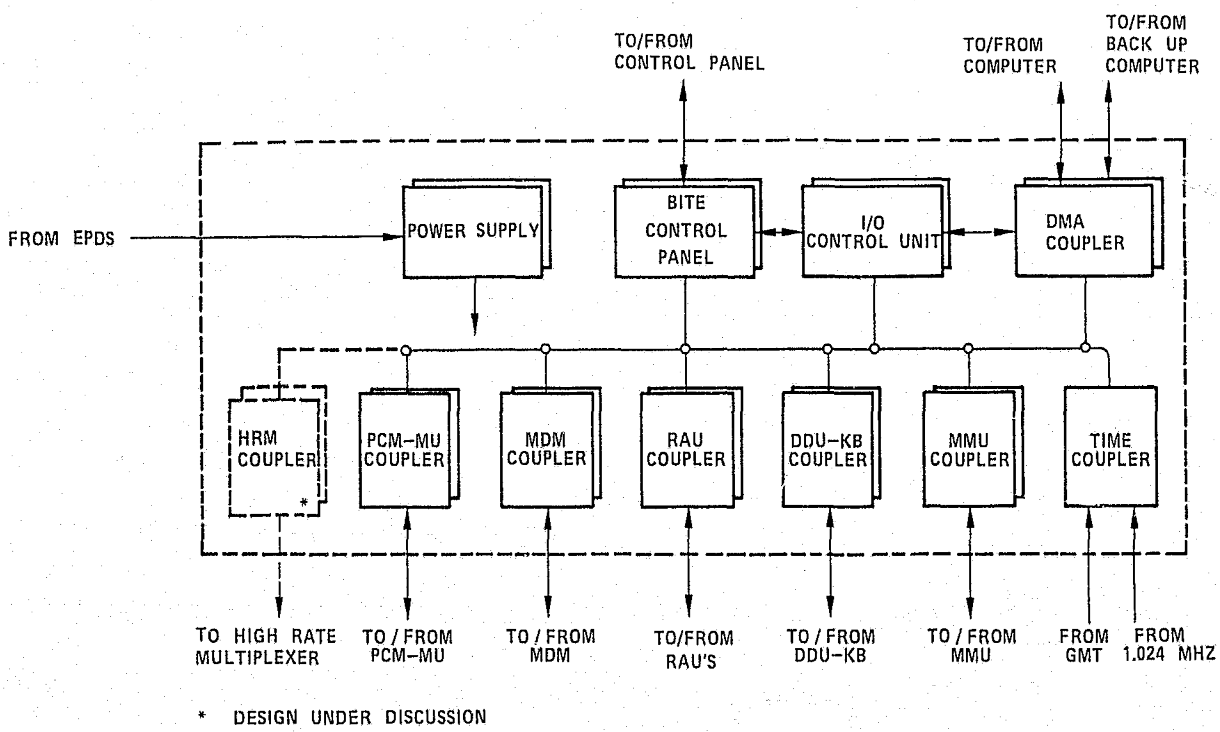


Figure 4.4 - 11 Simplified Block Diagram of the I/O Unit

The interface between the I/O unit and the prime (and back-up) computer is performed by the redundant Direct Memory Access (DMA) coupler. This coupler receives and generates control discretes from and to the computer memory interface and receives and transmits address and data over a 16 bit parallel memory bus. Only one DMA coupler is powered at any time, corresponding to the prime or back-up computer being powered.

Each peripheral coupler incorporates a micro-processor to supervise the transfer of data to or from the computer memory. It is capable of performing simple tests to ensure the validity of the data such as parity checks, word count and time out.

A coupler in the I/O unit is initialized by the transfer of two words (Status Table) from the computer memory. It then uses these words to point to an instruction list in the computer memory consisting of a number of word triplets (Command Table), each one defining one transaction for that coupler. It executes these to transfer data into or out of a data table to perform its interface function. Once initiated, this activity can proceed in parallel with the Central Processor Unit (CPU) use of the memory, although only one access to the memory can be accommodated at any instant. Because of the serial data transfer through the couplers and parallel data transfer with the memory, up to five couplers can effectively operate simultaneously.

The I/O unit has priority over the CPU memory access. If more than one coupler is queued for memory access then memory data are transferred in multiple word blocks which is more efficient (in time) than single word transfer.

Coupler access to the computer memory is controlled on a priority basis by the I/O Control Unit. The PCM Master Unit has been assigned the highest priority because of response time constraints imposed by the PCM Master Unit. At present the order of the rest of the coupler priorities has not been established.

4.4.2.3 High Rate Multiplexer

Editorial note: The High Rate Multiplexer (HRM) is still in the technical definition phase and is not yet included into the Spacelab baseline. All information and data given here are still preliminary.

The purpose of the High Rate Multiplexer is

- to multiplex serial data from differential sources with different data rates
- to transfer the multiplexed data stream to the Orbiter Ku-Band processor for the transmission to ground during TDRSS coverage periods
- to transfer the multiplexed data stream to the High Rate Digital Recorder (HRDR) for storage during TDRSS non-coverage periods
- to interleave the stored data during playback of the HRDR into the real time data stream.

The multiplexer system includes a real time demultiplexing at the TDRSS ground station. The composite data stream is split into the original experiment channels. The HRDR playback output channel, containing multiplexed data, needs subsequent second stage demultiplexing (Figure 4.4-12).

The multiplexer system will be completely transparant, i.e. the experiment data of each channel will be available on ground as they have been submitted on-board of Spacelab.

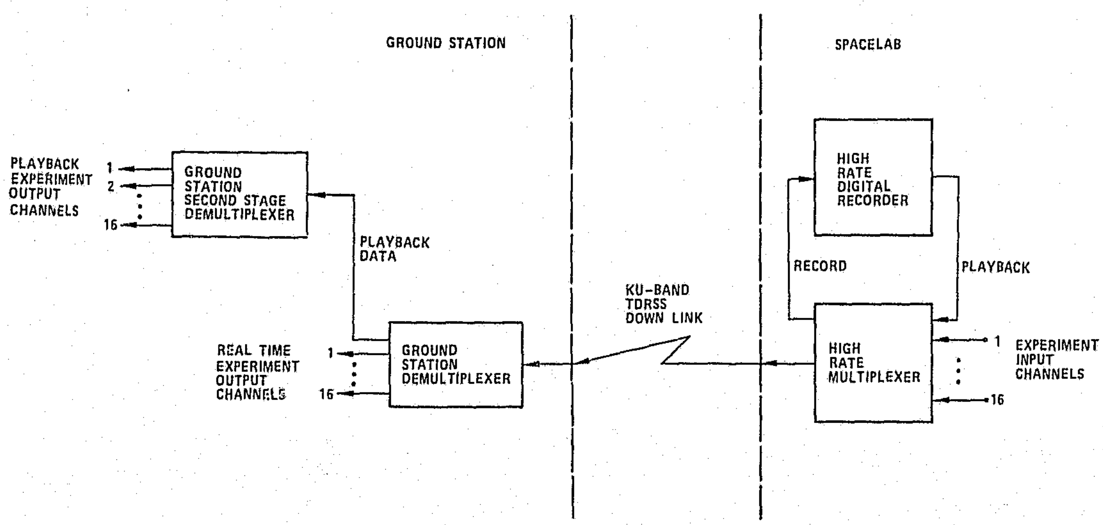


Figure 4.4-12: Logic Diagram of Multiplexer/Demultiplexer

The HRM will be programmable with respect to input channel allocation, input bit rates and output bit rates. The programmed modes will be computer controlled via RAU.

The HRM provides 16 experiment input channels with allocated bit rates variable in binary ratios. Four (4) different types of experiment input channels with different dynamic range are defined:

- 2 inputs from 16 Mb/s to 62.5 kb/s
- 2 inputs from 8 Mb/s to 62.5 kb/s
- 4 inputs from 4 Mb/s to 62.5 kb/s
- 8 inputs from 2 Mb/s to 62.5 kb/s

The code for the input data is NRZ-L plus clock.

In case of synchronous input data a clock frequency will be available to the experiment input channel which corresponds to the allocated bit rate.

In addition there are five dedicated input channels:

- 2 input channel for subsystem and experiment I/O unit outputs to HRM
- 1 input channel for HRDR playback data (max. 32 Mb/s)
- 1 input channel for direct access to the HRDR only (max. 32 Mb/s) or to the HRDR and the Ku-Band Signal Processor in parallel
- 1 input channel for direct access to the Ku-Band Signal Processor only (max. 50 Mb/s) or to the Ku-Band Signal Processor and the HRDR in parallel.

The possibility of interleaving the output data stream with GMT and flight number information is under discussion. There is no special voice annotation facility planned but the user may route his digital voice annotation to anyone of the 16 experiment input channels.

The output rate of the HRM is independently variable in binary ratios. The output rate will be selected in accordance with the mission phase dependent down link capability and in accordance with optimized bandwidth margins improving the bit error rate of the down link. The CDMS computer controlling the HRM/HRDR system receives information concerning the Orbiter selected Ku-Band mode via the Orbiter MDM channel.

The output rate imposes constraints to input channel allocation and input bit rates. The sum of the input bitrates must not be greater than the output range.

Table 4.4-2 lists the discrete input and output bit rates:

Table 4.4-2: HRM Input and Output Bit Rates

Input Bit Rate Mb/s	
32.0	HRDR
16.0	2 exp. channels
8.0	2 exp. channels
4.0	4 exp. channels
2.0	8 exp. channels
1.0	
0.5	
0.25	
0.125	
0.0625	

Output Bit Rate Mb/s
48.0
32.0
16.0
8.0
4.0
2.0
1.0

As described above the HRDR is integrated into the Multiplexer System to record data during TDRSS non-coverage periods. In addition a recorder provided by the Orbiter (Orbiter Payload Recorder) can interface with the HRM in the same manner, acting as a back up for data storage with a degraded recording performance of 1 Mb/s. Figure 4.4-13 depicts the HRM/recorder assembly.

It is under discussion to have a multiplexer system operating in a synchronous mode or a system operating in an asynchronous mode. In addition it is under discussion to deliver experiment data into the HRM as a continuous bit stream or as bit bursts, each burst forming a 16 bit word. If the input data are delivered in a continuous bit stream an asynchronous system avoids synchronizing problems. There are minor synchronization problems if the input data are delivered in bursts. In this case synchronization could be performed by a sync pulse, delivered by the multiplexer for each word.

The HRM will be located in the Control Center Rack in the Spacelab Module or in the Igloo in the pallet-only mode. The cabling from the HRM to the experiments has to be provided by the user.

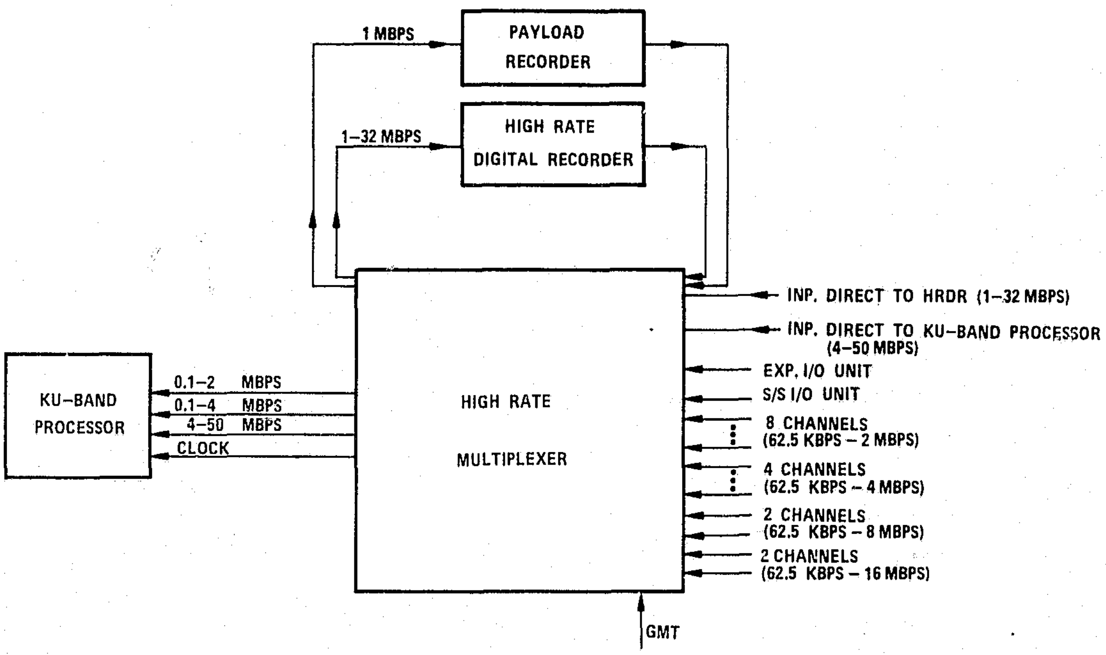


Figure 4.4-13: High Rate Multiplexer Input/Output Schematic

4.4.2.4 High Rate Digital Recorder

The principal function of the High Rate Digital Recorder (HRDR) is to provide for intermediate recording of experiment data during interrupted Orbiter to ground TDRSS transmission times. Beside this, the experimenter may record his experiment or housekeeping data for on-board storage.

The HRDR and the HRM will form an integrated system. Both are controlled by a CDMS computer in a coordinated manner. Only multiplexer outputs will be fed into the High Rate Digital Recorder. During record and playback the HRDR will be externally synchronized by the HRM clock.

The HRM provides an output channel to the HRDR, containing multiplexed data from more than one HRM input channel ($\leq$16 Mb/s each, up to a total of 32 Mb/s). Alternatively a dedicated HRM input may be switched directly to the HRDR. However, the recorder will accept only one of these input channels at a time.

The HRDR will be used as a buffer during TDRSS non-coverage times or Ku-Band modes with bit rates below the HRM output bit rate. During playback the recorded data are interleaved into the real time data stream through a recorder dedicated input channel of the HRM.

Data recording for on-board storage without transmission to the ground is only possible during periods when non-buffer capacity for transmission gap times is required. In this case the tape change capability of the HRDR may be useful for the experimenter.

In any case the function of the HRDR will be determined by the Orbiter transmission mode (see Section 4.4.3).

As back-up for the HRDR the Orbiter provided Payload Recorder can be used (see Figure 4.4-13). This recorder will have a storage capacity of 3.44×10^9 bits at a maximum input bit rate of 1024 Kb/s.

The HRDR characteristics are given in Table 4.4-3. The power consumption as indicated in Table 4.4-4 depends strongly on the actual operating mode.

It is under discussion to locate the HRDR in the Orbiter AFD, to make use of the tape change capability in all Spacelab configuration.

Table 4.4-3: High Rate Digital Recorder Characteristics

Record Technique	longitudinal, 28 tracks
Data Tracks	26
Data Storage	3.6×10^{10} bits
Bit Density/Track	12.5 Kb/inch
Data Rate Input	1,2,4,8,16,32 Mb/s
Data Rate Output	2,4,8,16,32 Mb/s
Total Record Time	600, 300, 150, 75, 37.5, 18.75 min
Data Type	Serial in, Serial out, NRZ-L + clock
Bit Error Rate	1×10^{-6}
Start Time	5 s
Stop Time	2.5 s
Tape Handling	tape cartridge, automatic threading
Tape Loading Time	0.4 min.
Rewind Time	4.6 min.
Tape Width/Reel Diameter	1" / 14"
Tape Reel with Tape and Cartridge	4.8 kg

Table 4.4-4: Power Consumption of the HRDR

		DC Power
Record	30 Mb/s	70 W
	6 Mb/s	55 W
	1 Mb/s	53 W
Playback	30 Mb/s	154 W
	2 Mb/s	138 W
Wind / Rewind		93 W
Standby		19 W

4.4.2.5 Closed Circuit TV System

As an extension of the Orbiter CCTV system, Spacelab provides accommodation for experimenter (or Obiter) provided TV monitor and the electrical interface to operate experiment provided TV cameras.

The experiment video signals should have a voltage range of $\pm$ 1.0 V $\pm$ 0.1 V and have to be compatible with EIA standards RS 170 and RS 330.

There will be:

- one input for experiment provided video signals
- one output for synchronization of experiment TV cameras
- one output for video signals used in the Spacelab located TV monitor

Each input or output will be a coaxial connector, located at the Control Center Rack.

4.4.2.6 4.2 MHz Analog Channel

The Spacelab provides a 3 Hz - 4.2 MHz analog channel for experiments. This channel can be used to for special non-EIA standard TV signals, if the Orbiter CCTV system is not able to accept these signals.

The voltage range of this analog input will be $\pm$ 1 V.

It is still under discussion to establish a standard of quality for the analog channel. First analysis shows that the following signal-to-noise ratios at the Spacelab/experiment interface will generally provide acceptable data quality on ground:

- a signal-to-noise ratio of 30 db will provide acceptable black and white TV.
- a signal-to-noise ratio of 40 db for analog data will be acceptable for most applications.
- a signal-to-noise ratio of 20 db (predetected) for fm - fm subcarrier oscillator data will be acceptable. This will yield a signal-to-noise ratio of better than 40 db after detection.

The analog channel will be a 75 Ω coaxial cable routed directly to the Orbiter Ku-Band Signal Processor. The connector is located at the forward end cone feedthrough plate.

4.4.3 Data Transmission

4.4.3.1 Network System

Figure 4.4-14 shows the possible transmission links to the ground. Two down link facilities are available to Spacelab:

- the Space Tracking and Data Network (STDN) linking the Orbiter directly to various ground stations via S-Band,

and

- the Tracking and Data Relay Satellite System (TDRSS), which has two relay satellites and one ground station. The TDRS link to the ground station is performed by Ku-Band. The TDRS/Orbiter link normally uses the Ku-Band while the S-Band is operated only during first antenna adjustment procedures.

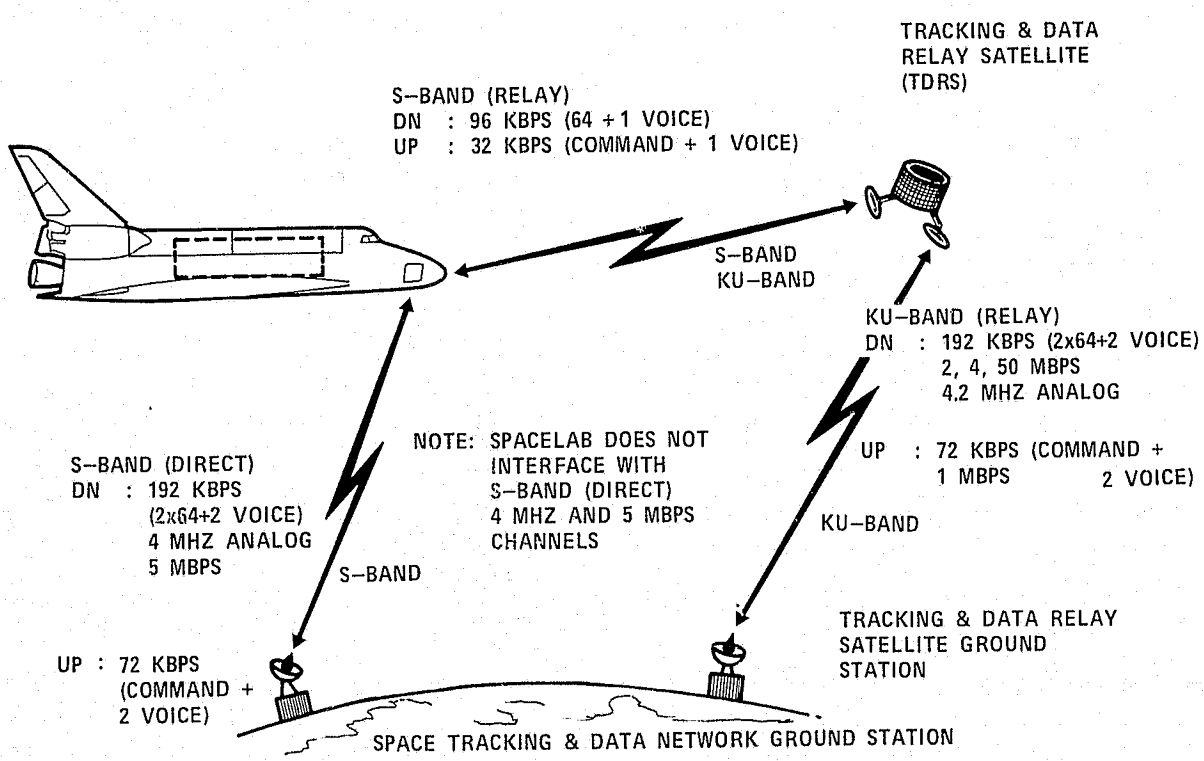


Figure 4.4-14 Orbital Communication Links

The present NASA STDN ground stations available for direct down links are shown in Figure 4.4-15. The coverage using these stations depends on the orbit and is, in any case, less than 30 %. With TDRSS operational the number of STDN ground stations will be reduced to those five stations underlined in Figure 4.4-15. This would decrease the nominal STDN coverage to approximately 5 %. It should be noted that Spacelab interfaces with the STDN link only for transmission of housekeeping data and low speed scientific data from the experiment computer.

The TDRSS consists of two geo-stationary satellites at 44^{o} W and 171^{o} W and one ground station located at White Sands, New Mexico. This configuration has the advantage of routing all data and commands to and from the Orbiter through one single ground station. However, this configuration cannot provide a 100 % coverage.

The dotted area in the lower part of Figure 4.4-15 shows Orbiter positions where no transmission via the TDRSS is possible.

For Orbiter altitudes of 200 km and 1000 km the hatched and cross hatched areas respectively in the upper part of Figure 4.4-15 indicate regions on the earth surface, which cannot be observed with direct TDRSS link.

The nominal TDRSS coverage using the Orbiter antenna and an additional payload chargeable antenna kit is approximately 85 % over 24 hours. This does not exclude that for some orbits the coverage per orbit may be about 50 %.

The exact coverage will depend on the actual mission profile and computer programs are developed to perform the necessary analysis of the various factors affecting coverage, such as

- TDRSS gaps over the Indian Ocean
- Ku-Band antenna masking by Orbiter, Spacelab and payload (beam blockage)
- Flux density limitations (transmission constraint)
- Ku-Band acquisition dependence on S-Band link
- RF transmission factors - bit error probability and data rate vs power gain/loss factors (quality)
- Solar interference - TDRSS data reception interruption

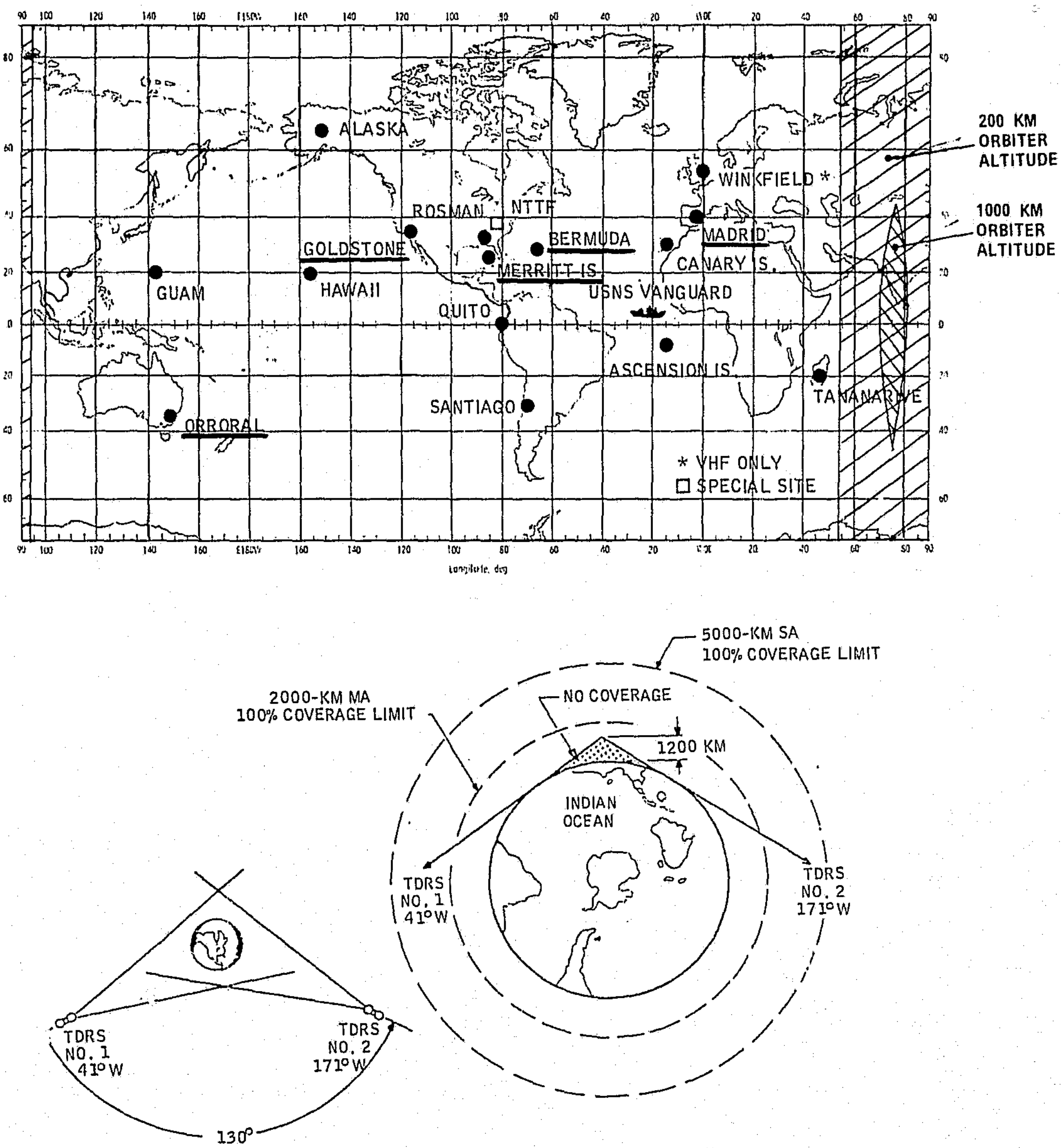


Figure 4.4 - 15: STDN and TDRSS Coverage

4.1.3.2 Down Link

The transmission of data generated by Spacelab or Spacelab payload is performed by the Orbiter Avionics (see Figure 4.4-16. There are two different types of Spacelab data treated by the Orbiter Avionics in different ways.

- Housekeeping and Low Rate Scientific Data

For Spacelab Housekeeping and Low Rate Scientific Data, routed through the subsystem and experiment I/O units the 192 kb/s telemetry channel, interleaved with Orbiter data, is available. This 192 kb/s data stream is split up into:

- two voice channels, 32 kb/s each
- Orbiter telemetry data, 64 kb/s nominal
- Spacelab data from experiment and subsystem I/O unit outputs, 64 kb/s nominal

The composition of the data in this 192 kb/s telemetry channel is software controlled through the PCM Master Unit. The PCM Master Unit acquires the data from different sources (Orbiter GPC, subsystem I/O, and experiment I/O) in a demand and response manner. As the Orbiter telemetry data will not need 64 kb/s all the time, it might be possible that experiment and subsystem data can be transmitted at more than 64 kb/s via this telemetry channel.

Controlled by the Network Signal Processor from the PCM Master Unit the 192 kb/s telemetry channel is transmitted to ground either via STDN to the appropriate STDN ground station orvia TDRSS to the TDRSS ground station. From the TDRSS ground station in White Sands, New Mexico, the 192 kb/s telemetry data are sent to the Payload Operation Control Center (POCC) in Houston via ground lines.

To bridge TDRSS non-coverage periods the 192 kb/s telemetry data are buffered on the Maintenance/ Loop recorder in the Orbiter.

- Wide Band Scientific Data

The term Wide Band Scientific Data covers the digital data from the HRM output, CCTV signals, and the analog data of the 4.2 MHz channel. This Wide Band Scientific Data is transmitted to ground only via the Ku-Band of the TDRSS. For the digital data TDRSS non-coverage periods are bridged by the Spacelab HRDR and the Orbiter Payload Recorder (see Section 4.4.2.3). Means to bridge the transmission of CCTV and analog signals are not provided.

The Orbiter controlled mode selection and channel allocation of the Ku-Band down link is performed by the Ku-Band signal processor.

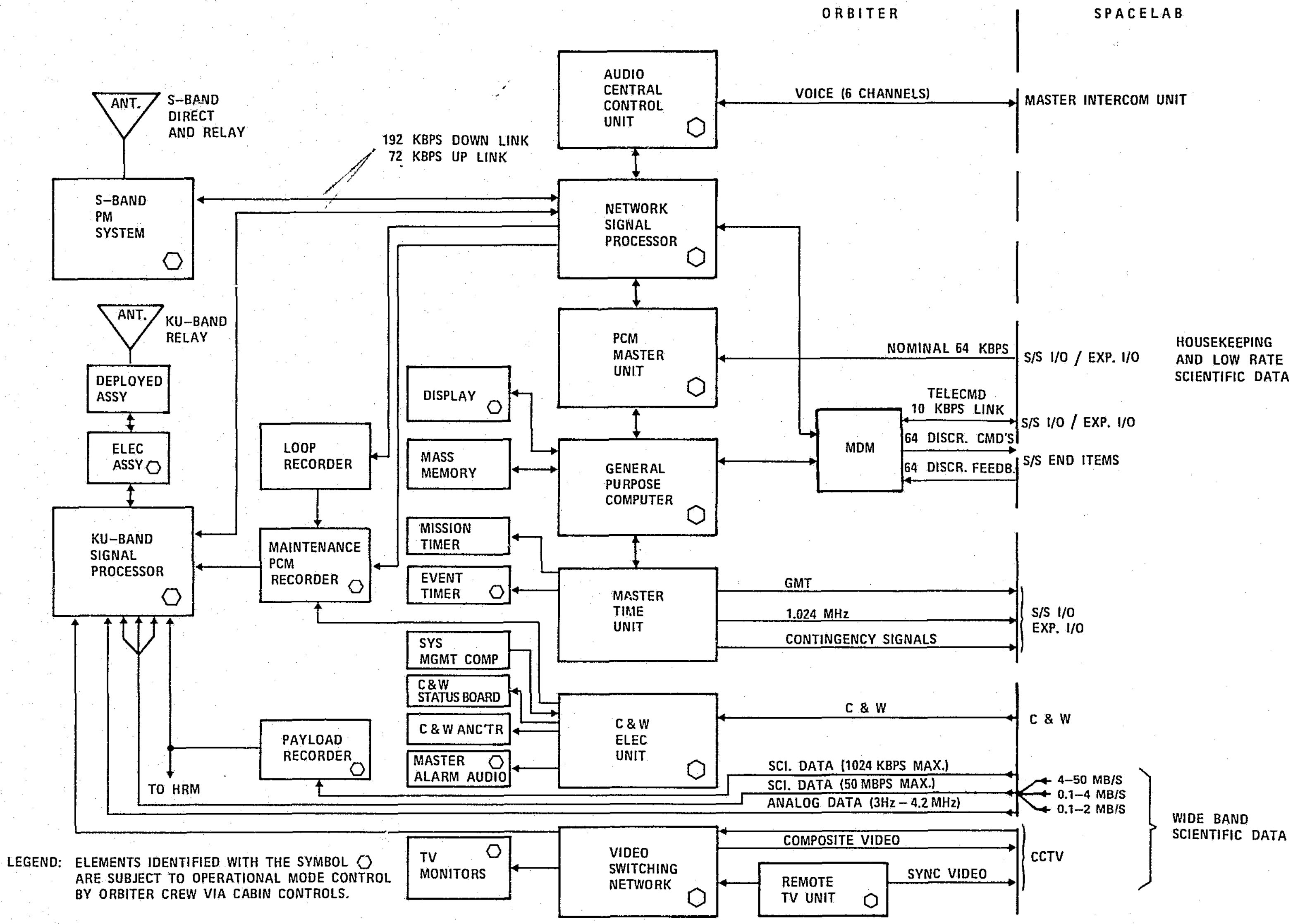


Figure 4.4 – 16 Orbiter Avionics Functional Diagram for Payloads

The functional flow chart in Figure 4.4-17 indicates the switching capabilities to combine the various inputs to the Ku-Band signal processor.

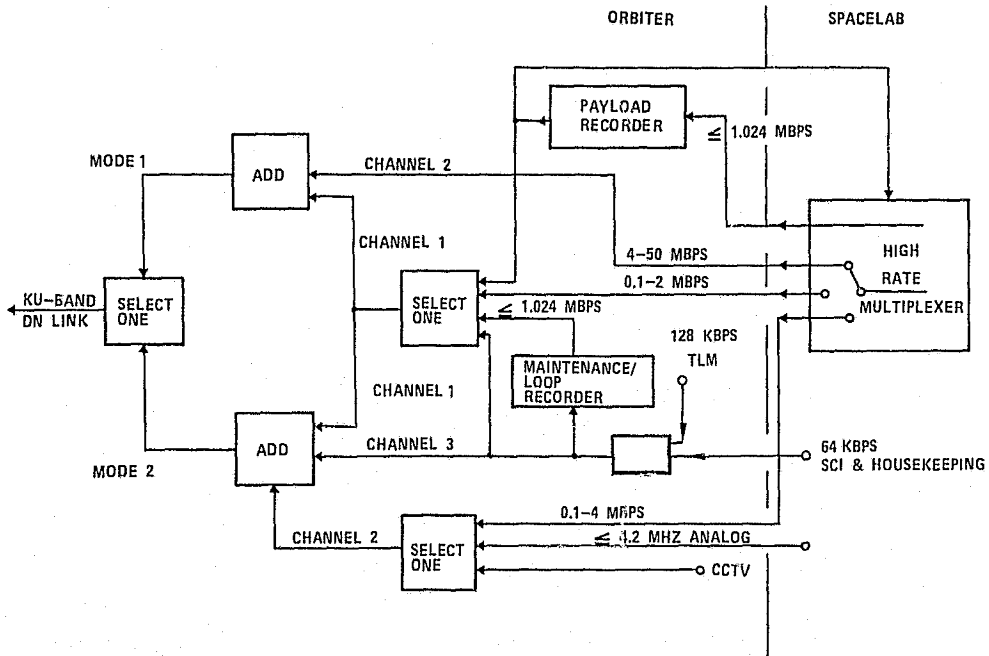


Figure 4.4-17: Functional Ku-Band Data Processing

The channels available in the two Ku-Band modes are summarized in Table 4.4-5.

Mode 1 is a phase modulated transmission line providing a 4-50 Mb/s channel interfacing with the HRM output together with another 0.1-2 Mb/s channel operating in parallel. This 0.1-2 Mb/s channel normally will be occupied with the 192 kb/s telemetry data.

Mode 2 is a frequency modulation transmission line providing one 192 kb/s channel for telemetry data, one 0.1-2 Mb/s channel interfacing with the HRM, or the Payload Recorder and one channel accepting either digital or analog signals. The digital data (0.1-4 Mb/s) are delivered from the HRM output. The analog data (3 Hz - 4.2 MHz) are delivered from the CCTV or from the 4.2 MHz analog channel directly.

Table 4.4-5: Ku-Band Mode Description (Orbiter to TDRSS)

MODE	CHANNEL		
	1	2	3
1 (PM)	Digital: 0.1 – 2 Mb/s	Digital: 4 – 50 Mb/s	Not available
2 (FM)	Digital: 0.1 – 2 Mb/s	Digital: 0.1 – 4 Mb/s or Analog: CCTV or 4.2 MHz Channel	Digital: 192 Kb/s

The further transmission and processing of Wide Band Scientific Data after receiving at the TDRSS ground station at White Sands, New Mexico, is still under discussion.

4.4.3.3 Uplink

The Orbiter Avionics System provides an uplink channel of 72 kb/s for command and voice via the STDN S-Band and via TDRSS Ku-Band. The TDRSS Ku-Band uplink in addition provides a 1 Mb/s uplink channel that, however, does not interface with Spacelab CDMS. There is no TV uplink to the Orbiter or Spacelab. The 72 kb/s uplink line is split up into two voice channels with 32 kb/s data rate each and one command channel allowing the user to transmit a 2 kb/s information rate. The remaining 6 kb/s are filled up with 1.6 kb/s sync overhead and 4.4 kb/s coding information.

Primarily these 2 kb/s tele-commands are routed to the Orbiter GPC. From there they are sent to the Spacelab subsystem or experiment computer via the 10 kb/s link, Orbiter MDM and Spacelab I/O units.

All commands sent to Orbiter or Spacelab have to be generated at the Payload Operation Control Center (Houston). That means that the user has to deliver his experiment commands and uplink data at Houston where the complete telemetry frames will be generated and transmitted to the Orbiter.

4.4.3.4 Detached Payload Link

For detached payloads (e.g. subsatellites) a dedicated S-Band link is provided by the Orbiter avionics.

The command link (Orbiter to detached payload) provides an information rate of 2 kb/s (8 kb/s bit rate). The data link (detached payload to Orbiter) is capable of accommodating a 16 kb/s data rate.

4.4.4 Data Processing

4.4.4.1 Computer

The CDMS has three identical MITRA 125 S general purpose computers with characteristics as shown in Table 4.4-6.

These computers have the inherent potential of an interrupt capability. However, it is still under discussion how to mechanize this interface to experiments. Eight hardware interrupts are wired from the computer to the I/O unit. Only four of these are presently required to support I/O unit activities. It may be possible to provide experiment access to the remaining four interrupts to enhance experiment use of the CDMS. The basic software of CDMS is non-synchronous and can be adapted to handle these interrupts.

The three computers are used as S/S Computer, Experiment Computer and Back-up Computer. S/S and Experiment Computers are connected to the dedicated CDMS equipment each via their own I/O unit, Data Bus and RAU's. There is no direct link between each computer.

The third computer is available as a back-up either for the S/S or the Experiment Computer and can be switched over manually.

Due to the concept of routing all S/S and Experiment peripherals through dedicated I/O units this switching connects the Back-up Computer to the appropriate I/O unit and all peripherals.

Normally the Back-up Computer is loaded with subsystem software (operating system and application software) since a S/S Computer failure is more critical with respect to the overall performance of Spacelab. However, in case of Experiment Computer failure the experiment software may be loaded from the Mass Memory Unit (MMU) by an operator command (see Section 4.4.4.2).

Table 4.4 – 6 : Computer Characteristics

Formats

Operands: 8, 16, 32 and 24 + 8 (floating points) bits

Instructions: 16 bits

Control Unit

Micro-programmed control unit

Cycle time 300 ns

Micro-interrupt capability

Micro-instructions 4 K words of 16 or 20 bits

Instruction Set

- Number of instructions 128
- Format 16 bits
  Immediate 8 bits
- Addressing capability
  Direct 256 Bytes
  Indirect memory double word
  Relative 512 bytes
  Based 256 bytes
  Indexed 64 K bytes
- Type
  Call and store
  Logic and comparison operations
  Shift operations
  Fixed-to-floating and floating-to-fixed conversions
  Conditional and unconditional jumps

Addressing Modes

Immediate, direct, indirect, relative to a base, indexed, relative to a program counter, half word, word, character, double word

- Addressing capability
  Byte, word, double word

Number of Adressable Registers

4 Specialized registers

62 Dedicated registers

7 Base registers

Computing Speed

Fixed Point 16 Bits

Add/Sub	Direct	2	µs
	Indirect	3	µs
Mul/Div	Direct	4	µs
	Indirect	5	µs

Fixed Point 32 Bits

Add/Sub	Direct	5.5	µs
	Indirect	6.5	µs
Mul/Div	Direct	8.3	µs
	Indirect	9.3	µs

Floating Point 32 Bits (24 + 8)

Add/Sub	Direct	5	µs
	Indirect	6	µs
Mul/Div	Direct	6	µs
	Indirect	7	µs

Gibson Mix $3.5 * 10^5$ Operations/Second

Input/Output

- Interrupts

– Number of external	8 Levels
– Number of internal	5 Levels
– Number of software	Program dependent
– Interrupt control	Microprogram + Software
– Priority scheduler	Software

- Data transfer mode

– Program controlled	
data rate	60 µs /word
no of addressable periferals	65 k
– Direct memory access	
data rate	400 to 800 K word/sec
control	direct

• Word length	16 bits plus 1 parity + 1 protection
• Discretes	8 inputs and 8 outputs
• Real time work	1 µs to 2^{32} msi

Memory

- Type: 18 mil ferrits cores, 2 1/2 D configuration
- Capacity: 64 K 16-bit words (plus 1 parity bit and 1 protection bit) extendible to 512 K 16-bit words
- Modularity: 16 K words
- Cycle time: 920 ns
- Addressing, Quantum: Byte, word
- Access time: 420 ns
- Ports : 2

In the Module or Module/Pallet mode the computers are located in the Work Bench Rack. The location in the Pallet-only mode is the Igloo.

The computer facilities allow general purpose processing by user provided software written in HAL/S (see Section 4.5) such as:

- Checkout of Experiments
- Sequencing of Experiment Operations
- Monitoring and Control of Experiments
- Processing of DATA Acquired by Experiment RAU's

Examples of Data Processing are:

- Filtering
- Data Reduction
- Wave Form Analyses
- Histograms
- Averaging
- Interpolation, etc.

The processed data may be delivered back to experiments, displayed on-board or transmitted to ground, depending on the mission requirements.

For experiment sequencing the user may provide several program packages for each experiment stored in the MMU. Depending on actual experiment results or data and information from ground the experimenter may stop or change via keyboard entries a running sequence of operation steps or initialize a new program to be executed in the Experiment Computer.

4.4.4.2 Mass Memory

The mass memory unit (MMU) is a tape recorder for storage of all basic and flight application software for the S/S- and the Experiment Computer, thus enabling the CDMS to reload the computer memories from the MMU. Besides this the MMU will be able to accommodate experiment data for comparison purposes or usage within experimenter provided programs. The MMU will also provide the capability to overlay S/S or experiment programs exceeding the computer core memory size. Although there may exist a limited possibility for on-board writing into the MMU the use of this feature is not encouraged since problems will arise due to hardware reliability degradation and data protection. Table 4.4-7 shows the characteristics of the MMU.

Table 4.4-7: Mass Memory Unit Characteristics

Total Storage Capability	1.31×10^8 bits
Organization:	
Files	8
Sub-files	64
Blocks	2048 of 512 words
Transfer rate	1 Mb/s
Access time	2 s average within any file
Start time	0.5 s to the first data block
Bit error rate	less than 1 in 10^8 bits

4.4.4.3 Data Display Unit and Keyboard

The operator/computer interface is performed via a data display unit (DDU) and an associated keyboard. There are two DDU/keyboard units in the Module and one in the Orbiter AFD, all connected both to the subsystem I/O unit and the experiment I/O unit. Thus by manual switching each DDU/keyboard unit can communicate with the subsystem or the experiment computer.

Each keyboard station has the ability to call and display either subsystem or experiment information on any of the three DDU's.

The keyboard functions are shown in Figure 4.4-18. The alpha-numeric keys cover the full ASCII capability. In addition, 25 function keys are provided. The actual functions of these keys can be chosen according to the user requirements.

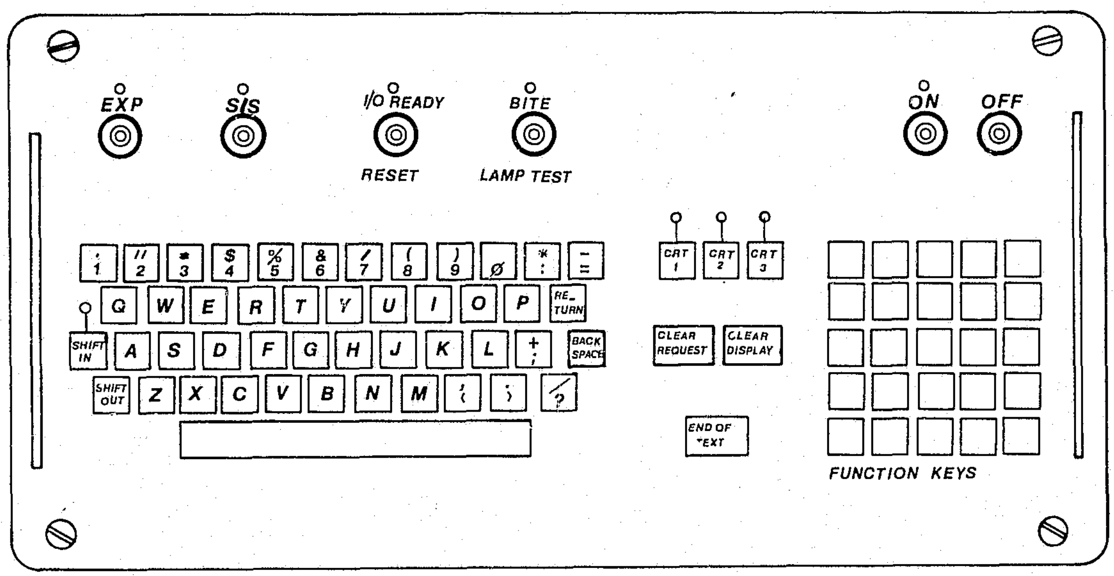


Figure 4.4 – 18 : Spacelab CDMS Keyboard

The DDU´s are tricolor (red, green, yellow) CRT´s with a 12 inch diagonal screen. The DDU´s are capable of displaying the following types of information:

- alpha-numeric formats
- vectors
- graphics
- formatted displays

Alpha-numeric information can be displayed in up to 22 lines with a maximum of 47 characters per line. Each character is presented by a 4 x 6 point matrix.

For graphic displays about 1024 x 760 x, y resolution elements are available.

Figure 4.4 - 19 shows a complete DDU/keyboard unit

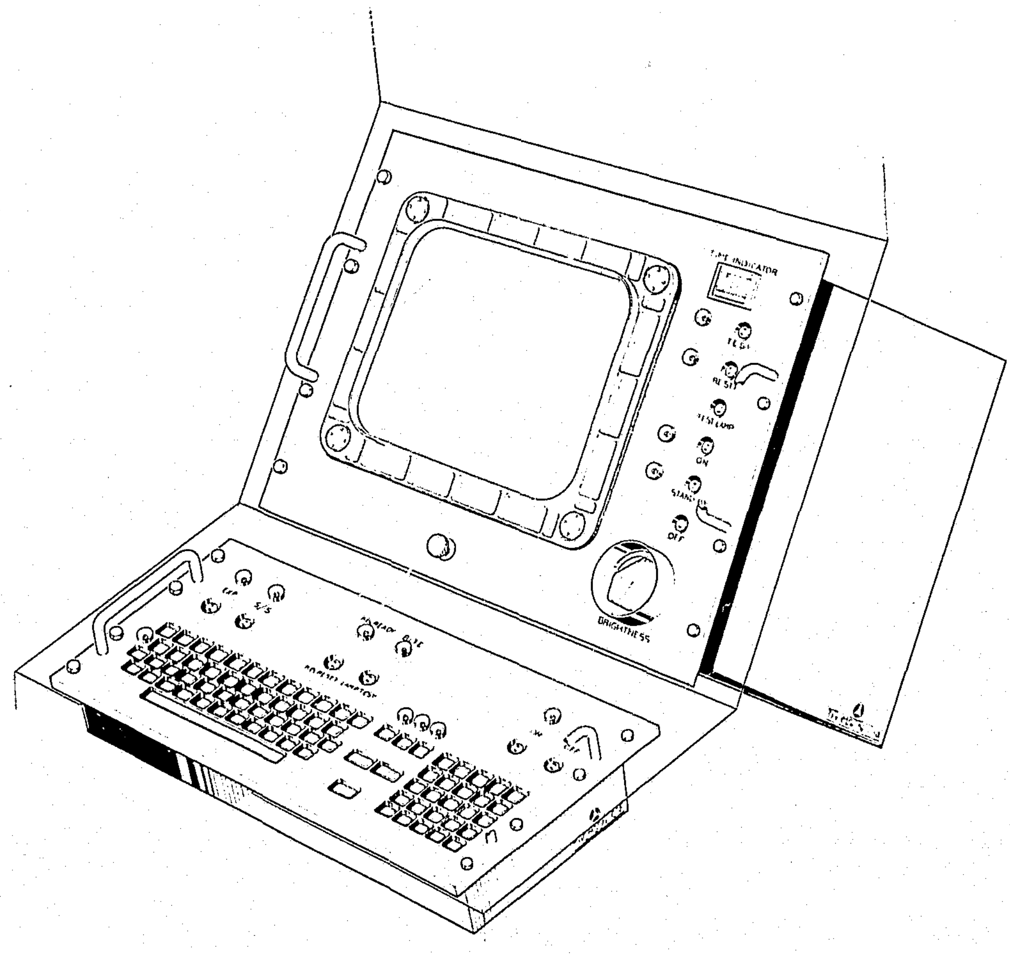


Figure 4.4 - 19 : DDU/Keyboard Unit

4.4.5 Subsystem Control

4.4.5.1 Control Concept

The main part of subsystem equipment is controlled during normal operation via the subsystem computer, its associated data bus, and subsystem remote acquisition units (RAU).

The RAU, under the control of the subsystem computer, is capable of issuing on/off commands or serial PCM command words to subsystem equipment. Similarly, the RAU is capable of receiving analog or digital monitoring signals.

Those functions which require control prior to the activation or after deactivation of the subsystem computer are performed via Orbiter multiplexer-demultiplexer (MDM) or by manual switches located on control panels in the Aft Flight Deck (AFD). The MDM under the control of the Orbiter computers is capable of transmitting on/off commands to Spacelab subsystem equipment. The MDM is similarly capable of receiving analog or digital monitoring signals.

For back-up purposes, or where direct control is preferable, local manual switches or valves are available.

Control from points external to Spacelab, for those items which are not connected to MDM, is made possible by the Orbiter/Spacelab computer links, which permit:

a) the transfer of commands from the Orbiter computer systems to the Spacelab subsystem computer

b) transmission of monitoring signals from Spacelab to Orbiter.

Many operation modes are possible, however; precise control modes will be established for each Spacelab mission, supported by documented procedures.

4.4.5.2 Activation Sequence

A typical sequence to activate a powered-down Spacelab is described as follows:

a) The liquid and air cooling loops, i.e., pumps and fans of the ECS, are turned on via Orbiter MDM commands. These can be initiated by an operator from computer keyboards located in the Orbiter.

b) As soon as the correct operation of the cooling system is verified through MDM monitoring functions, the Spacelab subsystem computer is activated, including I/O Unit, RAU's and Mass Memory Unit (MMU).

c) After computer operations have been verified and the subsystem software has been initiated, the subsystem computer with I/O unit and RAU's is used to distribute further commands to the subsystem equipment. These types of command are normally operator-initiated from the keyboards, one of which is located on the Orbiter AFD and two of which are located in the Spacelab module. Functions such as power ON/OFF switching of :

- cabin temperature control
- cabin oxygen control
- humidity control, cabin fan
- experiment computer including I/O unit
- experiment RAU's and experiment power distribution boxes

are controlled from the keyboards via the subsystem computer and RAU's.

4.4.6 Intercom

The Spacelab Intercom provides audio intercommunication between Spacelab/Orbiter, Spacelab/ground, Spacelab/air and internal Spacelab stations. In addition, an emergency communication link between Spacelab and Orbiter is provided.

The Spacelab Intercom consists of the following units:

- 1 Master Station
- 4 Remote Stations
- 5 Headsets/Umbilicals

The mode and channel switching capability is depicted in Figure 4.4-20.

There are five links between the Master Station and the Orbiter Audio Central Control Unit (ACCU). Each of the three Intercom channels can be connected to one of these five links by mode selection in the master station. Each of the five Headsets can be connected to one of the three channels by channel selection. In addition a page capability overriding all other audio communication is provided.

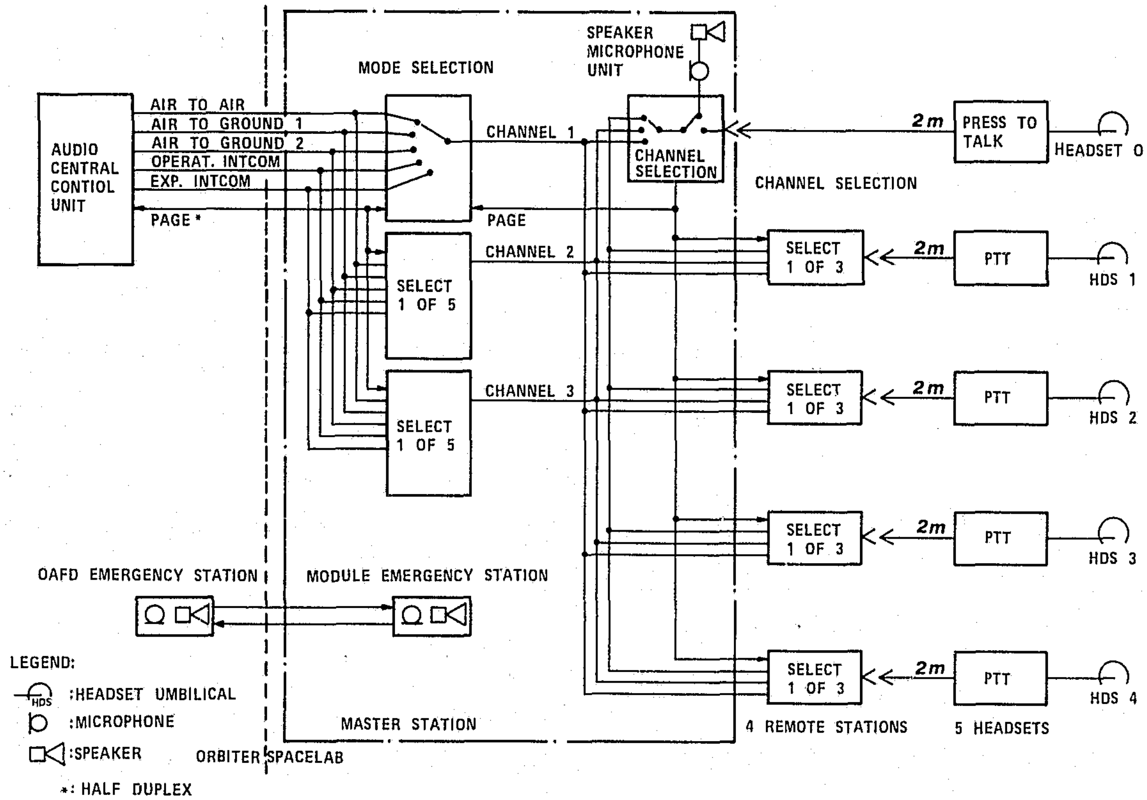


Figure 4.4 – 20 : Functional INTERCOM Block Diagram

4.4.7 Caution and Warning (C & W)

Spacelab has to provide to the Orbiter Caution & Warning System data which is critical to the safety of the Orbiter/Spacelab flight personnel. Caution & Warning signals are classified as follows:

1. Emergency – Crew hazard, requiring immediate instinctive crew action
2. Warning – Actual or impending anomalous condition which in itself is hazardous and requires immediate crew action
3. Caution – Actual or impending anomalous condition which in combination with other failures constitutes a system configuration that could be hazardous to the vehicle or crew and requires action or procedural change for corrective measures.

The Spacelab C & W system is integrated into the Orbiter C & W system. The level detection of analog C & W signals is performed (software controlled) in the Orbiter GPC and in the Spacelab Subsystem Computer for redundancy.

4.4.7.1 Emergency Signals

Emergency signals of Spacelab apply only to fire and rapid pressure loss in the Module.

Two types of sensors are foreseen. Fire and Smoke sensors and $\Delta P / \Delta T$ sensors indicating rapid cabin depressurization.

Three redundant pairs of Fire/Smoke sensors are located in the left and right avionics air loops and in the cabin. These sensors are hardwired to the Orbiter R-7 panel and fed into the Orbiter Caution and Warning Electronic Assembly (CWEA). The CWEA activates in case of Fire/Smoke inputs the Master Alarm and generates a Siren tone which is routed to Spacelab via a single hardwired output from the Audio Central Control Unit (ACCU).

The Orbiter System Management GPC will get Fire/Smoke information from the R-7 panel via a MDM and, for redundancy, from Spacelab via RAU´s, S/S Computer and PCM Master Unit. Emergency conditions are displayed on Orbiter and Spacelab CRT´s.

The Spacelab $\Delta P / \Delta T$ sensor output is only hardwired to the CWEA and will be disconnected during ascent and descent.

The Orbiter will provide redundant safing commands to act on the detection of a Spacelab Fire/Smoke condition. These commands (28 V discretes) will be initiated via the keyboard located in the Orbiter Forward Station.

4.4.7.2 Warning and Caution Signals

The Orbiter Caution & Warning System is shown in Figure 4.4-21. A maximum of 50 Warning and Caution sensor outputs may be routed via hardwired lines as discrete or analog signals to the dedicated Orbiter MDM. Presently 12 of these lines are used by Spacelab. Eight channels are connected to dedicated subsystem sensors and four are still free. In addition five direct Warning Inputs to the CWEA are possible.

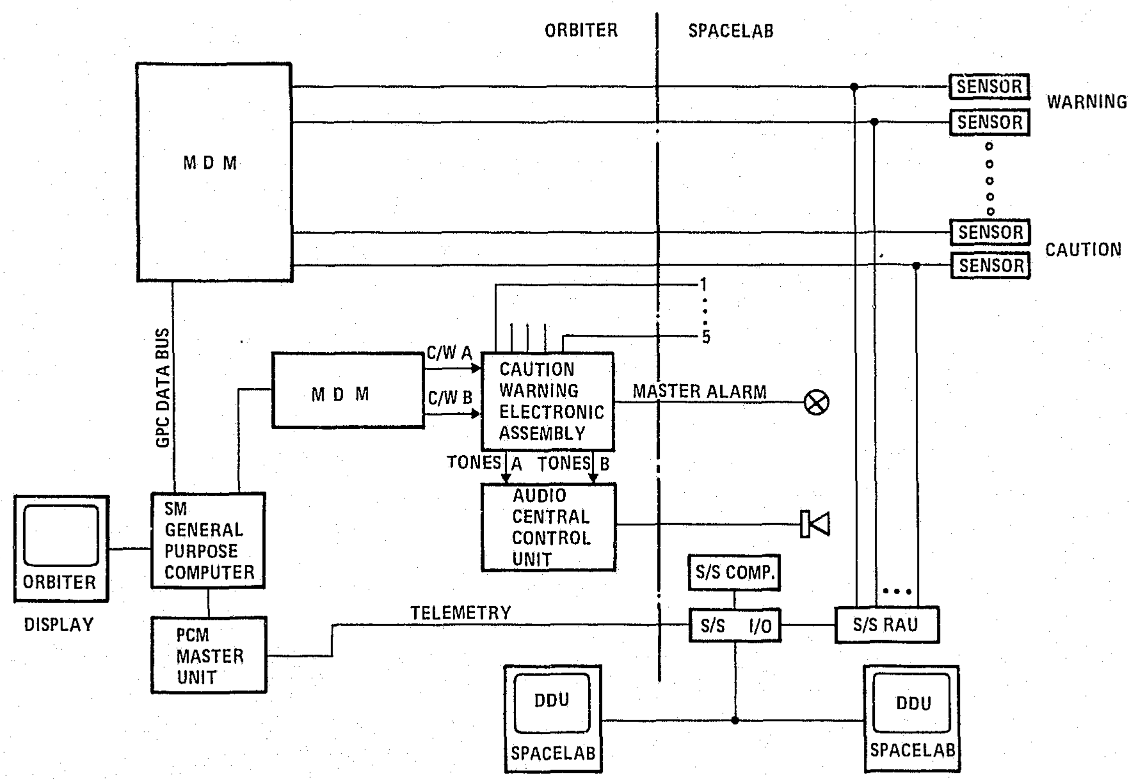


Figure 4.4 - 21 : Caution and Warning Flow Chart

Spacelab Warning and Caution signals are routed through the Orbiter MDM to the Orbiter System Management GPC. A second redundant path leading from Spacelab sensors to the GPC will be provided via S/S RAU's, S/S Computer and PCM Master Unit.

The GPC will detect warning and/or caution conditions and initiate the CWEA to activate the Master Alarm and generate Caution and Warning tones which are transmitted to Spacelab via the single hardwire output from the ACCU.

Caution and Warning conditions are displayed on Orbiter and Spacelab CRT's.

The Orbiter will provide a maximum of 36 safing commands to be used in response to a Spacelab Caution & Warning condition. These safing commands will be initiated by a keyboard entry to the GPC. The GPC issues the appropriate safing commands (discretes at voltage levels of 5 V or 28 V) to Spacelab via a MDM.

4.4.7.3 Experiment/Caution & Warning Interface

To interface with the C & W system through the four MDM and RAU inputs mentioned above, the experimenter has to provide his own sensors. To achieve a discrete or analog signal with the required characteristics it may be necessary to provide, in addition, an active signal conditioner. The signal conditioner has to be powered through the emergency bus.

The preliminary interface characteristics given in Table 4.4-8 are only for information.

The physical location of the MDM C & W interface is the Forward Endcone Feedthrough.

Table 4.4 – 8 : Preliminary C & W / Experiment Interface Characteristics

	MDM				RAU			
	SOURCE (SENSOR)	LOAD (MDM)	SOURCE (SENSOR)	LOAD (MDM)	SOURCE (SENSOR)	LOAD (RAU)	SOURCE (SENSOR)	LOAD (RAU)
TYPE	SINGLE ENDED	DIFFER. ISOLATED	ISOLATED	SINGLE ENDED	SINGLE ENDED	DIFFER.	SINGLE ENDED	DIFFER. ISOLATED
Line Length	max. 50 m		max. 50 m		max. 10 m		max. 10 m	
Analog Range	min. 0 V max. 5 V	min. – 5.12 V max. + 5.11 V	NA	NA	min. 0 V max. 5 V	min. – 5.12 V max. + 5.08 V	NA	NA
Discrete False	NA	NA	min. – 0.5 V max. + 0.5 V	≤ 2.0 V	NA	NA	min. – 0.5 V max. + 0.5 V	min. – 0.5 V max. + 2.0 V
Discrete True	NA	NA	min. + 4.0 V max. + 6.0 V	≥ 2.5 V	NA	NA	min. + 4.0 V max. + 6.0 V	min. + 2.5 V max. + 6.0 V
Ripple & Noise	TBD	NA	max. 20 m Vpp	NA	max. 20 m Vpp	NA	max. 20 m V	NA
CMV	NA	max. 10 V	TBD	NA	max. ± 1 V	max. ± 6 V	TBD	max. ± 6 V
Rise/Fall Time	NA	NA	min. 1 µs max. 2 ms		NA	NA	min. 1 µs max. 2 ms	
Roll-off Rate	NA	NA	NA	min 1 db/ Octave	NA	NA	NA	TBD
CMRR	NA	60 db within Filter	NA	NA	NA	min. 40 db	NA	min. 40 db
Transfer	DC-Coupled	DC-Coupled	DC-Coupled	DC-Coupled	DC-Coupled	DC-Coupled	DC-Coupled	DC-Coupled
Impedance	≤ 100 Ω	≥ 500 KΩ	≤ 100 Ω	min. 14 kΩ max. 21 kΩ	≤ 1 kΩ	≥ 500 kΩ	≤ 1 kΩ	≥ 500 kΩ
Power-Off Impedance	≤ 1 MΩ	≥ 100 kΩ	≤ 1 MΩ	10 kΩ	≤ 1 MΩ	2 kΩ Line-to-Line	≤ 1 MΩ	2 kΩ Line-to-Line
Capacitance	max. 200 pF	within Input Filter	max. 200 pF	within Input Filter	max. 200 pF	max. 200 pF	max. 200 pF	max. 200 pF
Drive Capability Load Current	1 mA	NA	1 mA	NA see Load Imp.	1 mA	0.05 mA	1 mA	0.05 mA
Overvoltage Protection	max. ± 20 V	max. ± 32 V	max. ± 20 V	max. ± 32 V	max. ± 32 V	max. ± 10 V	max. ± 15 V	max. ± 10 V
Fault Voltage Emission	max. ± 32 V	max. ± 15 V	max. ± 15 V	max. ± 15 V	max. ± 10 V	max. ± 15 V	max. ± 10 V	max. ± 15 V
Fault Current Limitation	No Limitation	max. 1 mA	max.±10 mA	max. 2 mA	max. ± 15 mA	max. ± 15 mA	max. ± 15 mA	max. ± 15 mA

4.5 Software

The Spacelab Computer Software comprises the software used for Spacelab during software developments, integration, testing, and operation. This includes subsystem testing, integration, checkout, on-board data handling for subsystems, on-board data handling support for experiments, and checkout for the CDMS portion of the experiment interfaces. Also included is certain support software used in the generation and validation of software and for the off-line reduction and analysis of checkout data.

Software especially dedicated to experiments is not included in the Spacelab computer software.

The Spacelab computer software is made up of sets, each of which is the assembly of software, used for a particular phase of the Spacelab program, with a specific computer system (experiment computer, S/S computer, EGSE, or Software Development Facility).

A set is made up of a number of packages.

A package consists of a group of software modules which are used together to perform some clearly definable functions.

Fig 4.5-1 thru-3 give an overview about SL computer software and the interrelationship between packages.

The Spacelab computer software is designed in a modular way in order to allow for good testability and maximum use of common functional units. Thus commonality can be achieved between the experiment and subsystem computers concerning the operating system and general facilities, such as operator interface, monitoring, fault isolation, subroutine library, etc.

Packages relevant for the experimenter are:

- CDMS Computer Operating System Packages

This package consists of the subsystem computer operating system (SCOS) and the experiment computer operating system (ECOS). For details see 4.5.1.1 -

- Support Software Packages.

The experiment application software packages running in the experiment computer under the ECOS will be supplied by the experimenter and/or the payload integrator.

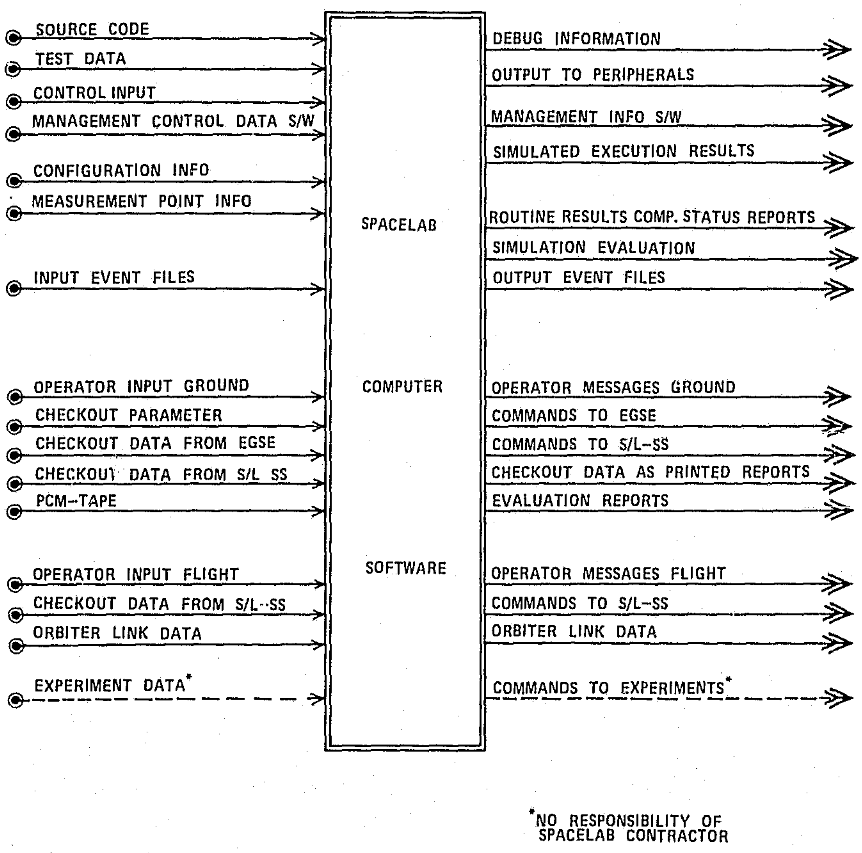
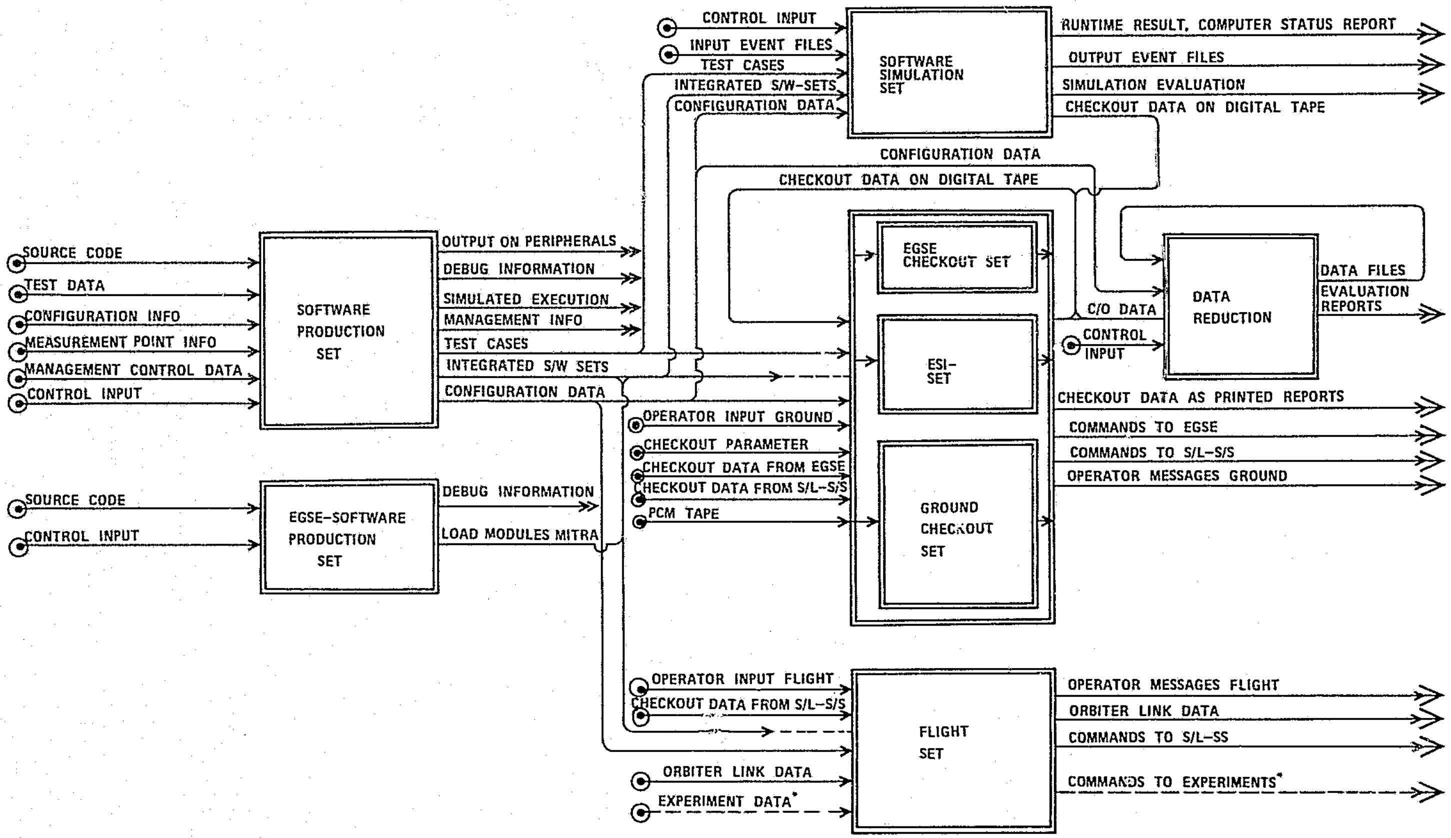
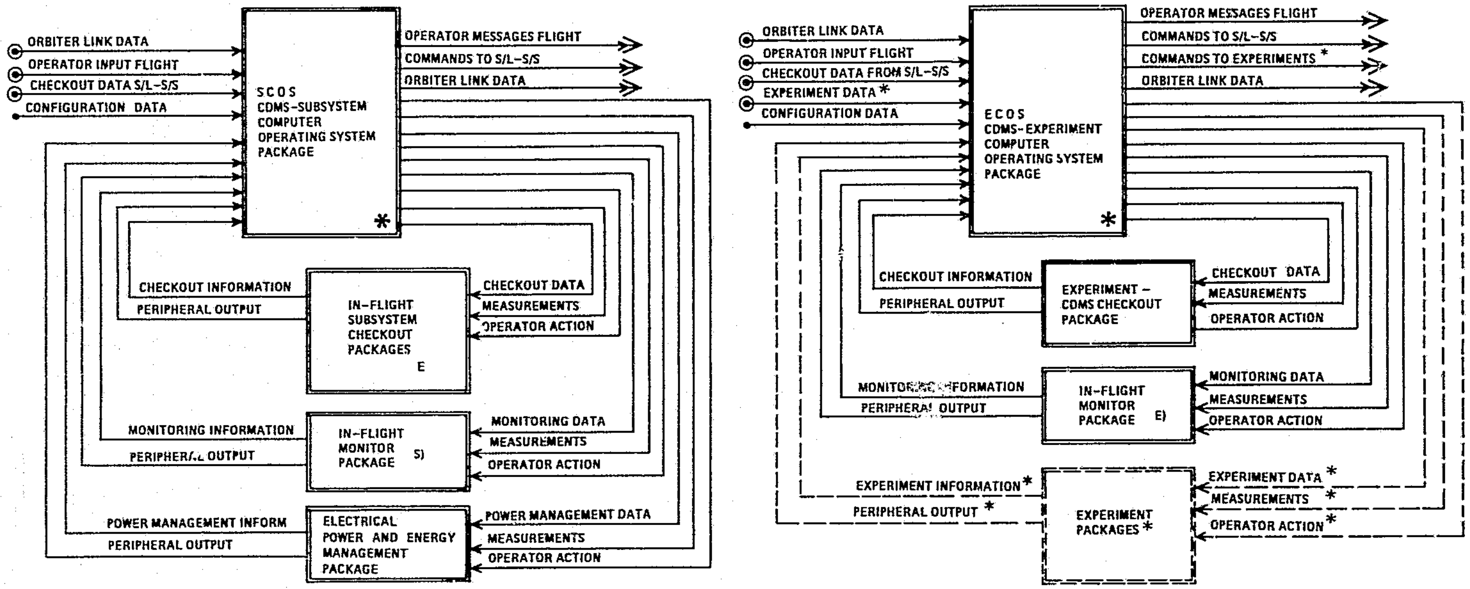


Figure 4.5 - 1 Spacelab Computer Software



* NO RESPONSIBILITY OF S/L-CONTRACTOR

Figure 4.5 - 2 Sets of Spacelab Computer Software



E: THE FOLLOWING IN-FLIGHT SUBSYSTEM CHECKOUT PACKAGES EXIST:
- EPDS - C/O
- ECS C/O
- S/S CDMS- C/O

*REMARK: FOR THE SAKE OF CLARITY THE CHECKOUT-INTERPRETER IS CONSIDERED TRANSPARENT. THEREFORE ALL INPUTS/OUTPUTS THE CHECKOUT APPLICATION SOFTWARE HAS TO DEAL WITH THROUGH THE INTERPRETER ARE ROUTED TO/FROM THE CHECKOUT APPLICATION SOFTWARE PACKAGES

S) MEANS CDMS - S/S-COMPUTER RESIDING MODULES OF PACKAGE

E) MEANS CDMS-EXP-COMPUTER RESIDING MODULES OF PACKAGE

* NO RESPONSIBILITY OF SPACELAB CONTRACTOR

Figure 4.5 - 3 Flight Software Set

4.5.1 Spacelab Software Environment

The experimenter – when linking up his experiment software with the Spacelab computer software – has to deal with the CDMS computer operating system package running in the experiment computer (ECOS), and with those modules of the flight application software packages (FLAP) which are also applicable to the experiments and which can be regarded as facilities available for applications. Furthermore, means are provided to support the experimenter in compiling, testing and integrating his software.

4.5.1.1 CDMS Computer Operating System

The CDMS computer operating system is at present the same for the subsystem (SCOS) and the experiment (ECOS) computers. However, because of the requirement that the experiment computer operating system accommodates a variety of experiment applications the ECOS may eventually grow to include greater capability in the areas of control and data processing.

The ECOS allows for asynchronous as well as synchronous tasks to be performed. The executive performs initialization, scheduling and termination of tasks. It assures time scheduling, loading of tasks including overlay and memory allocation to them, management of the various data tables in the data base for program control and housekeeping. It controls the allocation of the computer peripherals, such as memory, keyboards, data display unit, telemetry channels, and data bus. The executive allows for initialization of the computer system and for convenient recovery after system failure. The executive includes a computer self check which is executed periodically, providing a message in case of failure.

The input/output functions provide all services necessary to operate the remote acquisition units (RAU's para. 4.4.2.1). They format and transmit data to the CRTs for presentation to the crew and experimenters, receive and process external event messages based upon usage of the keyboard and inputs from experimenters. They permit communication with the Orbiter for reception of commands, state vector and timing data. They perform the transmission of data to the Orbiter for inclusion in downlink telemetry. They check the status of the peripherals (parity checking, data ready bits, data available bits as applicable).

The functions summarized as general facilities are functions common to all or most of the application programs and include services such as converting of raw data into engineering units, library of mathematical functions, etc.

The ECOS is able to monitor experiments, and to perform limit checking and calibration of data for display.

The ECOS is considered to be core resident with the exception of display formatting routines which will be stored on the mass memory (ref. para. 4.4).

The size of the ECOS is TBD.

Average operating system overhead is estimated to be 5 % of CPU time. Reaction time of external events is estimated to be 100 μsec maximum.

The S/W - S/W interface between the ECOS and experiment application packages is managed by supervisor calls and data tables.

The S/W - H/W interface between the ECOS and the peripheral hardware is handled via drivers in the ECOS which perform activation, status check data transfer and termination on the peripheral.

A keyboard language for communication between operator and experiment computer will be provided, thus the ECOS provides the interface between the operator and the computer system.

The functions involved are calling for data display and computer status display, initiating and termination of experiment modules at predefined points, interpretation of keyboard commands and changes to experiment modules.

The ECOS will be capable of displaying on CRT structures representing all the groups of data which may be selected for display. In addition, the capability will be provided to generate and display on-line a specific list of data on operator request.

4.5.1.2 Facilities Available for Application

The experiment application software for the experiment computer is the software executed by the ECOS and consists only of the monitor and fault isolation module for the experiment portion of the CDMS. All experiment related software packages to be loaded in the experiment computer are produced by the experimenters.

Only some of the modules of the Spacelab flight application software (FLAP) can also be used by the experimenter.

Within the FLAP there are modules available for management of electrical power and energy which make the respective information available for the experimenter on CRT by request via a keyboard entry to the subsystem computer. In addition, it will be possible to update-in predefined areas of memory-values and limits per telecommand and/or keyboard.

4.5.1.3 Software Integration

For integration of his experiment application software, the experimenter will be supplied with the following software (para. 4.9.2):

- CDMS - simulator
  This software will simulate on a host computer the CDMS environment

- Interpretive computer simulator (ICS)
  This software simulates the Mitra 125 S/MS on a host computer

- Experiment computer operating system (ECOS)

The CDMS environment simulator and the ICS can be integrated in order to simulate the complete CDMS on the host computer.

4.5.2 Software Development Aids

This software is part of the support software packages to be used for the effective development and maintenance of all Spacelab software, i.e. not only for the experiment software but also for operating systems, ground checkout packages and the flight application packages.

This means the experimenter, in developing his experiment software, shall utilize the facilities provided as far as possible. Experiment software shall be written in one of the languages explained in para. 4.5.2.2 which are available with the host software system (see para. 4.5.2.1). For debugging the simulator software shall be employed.

4.5.2.1 Host Software System

The host software system comprises all that support software necessary for the development of all experiment software and executed on a host computer (IBM/370). The following items will be available.

- HAL/S - 370 Compiler System
  This compiler system can be used to test programs written in HAL/S on an IBM 370. The compiler will compile HAL/S statements into code executable on an IBM 370 computer. The system also includes an execution monitor under which the compiled code can be executed.

- HAL/S - CII Compiler
  This compiler will compile HAL/S statements into code executable on a Mitra 125S/MS computer. The compiler itself will run on an IBM 370.

- GOAL Compiler

  The GOAL compiler will compile GOAL checkout statements into interpretative code. The interpretative code can be executed by an interpreter running on a Mitra 125 S/MS computer. The compiler itself will run on an IBM 370.

- Interpretative Computer Simulator (ICS)

  The ICS will simulate the Mitra 125 S/MS. This simulator will execute on an IBM 370.

- Mitra 125 S/MS Macro Assembler (MAS)

  Two versions of the assembler will be available. One will execute on IBM 370 and one will execute on the Mitra 125 itself. Code generated by either one can be processed by the EDL (see below).

- Mitra 125 S/MS Linkage Editor (EDL)

  Two versions of the EDL will be available. One will execute on IBM 370 and one will execute on the Mitra 125 itself. Code generated by either one can be processed by the preloader.

- I/O Box and Peripheral Simulator (IOBPS)

  This simulator will simulate the reactions of all CDMS hardware (except the computer) with respect to computer input/output and outside events. The IOBPS can work together with the ICS. It will execute on an IBM 370.

4.5.2.2 Programming Languages

- HAL/S

  Experiment software may be written in the programming language HAL/S. This is a real time programming language which allows the scheduling and synchronization of programs. The language also allows the manipulation of vectors and matrics and data structures in a simple manner.

  A wide range of mathematical functions is available with HAL/S.

- Experiment software may be written in CII MITRA 125 S/MS assembler language.

- Checkout software for experiments may be written in the checkout language GOAL. This language is oriented towards the convenient specification of checkout procedures by scientists and engineers.

4.5.3 Software Development Guidelines

Software development guidelines and standards, as well as procedures for the technical management, will be provided within the Software Standards Manual (Doc. No. MA-ER-0001).

There are two main topics: One covers the part of technical management such as verification (reviews and acceptance) and configuration control. The other specifies the necessary quidelines and standards to be followed during software development (design, implementation, test and documentation) to satisfy the requirements of software control.

As far as the user's interaction with NASA/ESA is concerned and to enable NASA/ESA to control and integrate the experiment software, the user will also have to follow some of the corresponding procedures and guidelines within the Software Standards Manual.

The relevant topics will be referenced in a manual of guidelines for experiment software. Additional guidelines, e.g. safety requirements, constraints on size and frequency and overall memory requirements, will be included.

4.6 Common Payload Support Equipment

The Common Payload Support Equipment (CPSE) consists totally of mission dependent equipment (Table 3 – 5). It is composed of the following items:

- one (1) top airlock (1 m dia., 1 m length)
- one (1) aft airlock (1 m dia., 1.5 m length)
- one (1) high quality window/viewport assembly
- one (1) modular film vault
- one (1) high vacuum vent facility

Provisions for the installation of this equipment are indicated in Fig 4.6 – 1. The aft end cone provides

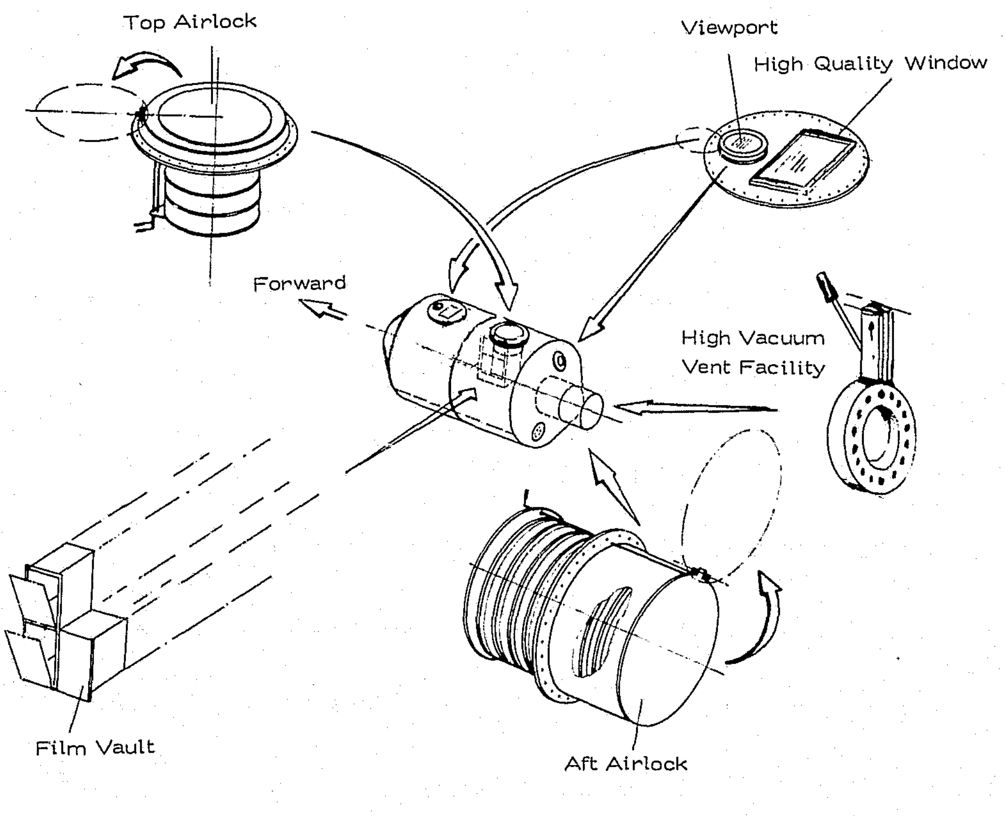


Figure 4.6 – 1 Common Payload Support Equipment For Module

the capability for installing the 1.5 m long airlock, or the high vacuum vent facility. There is a flanged cutout at the top of each cylindrical segment of the module to allow installation of either the viewport and the high quality window or the 1 m long airlock or the high vacuum vent facility. When not used, these cutouts are covered by coverplates.

Both the top and aft airlocks can be mounted into a one-segment module configuration, exchange of equipment in the airlocks during orbital operations will have to be done sequentially, and ground operational aspects (e.g. late access) may prohibit use of the top airlock in the core segment.

The modular film vault containers are mounted in standard 19" experiment racks, or optionally, on the center floor.

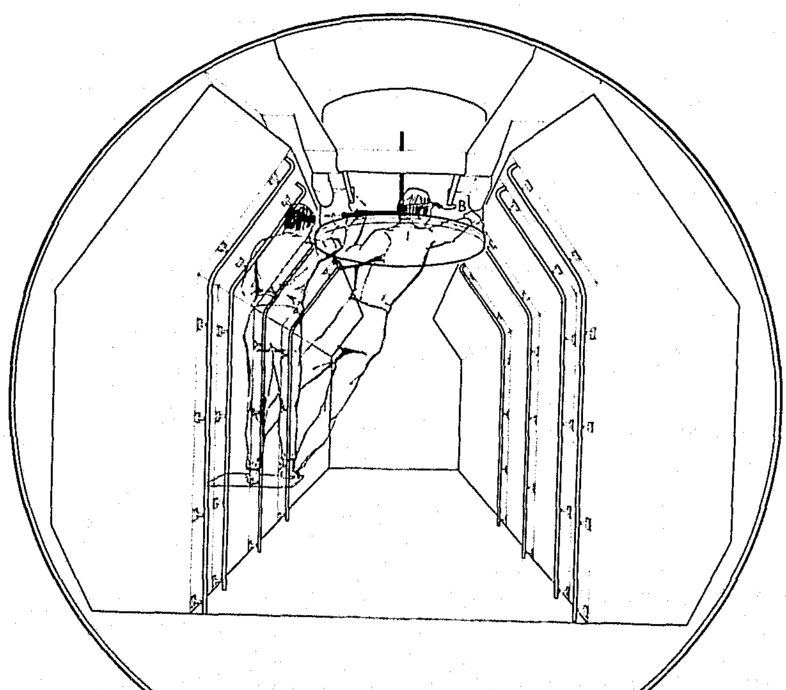


Figure 4.6-2: Top Airlock

4.6.1 Top Airlock

4.6.1.1 Description

This airlock is mounted in the top of the module and has the following major features (see Figure 4.6-2):

- Internal length: 1 m
- Internal diameter: 1 m
- Experiment mounting on sliding platform
- Manual operation of airlock
- Window in the inner hatch

The experiments will be mounted on a sliding platform parallel to the airlock axis. All controls of the airlock are manually operated. A control panel with control lights and a pressure gauge is provided for monitoring the operations of the airlock (Figure 4.6-3).

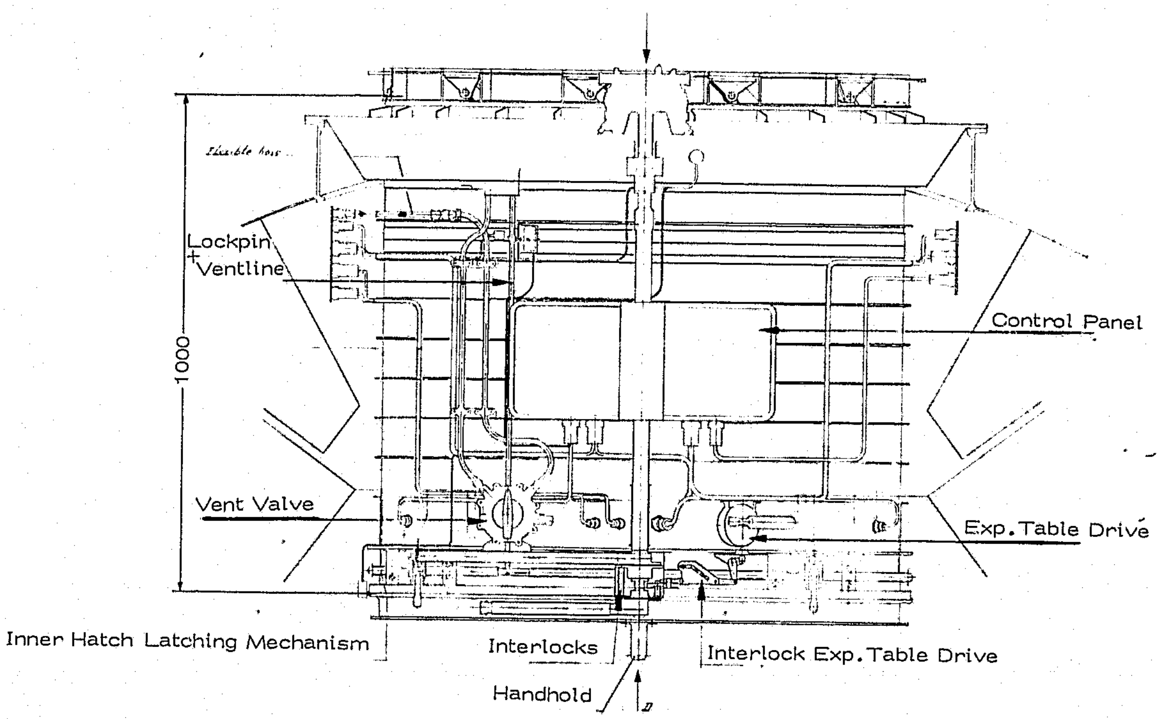


Figure 4.6-3: Top Airlock Inboard Profile Looking Aft

Electrical and mechanical interlocks prevent dangerous operational sequences. The controls and control panel are located at the lower part of the cylindrical shell.
Monitoring and display of airlock status is also provided by the CDMS.

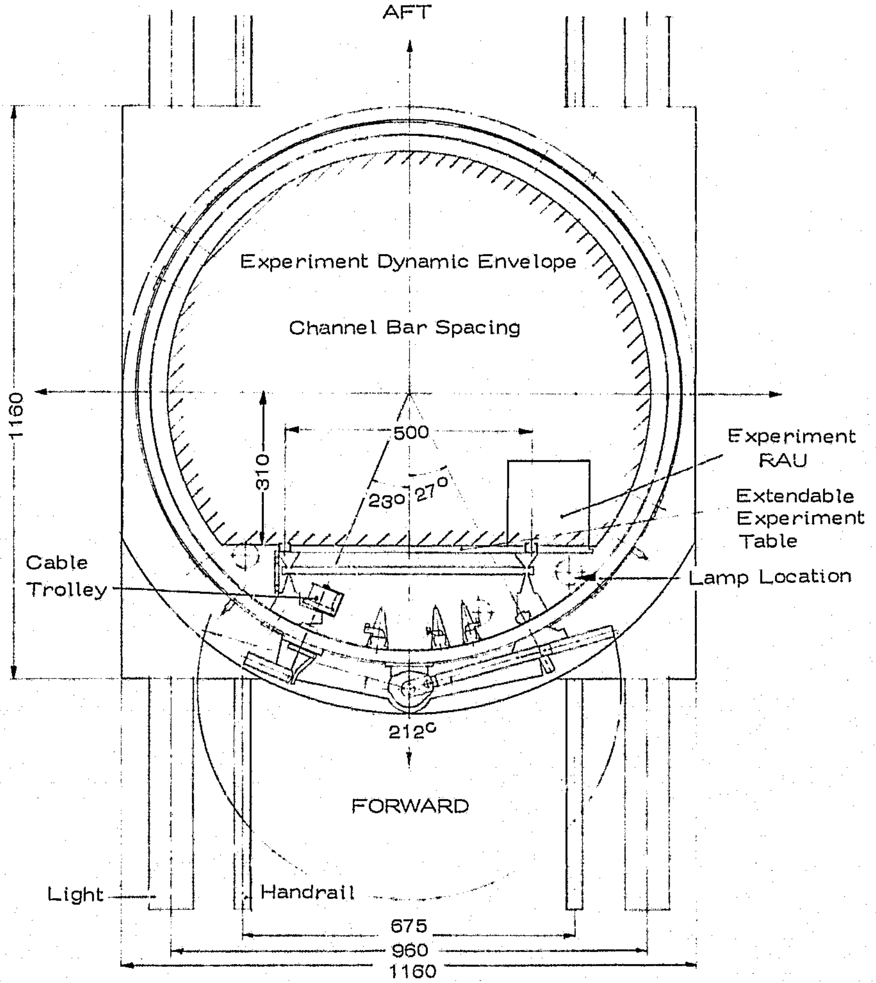
The experiment sliding platform, when extended, and the outer hatch penetrate the Orbiter cargo bay envelope (see Figure 2-5). In order to prevent a critical situation in the case of a malfunction of the retraction or hatch mechanism, safety precautions require the capability of jettisoning both the sliding platform and the hatch by means of the Orbiter Remote Manipulator (para. 2.6.1.6).

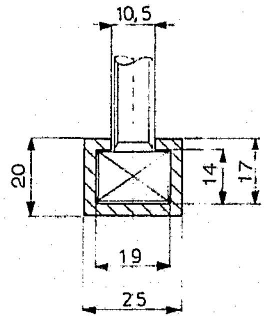
4.6.1.2 Experiment Accommodation

The dynamic envelope available for experiments is shown in Figure 4.6-4. Lamps inside the airlock may protrude into this envelope but these lamps can be removed when a voluminous experiment is flown.

Figure 4.6-4:

Top Airlock – View Up





Detail Channel Bar
With Bolt M10

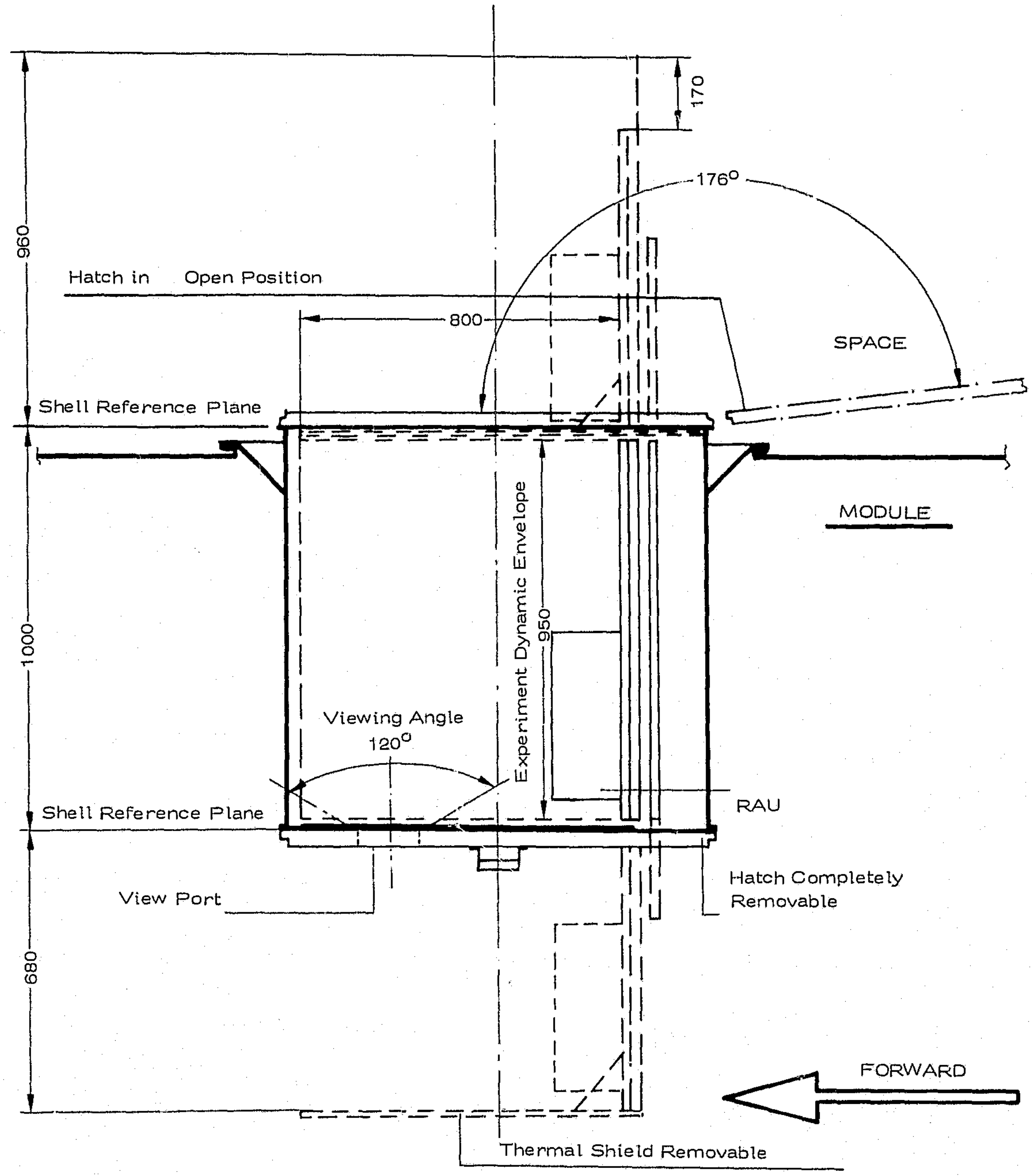


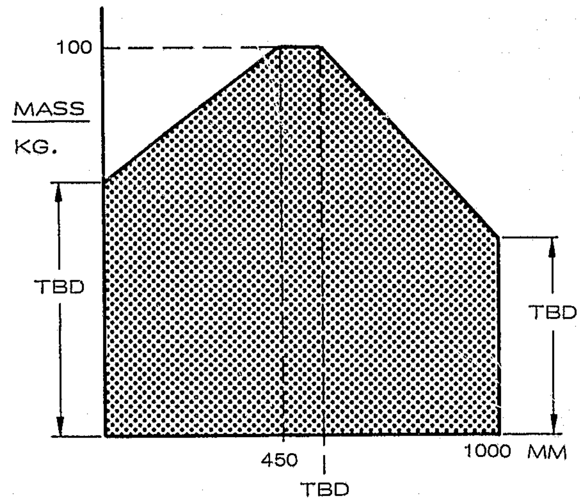
Figure 4.6-5: Top Airlock - Experiment Dynamic Envelope

Mounting provisions

The experiments will be mounted on the platform on two parallel channel bars 500 mm apart and 310 mm aft of the airlock axis by means of M 10 bolts. This interface, as depicted in Figure 4.6-4, allows a variety of convenient experiment attachments along the channel bars. For experiment mounting and checkout, the platform can be extended 0.7 meter into the module. For experiment mounting on the ground, however, the platform must be supported by a one-g kit, since the drive mechanism is not designed for one-g full load operation.

The airlock is designed to carry an experiment of 100 kg during launch and descent. Experiment mass and c.g. limitations are shown in Figure 4.6-6 and 4.6-7.

TOP AIRLOCK, LAUNCH & DESCENT



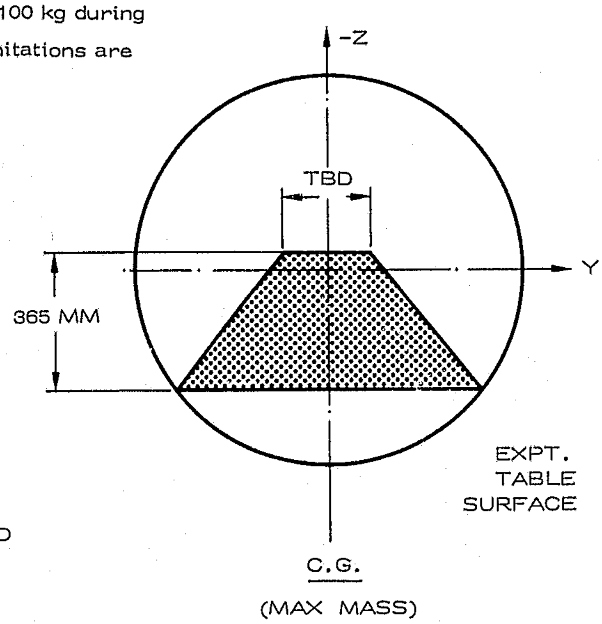
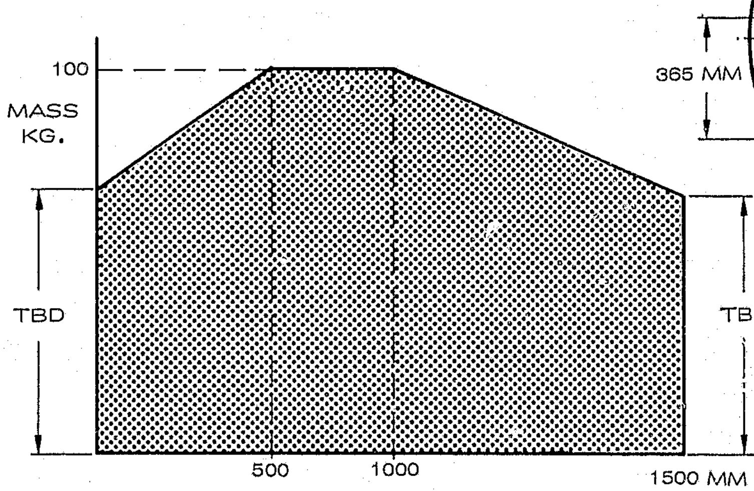


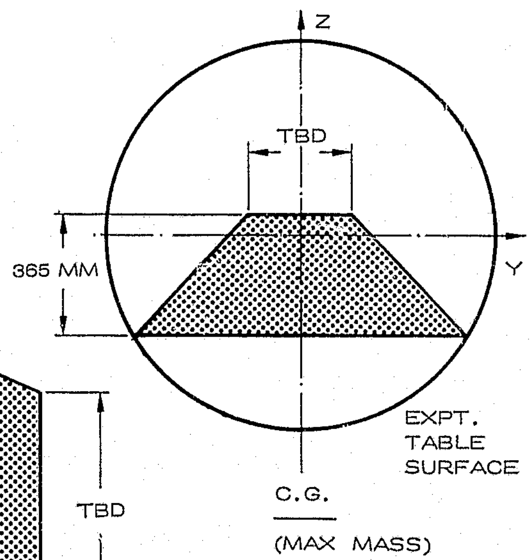
Figure 4.6-6: EXPERIMENT MASS AND C.G. LIMITATIONS

C.G. DISTANCE FROM INNER HATCH MM.

AFT AIRLOCK, LAUNCH AND DESCENT

Figure 4.6-7: EXPERIMENT MASS AND C.G. LIMITATIONS





C.G. DISTANCE FROM INNER HATCH

4.6.1.3 Platform Extension

After opening the outer hatch, the experiment may be extended up to 0.96 meter or any intermediate position. In this position, a thermal shield, as described in para. 4.6.1.6, closes the airlock opening thus protecting the airlock interior and module from thermal exposure.

Depending on still TBD thermal properties of the airlock and the thermal interface between airlock and module as well as on mission details such as viewing directions and times, it might be necessary for thermal reasons to always extend the platform for experiment operation.

The inner hatch can be completely detached for payload installation and access. The storage location for the inner hatch is TBD.

The inner hatch can be completely detached for payload installation and access.

4.6.1.4 Power Supply

Power connectors are provided at the platform for

- 28 V DC primary, 200 Watt
- AC 115/200 V, 400 Hz / 3 phase, 3 Amps

4.6.1.5 Data and Commands

Experiment data handling and control can be performed by either the CDMS (via an experiment RAU mounted on the airlock platform) or via experiment hard wired lines through the airlock shell to payload equipment in the module.

A connector for 3 times 2 twisted shielded pairs for payload hard wired lines is also available on the platform.

A flexible cable harness connects the platform with feedthrough connectors in the airlock shell, thus experiment equipment can be checked out while the platform is extended into the module.

4.6.1.6 Thermal Control

The detailed thermal characteristics of the airlock are still being evaluated. To maintain satisfactory operation of airlock mechanisms and seals heaters are incorporated with a total power consumption of around 77 W (the exact power at any time will depend on the particular mission operating conditions). While the airlock will be able to absorb TBD experiment generated heat, it will be the responsibility of the experimenter to provide the thermal control of his experiment.

In order to avoid excessive heating at the airlock cavity from solar radiation or earth albedo a removable thermal shield is provided, as shown in Figure 4.6-5. There are two holes provided in the cylindrical shell of the airlock (opposite the top surface of the platform) and these are available for payload provided feedthroughs and cooling lines (e.g. to utilize the experiment heat exchanger).

4.6.1.7 Viewing and Illumination

Experiments can be observed through a window (15 cm diameter) in the inner hatch, as indicated in Figure 4.6-5, providing a viewing angle of 120°. A respective cut-out is foreseen in the removable thermal shield to prevent obstruction of free viewing.

Illumination inside the airlock will be 100 lumen/m^2; the total power provided for the lamps is 25 W. Airlock lights and intensity can be controlled from the airlock control panel. An additional master switch is located at the subsystem control station in the core segment.

4.6.1.8 Airlock Operation

The airlock enables experiments to be exposed to space environment and a total of seven re-pressurization cycles is possible per mission, using the basic ECS nitrogen resources for seven day missions . However, additional re-pressurization may be possible depending on the usage of nitrogen by the ECLS.

The operational timeline of the airlock has been analyzed as per Table 4.6-1; details may be subject to change:

Table 4.6 - 1 : Airlock Operation

AIRLOCK OPERATION	TIME MIN	TIME SEC
Start with Launched Experiment in Airlock	00	00
Activate Airlock		
1. Check pressure gage for equalized pressure-no. leakage		05
2. Switch on airlock power		05
3. Open N_2 valve up-stream of airlock		05
4. Check housekeeping on airlock and CDMS	1	
Unlatch Inner Hatch		10
Remove & Stow Inner Hatch		
1. Pull hatch inward into module by hand hold on center of hatch		10
2. Translate to stowage area (TBD) & secure hatch (TBD). (Note: Removal of hatch, lock latch handle in open position.)		50
Retract Exp. & Exp. Table into Module		30
Activate Power to RAU and Experiment & Perform Experiment Checkout. TBD - Dependent on each specific experiment and its requirements. Power is switched on at CDMS/EPDS control panels		TBD
Note: For experiments using electrical power it may be desirable to switch off exp. power after checkout and perform deployment before power is switched back on.		
Return Exp. & Exp. Table to Stowed Position in Module		30
Retrieve Stowed Hatch & Replace on Airlock		60
Latch Inner Hatch		10
Vent Airlock	17	15
Unlatch Outer Hatch		40
Open Outer Hatch		35
Extend Experiment Table		30
Note: Further extension of experiment will be done from CDMS panel after verification of experiment table extended.		
Retract Experiment Table		25
Close Outer Hatch		30
Latch Outer Hatch		30
Pressurize Airlock	24	05
Remove & Replace Experiment	1	15

4.6.2 Aft Airlock

Editorial note: The deletion of the aft airlock is currently under discussion.

In many aspects, the aft airlock to be mounted in the aft end cone is identical to the top airlock. The resources to support experiments are the same. The main difference is the internal length of 1.5 m. The sliding experiment mounting platform in the airlock is capable of extending up to 1.46 meter to the outside or 1.2 meters to the inside of the module

The outer hatch pivots on a hinge at the lower outer part of the airlock and, when open, remains in a 175 degrees position.

4.6.3 High Quality Window - Skylab S 190 A Window

4.6.3.1 Mechanical Assembly

It has been decided to adapt spare units of the S 190 A window (left over from the Skylab program) for use with Spacelab and an adapter plate is being designed to enable mounting of the window assembly in the 1.3 m diameter holes available in the module. The design of this adaptor plate is in an early stage and two concepts are currently under consideration. One concept provides for inclusion of the 30 cm viewport assembly into the adaptor plate while the alternative concept is for the incorporation of only the high quality window into the adaptor plate (Figure 4.6-8 and 4.6-9).

The high quality window itself consists of a single pane of BK-7 glass of rectangular (41 x 55 cm) shape and 4.1 cm thick enclosed in a molded seal and supported by a flexible spring system in an aluminium frame. A detail window section is shown in Figure 4.6-10. The window is equipped with a heater system that controls window temperatures to minimize thermal gradients across the glass and to prevent condensation. The outer surface of the glass has a 40-watt electro-conductive film (ECF) heater and two 100-watt heater elements are mounted in the frame surrounding the glass.

An external, manually operated, cover protects the window glass from radiation, meteoroid impact, contamination etc. during periods when the window is not in use. A removable glass safety shield mounted to the inside of the window protects the window surface from impacts and provides a redundant pressure seal when the window is not in use.

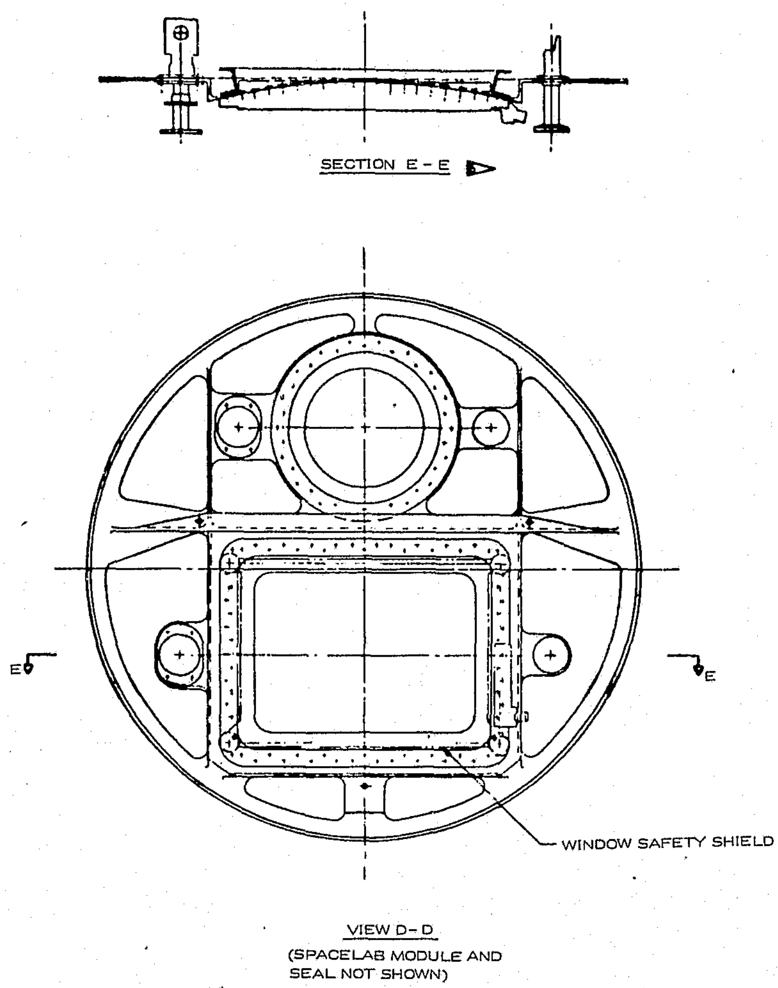


Figure 4.6-8: Window Design Concept 1 - Skylab Window/Viewport Combination

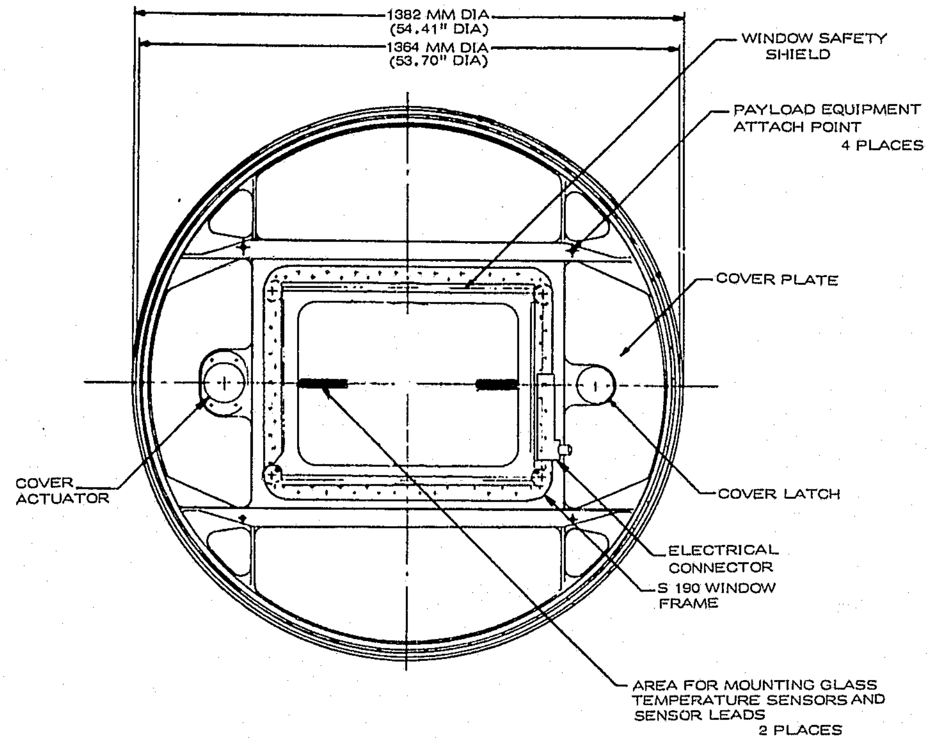


Figure 4.6-9: Window Design Concept 2 - Skylab Window Only

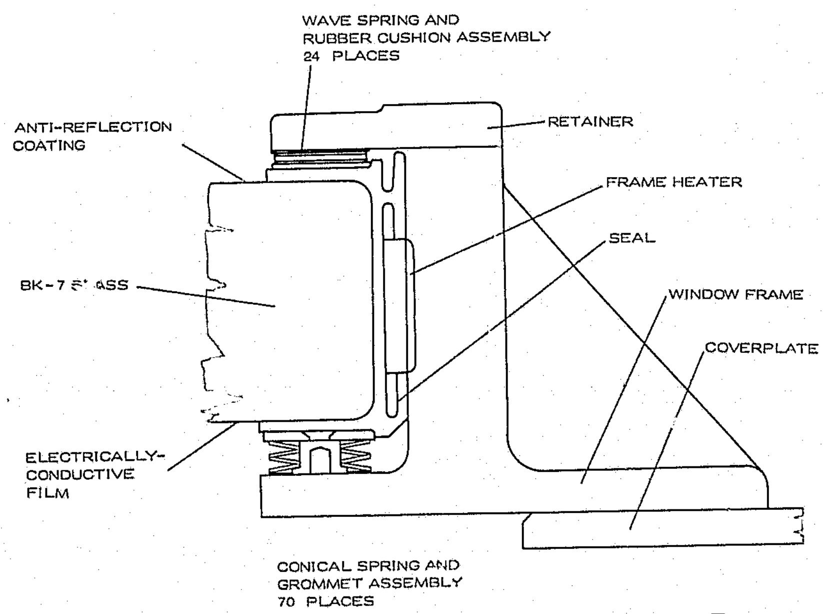


Figure 4.6-10: Detail Section Through the Window Frame

4.6.3.2 Optical Performance of Window

Transmissibility

The window transmission characteristics from 300 to 1000 nm are shown in Figure 4.6-11.

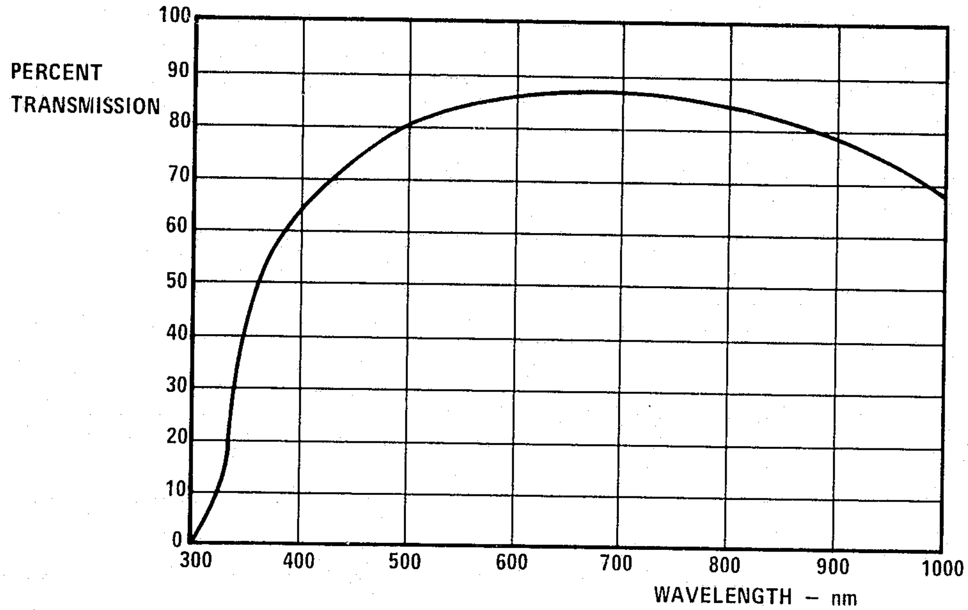


Figure 4.6-11: S 190 A. Window Transmission

Viewing Area

There are two areas 1 x 19 cm, as shown in Fig.4.6-9, where temperature sensors and their associated wiring are mounted to the outside of the glass. Relocation of these sensors is not possible without damaging the ECF coating on the glass and also affecting in the performance of the entire window.

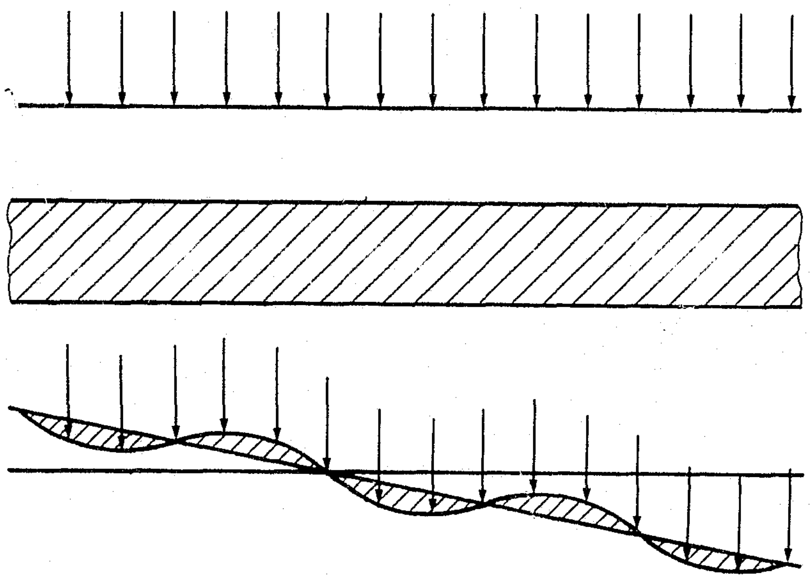


Figure 4.6-12: Wavefront Variation Indicating the Mean Deviation and Best Fit Plane After Passing through the Glass

Wavefront Variation

Wavefront variation is an optical performance criterion which affects distortion, resolution, contrast and registration. It depends on the intrinsic quality of the transmittant material and its final polishing and optical coatings and is also affected by factors such as pressure deflection and thermal gradients. Some preliminary measurements have been made for the S 190 A window under various thermal conditions simulating the different possible viewing conditions (see Table 4.6-2).

The measurements were made using a laser beam with a wavelength of 632.8 nm and refer to a circular area of 7.6 cm diameter. RMS values of wavefront variation are stated and are measured from two reference planes - the plane of best fit and the mean-deviated plane. Fig.4.6-12 illustrates the difference between these two planes.

Table 4.6-2: Wavefront Variation through Window

Viewing Conditions	Wavefront Variation (nm rms)	
	Best Fitting Plane	Mean Deviated Plane
Deep space viewing	≤ 13	≤ 26
Earth viewing	≤ 13	≤ 25
Sun viewing*	≤ 53	≤ 180

*There may be some constraints on sun viewing due to thermal stresses in the window glass.

Other Optical Characteristics

Table 4.6-3 shows some further optical characteristics of the window.

Table 4.6-3: Optical Characteristics of Window

Optical Characteristics	Window Performance
Parallelism	2 arc sec
Reflectance	2 % on inside 4 % on outside
Seeds and bubbles	total area: 0.1 mm^2/100 cm^3 of glass maximum dimension: 0.76 mm of single imperfection
Surface quality	60 - 40 or better as defined in MIL-13830

Mounting Provisions

The provisions for mounting cameras and other viewing instruments, as well as permissable envelopes, loads and c.g., are TBD.

Operations

The operational procedures for opening the external cover and for removal of the the internal safety shield are TBD.

The operation of the window heaters is automatic and will result in different levels of power consumption (chargeable to column B of Table4.2-2) for different viewing conditions as the heaters correct for equalization of temperature across the window. Table 4.6-4 indicates a preliminary estimate of the power consumption.

Table 4.6-4: Power Consumption of Window Heater

Viewing Conditions	Window Heater Power (W)
Deep space viewing	69
Daytime earth viewing	94
Nighttime earth viewing	15
Sun viewing*	200

* There may be some constraints on sun viewing due to thermal stresses in the window glass.

4.6.4 Viewports

The two viewports, the one located in the top of the module and the one located in the aft end cone are of identical design. Each viewport consists of two panes of 30 cm diameter, the outer one being quartz glass, the inner pane being safety glass (Figure 4.6-13).

The viewport glasses are exchangeable. Variances in thickness will require appropriate spacer rings.

The cavity between both panes is equipped with vent and purge devices. A manually operated external window cover is provided. Experiment-mounting provisions on the interface flange are foreseen; details are TBD.

The optical characteristics of the viewport are as follows (Table 4.6-5):

Table 4.6-5: Optical Characteristics of the Viewport

Spectrum	Transmission (%)
Ultraviolet, 200 to 300 nm	0.01
Visible range, 400 to 700 nm	65
Infrared, 800 to 1200 nm	10

The gaps from 300 thru 400 nm and 700 thru 800 nm, respectively, are of transient nature and not specifically defined.

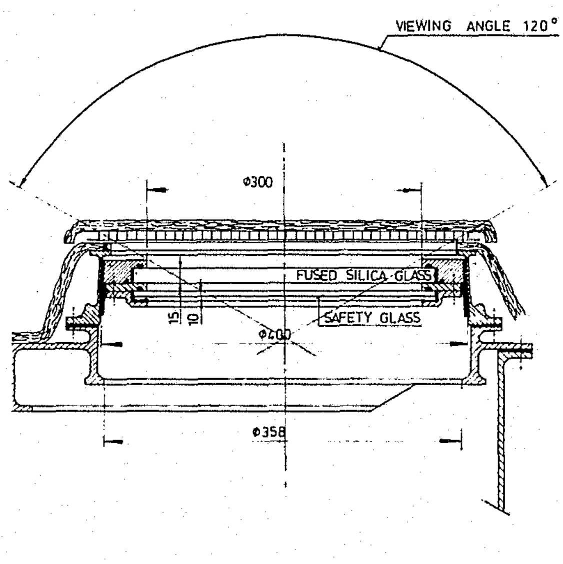


Figure 4.6-13: Viewport

4.6.5 Film Vaults

Editorial note: The film vault is subject to an engineering change proposal; subsequent description reflects only a preliminary design status.

The film vault consist of three modules that fit into the lower parts of standard racks as shown in Figure 4.6-14. The additional capability to mount the film vault on the center aisle is under consideration. Since the various missions require different film storage capabilities, the film vaults are of modular design to minimize ineffective volume utilization and unnecessary weight in orbit.

The useful volume of the film vault modules is at least 270 liters to accommodate approximately 150 kg of film.

Each vault has interchangeable drawers of three different heights to accommodate the various sizes of film cartridges and cassettes in an efficient manner. Each drawer is provided with film type as well as payload/experiment allocation indexes and can be secured by locks.

Each drawer will be filled with foam material; cut-outs will be custom made prior to each mission to suit the individual needs of the experimenter and to provide a tight fit for the film shapes to be accommodated. The loaded drawers can be installed in the film vault shortly before closing Spacelab during Level I integration, in order to maintain a controlled environment for as long as possible for the film material. For special heavy cassettes, loading of the drawers after installation will be necessary.

TBD

Figure 4.6 - 14 Modular Film Vault

Inside the Module, the film vault temperature will depend on the cabin air loop which is controllable within 291 and 300 K (18^{o} - 27^{o} C). No thermal insulation is necessary in the case of center aisle mounting of the film vault. For rack mounted units, some TBD user provided thermal insulation from nearby high heat rejecting experiments may be required.

The relative humidity is maintained between 25 and 60 per cent by means of desiccant salt bags installed in the film vault doors.

For seven day missions, no extra radiation shielding may be necessary for most film types, since the ambient masses plus film vault doors and drawer face plates provide sufficient protection. The average shielding effect corresponds to 10 - 12 g/cm^2 stopping power which, e.g. at a 57 degrees and 400 km orbit, would result in a dose rate of approximately 0.06 - 0.08 rads/day inside the film vault. For extended missions and in the case of sensitive materials, additional modular shielding is provided for one film vault module in the form of plates attached to the vault walls. User supplied internal slabs can also be installed within the drawers.

As a further measure, the experimenter may locate the most sensitive material in the middle of centrally located drawers so that ambient film material will provide additional protection.

Detailed data on dose rates to be expected will be subject to specific radiation shielding analyses for each individual mission. Due to the penetrating cosmic ray component, however, a dose rate less than in the order of 0.01 rads/day cannot be ensured by any practical means.

4.6.6 High Vacuum Vent Facility

Editorial note: The high vacuum vent facility is presently being defined; thus information contained in this paragraph is only preliminary.

4.6.6.1 Facility Components

The purpose of the high vacuum vent facility is to provide space vacuum inside experiment vacuum chambers and processing chambers which are mounted inside the module.

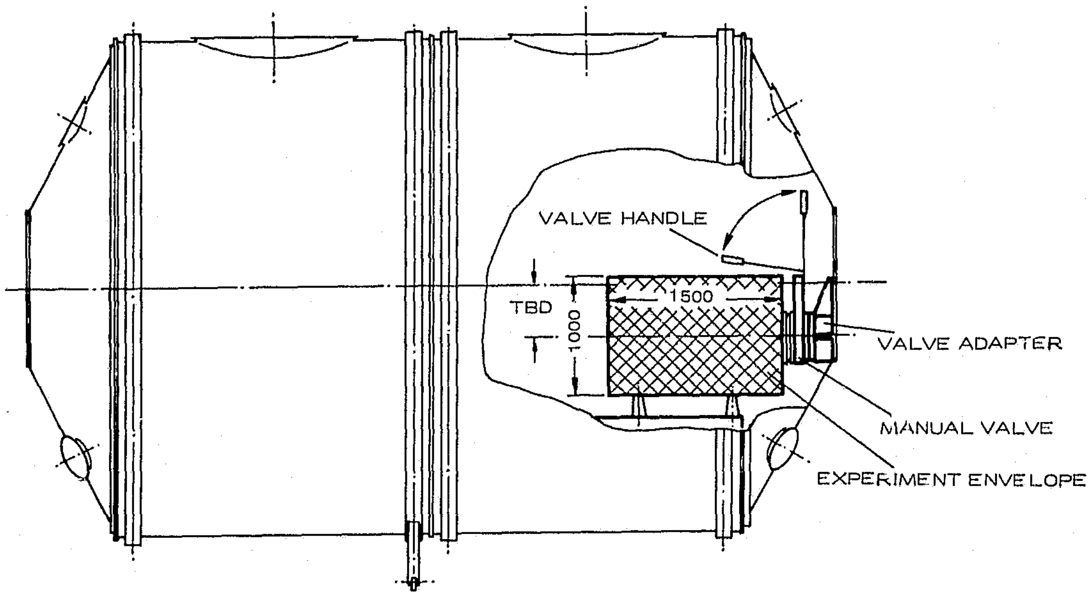


Figure 4.6-15: Aft Mounted High Vacuum Vent Facility

The high vacuum vent facility consists of a manually operated high vacuum valve assembly and ancillary equipment such as a pressure gauge and pressure equalizer. As depicted in Figure 4.6-15 and -16, the facility may be mounted, in accordance with the specific experiment requirements, either at the top or at the aft airlock interface flange.

A further possibility is shown in Figure 4.6-17 for rack mounted experiments.

The clear diameter of the valve assembly is approximately 300 mm; the distance from the experiment interface flange to the outside opening is kept to a minimum to provide a minimum pumping speed of 1.33 Pa liters per second (10^{-2} Torr l/s) at a pressure of 1.33×10^{-3} Pa (10^{-5} Torr).

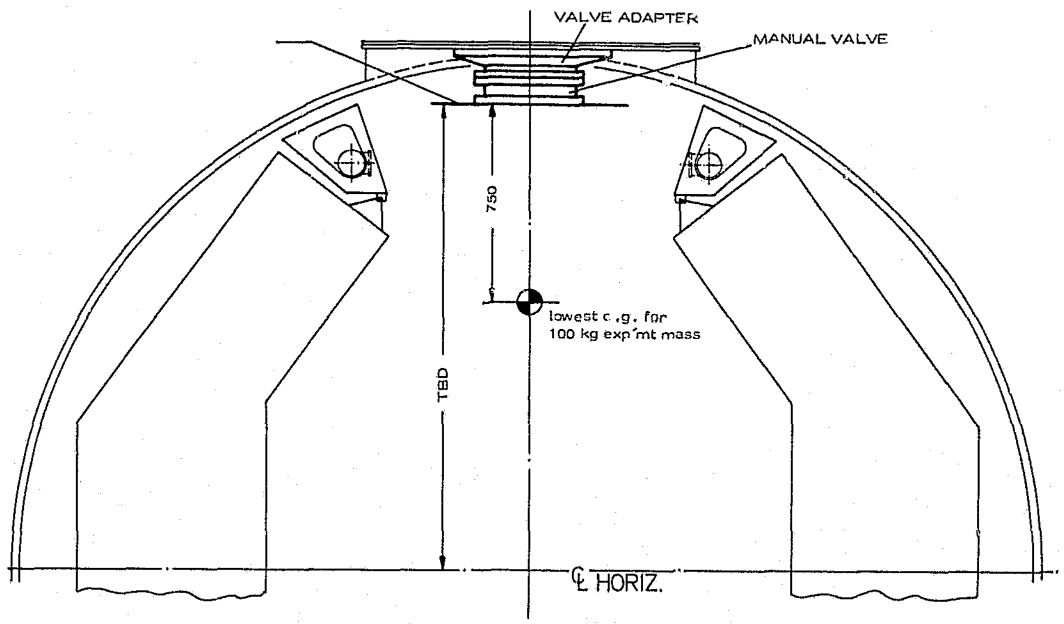


Figure 4.6-16: Top Mounted High Vacuum Vent Facility

4.6.6.2 Experiment Accommodation

The interface to the experiment vacuum chamber is given by a standard TBD (PNEUROP or DIN) high vacuum flange (Figure 4.6-17). The flange assembly will be designed to carry an experiment mass of 100 kg with a c.g. at a distance of 75 cm from the flange interface plane.

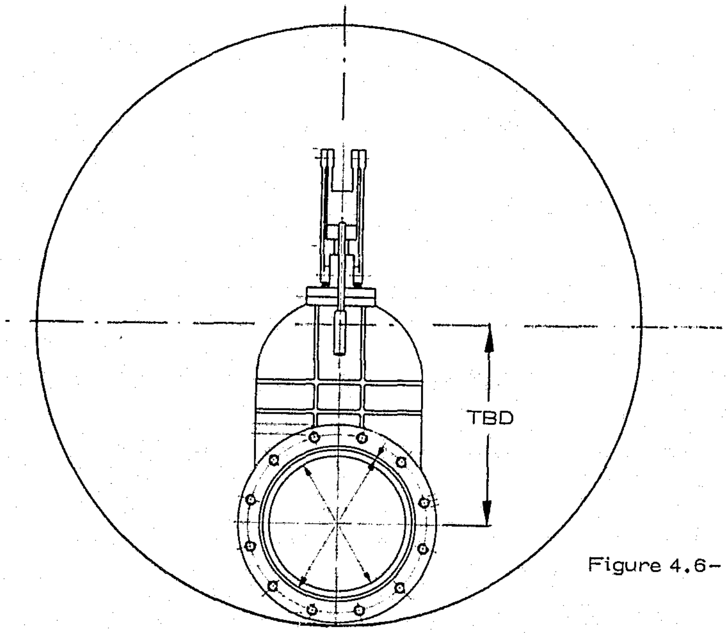


Figure 4.6-17: Typical High Vacuum Vent Valve Mounted to the Aft Interface Flange (View Aft)

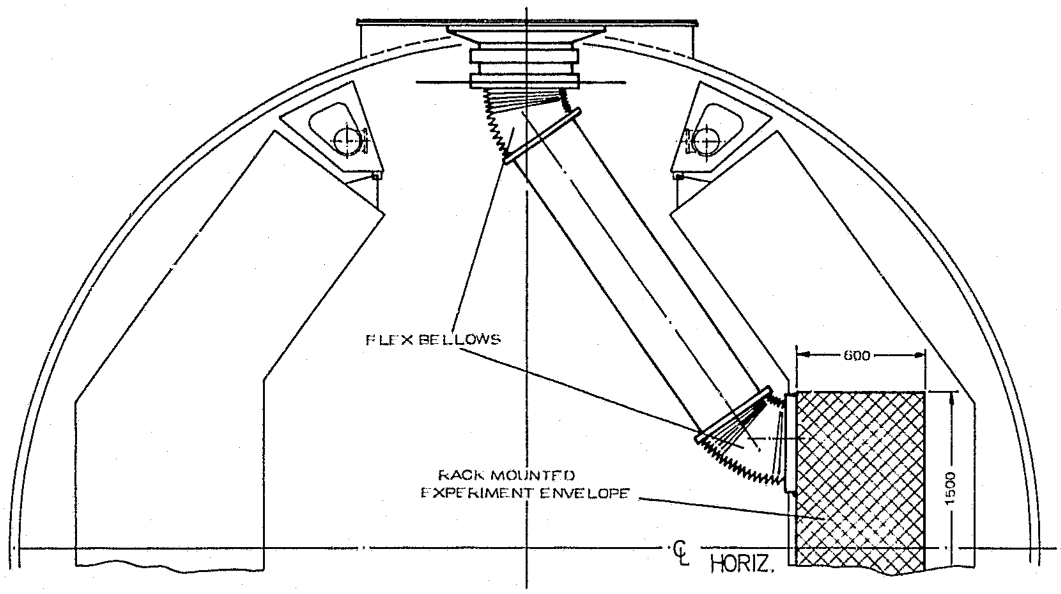


Figure 4.6-18: Rack Mounted High Vacuum Equipment

The maximum experiment envelope foreseen is 0.60 x 1.00 x 1.50 m for either center floor or rack accommodation (Figure 4.6-15 and -16). In both cases, the experiment mass may be increased up to 200 kg for additional support will be given by either the center floor structure or the rack.

For rack mounted equipment a user supplied connecting pipe must be routed to the lower portion of the double rack adjacent to the interface flange, as shown in Figure 4.6-18. The use of this extra connecting pipe, however, involves a TBD reduction of the pumping speed, hence an increase of time required for outgassing will have to be taken into account by the experimenter.

4.6.6.3 Utilities

Utilities provided for experiments connected to the high vacuum vent facility are as follows:

- Top airlock interface mounting: Use of top airlock utilities
- Rack mounted experiments: Use of rack provided power and RAU interface
- Aft airlock interface mounting: Use of aft airlock utilities
- Center floor mounting: Use of center aisle support utilities (presently subject to engineering change proposal)

4.6.6.4 Operation

The experiment equipment will be attached to the high vacuum vent facility throughout the entire mission. Detachment and/or assembly in flight is, under normal operational conditions, not foreseen.

Venting will be initiated by opening a pressure equalizer bypass valve. Whenever the pressure gauge (monitor) indicates a TBD pressure, the high vacuum valve will be manually opened. Experiment operation may be initiated whenever the experiment supplied vacuum meters indicate the vacuum desired.

Repressurization starts with closing of the high vacuum valve; then repressurization will be achieved by means of a valve system and nitrogen supplies, as used for the airlocks.

4.7 Mechanical Ground Support Equipment

4.7.1 General Overview

The Mechanical Ground Support Equipment consists of the operational tools used for the handling, transportation, servicing, alignment, and environmental protection of the assembled Spacelab, its subsystems and modulor elements. The modular Spacelab requires a flexible MGSE capability to accommodate all alternate Spacelab configurations.

To provide the flexibility, the MGSE incorporates individual supports for each major Spacelab element (core module, experiment module and pallet segment). These supports can be interlocked to accommodate the various assembled groupings of module and pallet segments. It also incorporates a matched-rail assembly stand system, which is adaptable to the longitudinal arrangement of the configuration. In addition, a family of handling and assembly aids is provided for the Igloo, equipment racks, floor structure, the utility bridges, airlocks, aft flight deck equipment, insulation, and other subsystem units. Figure 4.7-1 gives examples of typical handling support equipment provided.

Turn-around scheduling is aided by the ability to perform off-line integration of experiment groups. They can be inserted into the module after all subsystem functions for the experiment package have been verified. This is accomplished by rolling the experiment racks and floor structure into the open module cylinder and then securing the end cone bulkhead.

MGSE servicing equipment fills, drains, and leak checks the fluid loops of the thermal control system. Leak check equipment will be used to detect and locate possible leakage. The optical properties of the passive thermal control system insulation will be measured to verify that no unacceptable degradation has occured. After each mission, it will likewise be verified whether or not environmental degradation has precluded its further use. Figure 4.7-2 gives an overview of the servicing equipment.

Access platform workstands, transportation covers,shipping platforms are also provided, as shown in Figure 4.7-3. Access devices are foreseen to guide and support men and equipment as they enter the Spacelab module interior being in either horizontal or vertical position, the latter after the Spacelab has been "stacked" on the Shuttle Orbiter.

Capability to remove the Spacelab from the vertical Orbiter will be provided. The method of accomplishing these operations is presently under study. The current preliminary concept is depicted in Figure 4.7-4.

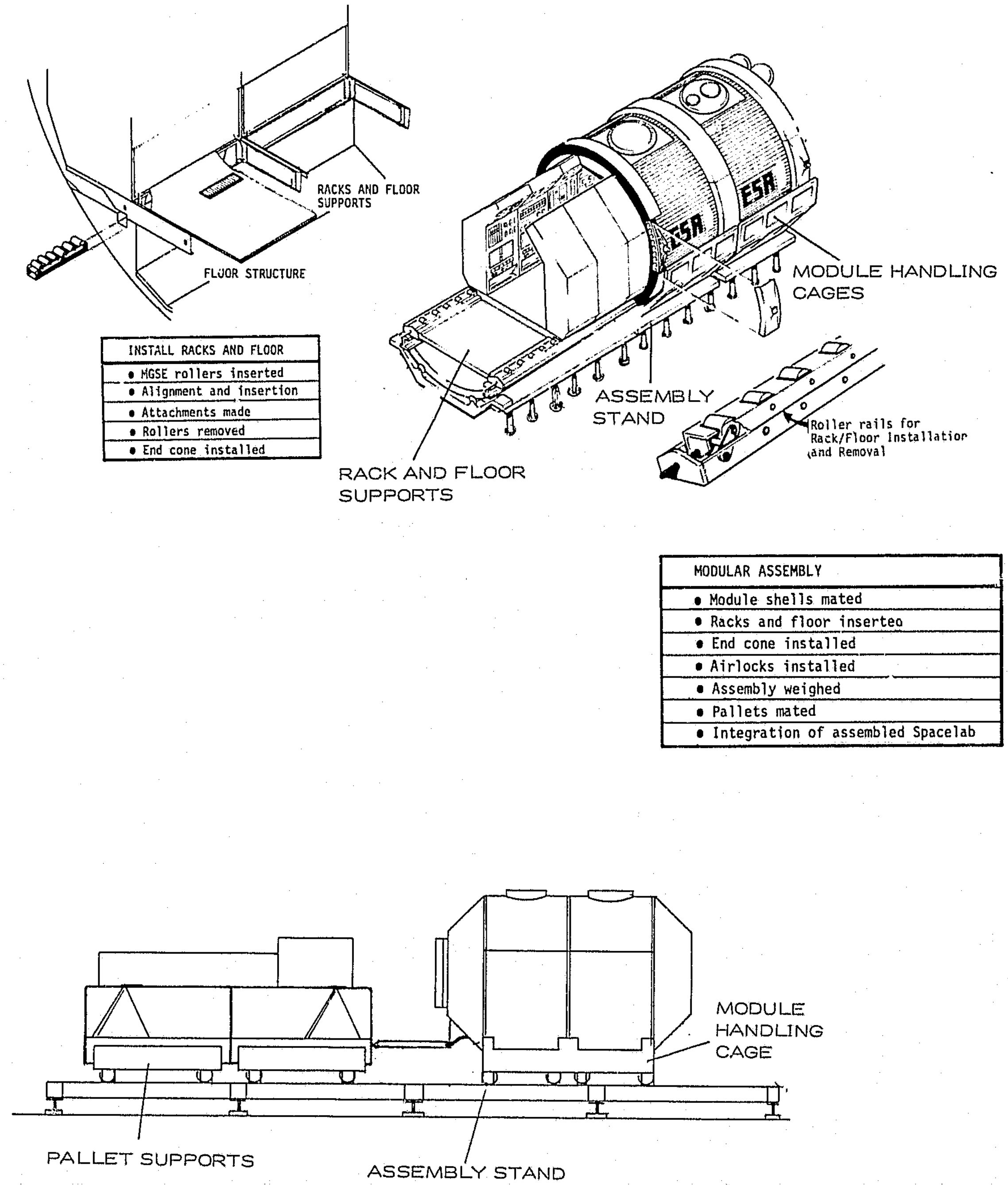


Figure 4.7-1: Matched-Rail Assembly Stand

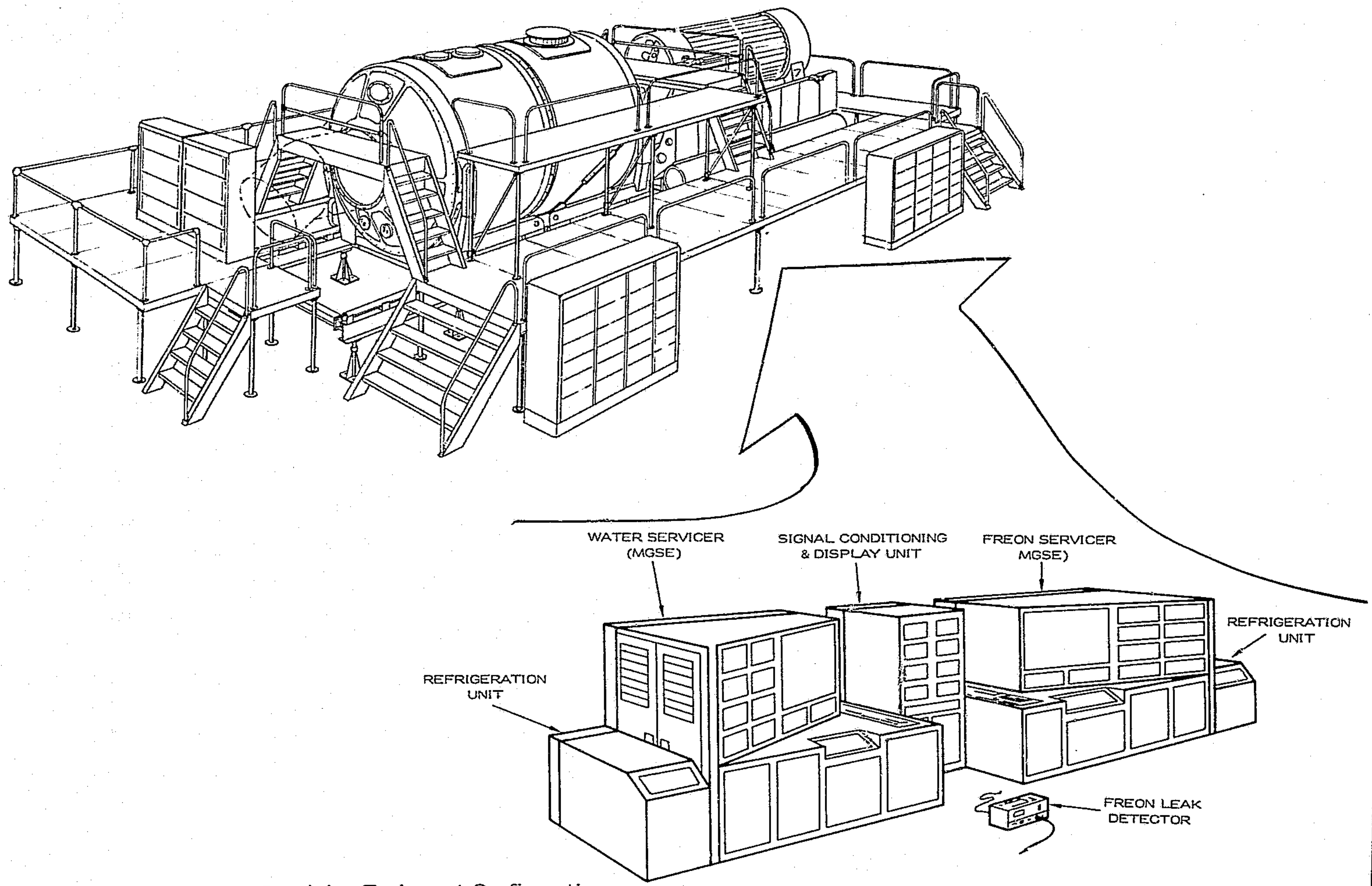
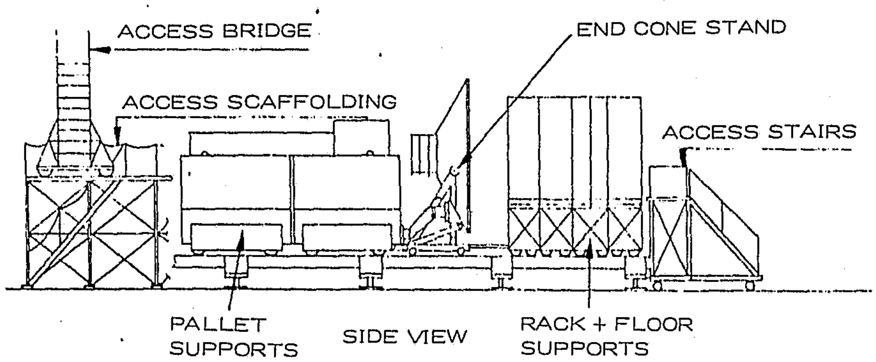


Figure 4.7-2: Servicing Equipment Configuration



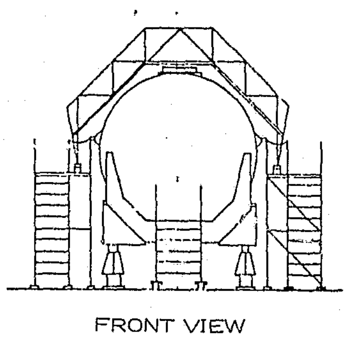


Figure 4.7-3: Example of Access Equipment

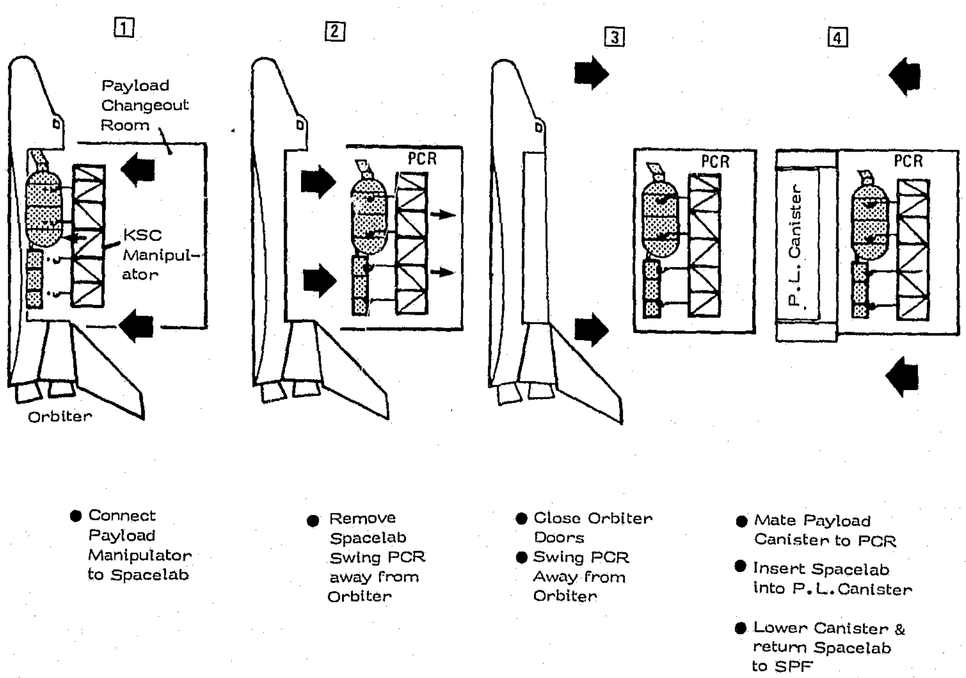


Figure 4.7-4: KSC Payload Changeout Concept

The Spacelab module and pallet are equipped with interface fittings to provide attachment points for the Shuttle payload ground removal manipulator and for possible use with the Shuttle payload canister. Additional NASA fittings and adapters will afterwards remove the Spacelab tunnel, utilities and possibly the EVA module (Orbiter airlock and tunnel adapter). The Spacelab interface fittings will be instrumented to determine the extent of load transfer at each point to avoid premature release of the Orbiter payload fittings. Personnel access to attach points will be provided in the payload changeout room which is pre-configured to the respective Spacelab configuration installed.

Editorial note: The detailed hardware content of the mechanical ground support equipment is currently under re-evaluation to adjust the MGSE support capability in accordance with revised operational ground rules.

4.7.2 Payload Support Capability

The MGSE is designed to provide support to the Spacelab system, its subsystems and various equipment elements. It is not specifically designed for the purpose of Spacelab payload support. However, it has payload support capability in the areas of transportation and access support.

4.7.2.1 Transportation Support

The MGSE is designed to support Spacelab and parts of Spacelab when they are integrated with experiments.

The Spacelab and its elements can be transported on-base on a special NASA transporter which provides the required shock and vibration protection. The transporter will be equipped for overhead loading and unloading operations. When off-base highway transport is desired, the Spacelab envelope can be reduced by removing the module cylindrical structure and placing the pallets, racks and floor structure on a special low-board transporter selected for the specified highway envelope. The required clearance envelope is shown in Figure 4.7-5.

Air transportation of the large envelope sections of Spacelab such as pallets or the experiment dedicated racks and floor structure is possible within the cargo envelope of the C 5-A. Individual segments and subsystems with lesser envelope requirements can be carried by the Transall C 160 or other suitable aircraft.

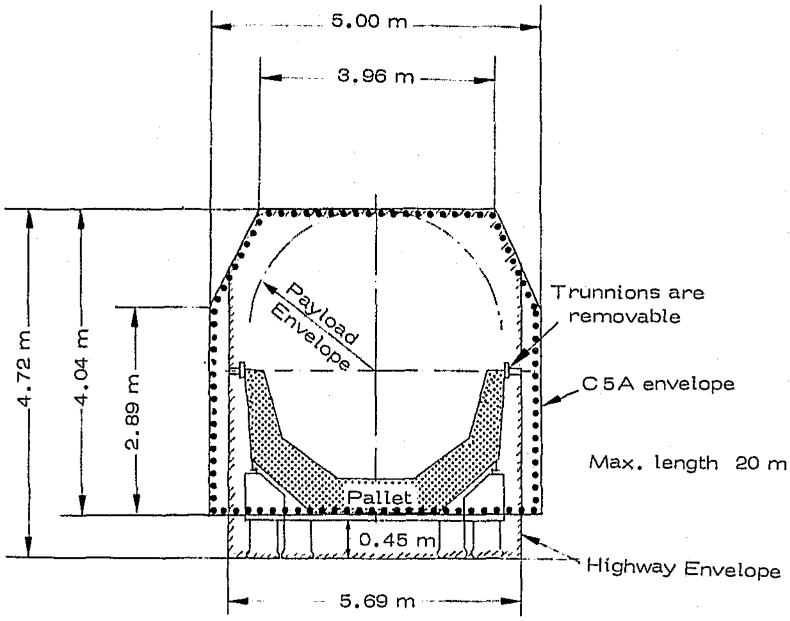


Figure 4.7-5: Specified Highway/C 5A Cargo Envelope for Pallet/Experiment Transportation

4.7.2.2 Access Support

The MGSE access equipment is designed to support the access to Spacelab subsystems and to its integrated payload externally and internally to Spacelab. This capability will include the internal access to the Spacelab Module during the period when Spacelab is integrated into the Orbiter with the Orbiter in the horizontal position and the cargo bay doors open. Access is supported by KSC provided facilities.

Access to Spacelab on the pad with the Orbiter in vertical position is planned to be supported by MGSE. The present baseline is restricted to contingency access through the Orbiter and the tunnel.

However, it is under discussion to provide late access to experiments in Spacelab through the core segment CPSE opening with the Orbiter cargo bay doors open.

This concept would exclude the use of the top airlock in the core module.

Access through the tunnel, nevertheless, will not be precluded. Figure 4.7-6 depicts one possible concept for MGSE supporting access into Spacelab in vertical position.

- Personnel and equipment are tethered for safety
- Emergency breathing air can be provided by back pack
- Forced air circulation through utility tray duct
- Dolly carries tools using ladder-mounted track guides
- Internally erected scaffolding required for many access tasks (LRU replacement, late stowage)
- All operations constrained by safety
- Spacelab lighting & communications usable for ground operations
- Visual contact with a safety observer is maintained at all times
- A portable winch assists in lowering equipment and personnel

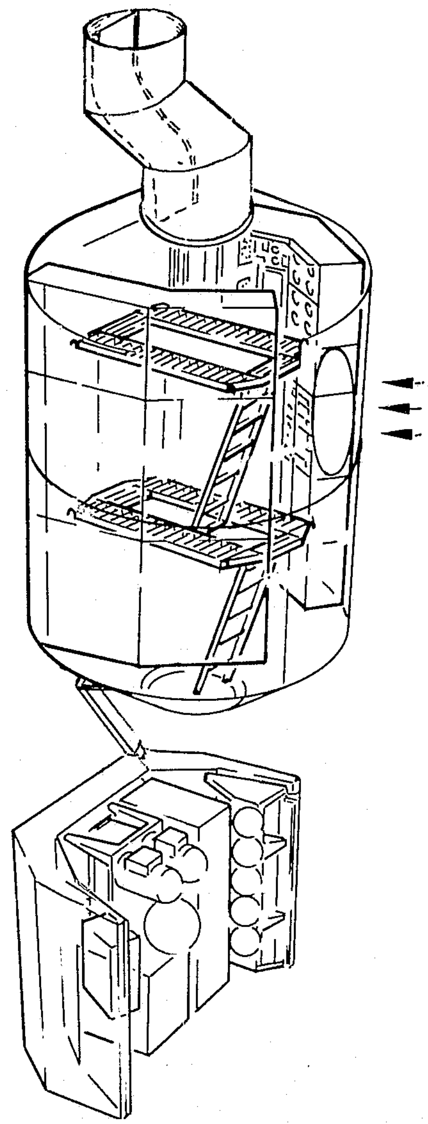


Figure 4.7-6: Access in Vertical Position

4.8 Electrical Ground Support Equipment

4.8.1 General

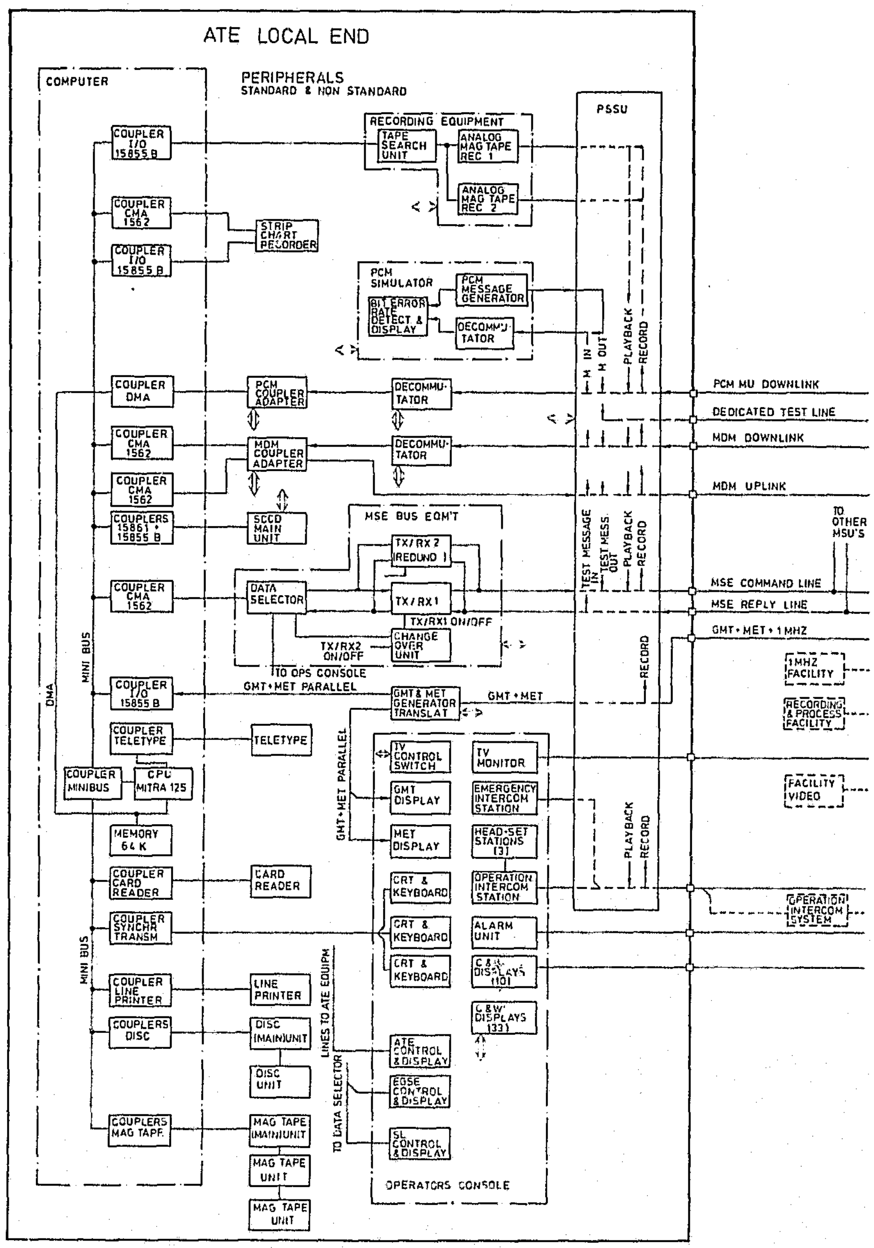
The Electrical Ground Support Equipment (EGSE) design is based on the use of computer controlled Automatic Test Equipment (ATE) augmented by simulators, in particular the Orbiter Interface Adapter (OIA) and the Core Segment Simulator (CSS). It is designed to support the Spacelab and its payloads during the integration Levels II and III. The primary purpose is to assure that the Spacelab subsystems are operating within their design limits; in addition, experiment integration and final verification is supported in conjunction with special experiment GSE, the latter not being part of the Spacelab program baseline.
Overall test control is implemented via the EGSE computer, the checkout software, and the EGSE to Spacelab and experiment interfaces. EGSE measurement, stimuli, recording and processing capabilities allow detailed testing and fault isolation of Spacelab and experiment equipment, and tasks such as data reduction, test result print-out and display. The EGSE to Spacelab communications are via simulated Orbiter I/O links. The Operator Console provides the principal man-machine interface to the EGSE as well as to the Spacelab during ground checkout.

4.8.1.1 Automatic Test Equipment

The ATE is that portion of the EGSE which, under computer control, configures the test set-ups for the different Spacelab integration levels, controls the test sequencing and performs the data acquisition, recording, decommutation, evaluation, display, and print-out. It also controls the generation and verification of commands, stimuli, and encoded data.

In support of minimum turn-around time requirements, the ATE self-tests and isolates malfunctions within EGSE to a line replaceable unit (LRU) level and is capable of isolating malfunctions in the Spacelab subsystems to the LRU. Identification of failed experiment LRU's is supported but depends on the test points provided within the experiments and on the software loaded in the ATE computer.

Figure 4.8-1 depicts an overall ATE block diagram.



Legend applicable for Figures 4.8 – 1 thru – 6

BOX	DATA TO S/S RAU FROM BOX
BOX	COMMAND FROM S/S RAU TO BOX
BOX	DATA TO MSU FROM BOX
BOX	COMMAND FROM MSU TO BOX
BOX	DATA TO SCCD FROM BOX
BOX	COMMAND FROM SCCD TO BOX
BOX	BOX CONTROLLED BY CSS CONTROL & DISPLAY PANEL
	INDEPENDENT CIRCUITS IN THE SAME BOX
	S/S AC POWER LINE
	S/S DC POWER LINE
	S/S EMERGENCY POWER LINE
	S/S ESSENTIAL POWER LINE
	C&W OR ADVISORY SENSOR
	INTERFACE
	MISSION DEPENDENT EQUIPMENT
	U S FACILITY SOURCES
I/S	INTERCONNECTING STATION THE MAXIMAL POSSIBLE NUMBER OF I/S'S IS SHOWN
NOTE	THIS DIAGRAM SHOWS FUNCTIONAL INTERFACES BETWEEN SL, ORBITER AND APPROPRIATE SIMULATORS THIS DIAGRAM REFLECTS THE CURRENT WORKING-LEVEL AND INCLUDES ALREADY EXPECTED CHANGES

Figure 4.8 – 1 Automatic Test Equipment

Editorial note: Diagrams are presently being updated and do not necessarily reflect the current status in all details

4.8.1.2 Orbiter Interface Adapter

The OIA acts as the primary EGSE to Spacelab interface during all test phases when the Spacelab is outside the payload bay. It simulates Spacelab related electrical Orbiter resources; functions not provided by the OIA, but generated in the ATE are routed through the OIA. The OIA interfaces directly with the Spacelab feedthrough plates.

The OIA includes a simulated Orbiter aft flight deck (AFD), by which Spacelab as well as experiment AFD equipment can be accommodated.

Fig 4.8 - 2 depicts an overall OIA block diagram with Spacelab AFD equipment included.

4.8.1.3 Core Segment Simulator

Editorial note : The core segment simulator (CSS) acts as primary EGSE to experiment train interface during all test phases, when the experiment train is outside the Spacelab, i.e. not connected either to the core segment or the igloo. The CSS capability is presently under extensive review.

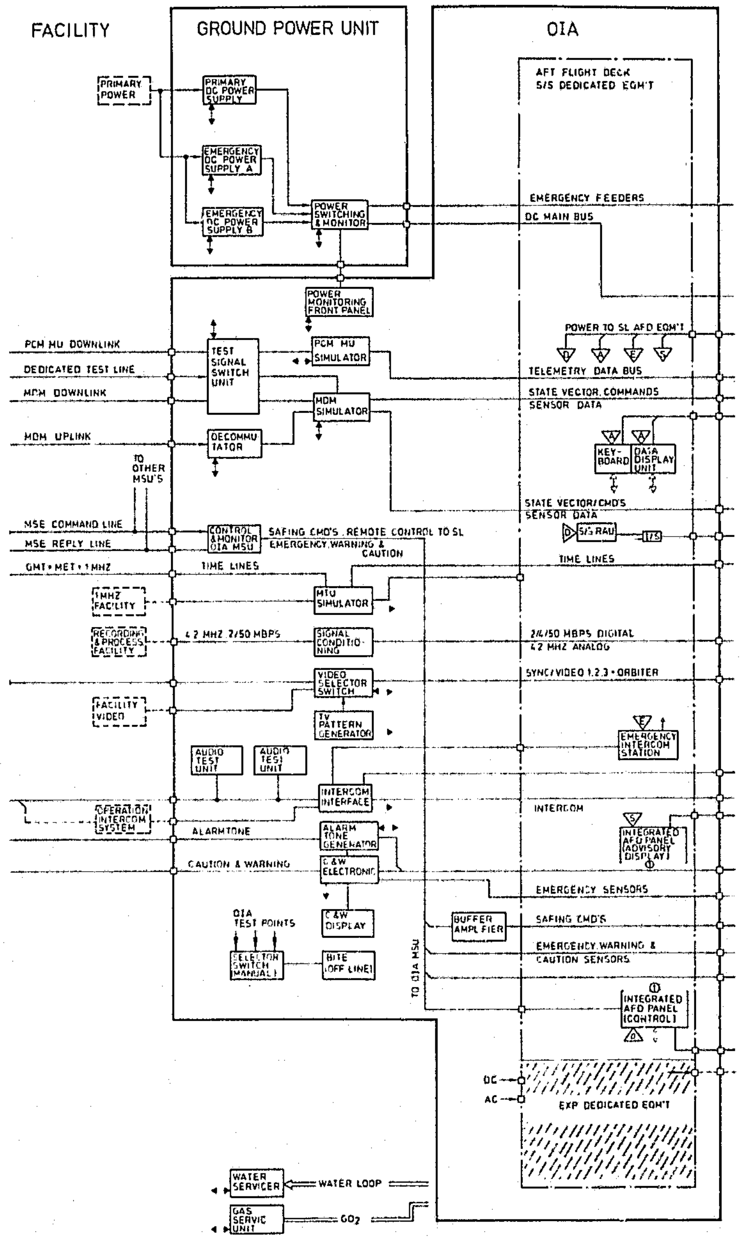


Figure 4.8 – 2 Orbiter Interface Adapter

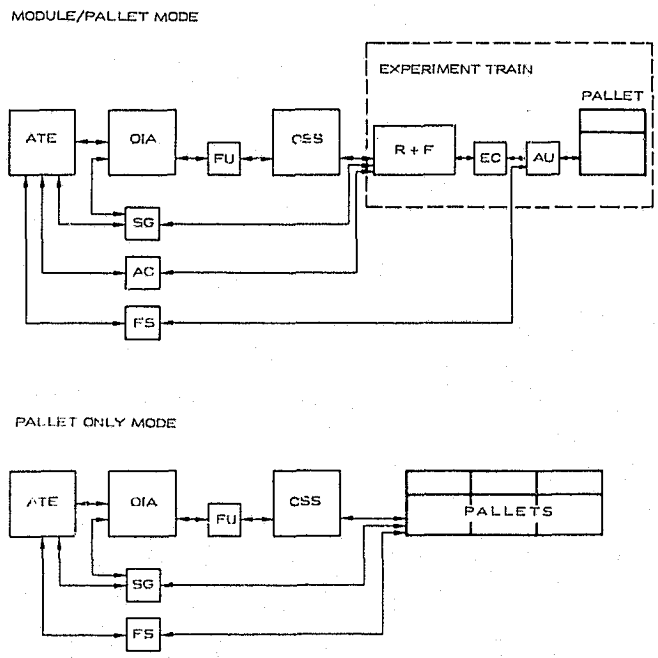
Editorial note: Diagrams are presently being updated and do not necessarily reflect the current status in all details

4.8.2 Utilization of EGSE

In support of the payload integration and test EGSE is used in various configurations for testing of different Spacelab modes at the following levels of integration. These integration levels are defined in the "Spacelab reference flow" and are described in more detail in Section 6 .

4.8.2.1 Level III Integration

Experiment assembly to verification will take place within the Level III - activities.
The applicable test configurations are presented in Fig 4.8 - 3 ; these configurations are capable of performing a complete hardware/software verification prior to the assembly of the Spacelab with the experiment train.



AC	-	Avionics Cooling Unit (GSE)
AU	-	Aft Utilities
EC	-	End Cone Simulator (GSE)
FS	-	Freon Servicer (GSE)
FU	-	Forward Utilities (GSE)
SG	-	Special Experiment GSE

Figure 4.8 - 3 Level III Test Configuration

In above test configurations, the experiments interface through the normal experiment-to-Spacelab interfaces with the EGSE, which replaces the core segment or igloo by its CSS. Experiments may also interface with special experiment GSE. The EGSE hardware provides in addition the capability to interface with its Measuring and Stimuli Unit (MSU) input/output channels directly with the special experiment GSE or with the experiment itself (see Figure 4.8 – 4). However, the data acquisition and stimuli generation capability depends on the availability of proper experimenter provided test software. The MSU output/input channel characteristics are defined in detail in para 4.8.2.4 .

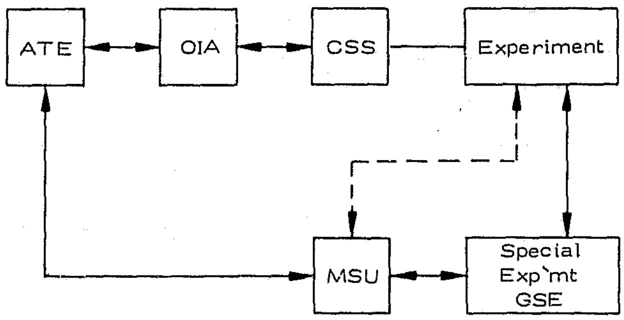
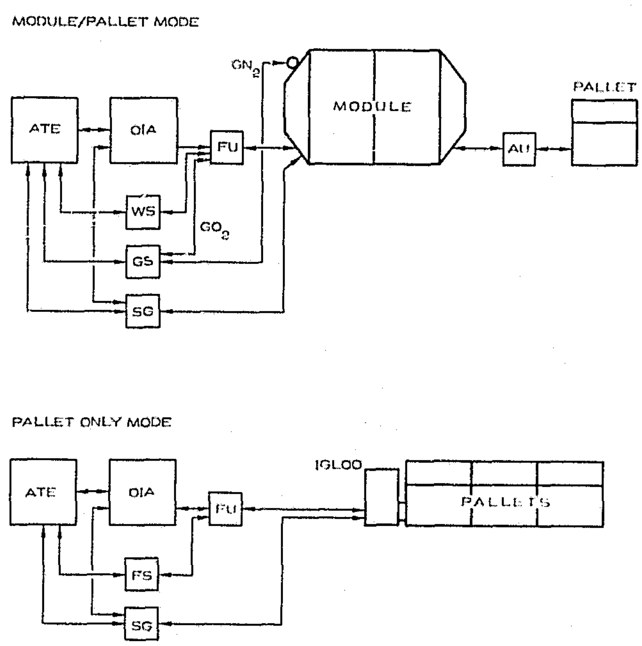


Figure 4.8 – 4 Test Configuration with Measuring and Stimuli Unit (MSU)

4.8.2.2 Level II Integration

In the pre-launch phase Spacelab/Experiment Assembly to Spacelab Close-Out will take place within the Level II - activities. The applicable test configurations are presented in Figure 4.8 - 5.



AU	-	Aft Utilities
FS	-	Freon Servicer (GSE)
FU	-	Forward Utilities (GSE)
GS	-	Gas Servicing Unit for GN_2 and GO_2 (GSE)
SG	-	Special Experiment GSE
WS	-	Water Servicer (GSE)

Figure 4.8 - 5 Level II Test Configurations

In above test configurations, the experiments are interfacing through the normal experiment to Spacelab and Spacelab to Orbiter channels with the EGSE; they may also interface with special experiment GSE. In addition, the EGSE hardware provides the capability to interface with its measuring and stimuli unit (MSU) input/output channels directly with the special experiment GSE or with the experiment itself, however, the data acquisition and stimuli generation capability depends on the availability of proper checkout software from experimenters.

In the post-flight phase EGSE supports Level II contingency activities in test configurations as described before.

Figure 4.8 - 6 illustrates the Spacelab Assembly Stand with associated EGSE. It encompasses the Local End EGSE: ATE, operator consols, computers etc, and the Far End EGSE : the ground power unit (GPU) and OIA . The Local End EGSE is physically apart from the actual assembly stand.

4.8.2.3 Level IV Integration

The EGSE is not designed to support Level IV activities, however, this support can be provided to the extent possible and practicable with the Level III support configuration.

4.8.2.4 MSU Interface Characteristics

MSU's are connected to a 256 Kbps up-/downlink bus. This data bus is to a great extent busy with GSE and Spacelab control and monitoring tasks, however, a limited amount of payload data can still be transferred via this bus assuming that the proper software is provided. The following three principal operating modes are possible:

- software controlled
- manual access via keyboard and CRT data display (software controlled)
- manual access via function keys and status indicators (hardware controlled) to discrete inputs/outputs

C-4

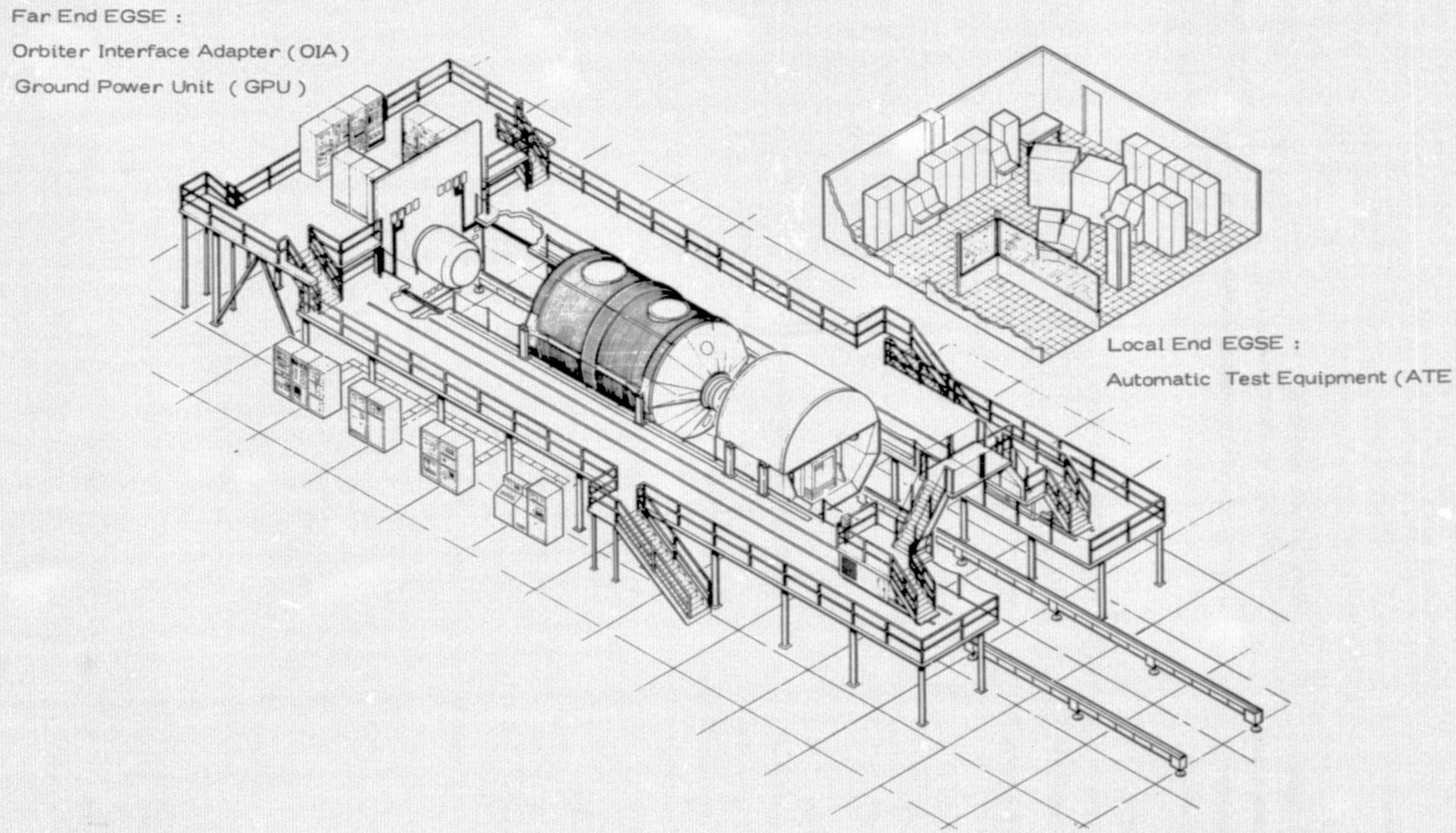


Figure 4.8 - 6   Spacelab Assembly Stand With Associated ESGE

MSU's provide the following user interfaces:

a) Measuring

Analog

Input type:	differential, isolated
Voltage range:	− 5.12 V to + 5.11 V line to line
Accuracy:	0.5 % full scale (for source impedances less than 500 Ohm)
Resolution:	10 bits + 1 sign bit

Discrete

Input type:	differential, isolated
Voltage level "1" :	$V_{IN} = 4\ V^{+2\ V}_{-1.5\ V}$
Voltage level "0" :	$V_{IN} = 0 \pm 0.5\ V$

b) Stimuli

Analog

Output type:	single ended, sample and hold
Voltage range:	− 5.12 V to + 5.11 V line to ground
Driving capability:	2 kiloOhms
Refreshing cycle time:	1 sec
Accuracy:	0.5 % full scale

Discrete

Output type:	single ended
Voltage level "1":	$V_{OUT} = 5\ V^{+1\ V}_{-0.7\ V}$ for $I_{OUT} \leq 20$ mA
Voltage level "0":	$V_{OUT} = 0 \pm 0.5\ V$ for $I_{OUT} = 0$

4.9 Instrument Pointing Subsystem

The Instrument Pointing Subsystem (IPS) provides precision pointing for payloads which require greater pointing accuracy and stability than is provided by the Orbiter. The IPS can accommodate a wide range of payload instruments of different sizes and weights.

Editorial note: A formal commitment by ESA on the implementation of the IPS has been made. Although the IPS will be handled somewhat differently from other Spacelab subsystems, it will be delivered under the same general terms as the Spacelab and will be available for use on the second and subsequent flights.

4.9.1 IPS Description

The IPS contains a three-axis gimbal system (azimuth-, cross elevation- and elevation-axis) as illustrated in Figure 4.9 - 1 and - 2 . It is mounted to the floor of a pallet. The payload requiring high precision pointing is attached to the IPS via a Payload Integration Ring. A Soft Mount consisting of springs and dampers is inserted between the gimbal system and the pallet in order to reduce attitude disturbance caused by Orbiter thruster firings, crew motion etc. During launch and return, the Soft Mount is inoperative and locked by the soft mount clamp. The IPS concept, as shown in Figure 4.9 - 1 , has the capability to accommodate easily different sizes and types of payloads.

During launch and return, the payload is physically separated from the IPS by the Payload/Gimbal Separator Mechanism in order to avoid inputs of flight loads to the payload from the IPS. The payload is supported by a Payload Clamp Assembly which distributes the flight loads exerted by the payload to pallet hardpoints (Figure 4.9 - 3). This assembly consists of three clamping devices, one half of each clamp being attached to the payload, the other half being supported by the load carrying members of the Payload Clamp Assembly. The three clamping devices need to be arranged in a plane containing the C.G. of the payload. Payloads with various dimensions can be accommodated by utilizing differently sized load carrying members of the Payload Clamp Assembly.

Overall attitude control of the payload is based on rate integrating gyros (RIG) error signals processed within the Spacelab computer (CDMS) to generate command signals to DC torquers in each gimbal axis.

The RIG package is located on the outer gimbal and hence, aside from distortion or flexures occuring within the payload, maintains the payload as an inertially stabilized platform.

To correct for gyro drift and to provide an absolute attitude reference, a package of optical sensors is also included. In a stellar mission this would comprise three star trackers, and in a solar mission one star tracker would be replaced by a solar sensor. The package will normally be located on the payload to assure the optimum viewing configuration and to minimize misalignments between star-tracker and experiment line-of-sight axes for those payloads relying on mechanical alignment. Mechanical mounting methods do not allow pointing accuracies of the order of a few arc seconds to be achieved. Therefore, the optical sensors allow for the input of a simulated star image or an electrical signal from the payload into the sensor as an indication of the experiment line-of-sight (LOS).

A correction for the offset from the star-tracker LOS is then made in the software processing of the star-tracker data in the CDMS. Alignment errors between the experiment line-of-sight and the reference guide star are in this way minimized, and the best possible pointing accuracy within the inherent capabilities of the star-tracker and experiment optics is achieved.

The major design features outlined here have incorporated considerable flexibility for payload physical parameters and have utilized the zero-gravity operating environment to provide a compact, low-weight system capable of controlling payloads which far exceed it in size. As an outcome of this, however, constraints and limitations are placed on the extent of 1 g testing and pre-launch check-out which can be performed. Removal of the payload and the addition of balancing weights allow all slewing and pointing functions to be exercised under computer control, as in orbit. However, it is expected that the increased friction torque in the drive-assemblies as well as imperfect balancing will allow only a limited assessment of the IPS zero-g performance capability.

Preliminary data of the mass and power budget are presented in Table 4.9-1. The mass data apply to an IPS configuration to accommodate a nominal 2000 kg payload.

Table 4.9-1: Mass/Power Budget of IPS

Assembly	No.	Mass (kg)	Power (W) Mean
Gimbal Structure	1	256	0
Drive	3	126	21
Thermal Control		10	50
Payload Clamp	1	144	0
Attitude Measurement Sensors	1	57	88
Power Electronics	1	27	90
Data Electronics	1	24	28
MARGIN		106	43
TOTAL		750	320

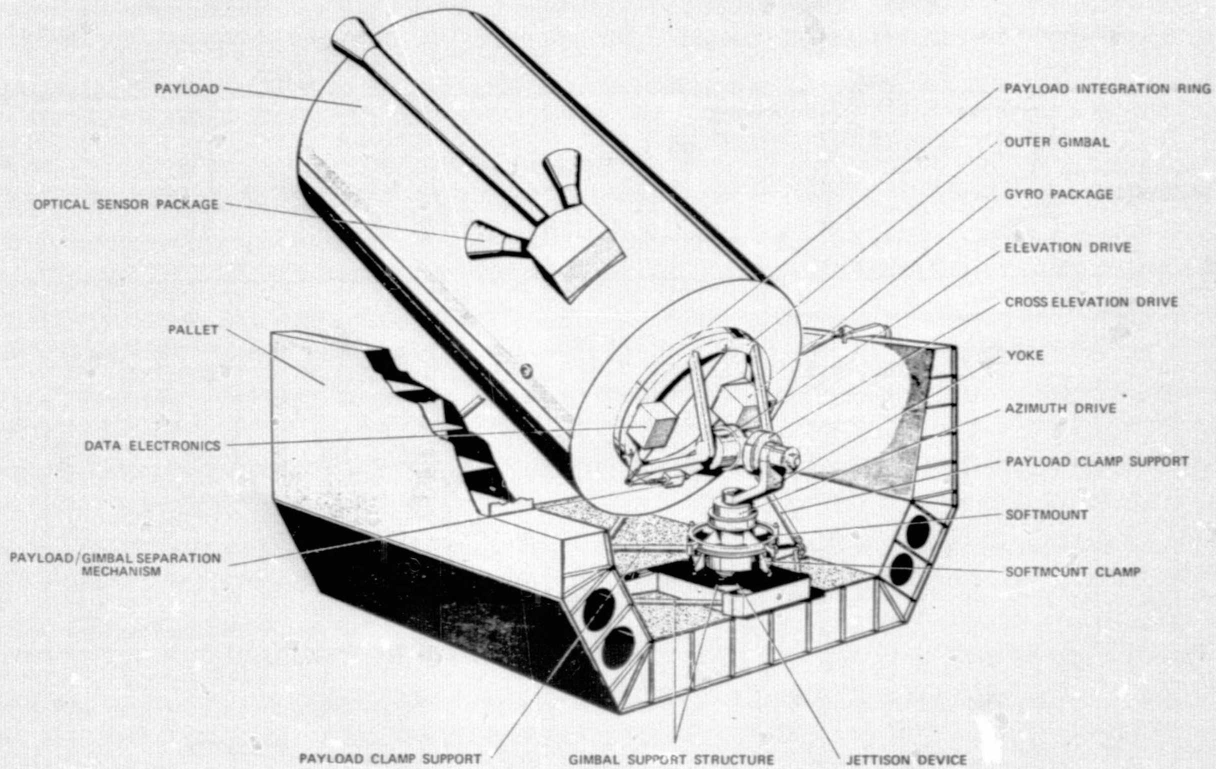


Figure 4.9-1: Instrument Pointing Subsystem

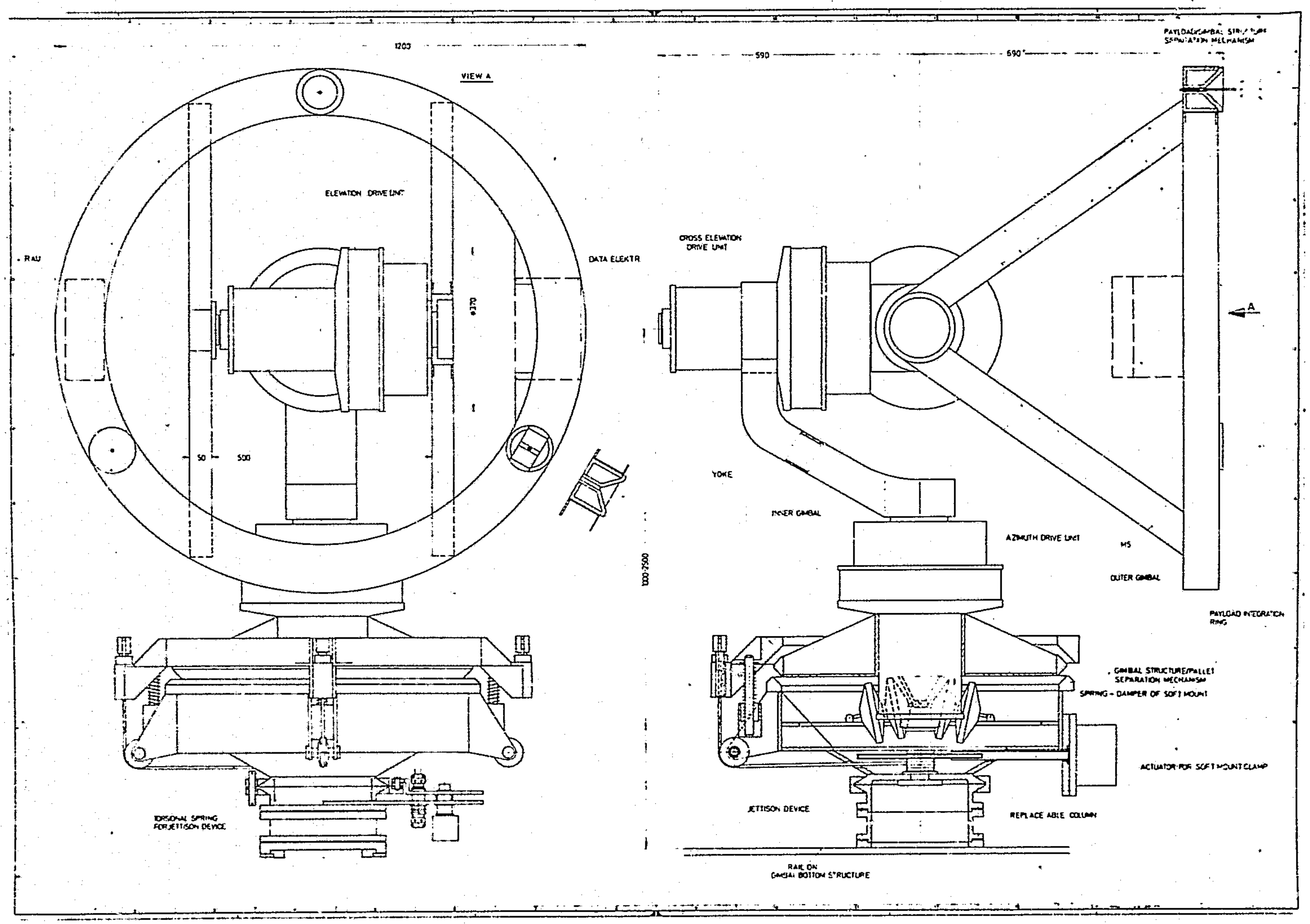


Figure 4.9-2: Gimbal Structure

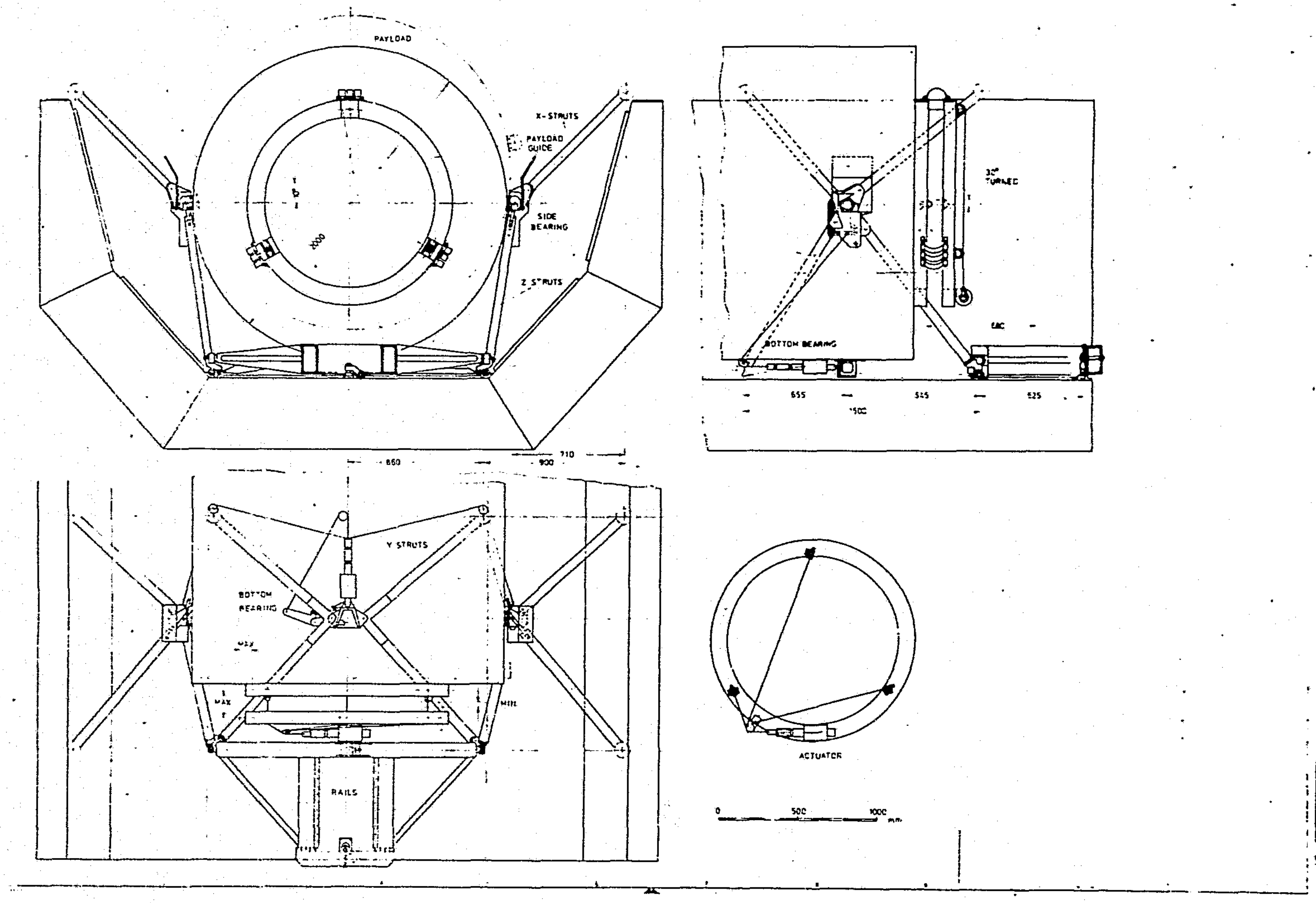


Figure 4.9-3: IPS Payload Clamp Assembly

4.9.2 Payload Accommodation

4.9.2.1 Payload Mass Properties

The IPS is capable of mounting and distributing the load of a nominal 2000 kg payload and the IPS into a single unmodified pallet without exceeding safe loading conditions. The non-removable structural elements of the Payload Clamp Assembly are capable of accommodating payload masses up to a maximum of 3000 kg. In addition, the IPS is capable of performing all normal pointing functions for payloads of mass greater than 3000 kg, if they are supported throughout launch and landing by a separate clamp system.

The IPS is capable of accommodating payload CG locations equivalent to a 5 cm radial displacement in the Y_c–Z_c–plane of the CG of 3000 kg payload from the center of a 3 m diameter circle supported in the clamp system, and is capable of accommodating a 10 cm offset in X_o–direction between the CG of a 3000 kg payload and Y_o–Z_c–clamping plane.

4.9.2.2 Payload Dimensions

The IPS payload clamp assembly is capable of accommodating payloads from 0.5 to 3 m diameter in the Y_c–Z_c plane. It is a design goal to minimize the dimension constraints imposed on the payloads by the clamp assembly. The clamp assembly is capable of adaption to different payload Y_o–Z_o dimensions by the replacement of structural elements within the assembly. The IPS is capable of accommodating distances between the payload CG and the IPS/payload interface plane ranging from 0.5 to 3 m.

The overall dimensions of the IPS gimbal structure including drives are smaller than 1.5 m x 2 m x 3.5 m in x_c–, y_o– and z_o–directions at maximum distance between the center of rotation and the pallet floor.

(see para 4.9.2.5)

4.9.2.3 Pointing and Stabilization

The IPS provides 3-axis attitude control and stabilization for experiments. The characteristics of nominal 2000 kg and 200 kg payload are given in Table 4.9-2 and these are used as design reference payloads except when a requirement specifically states otherwise. Error requirements apply during solar and stellar fine pointing of the IPS, with the Orbiter in either an inertially stabilized mode or a free-drift mode with angular rates up to 10 deg/min with respect to inertial space. The values of bias and quiescent stability error shall mean, in each case, that the probability of the error being less than the required value is 67%. Bias error shall include all sources of error witn time constant equal or greater than one orbital period.

Bias Error

The bias or reference error of the experiment LOS with respect to a reference star or an idealized solar disk is less than 2 arc-sec (design goal 0.8 arc-sec) in the two axes perpendicular to the experiment LOS and less than 40 arc-sec (design goal 15 arc-sec) in roll about the experiment LOS.

Table 4.9-2: Characteristics of Nominal 2000 kg and 200 kg Payloads

	LARGE PAYLOAD	SMALL PAYLOAD
Mass	2000 kg	200 kg
Dimensions	2 m Ø x 4 m	1 m Ø x 1.50 m
Moment of inertia about payload CG:		
about axis perp. to LOS	1200 kgm^2	20 kgm^2
about LOS axis	1000 kgm^2	25 kgm^2
CG offset from center of rotation of gimbal axes:		
along LOS	2.5 m	1.50 m
perp. to LOS	0.30 m	0.10 m

Quiescent Stability Error

The quiescent stability error shall be less than 1.0 arc-sec (design goal 0.33 arc-sec) in the two axes perpendicular to the experiment LOS, and 3 arc-sec (design goal 1.6 arc-sec) in roll about the experiment LOS. These values apply for all angles within the LOS range and for both the nominal 2000 kg and 200 kg payloads without disturbances from the Shuttle.

Disturbance Response Errors

The disturbance response errors discussed herein are defined as including the quiescent stability error and apply for the nominal 2000 kg payload, with the IPS located at the forward end of a five pallet train. For a nominal 200 kg payload the disturbance error is less than 1.7 times higher.

Man Motion Disturbance

The disturbance error (peak value) due to a standardized man motion disturbance is less than 3 arc-sec (design goal is 1 arc-sec) in the two axes perpendicular to the experiment LOS and less than 10 arc-sec (design goal is 4 arc-sec) about the roll axis.

These values are achieved at any attitude within the LOS range.

Orbiter Limit Cycle Disturbance

The limit cycle errors (peak value) in each axis due to Shuttle limit cycle motion of $\pm$ 0.1 degree and 30 m sec duration thruster firing (for a nominal 2000 kg payload under the same IPS location constraint and LOS pointing range conditions as given above) are still under study. Preliminary analyses have shown the errors are not greater than those caused by man-motion disturbances.

Stability Rate

During fine pointing, the peak stability rate is less than 2 arc-min/sec for a nominal 2000 kg payload. This is an instantaneous value occuring during the response of the IPS to a man-motion disturbance; normally occuring values are significantly lower.

Pointing Range

The IPS has a LOS pointing range of at least π steradians without a payload. The range of roll angle about the experiment LOS is at least π radians at any position within the π steradians LOS pointing range.

Scan and Earth Pointing Modes

The performance in these modes is based on the performance as listed above.

4.9.2.4 Payload Supporting Services

The IPS will meet the pointing requirements of para. 4.9.2.3 while providing the following supporting service interfaces across the gimbal system for the use of payloads:

- wiring for three independent 200 watt power loads at either 28 VDC or 115 VAC
- wiring for three remote acquisition units (RAU)
- 20 coaxial cables (equivalent to type RG 178b) each adequate for transmission of the Orbiter max. data rate.

Mounting provisions are provided on the payload side of the gimbals for mounting of 3 RAU's dedicated for payload use.

4.9.2.5 Flexibility and Growth Potential

The IPS is adaptable, by the replacement of simple structural elements, to mounting with the center of rotation at different distances above the pallet floor within the range of 1.3 m to 2.5 m. Mechanical provisions are available at appropriate locations on each side of the gimbal system for the addition of other payload supporting services, beyond those specified in 4.9.2.3 with some resultant degradation of performance.

The IPS optical sensor package is capable of operation when mounted on the IPS structure or on a reference surface provided by the payload.

The optical sensor package includes the capability to have the two roll sensors LOS at a skewed angle of either 45 degrees or 12 degrees with respect to the LOS of the centrally mounted optical sensor. The LOS's of all three optical sensors are arranged in one plane. Provision is also made for the mounting of a light baffle system, designed for specific mission conditions, at the aperture of each optical sensor but structurally decoupled from the sensor.

4.9.3 IPS Interfaces

4.9.3.1 Spacelab/Orbiter Interface

- Mechanical Interface

  The IPS provides a retention point for the Orbiter Remote Manipulator System, in such a position as to allow replacement of the payload into the stowed configuration in case of gimbal disablement, and controlled release during an emergency mode jettison.

- Software

  The IPS software is capable of interfacing via CDMS with the Orbiter data handling system to allow:

  - determination of the angular relationship between the IPS coordinate system and the Orbiter IMU coordinate system.
  - transmission via CDMS to the Orbiter of IPS orientation and angular rates..
  - receipt from the Orbiter of ephemeris data, Orbiter state vector data and timing data.

4.9.3.2 Spacelab/Payload Interfaces

- Mechanical Interfaces

  The IPS/Payload mechanical interface provides maximum access to the payload within the constraints imposed by the launch and landing environments, the pointing requirements, and the mounting of IPS components.

  The IPS design does not impose the loads of the IPS gimbal system on the payload during launch and landing, within a reasonable limitation that some IPS hardware (in addition to the clamping system and the optical sensor package) will remain attached to the payload during these periods.

- Electrical

  The IPS provides the interface and the capability to accept experiment generated control signals for use in bias error control.

4.9.3.3 Spacelab Ground Support

- EGSE

  The IPS EGSE is capable of performing all test, checkout and integration activities necessary prior to IPS/Spacelab integration or IPS/payload integration and will support maintenance of the

IPS to intermediate maintenance level (one level lower than line replaceable unit). Stimulation and verification of all IPS/payload and IPS/Spacelab interfaces is possible. Simulation of the Spacelab or payload hardware is not provided, except as required for adequate IPS check-out.

- MGSE

  MGSE is provided for IPS storage, transportation,handling and mating with Spacelab subsystems and payload. Any special purpose tools or fixtures required to mate IPS assemblies or align critical equipment are provided, but normal laboratory or capital equipment may be required in addition.

4.9.3.4 Spacelab Subsystem Interfaces

The IPS interfaces with the CDMS for data management and transfer functions. The IPS control system utilizes the subsystem CDMS for all normal operations, with emergency control being exercised from a separate IPS control panel.

4.9.4 Habitability and Cleanliness Requirements

The IPS specific habitability and cleanliness requirements are covered by the general Spacelab system requirements.

4.9.5 Environment

During orbital operations, the IPS is capable of continuously operating in full solar illumination or in a completely shadowed configuration.

4.9.6 Testing

Adequate and comprehensive testing of the IPS will be performed under the following guidelines:

4.9.6.1 Testing During Design and Development

- No zero-gravity testing of a completely assembled IPS will be required.
- Performance testing of a completely assembled IPS is considered highly desirable. Pointing performance testing may be limited to those configurations and those parameters for which the one-g performance is accepted as relateable to the zero-g performance but will include, as a minimum, closed loop performance under gyro control, sensor stimulation, slewing verification, verification of range/rate safety overrides, emergency control mode and all redundancy switching functions.
- All performance parameters entering into mathematical simulation of sub-system performance will be measured at the component level in one-g in such a manner as to provide the best insight into their zero-g performance.

4.9.6.2 Pre-Launch Test and Check-Out

- Complete performance testing of the fully assembled IPS will not be required prior to each launch.

- Off-line functional testing of the fully assembled IPS prior to each launch will be possible in one-g using only IPS GSE . This test will exercise all functions, (although not over their complete range) and it may be performed without a payload if desired.

- Level III Payload Integration

  After IPS integration with the experiment train (Spacelab experiment racks and pallets) as a minimum, interface verification will be required. Spacelab system level GSE will be used to perform LevelIII payload integration, augmented by IPS subsystem GSE as required.

- Level II Payload Integration

  After integration of the experiment train (including IPS) with the Spacelab core segment, as a minimum verification of interfaces with the Spacelab subsystems will be required, and status check-out of IPS for fault isolation to the LRU level will be required.These tests will be possible using Spacelab System level GSE only.

  Functional testing of IPS at this stage of integration, is considered desirable within the constraint that only LevelII Spacelab System Level EGSE is required.

- Level I Payload Integration

  After installation of a fully integrated Spacelab into the Orbiter, as a minimum, verification of any interfaces with the Orbiter will be required, and status checkout for fault isolation to the LRU level will be required. It is necessary that no IPS subsystem GSE or Spacelab System GSE be required for this task.

  Functional testing of IPS at this stage of integration, is considered desirable desirable within the constraint that no Spacelab or IPS GSE will be required. This task will be performed by the Launch Processing System through Orbiter and Spacelab Subsystems.

4.9.7 Software

For development of the IPS, a three-axis IPS simulation will be provided, which includes the effects of payload physical and structural characteristics.

For IPS operations, two software sets will be provided:

a) Ground Operating Software Set
This set will allow verification of the IPS function and partial performance at the different integration levels and prior to each launch.

b) Flight Operating Software Set
This set will serve for on-orbit checkout and monitoring, and for normal on-orbit operations during the different IPS operational modes.

All IPS software, which is processed in the CDMS, will be written in the high order language HAL/S except where a special requirement makes the use of assembler language coding necessary . The Flight Operating Software will be capable of interfacing via the CDMS with the Orbiter data handling system.

4.9.8 Operations

The IPS will be designed for an operational life of 10 years or 50 missions. To meet this requirement, periodic ground maintenance of critical IPS components and their replacement/refurbishment as required, is allowed.

Normal on-orbit operation of the IPS will be exercised from the CDMS console and will be possible from the Spacelab module or the Orbiter aft flight deck.

4.9.8.1 Operating Modes

The IPS is capable of being operated in the following modes:

- Stowed mode
- On-orbit check-out
- Acquisition of a reference (guide) star or sun
  The probability of achieving pointing to any position on the celestial sphere for any given roll orientation will be greater than 95 % when all sensors have an unobstructed field of view, and the probability of false acquisition will be less than 5 % when the average background illumination at the star-sensor aperture is equivalent to 900 stars of 10th magnitude per square degree.
- Inertially fixed fine pointing (including solar pointing) for periods of up to 90 minutes.
- Solar off-set pointing over an angular range of at least $\pm$ 25 arc-min with respect to the center of the solar disk.
- On-orbit alignment measurement
  Measurement of the misalignment between the payload LOS reference and the optical sensor package reference axis, and use of these misalignment angles to achieve correct payload LOS pointing will be possible. This will be achievable in two ways:
  a) The IPS accepts a payload-generated electrical signal to be compared with IPS reference star sensor signals.
  b) The IPS accepts a payload-generated simulated star image(s) into the star sensor field of view.

- Manually controlled slewing using the CDMS keyboard.
- Scan mode. The IPS will be capable of executing programmed scan motions of dimensions up to the size of the optical sensor FOV while maintaining continuous attitude control utilizing a celestial object. When the scan motion exceeds the reference optical sensor capabilities, control will be by the gyros only.
- Earth pointing mode
  The IPS will be capable of tracking an earth-fixed target using payload supplied attitude error signals with the Orbiter at a minimum attitude of 200 km in a z-local vertical orientation (within the IPS torque limitation of 20 Nm per axis)
- Emergency retraction into stowed configuration
- Emergency jettisoning

4.9.8.2 Emergency Control

The IPS will include dedicated equipment to assure the safety of the crew and vital Orbiter and Spacelab equipment. Provisions will include manual as well as automatic recognition and control of failures and unsafe operation. Emergency control will provide for

a) payload retraction and locking with positive locking indications

b) safe jettison of equipment which constitutes a hazard.

Provision will be made to perform these emergency control functions from the Orbiter aft flight deck, after emergency evacuation of the Spacelab module has occured with the IPS left in a normal operating mode, and without use of the CDMS console.

5. PAYLOAD ENVIRONMENT

This section describes the natural and induced environments that the Spacelab payload may be exposed to during operation with the Spacelab system. It must be emphasized that the induced environment data are preliminary and will be updated as Orbiter and Spacelab test results become available.

5.1 Module Equipment/Flight Environment

The environments described hereafter refer to all equipment which is located at all times inside the pressurized module.

5.1.1 Vibration

5.1.1.1 Sinusoidal Vibration

- System Level

Events such as gust loading, engine ignition and cutoff, separation and docking will induce low frequency transient responses in the Space Shuttle vehicle. The overall effect of these transients is on the assembled Spacelab / Payload system may be accounted for by a swept sinusoidal vibration environment . .Assuming 50 flights, the equivalent environment is as follows:

Frequency Range:	5 – 35 Hz
Level:	$\pm$ 0.25 g 0 to peak
Sweep Rate:	1 Oct/min. (1 sweep only up and down)
Axes:	3 principal axes

- Equipment Level

Assuming 50 flights, the equipment mounted in the Spacelab module will be exposed to an environment equivalent to the following:

Table 5-1: Sinusoidal Vibration Level

AXES	FREQUENCY	LEVEL
X	5 – 8.5 Hz 8.5 – 35 Hz 35 – 50 Hz	20 mm (0.80in) peak-to-peak 3 g 0 to peak 1 g 0 to peak
Y Z	5 – 8.5 Hz 8.5 – 35 Hz	20 mm (0.80in) peak-to-peak 3 g 0 to peak
Sweep Rate:	2 Oct/min.	5 sweeps up and down

- Equipment which will be flown less than 50 times will be exposed to more favorable environments than those outlined above. Details are TBD .

5.1.1.2 Random Vibration

Maximum vibration levels occur during the launch phase. Equipment within the Spacelab module will be subjected to vibration arising from the overall acoustic level inside the cargo bay, and to a very minor degree from vibration transmitted through the Orbiter/Spacelab mounting fixtures into the module structure.

The vibration level for any particular equipment depends on the location within the module, the mounting configuration and the equipment weight. The vibration level for equipment mounted in a standard rack is given in Table 5-2. The rack is assumed to be fully loaded..The vibration levels exist for approximately 6 seconds per mission (at lift-off).

Table 5-2: Random Vibration Environment for Module Mounted Equipment

LOCATION	FREQUENCY	LEVEL
Zone 1 a) Module Wall Mass of equipment < 15 kg	20 Hz 20 – 100 Hz 100 – 470 Hz 470 – 920 Hz 920 – 2000 Hz Composite:	0.00085 g^2/Hz + 9 dB/oct 0.10 g^2/Hz – 9 dB/oct 0.013 g^2/Hz 8.43 g RMS
Zone 1 b) Module Wall Mass of equipment ≥ 15 kg	20 Hz 20 – 80 Hz 80 – 470 Hz 470 – 920 Hz 920 – 2000 Hz Composite:	0.00085 g^2/Hz + 9 dB/oct 0.047 g^2/Hz – 9 dB/oct 0.0064 g^2/Hz 5.86 g RMS
Zone 1 c) Input to equipment mounted in Racks and Film Storage	20 Hz 20 – 60 Hz 60 – 300 Hz 300 – 590 Hz 590 – 1400 Hz 1400 – 2000 Hz 2000 Hz Composite:	0.00085 g^2/Hz + 9 dB/oct 0.021 g^2/Hz – 9 dB/oct 0.0027 g^2/Hz – 9 dB/oct 0.00092 g^2/Hz 3.3 g RMS
Zone 2 a) Endcones Mass of equipment < 15 kg	20 Hz 20 – 100 Hz 100 – 400 Hz 400 – 780 Hz 780 – 2000 Hz Composite:	0.00052 g^2/Hz + 9 dB/oct 0.064 g^2/Hz – 9 dB/oct 0.0084 g^2/Hz 6.37 g RMS
Zone 2 b) Endcones Mass of equipment ≥ 15 kg	20 Hz 20 – 85 Hz 85 – 400 Hz 400 – 780 Hz 780 – 2000 Hz Composite:	0.00052 g^2/Hz + 9 dB/oct 0.040 g^2/Hz – 9 dB/oct 0.0052 g^2/Hz 5.08 g RMS
Zone 3 a) Input to equipment mounted on floor hard points (beams) Mass of equipment < 15 kg	20 Hz 20 – 150 Hz 150 – 900 Hz 900 – 2000 Hz 2000 Hz Composite:	0.0028 g^2/Hz + 3 dB/oct 0.021 g^2/Hz – 12 dB/oct 0.00088 g^2/Hz 4.77 g RMS
Zone 3 b) Input to equipment mounted on floor hard points (beams) Mass of equipment ≥ 15 kg	20 Hz 20 – 60 Hz 60 – 900 Hz 900 – 2000 Hz 2000 Hz Composite:	0.0028 g^2/Hz + 3 dB/oct 0.0084 g^2/Hz – 12 dB/oct 0.00090 g^2/Hz 3.10 g RMS
Zone 3 c) Input to equipment mounted on floor honeycomb. Mass of equipment < 15 kg	20 Hz 20 – 150 Hz 150 – 350 Hz 350 – 2000 Hz 2000 Hz Composite:	0.010 g^2/Hz + 3 dB/oct 0.076 g^2/Hz – 6 dB/oct 0.0022 g^2/Hz 6.54 g RMS
Zone 3 d) Input to equipment mounted on floor honeycomb. Mass of equipment ≥ 15 kg	20 Hz 20 – 37 Hz 37 – 700 Hz 700 – 2000 Hz 2000 Hz Composite:	0.010 g^2/Hz + 3 dB/oct 0.019 g^2/Hz – 6 dB/oct 0.0022 g^2/Hz 4.63 g RMS

5.1.2 Acoustic Noise

The estimated acoustic spectrum inside the cargo bay during launch is assumed to be the same for all Spacelab locations (module - and pallet -locations) and is shown in Figure 5 - 1.

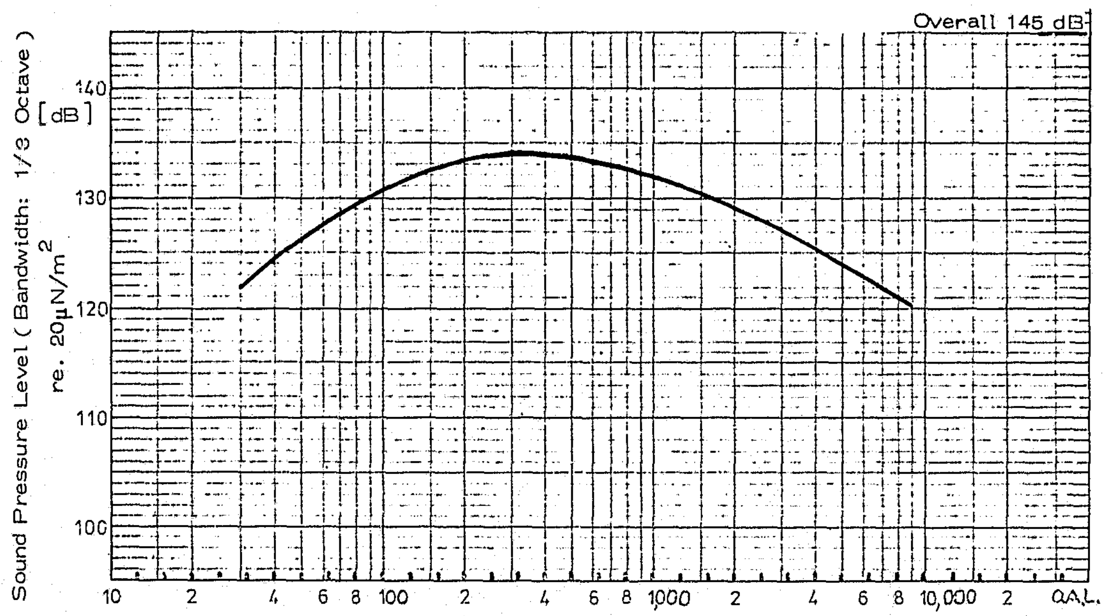


Figure 5 - 1: Analytical Prediction of the Acoustic Spectrum in the Orbiter Cargo Bay

This spectrum produces an effective (integrated) Overall Sound Pressure Level of 145 dB which can be considered to act for 6 sec per launch. The maximum acoustic level occurs at time of lift-off and decreases with time very rapidly as shown in Figure 5-2.

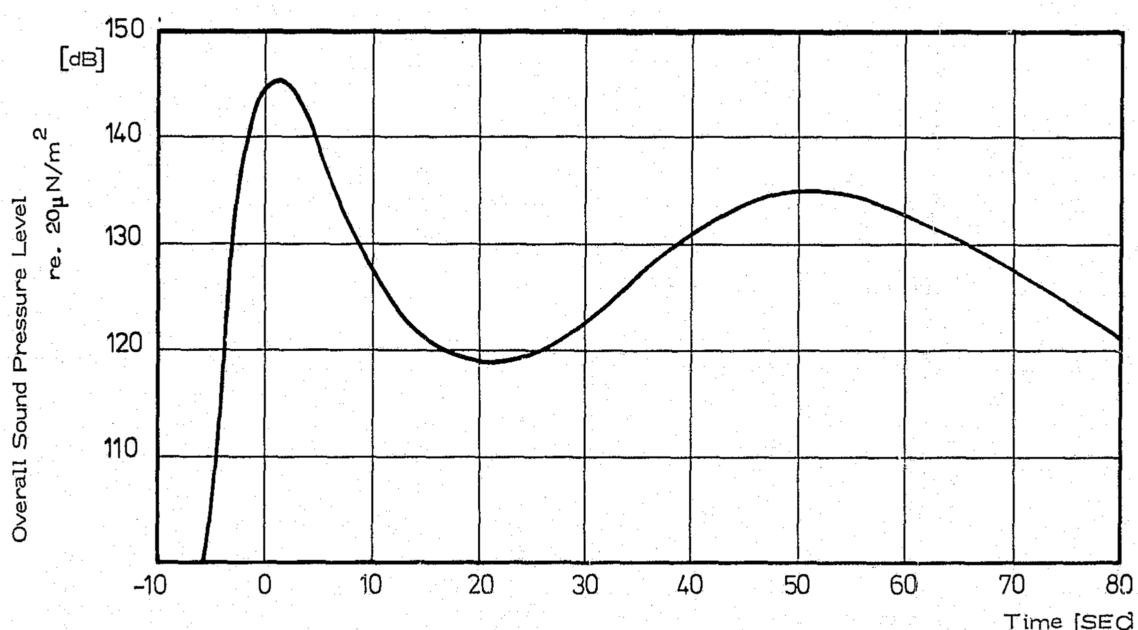


Figure 5-2: Acoustic Noise History in the Orbiter Cargo Bay

The Spacelab module shell and insulation will attenuate the acoustic vibration by approximately 9 dB overall, the attenuation being frequency dependent. Hence equipment mounted anywhere in the module will be subjected to acoustic spectra given in Figure 5-3, varying in time from launch in a similar manner as shown in Figure 5-2

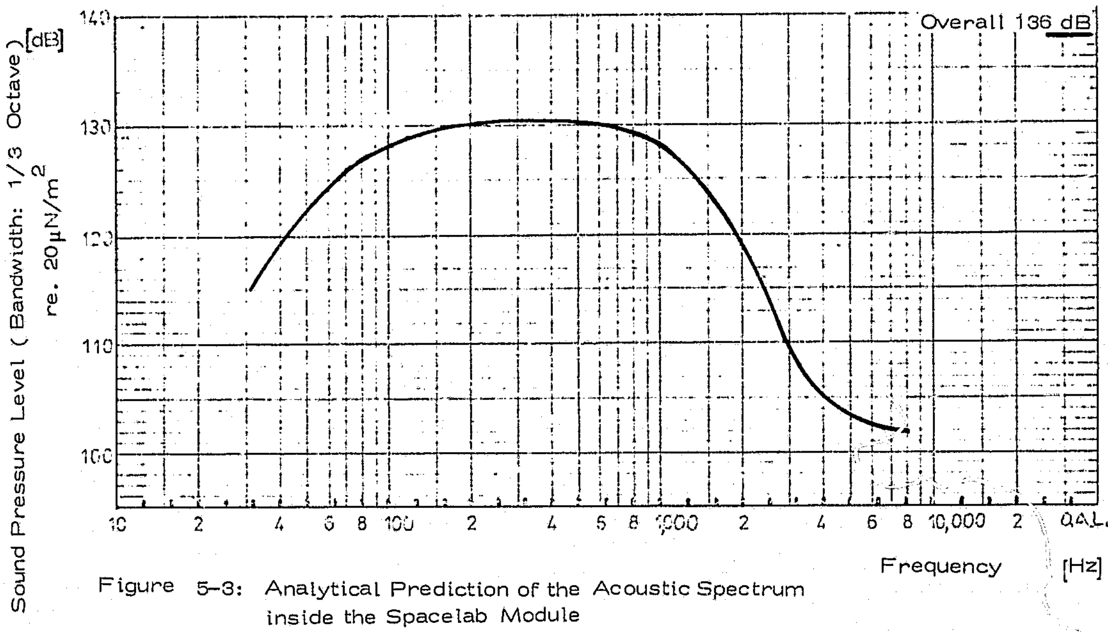


Figure 5-3: Analytical Prediction of the Acoustic Spectrum inside the Spacelab Module

5.1.3 Shock

a) Pyro Shock: TBD

b) Landing Shock:

Rectangular pulses of the following peak accelerations will be as defined in Table 5-3 which is generated by the Orbiter during landing operations. These levels are applicable at the equipment mounting plane in the $+Z_0$ direction only.

Table 5-3: Landing Shock

Acceleration (g Peak)	Duration (Milliseconds)	Applications per 50 missions
0.23	170	11
0.28	280	19
0.35	330	16
0.43	360	10
0.56	350	5
0.72	320	2
1.50	260	1

c) On-Orbit Operations

Loose Equipment

Equipment which will be unstowed and used by the crew during normal on-orbit operations may be subjected to various handling shocks. However, the zero-weight environment effectively ensures that these shocks will be less than those to be expected during normal ground handling and hence the values given in para 5.4.1 may be considered the most severe handling shocks for loose equipment.

Fixed Equipment

Equipment which is mounted in the module in a location which is immediately reachable by the crew in the zero-g environment, may be subjected to an inadvertent kick or push-off by them. All experiment equipment mounted within reasonable access of such a kick or push-off load should be capable of operating normally after such an event.

Editorial note: The worst case (kick load) and the "steady state" (push-off load) are not yet defined.

5.1.4 Linear Acceleration

5.1.4.1 Nominal Mission/Emergency Sequence

Maximum expected accelerations (limit load factors) for Spacelab missions are given in Table 5 – 4 .

The accelerations result from the dynamic response of the Spacelab to the acceleration factors experienced by the integrated Shuttle vehicle and are a function of the mass and stiffness distribution of the Spacelab and the type and location of its attachment to the Orbiter vehicle. Actual Spacelab accelerations of various Spacelab configurations will be established by means of coupled Shuttle/Spacelab dynamic analysis. Table 5 – 5 gives the anticipated angular velocities.

Steady state crash acceleration are given in Table 5 – 6 . These accelerations are applicable only to the mounting and attachment structure of large equipment items, their attachment fasteners and local surrounding structure (e.g. bearing structure around bolt-holes). Crew compartment interior crash accelerations are also included on this table, being applicable to mounting structures for equipment and crew provisions in the crew compartment.

Crash accelerations act separately and are ultimate. The longitudinal accelerations are directed within a cone of 20^{o} half angle from the longitudinal (X) axis.

Equipment should be capable of normal operation after being subjected to acceleration for each of the mission events except for crash landing.

Editoral note: Spacelab and payload equipment must be capable of surviving the "crash" landing acceleration loads with sufficient physical integrity to preclude hazards to the flight personnel after the application of crash loads.

Table 5-4: Spacelab Limit Load Factors (g's) and Angular Accelerations (RAD/SEC^2)

Condition	$\ddot{X}$	$\ddot{Y}$	$\ddot{Z}$	$\ddot{\Phi}$	$\ddot{\Theta}$	$\ddot{\psi}$
Lift off	-1.6 ± 1.3	± 0.8	-0.1 ± 1.0	± 0.15	± 0.1	± 0.1
High Q Boost	-1.8 ± 0.2	± 0.5	± 0.6	± 0.15	± 0.1	± 0.1
Max. Boost	-3.0 ± 0.15	± 0.2	-0.3	± 0.1	± 0.1	± 0.1
Orbiter Max. Load	-3.0 ± 0.15	± 0.2	-0.75	± 0.1	± 0.1	± 0.1
Entry & Descent Maneuvers						
+ Pitch	1.1	0.0	2.5	0.0	-0.1	0.0
− Pitch	0.6	0.0	-1.0	0.0	0.7	0.0
± Yaw	1.0	± 1.25	1.0	0.0	0.0	± 0.2
± Roll	0.9	± 0.2	1.5	± 2.6	0.3	± 0.2
Landing	0.2 ± 1.3	± 0.7	2.0 ± 2.0	± 0.2	± 0.2	± 0.1

Note 1 : x_o, $\dot{\Phi}$ etc are defined in Figures 2 − 1 and 2 − 2 .

Note 2 : The load factors and angular accelerations quoted for each case have to be superimposed.

Note 3 : Center of rotation of Orbiter : x_o = 1080"
y_o = 0"
z_o = 400"

Table 5-5: Assumed Angular Velocities

Condition	$\dot{\Phi}$ rad/s	$\dot{\Theta}$ rad/s	$\dot{\psi}$ rad/s
Ascent	0	0	0
+ Pitch	0	-0.2	0
− Pitch	0	$+0.52$	0
± Yaw	0	0	0
+ Roll	-1	$+0.3$	-0.3
− Roll	$+1$	$+0.3$	$+0.3$
Landing	± 0.3	± 0.3	0

Table 5-6: Ultimate Crash Load Factors

	TOTAL ACCELERATIONS (G)					
	+X	-X	+Y	-Y	+Z	-Z
Crash Landing Loads	9.0	1.50	1.50*	1.50*	4.5	2.0*
Crash (crew interior)	20.0	3.30	3.30*	3.30*	10.0	4.4*

* Superseded by 20° half cone angle misalignment of axial crash loads

Accelerations to act independently

5.1.4.2 On-Orbit Maneuvers

During normal Orbiter attitude-control activities, thrusting of the Orbiter RCS will cause slight acceleration to be exerted on Spacelab equipment depending on its location with respect to the center of rotation. Values are given in Table 5-7 for the RCS thrusters. The values shown are based on an Orbiter prior to the deorbit burn with a 14 515 kg (32 Klb) cargo.

All three angular acceleration may occur simultaneously and the linear acceleration at any point of Spacelab may be calculated based on the distance from the Orbiter's center of gravity. This location will vary to some extent with the particular payload weight distribution. The typical location of the Orbiter's centerof gravity without Spacelab and its payload is given in para. 2.2.7.

Table 5-7: Orbiter RCS Maximum Acceleration Levels

Direction	Translational, m/sec² (ft/sec²) 1)					Rotational, degrees/sec² 2)			
RCS System	$+x_o$	$-x_o$	$\pm y_o$	$+z_o$	$-z_o$	$\pm\ddot{\phi}$	$+\ddot{\theta}$	$-\ddot{\theta}$	$\pm\ddot{\psi}$
Primary Thruster 900 LB	0.169 (0.554)	0.129 (0.424)	0.212 (0.695)	0.381 (1.251)	0.309 (1.014)	1.168	1.320	1.482	0.738
Vernier Thrusters 25 LB	0	0	0.0021 (0.0070)	0	0.0024 (0.0080)	0.037	0.024	0.017	0.019

The time history of the specified levels is TBD

1) i.e. translational acceleration of the Orbiter's center of gravity

2) ϕ, θ and ψ are defined in Figure 2 - 2 . .

5.1.4.3 Orbit Atmosphere Accelerations

On-orbit acceleration levels resulting from atmospheric drag on the Orbiter while in a drift mode of operation are shown in Figure 5 - 4 . Perturbations such as crew movement, venting, etc., would affect acceleration levels in this mode of operation.

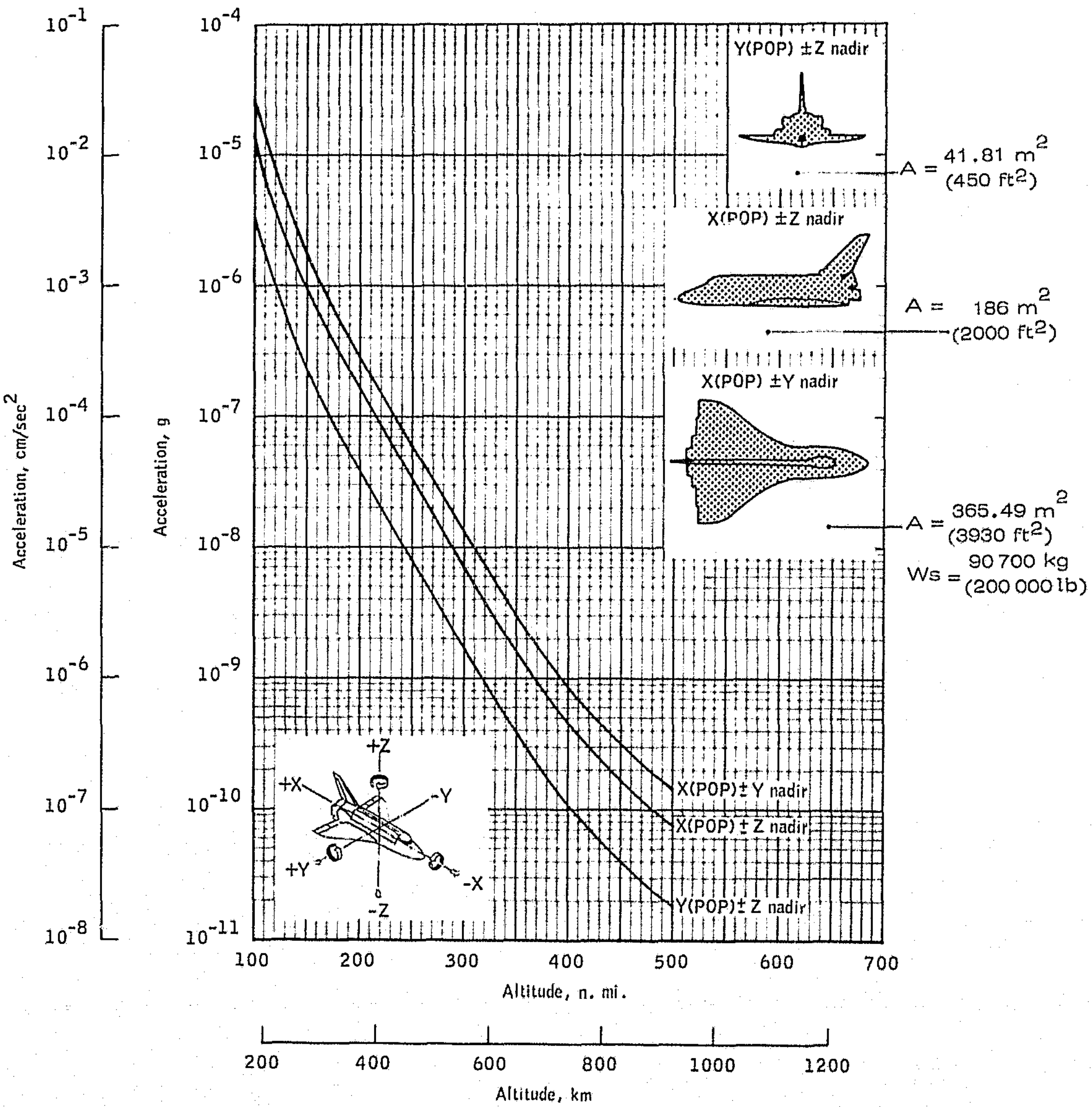


Figure 5 -4 : Effects Of Atmospheric Drag On The Orbiter

5.1.5 Temperature

During on-orbit operations, the temperature of module equipment will be controlled primarily by the forced flow of air in the equipment racks (avionics loop) and the cabin area (cabin air loop).

a) On-Orbit Cabin Air Temperature

The cabin air temperature may be adjusted within the range 18° C to 27° and will be controlled automatically to within $\pm 1^{\circ}$ C.

b) Ascent/Descent Phase Cabin Temperature

During ascent, descent and landing, the cabin air temperature is based on the extremes of the external temperatures. The estimated temperature limits for these periods are given in Table 5-8.

Table 5-8: Cabin Temperature Limits During Ascent/Descent

OPERATIONAL PHASE	T min.($^{\circ}$ C)	Tmax. ($^{\circ}$ C)
Launch/Ascent	7	50
Re-Entry	TBD	45
Post Landing	TBD	45

c) Avionics Loop Temperature

Equipment installed inside the avionics racks is cooled by forced air with the following temperature characteristics (estimated values) (Table 5 - 9):

Table 5 - 9: Avionic Rack Forced Air Cooling (Estimated)

Draw-Through Cooling	Maximum Inlet Temperature: Minimum Inlet Temperature:	35° C 10° C
Surface Cooling	Maximum Temperature: Minimum Temperature:	35° C 10° C

d) Surface Temperature

Under normal operations, the mean radiant temperature of the habitable module interior will not exceed 30° C and the temperature of any inside wall surface of possible crew contact shall not exceed 45° C unless protective guards are provided.

No external equipment surfaces, whether reachable or not, may be cooler than the dew point temperature of the module atmosphere (see Section 5.1.6.2).

5.1.6 Atmosphere

5.1.6.1 Pressure

During normal operations, pressure within the Spacelab module is maintained at 1.013 ± 0.013 bar. Under some emergency conditions, evacuation of the module may be required. The estimated pressure profile for this event is TBD.

5.1.6.2 Composition

The gaseous composition of the module atmosphere is given in Table 5 - 10 for the major constituents. The humidity is maintained within the range 25 % to 70 % by the ECS and is not adjustable. The range of temperature/humidity conditions provided within the module are shown in Figure 5 - 5.
The cabin atmosphere design requirements are presented by the dotted humidity-temperature range. Although these requirements allow relative humidity levels to be uncontrolled over a wide range acceptable to crew comfort, the performance of the cabin heat exchanger is expected to control to tighter relative humidity levels.

Table 5 - 10: Module Atmosphere Gas Composition

Cabin Pressure	
O_2/N_2 total pressure:	1.013 ± 0.013 bar
O_2 partial pressure:	0.220 ± 0.017 bar
CO_2 partial pressure:	0.0067 bar (nominal) 0.01 bar (maximum for normal operations)

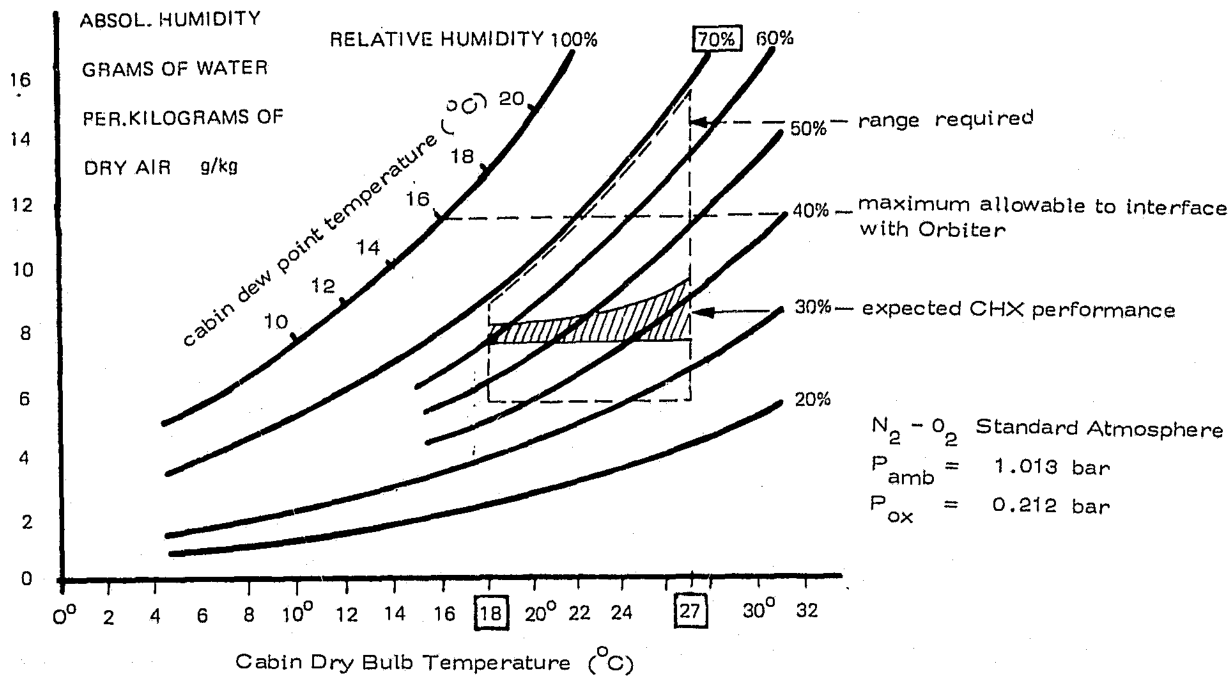


Figure 5 – 5 Crew Thermal Comfort Criteria

Figure 5 – 5: Crew Thermal Comfort Criteria

5.1.7 Cleanliness and Contamination

Prior to installation into the Spacelab module itself or into rack sections, all equipment surfaces shall be cleaned to a visible clean level. During operation of the Spacelab module the cleanliness will be maintained by the module ECS system, using circulated air filtered by 280 micron filters.

5.1.8 Electrical Environment - Module

5.1.8.1 Radiated Emissions

Orbiter Emissions

Shuttle contributions to the payload electro-magnetic environment will be limited to levels of not more than 0.5 volts/meter from 15 KHz to 10 GHz except at frequencies of the orbiter-installed transmitters which shall be as specified below (Table 5-11).

Table 5-11: Orbiter Transmitter Characteristics

Transmitter	Frequency	Mission Phase	Field Intensity	Field Density (W/M^2)
S-Band Hemi	2000 - 2300 MHz	Ascent, Orbit	4 V/m	
S-Band Quad	2200 - 2300 MHz	All	8 V/m	
S-Band Payload	2000 - 2200 MHz	Orbit	3 V/m	
EVA Backpack	250 - 300 MHz	Orbit	4 V/m	
Rendezvous Radar	13 - 15 GHz			
Comm. Mode		Orbit		2800 at 1 Meter* 980 at 3 Meters* 280 at 6 Meters* 119 at 9 Meters* 70 at 12 Meters*
Pulse Mode (Backlobe Radiation)		Orbit		1000 at 1 Meter* 112 at 3 Meters* 28 at 6 Meters* 12.3 at 9 Meters* 7 at 12 Meters*

All field intensity levels at cargo bay location $X_o = 702$, $Z_o = 460$, except for EVA backpack, which is 1 meter from backpack.

*From $X_o = 527$, $Y_o = 130.7$, $Z_o = 447.75$

It is expected that the module structure will provide an attenuation of at least 20 dB to electrical fields, and that the Orbiter RF emission levels seen by payload equipment within the Spacelab module will be similarly reduced.

Launch Environment

The launch environment RF levels expected to be encountered by the Spacelab are being defined by NASA at this point and are expected to become available during summer 1976.

Spacelab Emissions

RF emissions radiated by Spacelab subsystem equipment are controlled to the specification limits given in Figures 5 - 6 and 5 - 7 .

Composite Payload Environment - Module Equipment

Due to the variable amount of Spacelab subsystem and payload equipment which may be carried on any one mission, it is not possible to place an accurate limit on the composite spectrum radiated by these units operating simultaneously. Instead, a best-guess estimate of the RF environment expected for payloads within the Spacelab module places an upper limit of 0.1 V/m for narrowband emissions (peaking at 1-10 MHz and at S-Band) and 90 dB μ V/m/MHz for broadband emissions (with a peak in the 100 MHz range due to possible cavity resonance effects of the module structure).

5.1.8.2 Conducted Emissions

Orbiter Emissions

(1) Ripple

The Orbiter-generated ripple at the interface will not exceed:

a) 0.9 V peak-to-peak (30 Hz to 7 KHz) falling 10 dB per decade to 0.28 V peak-to-peak at 70 KHz, thereafter remaining constant up to 400 MHz.

b) 1.6 V peak-to-peak over a broad band of 30 Hz to 50 KHz, then decreasing linearly with frequency at a rate of 40 dB per decade to 500 KHz, the thereafter remaining constant to 50 MHz.

(2) Transients

Will not exceed + 30 volts above nor -30 volts below nominal power voltage level with rise and fall times greater than 1 microsecond and a pulse duration of not greater than 10 microseconds.

(3) Dedicated Fuel Cell Operation

The Orbiter-generated ripple and transients at the interface will not exceed 100 millivolts peak-to-peak (30 Hz to 10 MHz).

(4) Ground Power

The Orbiter ripples at the interface will not exceed:

a) 1.2 V peak-to-peak (30 Hz to 7 KHz) falling to 0.28 V peak-to-peak at 70 KHz, thereafter remaining constant up to 400 MHz.

b) 2.0 V peak-to-peak over a broad band of 30 Hz to 50 KHz, then decreasing linearly with frequency at a rate of 40 dB per decade till 500 KHz, thereafter remaining constant up to 50 MHz.

(5) Transients

Will not exceed + 30 volts above nor TBD volts below nominal power voltage level with rise and fall times greater than 1 microsecond and a pulse duration of not greater than 10 microseconds.

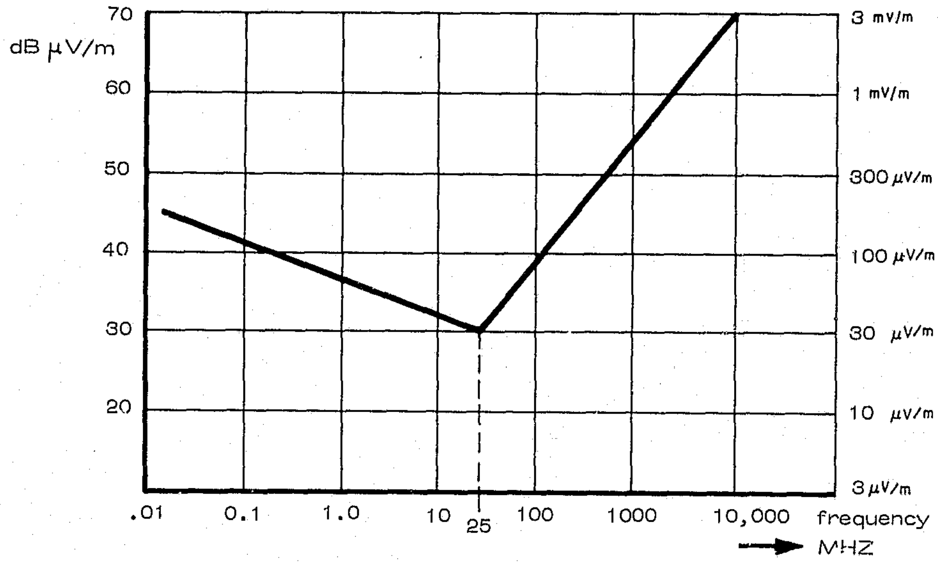


Figure 5 – 6: Limit for Narrowband Emissions

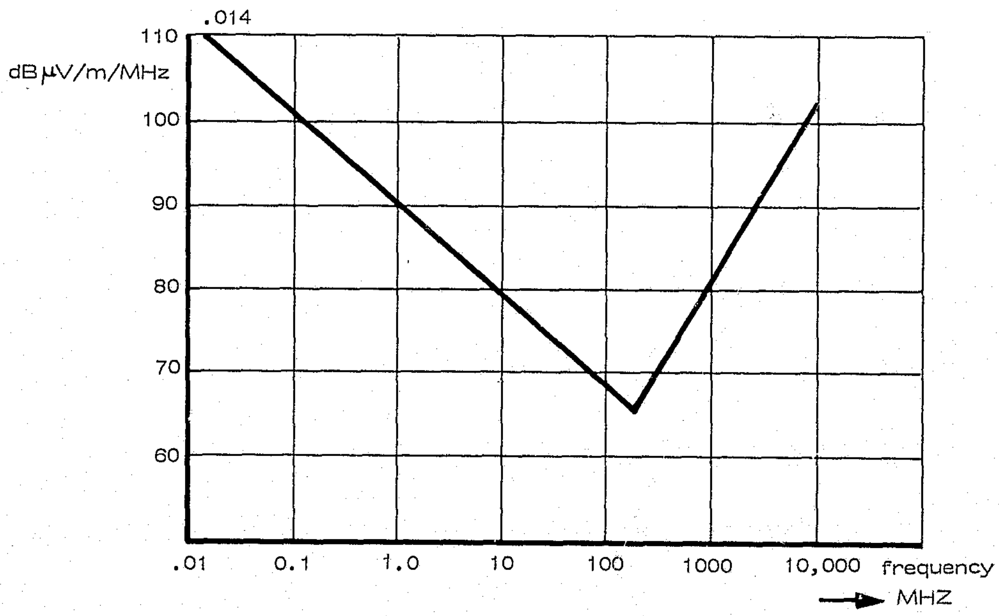


Figure 5 – 7: Limit for Broadband Emissions

Spacelab Emissions

Present limits on Spacelab-generated conducted interference levels, include the following :

- 1.5 volts (RMS) at any discrete frequency between 30 Hz and 3 KHz, and 1.5 volts (peak-to-peak) when measured in the frequency band extending from 3 KHz to 100 KHz.
- primary bus transients measured on the positive lead of the 28 VDC powerline shall not exceed $\pm$ 28 volts and 10 μ sec in duration.
- current rise and fall under load switching conditions shall be as slow as practicable and shall not exceed a rate of five (5) amperes per microsecond.

Composite Payload Environment

The composite conducted interference environment seen by the Spacelab payloads reflects the operation of the EPDS, i.e. dedicated operation with a fuel cell during normal on-orbit conditions, and non-dedicated fuel cell operation with Orbiter loads in parallel with Spacelab.

- Non-dedicated operation (ascent, power switchover, back-up power operation, descent): power bus characteristics will be determined principally by the Orbiter emission levels.
- Dedicated operation: power bus characteristics will be determined principally by Spacelab emission levels, although other mission payloads may add to these levels.

5.1.8.3 Bonding and Lightning Protection

The primary Spacelab structure is designed to provide a continuous, low-impedance, equipotential ground reference plane for the various separate, galvanically isolated secondary power networks. It is also designed to serve as a fault current return path for both the primary and the secondary power networks. Latest NASA/Rockwell calculations indicate that lightning - induced current in the Spacelab structures will be lower than the potential short circuit currents of the primary DC power distribution network.

Primary structure bonding requirements of 2.5 milliohms between adjacent metallic components have still been retained, however, in view of the expected 6000 ampere short circuit current capacity of the fuel cells.

The experimenter is also cautioned to protect his externally interfacing input and output circuits against transients induced on signal and power lines by the magnetic fields which accompany the lightning currents.

5.1.8.4 Electrical Surface Properties: TBD

5.1.9 Magnetic Environment

Orbiter Emissions

Shuttle contributions to the payload magnetic environment will be limited to less than 140 dB above 1 pico-Tesla(30 Hz to 2 kHz), falling 40 dB per decade to 50 KHz. The DC fields are TBD, and are not controlled by present specifications.

Editorial note: The above AC magnetic field values are still under review by NASA. They apparently represent worst case values near magnetic hot spots such as junction boxes.

Spacelab Emissions

Spacelab equipment magnetic field emission levels are controlled in such a way that emissions over the frequency range of 20 Hz to 50 kHz are limited to not more than 60 dB above a pico-Tesla.In accordance with Shuttle specifications. No requirements have been imposed to control magnetic fields below 30 Hz. However, the Spacelab project is studying the impact on the program of minimizing the amount of magnetic materials and number of current loops in the design of Spacelab. The conclusions of this study and the resulting actions will be available in the near future.

Composite Payload Environment

In estimating the magnetic field environment for Spacelab payload equipment, it must first of all be recognized that the magnetic field emission limits for on-board equipment are set for a measurement distance of one meter. Since the equipment racks are only one-half that width, equipment mounted in adjacent racks may be expected to encounter higher than specification level magnetic fields.

On the basis of the above considerations, it is recommended that payload equipment be designed to operate in the following AC magnetic field environment: 120 dB above a pico-Tesla at 30 Hz decreasing at a rate of 6 dB/octave to 80 dB above a picotesla at 3 KHz, and maintaining that value to 50 KHz.

5.1.10 Radiation Environment

The radiation environment external to Spacelab is described in para 5.2.10. The actual radiation level at a given equipment will depend on the mission profile and on the stopping power of the material surrounding the equipment. The minimum shielding is that of the module wall alone: 0.55 g/cm^2.

Equipment requiring stowage in very low radiation environments may be stowed in the specially shielded film vaults. The design of these vaults is discussed in para 4.6.5.

Editorial note: The attenuation of a typical external particle flux within the film vault will be supplied in a later issue.

5.2 Pallet Equipment/Flight Environment

The environments described hereafter refer to all equipment which is located on the pallet or outside of the pressurized part of the module.

5.2.1 Vibration

- Sinusoidal Vibration

  Same requirements are applicable as for module mounted equipment defined in para. 5.1.1.

- Random Vibration

  Maximum vibration levels occur during the launch phase. Equipment mounted on the pallet will be subjected to vibration arising from the overall acoustic level inside the cargo bay, and a very minor degree to vibration transmitted through the Orbiter/Pallet mounting fixtures into the pallet structure.

  - Panel Mounted Equipment/Small Equipment Frame & Hardpoint Mounted

    The vibration levels for equipment mounted directly to a pallet panel and small equipment mounted to pallet hardpoints or frames are given in Table 5-12.

Table 5-12: Random Vibration Levels for Pallet Mounted Equipment

LOCATION	FREQUENCY	LEVEL
Zone 4 a)	20 Hz	0.020 g^2/Hz
Input to equipment	20 – 50 Hz	+ 3 dB/oct
mounted on pallet	150 – 900 Hz	0.15 g^2/Hz
hard points.	900 – 2000 Hz	– 12 dB/oct
Mass of equipment	2000 Hz	0.0064 g^2/Hz
$<$ 30 kg	Composite:	12.9 g RMS
Zone 4 b)	20 Hz	0.020 g^2/Hz
Input to equipment	20 – 84 Hz	+ 3 dB/oct
mounted on pallet	84 – 900 Hz	0.088 g^2/Hz
hard points	900 – 2000 Hz	– 12 dB/oct
Mass of equipment	2000 Hz	0.0037 g^2/Hz
$\geq$ 30 kg but $<$ 65 kg	Composite:	10.6 g RMS
Zone 4 c)	20 Hz	0.020 g^2/Hz
Input to equipment	10 – 52 Hz	+ 3 dB/oct
mounted on pallet	52 – 900 Hz	0.053 g^2/Hz
hard points	900 – 2000 Hz	– 12 dB/oct
Mass of equipment	2000 Hz	0.0022 g^2/Hz
= 65 kg	Composite:	7.78 g RMS
Zone 4 d)	20 Hz	0.076 g^2/Hz
Input to equipment	20 – 150 Hz	+ 3 dB/oct
mounted on the	150 – 350 Hz	0.58 g^2/Hz
inner pallet honeycomb	350 – 550 Hz	– 6 dB/oct
panel	550 – 2000 Hz	0.23 g^2/Hz
Mass of equipment $<$ 15 kg	Composite:	23.9 g RMS
Zone 4 e)	20 Hz	0.067 g^2/Hz
Input to equipment	20 – 55 Hz	+ 3 dB/oct
mounted on the	55 – 350 Hz	0.18 g^2/Hz
inner pallet honeycomb	350 – 550 Hz	– 6 dB/oct
panel	550 – 2000 Hz	0.072 g^2/Hz
Mass of equipment $\geq$ 15 kg	Composite:	13.6 g RMS

The weight figures are understood as the sum of all components mounted randomly distributed on the sandwich panels of a pallet. The vibration levels exist for approximately 6 seconds per flight.

- Large Equipment Hardpoint Mounted Equipment

Equipment attached to pallet hardpoints is subjected to vibration generated by the adjacent panels, which are excited by the overall acoustic noise level within the payload bay. The actual vibration level to which hardpoint mounted equipment is exposed depends on equipment mass, number of hardpoints used, additional masses attached to adjacent panels, surface of considered equipment, etc., therefore, it is necessary to perform a payload accommodation study for large equipment mounted on pallet hardpoints to define the vibration levels. An example is given in Table 5-13 for a piece of equipment attached to all 24 hardpoints of one pallet, with a total mass of 1000 kg. No excitation of the experiment itself through acoustic noise is assumed.

Editorial note: Table 5 - 13 is based on respective calculations by ERNO.

Table 5-13: Random Vibration Level for Pallet Hardpoint Mounted Equipment

Location	Frequency	Level
Pallet hardpoint mounted equipment mass = 1000 kg	20 - 200 Hz	+ 8 dB/oct
	200 - 700 Hz	0.048 g^2/Hz
	700 - 900 Hz	- 18 dB/oct
	900 - 2000 Hz	0.011 g^2/Hz
	composite	6.8 g RMS

The vibration environment exists for approximately 6 seconds per flight.

5.2.2 Acoustic Noise

The acoustic sound pressure field inside the cargo bay can be considered a reverberant one. The spectrum varies during launch with respect to level and shape. The maximum Outer Acoustic Level occurs for a few seconds during main engine run up and launch. The time history of the overall sound pressure level in the Orbiter cargo bay is given in Figure 5-2. The acoustic spectrum in the Orbiter cargo bay is given in Figure 5-1. All pallet mounted equipment and equipment external to the pressurized part of the module will be subjected to this acoustic noise environment.

5.2.3 Shock

- Pyro Shock: TBD
- Landing Shock: As specified in paragraph 5.1.3, Table 5-3.
- On-Orbit Operations

During missions in which EVA to the pallet is a scheduled activity, pallet-mounted equipment may be inadvertently kicked or pushed by the crew. The worst case for a kick-load by an EVA-suited crew member is 9 kg m/s at a maximum velocity of 1.5 m/s (tested by means of a metal block corner with 5mm spherical radius mounted on a rigid base) and for a steady state load of 550 N. Equipment mounted in the vicinity of scheduled EVA activity areas must be capable of operating normally after such loads have occured. For missions in which EVA would only occur as as emergency activity, pallet-mounted equipment needs only be capable of surviving these loads intact.

5.2.4 Acceleration

Acceleration loads for pallet-mounted equipment are the same as for module equipment and are described in Section 5.1.4.

5.2.5 Temperature

5.2.5.1 Air Temperature

During the launch sequence and portions of the landing sequence, the temperature of Spacelab equipment is influenced by the air temperature in the cargo bay. The estimated extreme temperatures during the launch sequence, during re-entry and landing, as well as prior to reconnection of the GSE at the landing site are shown in Table 5-14

Table 5-14: Orbiter Cargo Bay Air Temperature

CONDITION	T min. (° C)	T max. (° C)
Purge on Launch Pad	7*	49*
Launch/Ascent (estimated values)	7*	49*
Re-Entry and Landing (estimated values)	Temperature profile as defined in Fig. 5-8	TBD
On landingsite prior to reconnection of GSE/ECS (estimated values)	TBD	TBD

*estimated values

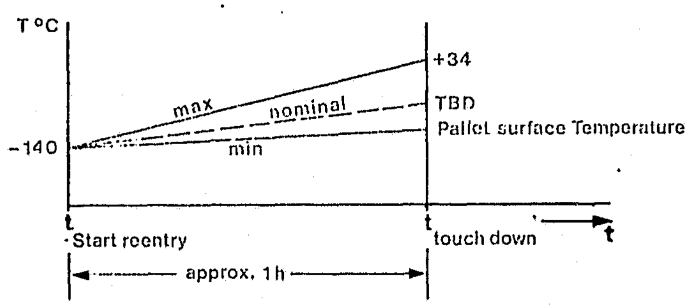


Figure 5 - 8: Estimated Temperature Profile of the Cargo Bay Air During Descent (Cold Case)

5.2.5.2 On-Orbit Conditions

The thermal environment for pallet-mounted equipment is dependent on many factors. The most important parameters are listed here, however, the manner in which they affect the equipment temperature requires a detailed analysis of the mounting configuration of all equipment in the vicinity and the mission profile.

5.2.5.2.1 Orbiter Cargo Bay Temperatures

While the Orbiter cargo bay doors are closed, the relative coupling with all exposed Spacelab surfaces is dependent on the inside surface characteristics and temperature, which in turn is determined by the attitude mode of the Orbiter.

The estimated surface temperatures for various modes are given in Table 5 - 15.

Editorial note: The maximum allowable heat rejection rate from a payload to the Orbiter structure, with the cargo bay doors closed, during various attitude modes in TBD.

Table 5-15: Orbiter Cargo Bay Wall Temperature

CONDITION	DESIGN MINIMUM	DESIGN MAXIMUM
Prelaunch	+ 4,5° C (+ 40° F)	+ 49° C (+ 120° F)
Launch	+ 4,5° C (+ 40° F)	+ 65,5°C (+ 150°F)
On-Orbit (doors closed)	See C & D	See A & B
Entry and postlanding	− 73° C (− 100° F)	+ 93,5°C (+ 200°F)
Heat leak criteria: TBD		
A. Total bay heat gain, average : TBD		
B. Heat gain, local area: TBD		
C. Total bay heat loss, average: TBD		
D. Heat loss, local area : TBD		

5.2.5.2.2

- Pallet Surface

The expected temperature range of the pallet structure surface is given in Table 5-16.

Table 5-16: Temperature Range of Pallet Structure Surface

OPERATIONAL PHASE	T min. (° C)	T max. (° C)
Launch / Ascent	4,5	65,5*
Orbit doors closed	− 50	70*
Orbit doors open	− 150	70*
Re-Entry	− 100	100*
Post Landing	− 100	100*

* estimated values

- Pallet Mounted Equipment

Pallet mounted equipment which is not thermally protected, is subject to the same temperature requirements as the pallet structure surface.

5.2.5.2.3 Space Environment

With the cargo bay doors open, the radiative environment for pallet-mounted equipment is determined by the incoming fluxes given in Table 5-17, the Orbiter attitude, and the equipment shadow/illumination configuration.

Editorial note: The cross-section of the pallet/Orbiter mounting arrangement to be used to determine the shadow/illumination configuration for pallet-mounted equipment will be supplied in a later issue.

Table 5-17: Space Thermal Environment

Environmental Parameter	Unit	Maximum	Nominal	Minimum
Solar Radiation	W/m^2 ($Btu/h\text{-}ft^2$)	1440.5 (457)	1352.2 (429)	1264.0 (401)
Earth Global Albedo	Percent (%) of Solar Radiation	42	30	18
Earth Thermal Radiation	W/m^2 ($Btu/hr\text{-}ft^2$)	278.6 (88.4)	236.4 (75.0)	194.2 (61.6)
Space Sink Temperature	K	-	2.7 K	-

5.2.6 Atmosphere

5.2.6.1 Launch Sequence

The Orbiter cargo bay is vented during the launch and entry phases and operates unpressurized during the orbital phase of the mission. Figure 5 - 9 and 5 - 10 define the cargo bay pressure history during ascent.

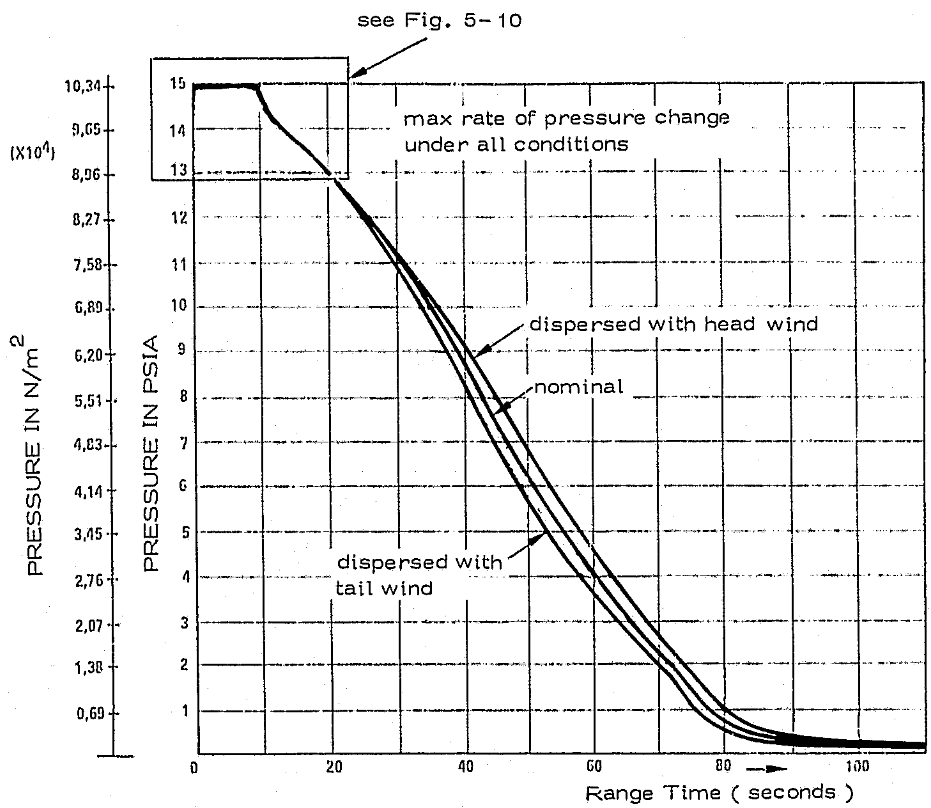


Figure 5-9: Orbiter Cargo Bay Internal Pressure History During Ascent

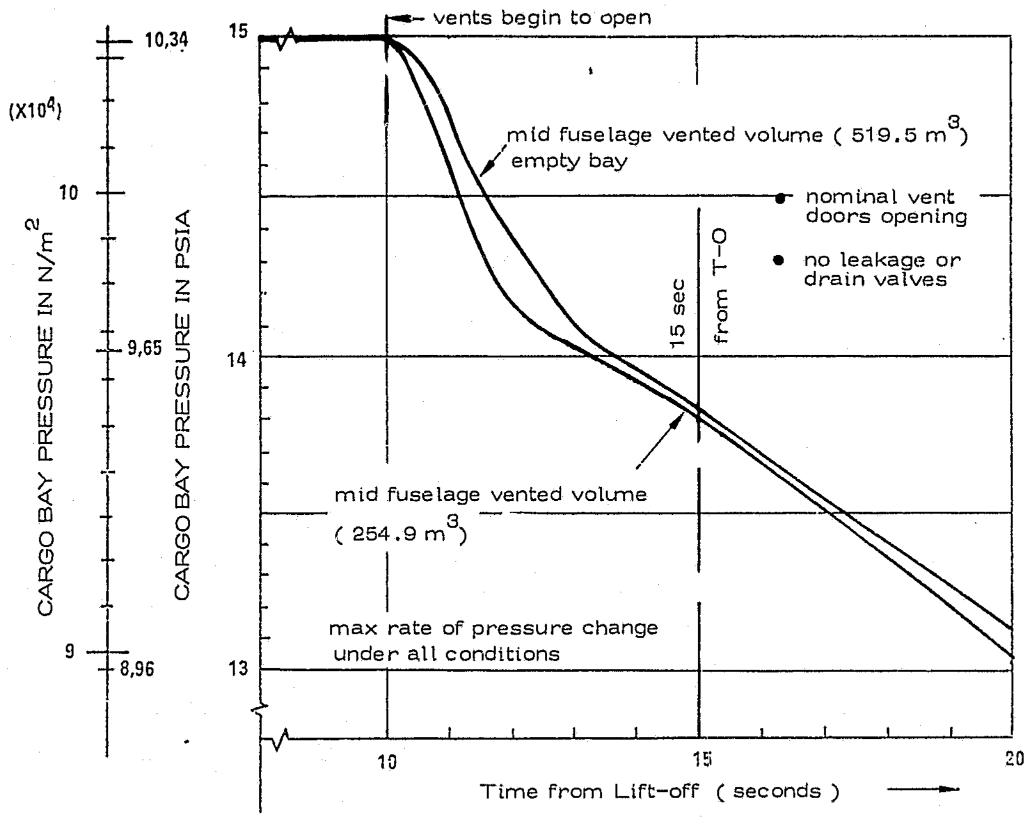


Figure 5-10: Orbiter Cargo Bay Internal Pressure History During Ascent Liftoff Detail

5.2.6.2 On Orbit

Respective values are given in Table 5-18.

Table 5-18: Atmospheric Pressure

Altitude	Max P (bar)	Min P (bar)
81 km	1.8×10^{-8}	1.06×10^{-8}
115 km	2.9×10^{-9}	6.3×10^{-10}
575 km	2.9×10^{-12}	6.3×10^{-14}
1,380 km	Approx. 1.33×10^{-14} in either case	

5.2.6.3 Re-Entry Sequence

During re-entry and landing the cargo bay will be increased to the landing site pressure by using filtered atmospheric air (35 microns absolute). Figure 5 - 11 and 5 - 12 define the cargo bay pressure history during re-entry.

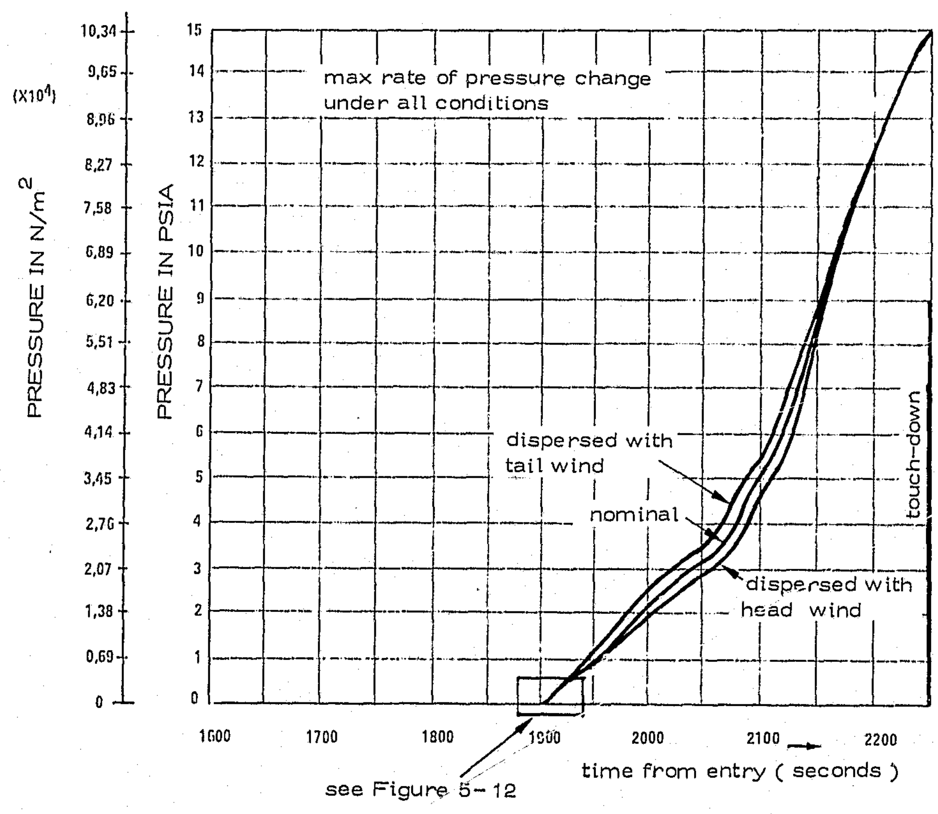


Figure 5-11: Orbiter Cargo Bay Internal Pressure History During Entry for all Flight Modes

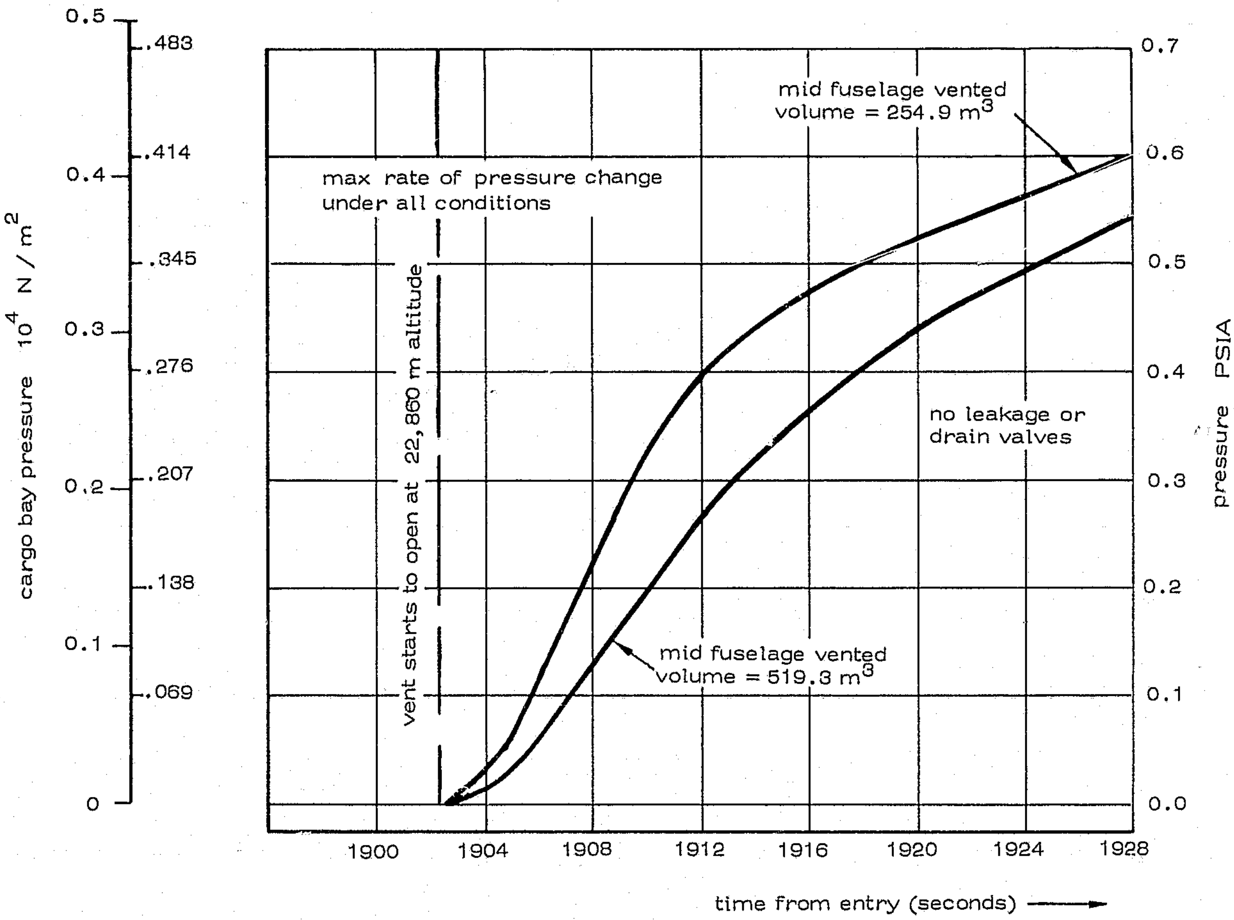


Figure 5-12: Pressure Profile Detail - Vent Doors Opening During Entry

5.2.7 Cleanliness and Contamination

The contamination aspects of the Space Shuttle are described in para. 2.2.8.

Prior to Spacelab loading into the Orbiter,the internal surfaces of the Orbiter cargo bay and Spacelab will be cleaned if required by the payload.

The contamination generated by Spacelab on-orbit is TBD.

5.2.8 Electrical Environment - Pallet

The electrical environment described in Section 5.1.8 is generally applicable also for pallet mounted equipment. The significant difference between the two environments is that the pallet mounted equipment can be expected to encounter the full Orbiter and Spacelab emission levels (ref. 5.1.8.1) due to the absence of any module shielding.

For equipment which may be susceptible to interference from particular types of subsystem or payload equipment-generated signals, the layout of cabling within a pallet is shown in para. 3.4, Figure TBD Pallet equipment may also be in the back-lobes or reflected signal paths of the various Orbiter RF-transmitters. These are listed in Table5-19with their frequency characteristics, indicating the detailed payload interrogator frequencies used by the Orbiter.

Table 5-19 shows the NASA-estimated worst case conditions for RF-field coupling to harness wiring. Orbiter antenny locations are shown in Figure 5-13.

Table 5-19: RF Field to Wiring Coupling (S-Band Worst Case)

	INDUCED POWER (Watts)	
FIELD INTENSITY	UNSHIELDED CIRCUITS	SHIELDED CIRCUITS
3.2 V/M	27×10^{-6}	27×10^{-9}
1.3 V/M	4.5×10^{-6}	4.5×10^{-9}
0.7 V/M	1.3×10^{-6}	1.3×10^{-9}

5.2.9 Magnetic Environment

Editorial note: This section is similar to Section 5.1.9 (Module Environment) and presently the subject of an ESA/NASA study.

Table 5-20: Orbiter Payload Interrogator Transmit and Receive Frequencies

Channel	Transmit	Receive
1	2028.1 MHZ	2202.5 MHZ
2	2032.7	2207.5
3	2037.3	2212.5
4	2041.9	2217.5
5	2046.5	2222.5
6	2051.0	2227.5
7	2055.7	2232.5
8	2060.8	2237.5
9	2065.0	2242.5
10	2069.6	2247.5
11	2074.2	2252.5
12	2078.8	2257.5
13	2083.4	2262.5
14	2088.0	2267.5
15	2092.6	2272.5
16	2097.2	2277.5
17	2101.8	2282.5
18	2106.4	2287.5
19	2111.0	2292.5
20	2115.6 MHZ	2297.5 MHZ

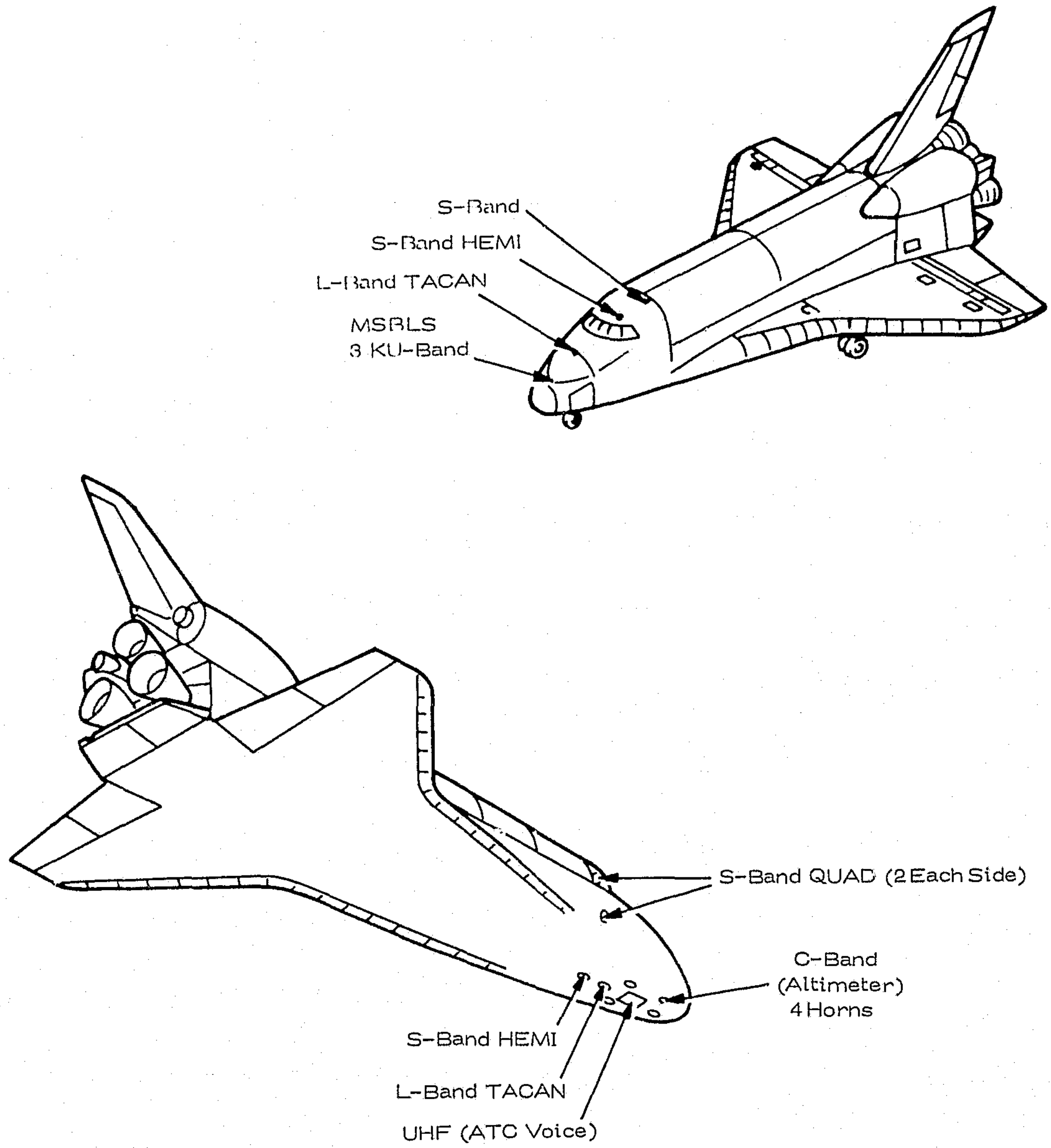


Figure 5-13: Orbiter Antenna Locations

5.2.10 Radiation Environment

Pallet equipment will be subjected to (1) galactic cosmic radiation, (2) geomagnetically trapped radiation, and (3) solar flare particle flows.

5.2.11 Meteoroids

Pallet equipment will be subjected to meteoroid impacts during the time in orbit when the cargo doors are open.

The meteoroid model encompasses particles of cometary origin in the mass range between 1 and 10^{-12} grams for sporadic meteoroids and 1 to 10^{-6} grams for stream meteoroids.

Average Total Environment:

Particle Density : $0.5\ g/cm^3$
Particle Velocity : 20 km/sec
Flux Mass Models :

(1) For $10^{-6} \leq m \leq 10^{0}$ $\log Nt = -14.37 - 1.213 \log m$

(2) For $10^{-12} \leq m \leq 10^{-6}$ $\log Nt = -14.339 - 1.584 \log m - 0.063 (\log m)^2$

Nt = no. particles/m^2/sec of mass m
m = mass in grams
Defocussing factor for earth, and if applicable, shielding factor are to be applied.

This model is only applicable if and when the cargo bay doors are open and the equipment is exposed to space.

5.3 Airlock and Airlock Equipment

The environments defined in this section are applicable to the airlock (top and aft) structure and all equipment which is attached to it, external and internal.

5.3.1 Vibration

- Sinusoidal Vibration

Same requirements are applicable as for module mounted equipment defined in para. 5.1.1.

- Random Vibration

Top Airlock

The major source for vibration generation at the top airlock is the module cylindrical wall vibration. The expected vibration levels for equipment mounted at the top airlock are specified in Table 5-21.

The vibration environment exists for approximately 6 seconds at each lift off based on equivalent damage criteria. This does not include any safety factors.

Table 5-21: Top Airlock Equipment Random Vibration Level

LOCATION	FREQUENCY	LEVEL
Zone 7 a)	20 Hz	0.00042 g^2/Hz
Input to equipment	20 - 68 Hz	+ 9 dB/oct
mounted at the	68 - 105 Hz	0.016 g^2/Hz
Top Airlock	105 - 150 Hz	+ 6 dB/oct
Mass of Equipment	150 - 2000 Hz	0.032 g^2/Hz
$\leq$ 15 kg	Composite:	7.82 g RMS
Zone 7 b)	20 Hz	0.00042 g^2/Hz
Input to equipment	20 - 68 Hz	+ 9 dB/oct
mounted at the	68 - 105 Hz	0.016 g^2/Hz
Top Airlock	105 - 120 Hz	+ 6 dB/oct
Mass of Equipment > 15 kg	120 - 2000 Hz	0.020 g^2/Hz
	Composite:	6.22 g RMS

- Aft Airlock

The major source for vibration generation at the aft airlock is the aft cone vibration. The expected vibration level for equipment mounted at the aft airlock is specified in Table 5-22.

The vibration environment exists for approximately 6 seconds per flight.

Table 5-22: Aft Airlock Equipment Random Vibration Level

LOCATION	FREQUENCY	LEVEL
Zone 8 a)	20 Hz	0.00052 g^2/Hz
Input to equipment	20 - 80 Hz	+ 9 dB/oct
mounted to the	80 - 2000 Hz	0.032 g^2/Hz
Aft Airlock		
Mass of Equipment < 15 kg	Composite:	7.88 g RMS
Zone 8 b)	20 Hz	0.00052 g^2/Hz
Input to equipment	20 - 75 Hz	+ 9 dB/oct
mounted at the	75 - 400 Hz	0.026 g^2/Hz
Aft Airlock	400 - 440 Hz	- 9 dB/oct
Mass of Equipment > 15 kg	440 - 2000 Hz	0.020 g^2/Hz
	Composite:	6.40 g RMS

5.3.2 Acoustic

The acoustic noise level inside the airlock is the same as inside the module pressurized area, specified in paragraph 5.1.2, Figure 5-3.

The part of the aft airlock extending out of the module structure and the outer hatch of the top airlock are exposed to the same acoustic environment.

5.3.3 Shock

- Pyro Shock: TBD
- Landing Shock: As specified in paragraph 5.1.3, Table 5-3.

5.3.4 Linear Acceleration

Same requirements as specified in paragraph 5.1.4 of this specification for module equipment.

5.3.5 Temperature

- Inner Hatch Open

Same temperature requirements as specified in paragraph 5.1.5 of this specification.

- Outer Hatch Open

During airlock operations on orbit with outer hatch open the temperature will meet extremes equivalent to module exterior.

Top Airlock

Tmin $\geq$ T dewpoint of module atmosphere

Tmax = 45^{o} C (max. wall touch temp.)

Aft Airlock

Tmin $\geq$ T dewpoint of module atmosphere

Tmax = 45^{o} C (max. wall touch temp.)

- Both Hatches Closed

The temperature of the airlock interior depends on the experiment temperature after the experiment is retracted into the airlock. Repressurization of the airlock will help to stabilize the temperature in the airlock to the requirements of the module interior, for protection against condensation and ECS-overload. So the same conditions as for module internal equipment described in para. 5.1.5 are valid.

5.3.6 Atmosphere

- Inner Hatch Open

During airlock operations with inner hatch open, the atmosphere requirements are as specified in paragraph 5.1.6 of this specification for module internal equipment.

- Both Hatches Closed

  - The pressure time history for depressurization before outer hatch opening and repressurization before inner hatch opening is 6.6×10^{-3} bar/sec maximum rate.
  - After experiment operation, the airlock is repressurized with dry nitrogen. The nitrogen flow is 4 g/sec.

- Outer Hatch Open

For experiment operation with outer hatch open, the vacuum conditions as specified in paragraph 5.2.6 of this document can be assumed.

5.3.7 Contamination

- Module Internal

  During inner hatch open, equipment inside the airlock is subjected to contamination and cleanliness requirements as specified in paragraph 5.1.7 of this document.

- Module External

  During outer hatch open, equipment of the airlock is subjected to contamination cleanliness requirements as specified in paragraph 5.2.7 of this document.

5.3.8 Electrical

The electrical fields specified for Spacelab internal equipment in paragraph 5.1.8 of this document are applicable.

5.3.9 Magnetic

The magnetic environment specified in paragraph 5.1.9 of this specification is applicable.

5.3.10 Radiation Environment

Same requirement as specified in para. 5.2.10 of this document.

5.3.11 Meteoroid Environment

During outer hatch open operation, equipment of the airlock is subjected to the meteoroid model specified in paragraph 5.2.11.

5.4 Ground Operations Environment

This section describes the environments to which experiment equipment may be subjected during transportation, integration, installation and pre-launch check-out activities.

5.4.1 Vibration/Shock

- Transportation

  During transportation and handling of experiment flight hardware, ground support equipment shall be provided which controls dynamic loads exposed to flight hardware to be lower than those specified for flight operations in paragraph 5.1, 5.2 and 5.3 of this document.

- Handling

  The shock environments experienced by the payloads during handling are represented by 20 g terminal sawtooth shock pulses of a 10 millisecond duration in both directions of each axis.

5.4.2 Atmosphere (Temperature, Pressure, Humidity, Contamination)

5.4.2.1 Integration Operations

- Central Integration Site

  Integration and Checkout Area

Temperature:	TBD
Humidity:	TBD
Pressure:	TBD
Cleanliness:	Class 100 000

- Operation and Checkout Building (0 & C)

  The 0 & C Building includes the

  - Experiment/Rack/Pallet Processing Area
  - Assembly and Checkout Area
  - Receiving and Shipping Area
  - Inspection and Cleaning Area

All areas have the same conditions:

Temperature:	$21^{\circ} \pm 3^{\circ}$ C ($70^{\circ} \pm 5^{\circ}$ F)
Humidity:	50 ± 5 % RH
Pressure:	TBD
Cleanliness:	Class 100 000

- Orbiter Processing Facility (OPF)

The area assigned for payload processing and checkout will provide the following conditions:

Temperature:	$24 \pm 1^{\circ}$ C ($75^{\circ} \pm 3^{\circ}$ F)
Humidity:	50 % RH $\pm$ 5 %
Pressure:	TBD
Cleanliness:	purged with Class 100 000 air

5.4.2.2 Orbiter Cargo Bay

While the Spacelab flight unit is installed in the Orbiter cargo bay and the cargo bay doors are closed, the Orbiter cargo bay atmosphere will be controlled to provide the following conditions:

Air Purge - Ground Transport, VAB and OPF

- Flow rate - (0 - 42.7 kg/min) (0 to 94 lbs/min)
- Temperature - Selectable within the range of ($18.3 - 29.4^{\circ}$ C)($65 - 85^{\circ}$ F) controlled to ($\pm 2^{\circ}$ F) $\pm 1^{\circ}$ C of the desired setting.
- Cleanliness - Nominally class 100 guaranteed class 5000 (HEPA filtered) air with 15 ppm or less hydrocarbons based on methane equivalent.
- Humidity - Equal to or less than 50 percent relative humidity.

Prior to the cargo bay doors being closed, the cargo bay environment will be maintained by providing a facility purge of:

- Temperature: $21 \pm 3^{\circ}$ C ($70 \pm 5^{\circ}$ F)
- Cleanliness: Nominally class 100 guaranteed class 5000 (HEPA filtered)
- Humidity: 45 ± 5 % RH

Air Purge - Launch Umbilical Panel

A conditioned air purge will be supplied to the cargo bay up to 30 minutes prior to propellant loading; subsequently, GN_2 will be supplied until lift-off. The purge capability at the launch umbilical panel is as follows:

Air Purge

- Flow rate - 0 to 50 kg/min (0 - 110 lb/min)
- Temperature - Selectable within the range 7.2 - 48.8° C (45 - 120° F) controlled to ± 1° C (± 2° F) of desired setting.
- Cleanliness: Nominally class 100 guaranteed class 5000 (HEPA filtered) air with 15 ppm or less hydrocarbons based on methane equivalent.
- Humidity: Equal to or less than 50 percent relative humidity.

GN_2 Purge - Launch Umbilical Panel

- Flow rate - 0 - 165.3 kg/min (0 - 364 lb/min)
- Temperature - Selectable between 7.2° and 48.8° C (45 - 120° F) controlled to ± 1° C (± 2° F)
- Cleanliness - Nominally class 100, guaranteed class 5000 (HEPA filtered) with 15 ppm or less hydrocarbons based on methane equivalent
- Humidity - 0 to 0.14 grams/kg of dry GN_2

Air Purge - Post Landing (Runway to OPF)

- Flow rate - 0 - 42.7 kg/min (0 - 94 lb/min)
- Temperature - Selectable within the range 7.2 to 48.8° C (45 - 120° F) controlled to ± 1° C (± 2° F) of the desired setting
- Cleanliness - Nominally class 100 guaranteed class 5000 (HEPA filtered) air with 15 ppm or less hydrocarbons based on methane equivalent
- Humidity - Equal to or less than 50 percent relative humidity

5.4.2.3 During Transportation

During transportation of flight hardware the transported article is protected by transportation equipment. The atmospheric conditions for the transported article are depending on the external atmospheric conditions limited only by passive protection. The transportation equipment is designed such that the following conditions are not exceeded for flight hardware when subjected to external atmospheric condition.

Temperature:	− 10 to + 55° C
Humidity:	< 50 % RH for integrated Spacelab, Spacelab Modular Elements and Assemblies
	< 90 % RH for Equipment
Pressure:	Sea Level Pressure to 0.27 bar

5.4.3 Cleanliness

Prior to incorporation into any Spacelab integration activities, all equipment must be cleaned to a visibly clean level, as defined in JSC Specification SN-C-0005. It will be visually inspected upon receipt for integration and will be maintained in a 100,000 class clean environment through all activities until launch. More detailed data is in paragraph 5.2.7.

5.4.4 Terrestrial Environment

Except under emergency conditions Spacelab equipment will not be exposed to uncontrolled terrestrial environments. If an emergency mate or demate of the Spacelab from the Orbiter is required on the launch pad, exposed equipment surfaces may be subjected to such environments. However, every reasonable effort will be made to avoid such exposure.

Editorial note: A listing of uncontrolled environmental exposures due to emergency conditions will be supplied at a later date.

6. OPERATIONS

This section describes the concepts and functional requirements for Spacelab operations. Operations, as described herein, encompass those activities to be performed after a complement of experiments (or payloads) has been approved to fly on a specific Spacelab flight. The activity involvement contained in this section includes those functions necessary to prepare the Spacelab and its payload for launch, their operation in orbit, and return for post flight assessment, refurbishment and maintenance.

6.1 Ground Operations

Ground operations are those activities directly associated with the premission physical preparation of Spacelab and experiment hardware for the flight, and the post mission activities associated with Spacelab and experiment recovery and Spacelab maintenance.

6.1.1 Introduction

6.1.1.1 Purpose

The purpose of this section is to describe the Spacelab payload ground operations concept and capabilities. The ground operations concept is mission independent and is applicable to all Spacelab processing operations. The operations described in this section address those operational processing events necessary to integrate payloads with the Spacelab and Shuttle during prelaunch, launch, post landing and refurbishment activities. Details of Spacelab operational interfaces and Spacelab ground support equipment and facilities interfaces are provided to allow experiment design, development and integration up to a level where a group of individual experiments are integrated into a complete Spacelab payload. Integration of a complete Spacelab payload with Spacelab subsystems and integration of Spacelab with the Orbiter are briefly described.

6.1.1.2 Scope

Addressed are those pre-flight and post flight ground processing activities directly associated with the preparation of Spacelab and its experiment payload. Not addressed are the experiment activities which occur at, or prior to experiment end item acceptance.

The contents of this section are arranged to provide the Spacelab user with a logical sequence of information necessary to prepare a Spacelab payload for a scientific mission or flight. The data presented include:

- Definition of levels of integration
- Description of the ground operation concept
- Ground operations processing activities
- Standard NASA Spacelab facilities

6.1.3 Groundrules and Assumptions

- The integration levels in ground operational processing of Spacelab, and of its experiment payloads are:

  Level I – integration and checkout of the Spacelab and its payloads with the Shuttle Orbiter, including the necessary pre-installation testing with simulated interfaces.

  Level II – integration and checkout of the combined experiments mounting elements (e.g. racks, rack sets, and pallet segments) with the flight subsystem support elements (i.e. basic module, igloo) and extension modules, when applicable.

  Level III – combination, integration and checkout of all experiment mounting elements (e.g. racks, rack sets, and pallet segments) with experiment equipment already installed and of experiment and Spacelab software.

  Level IV – integration and checkout of experiment equipment with individual experiment mounting elements (e.g. racks and pallet segments).

6.1.1.3 Groundrules and Assumptions

- Spacelab Level IV integration activities can be performed at remote sites within or outside USA.
  - Refurbished Spacelab pallets and racks will be shipped to remote sites as required to support Level IV integration activities.
  - Spacelab GSE provided to support Level IV integration activities at remote sites will be limited to equipment necessary to support the handling and transport of pallets and racks.
  - Refurbished Spacelab common payload support equipment (CPSE) will be shipped to remote sites on an as available basis dependent upon unique experiment requirements.
- Spacelab Level III integration activities will be performed at KSC and possibly also in EUROPE.
- Spacelab Level II integration activities will normally be performed at KSC.
- Spacelab Level I integration activities will be performed at the Space Shuttle launch site (KSC or WTR).

- Integrated Spacelab/Orbiter processing activities considerations and constraints include:

  - Turnaround times will be constrained by the Shuttle 160 hour turnaround timeline.

  - The tunnel will be installed into the Orbiter after Spacelab installation.

  - The installation of Spacelab in the Shuttle Orbiter will take place with the Shuttle Orbiter in a horizontal position. The contingency capability for on pad removal is provided which includes the capability to handle Spacelab in a vertical configuration with provisions for transfer from the vertical to the horizontal position. Spacelab vertical installation is not planned.

  - After Orbiter installation, access will be possible to the interior and to the exposed exterior of Spacelab. Spacelab and its GSE are designed to provide limited access for experiment servicing during ground operation.

    the exposed exterior of Spacelab. Spacelab and its GSE are designed to provide limited access for experiment servicing during ground operations in a vertical position. As a goal, access to experiments will be possible up to 4 hours before launch on a contingency basis. Access to the interior of the Spacelab during pad operations will be limited to functions which cannot be performed while Spacelab is in the horizontal position. Access to the pallet and the exposed exterior of Spacelab during pad operations is not normally planned, but can be made available if required. As a goal post landing access to the module interior will be possible at the landing strip. Access to the pallet will be possible approximately 9 hours after landing.

  - After Orbiter installation, power and monitoring capability will be provided consistent with the capabilities of Orbiter and GSE during ground flow. Experiments must be able to withstand periods of no STS power and monitoring capability of up to 26 working hours during the flow.

  - Normally Spacelab/Orbiter recovery will occur at the launch site; however, this does not preclude the possibility of recovery at an alternate or contingency landing site.

- Spacelab processing activities considerations and constraints include:

  - The post mission refurbishment of Spacelab hardware will be normally be accomplished at KSC.
  - Following completion of post flight checks the rack/rack set/floor assemblies and/or pallets will be demated from the Spacelab Module and/or Igloo for experiment removal and refurbishment.
  - Experiments removed from Spacelab flight hardware elements during post mission processing will be returned to the appropriate user.
  - Caution and warning indications required for experiments which have hazardous conditions will be displayed by GSE during active subsystems testing or operation.

- Experiment user considerations and constraints include:

  - Unique experiment GSE required for support of ground testing, monitoring, and servicing of experiments will be minimized by making maximum use of the Spacelab and experiment flight systems to support these functions. Instrumentation system capabilities and sensors required to support ground test activities must be included in the flight experiment wherever practical in order to minimize the requirements for ground support equipment.
  - All experiment unique test and servicing equipment and experiment GSE must be provided by the experimenter.
  - Experiment unique GSE will be operated by experimenter supplied personnel and used under the observation and scheduling of the Spacelab processing team.
  - Experiment unique GSE must be designed to interface with standardized interfaces.
  - The experimenter will be responsible for spares support of their respective deliverable hardware. Spares should precede or accompany the delivery of experiment related items.
  - Spacelab and Shuttle Orbiter payload bay requirements are based upon the need to maintain a Class 100,000 cleanliness level during all ground processing and mission phases. Specific Spacelab payloads may require Class 10,000 cleanliness level. It will be the responsibility of the experiment user to provide those unique provisions necessary to maintain experiment class 10 K cleanliness environments.

- User/NASA involvement in experiment calibration will be as follows:
  - User/experimenter will be responsible for experiment calibration.
  - NASA/KSC will provide standard calibration laboratory facilities to calibrate and repair test instrumentation when required.

6.1.2 Ground Operations Concept

This section defines the baseline concept for Spacelab ground processing. Included are a top level flow diagram, description of major processing activities and significant guidelines applicable to both initial and operational phases which are defined as follows:

- The initial phase includes the receipt and post transportation verification of new Spacelab flight hardware following its initial delivery from Europe and the ground processing of this hardware during the first two Spacelab missions.
- The operational phase consists of the routine turnaround ground processing of fully operational Spacelab hardware between subsequent STS missions.

6.1.2.1 Spacelab Ground Operations Philosophy

The Spacelab ground operations philosophy is influenced by two primary factors of the Spacelab program, that is (1) the reusable nature of the Spacelab carrier elements and (2) the Spacelab traffic model. To minimize Spacelab program costs and associated cost to the user, maximum usage of flight hardware, GSE, and associated facilities is planned. To achieve the optimized use of these resources requires a pre-planned flow of the hardware with well-defined user/Spacelab responsibilities and activities, thereby keeping to an absolute minimum the time in process of Spacelab elements.

The major user responsibility in the ground processing flow is for performance of the experiment, which requires that the experiment instruments function properly. Accommodation of this responsibility is provided through the Level IV integration function (see Figure 6- 1) where the user can ensure the proper installation and functioning of the instruments at his facility, or selected site.

The Spacelab responsibility, to ensure that the Spacelab meets its performance criteria, is accomplished separately to Payload Integration operations. Emphasis during integration operations is placed upon verification of interfaces not previously tested.

6.1.2.2 Ground Operations Processing Activities

The Spacelab operational phase Ground Operations encompass those operations associated with the normal turnaround processing of fully operational Spacelab flight hardware. The following sections provide a summary of the basic ground processing flow applicable to the operational phase, with exceptance for the initial phase as noted. The current sequential processing timeline and a brief summary of the nominal operational activities follow.

6.1.2.2.1 Functional Flow

This subsection presents the baseline Spacelab ground operations functional flow block diagram (Figure 6- 1) which shows the generalized top-level blocks of activities.

6.1.2.2.2 Timeline

The timeline (Figure 6- 2) identifies the sequence constraints and typical time allocations for each block activity shown on the functional flow block diagram.

Staging – Preparation of experiment elements, aft bulkheads and special experiment sections for integration.

Level IV – Integration and checkout of experiment equipment with individual experiment
Experiment mounting elements (e.g. racks and pallet segments).

Level III – Combination, integration and checkout of all experiment mounting elements (e.g. racks, rack sets and pallet segments) with experiment equipment already installed.

Level II – Integration and checkout of the combined experiment equipment and experiment mounting elements (e.g. racks, rack sets and pallet segments) with the flight subsystem support elements (i.e. core segment, igloo) and experiment segment when applicable.

Level I – Integration and checkout of the Spacelab and its payloads with the Shuttle Orbiter, including the necessary pre-installation testing with simulated interfaces.

Post Flight Processing –
Landing and safing operations, removal from Orbiter, disassembly, maintenance and reverification of support systems and pressure elements.

Experiment Element Post Flight Processing –
Element disassembly, experiment removal, refurbish and reverify element, bulkheads and special experiment sections.

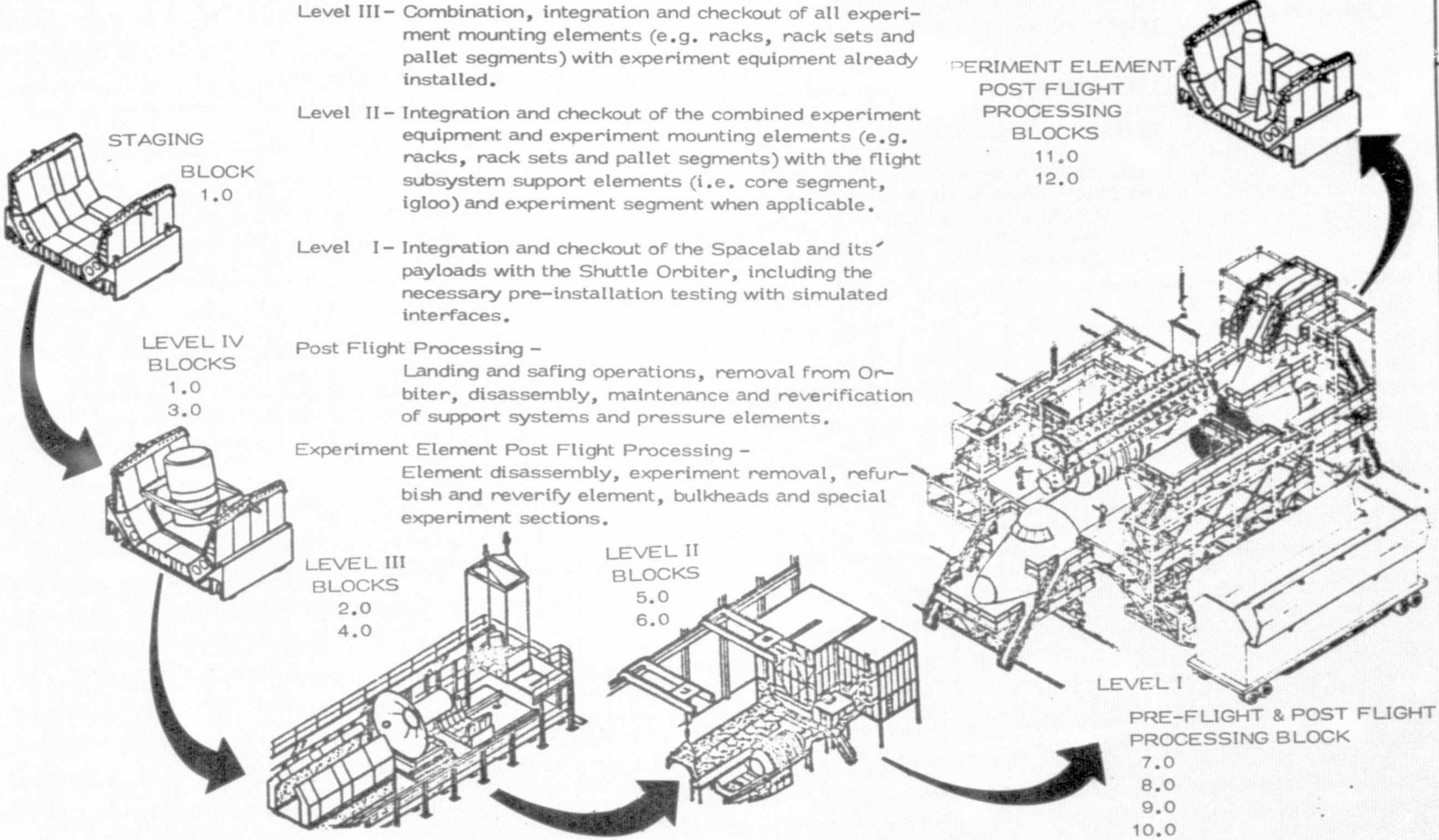
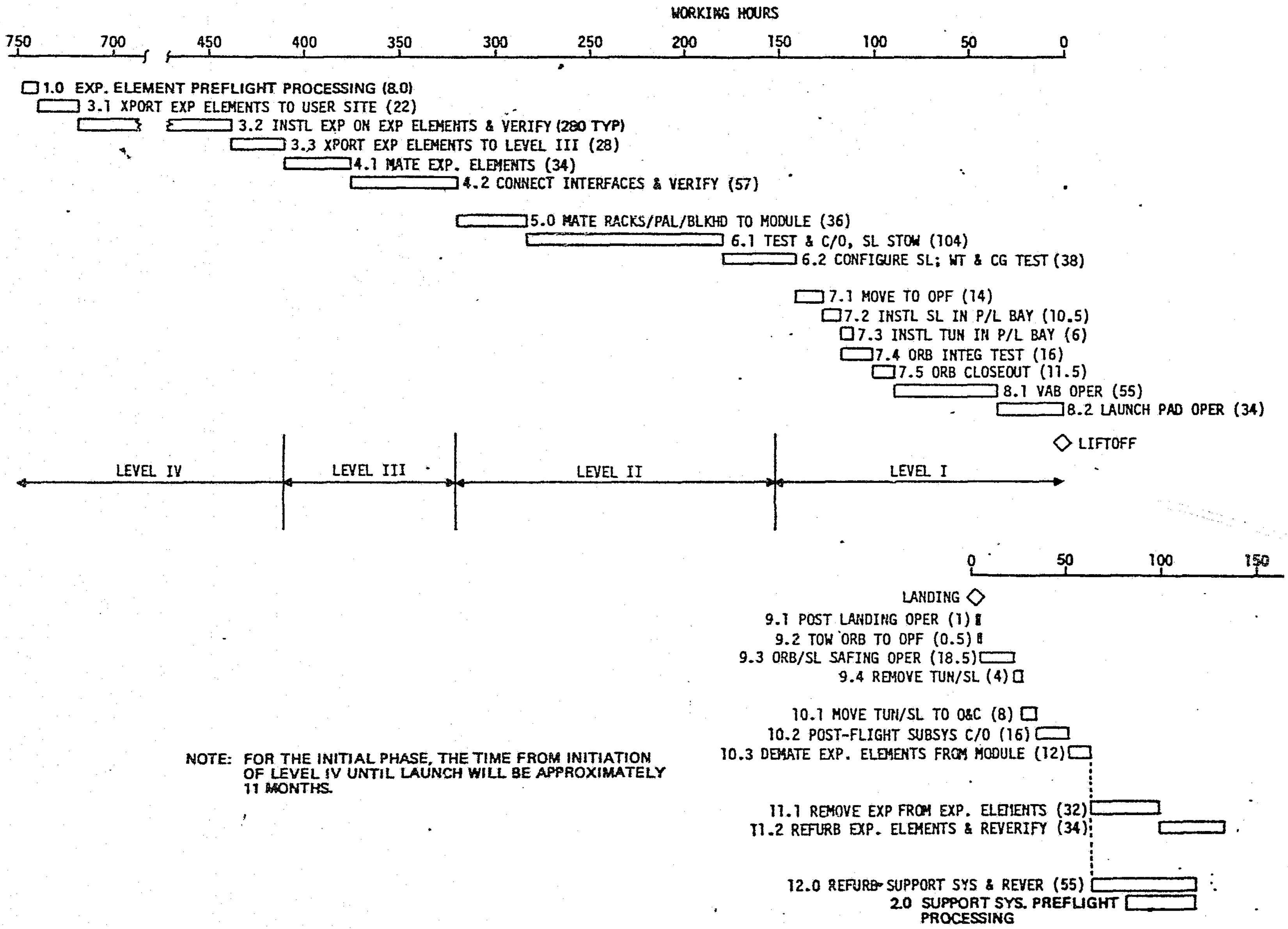


Figure 6 – 1: Typical Spacelab Flight Hardware Processing by Location



NOTE: FOR THE INITIAL PHASE, THE TIME FROM INITIATION OF LEVEL IV UNTIL LAUNCH WILL BE APPROXIMATELY 11 MONTHS.

Figure 6 – 2: Operational Phase Timeline, Spacelab Ground Operations

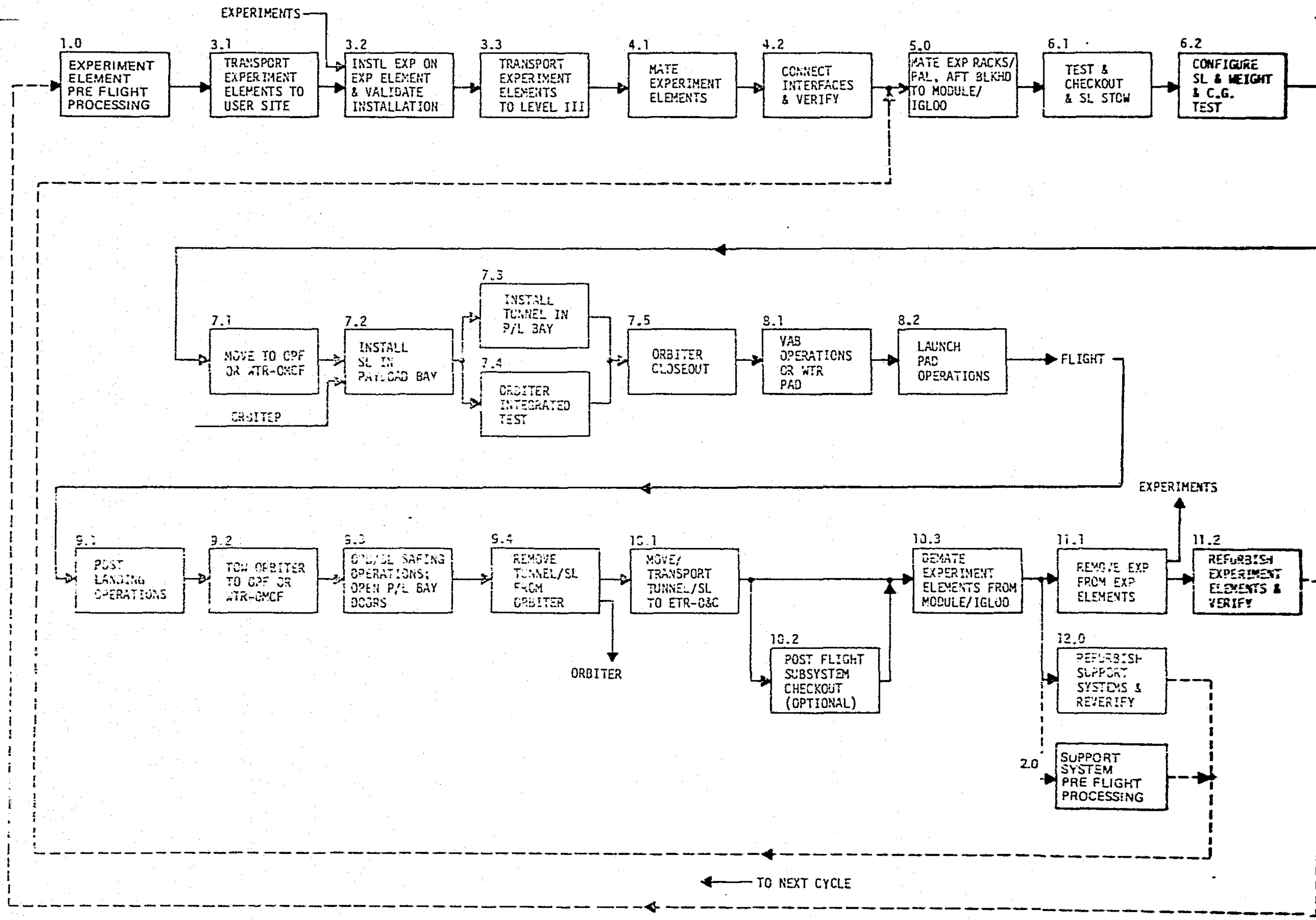


Figure 6 – 3: Ground Processing Flow – Operational Phase

6.1.2.2.3 Activity Descriptions* (see Figure 6 – 3)

Specific allocation of activities within blocks 1 (one) to 12 (twelve) currently under review.

(1) Experiment element pre-flight processing

- Specific missions preparations
- Prep for transportation

(2) Support systems pre-flight processing

- Mate or demate basic and extension segments as required
- Install subsystems, cables, etc., as required
- Install CPSE and mission dependent hardware

(3) Level IV integration (typical)

- Transportation to/from user sites
- Receiving inspection
- Install experiment element in work stations
- Install experiments on experiment elements
- Connect and verify experiment interfaces
- Perform special experiment tests
- Prep for transportation

(4) Level III integration

- Receiving inspection
- Install experiment elements in work stations
- Mate pallet segments, connect interfaces and functionally verify
- Mate rack sets, connect interfaces and functionally verify

(5) Level II assembly

- Inspect and install flight hardware in check out (C/O) stand
- Mate and verify GSE interfaces
- Install rack sets/floors in module and perform bonding checks
- Install and connect pallet/module interface harness
- Connect subsystem and experiment interfaces and functionally verify
- Mate aft bulkhead to module

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*Major paragraph numbers encompass all lower number functions i.e. 3.0 covers section 3.1, 3.2, 3.3.

(6) Level II checkout

- Install payload specialist station (AFD) panels in GSE and connect interfaces
- Load computer programs and perform confidence routines
- Power-up functional interface verification
- Experiment functional checks and calibration if required
- Electromagnetic interference/electromagnetic compatibility (EMI/EMC) testing
- Spacelab/simulated Orbiter verification
- Mission sequence check
- Stow non-time-critical flight items
- Perform final inspection and Spacelab closeout
- Perform weight and center of gravity checks
- Remove AFD panels from GSE

(7)/(8) Level I integration

Spacelab/Orbiter integration

- Move Spacelab to Orbiter processing facility (OPF) or transport to Western Test Range (WTR)
- Control, monitor and verify shipping environment
- Install utilities harness and piping
- Install Spacelab in Orbiter and verify interfaces
- Install tunnel in Orbiter and leak check interfaces
- Install AFD panels in Orbiter cabin and verify interfaces
- Provide access into Spacelab
- Orbiter integrated test - verify Spacelab/Orbiter functional interfaces
- Perform Spacelab closeout inspection
- Stow time-critical items in Spacelab
- Remove all non-flight protective covers
- Remove all access equipment
- Configure Spacelab cabin switches for launch

- Close payload bay doors
- Activate payload bay purge

Prelaunch/launch activities

- Orbiter activities: assemble Shuttle, move to launch pad
- Monitor caution and warning (C&W) system during launch readiness test
- Install time-critical items, if required at this time
- Perform experiment final servicing, if required

Ref.: Lift-Off, Flight and Landing

(9) Orbiter post flight processing

- Orbiter Activities: Deactivate and secure Orbiter, exchange crews
- Safe Spacelab, remove time critical items
- Initiate payload bay purge
- Open payload bay doors
- Remove or cover experiments, as required
- Install Spacelab access equipment (if required)
- Demate and remove tunnel
- Demate and remove Spacelab
- Remove AFD panels
- Prep Spacelab hardware for move or transport (for landings at sites other than KSC)
- Control, monitor and verify shipping environment
- Remove utilities (if required)

(10) Spacelab post flight processing

- Move or transport Spacelab to Eastern Test Range (ETR) – Operations & Checkout Building (O&C)
- Receiving inspection and clean flight hardware
- Install Spacelab in checkout stand and connect to GSE
- Remove experiment specimens and data
- Post flight subsystem checkout – troubleshoot flight anomalies and identify maintenance requirements

- Remove pallets
- Disconnect and remove rack sets from module

(11) Experiment elements disassembly and refurbishment

- Install experiment elements in work stations
- Disconnect and remove experiments from experiment elements
- Prep experiments equipment for shipment to user
- Control, monitor and verify shipping environment
- Disassemble rack sets and pallets
- Perform maintenance, refurbish, and reverify experiment elements, harnesses, etc.

(12) Support system refurbishment

- Perform maintenance, refurbish and reverify core/experiment segments, igloo, aft bulkhead, CPSE tunnel, AFD panels, and harnesses as applicable

Note: Operations activities as they related to European payload integration and processing is under review.

6.1.2.3 User Involvement

Figure 6-4 shows the typical user involvement in the Spacelab ground operations processing and the associated function in which in the item is used.

6.1.2.4 Vertical Payload Removel

Vertical payload removal from the Orbiter bay while the Shuttle is in the stacked on pad condition is a contingency mode which will be supported by the ground operations. Major requirements and constraints for the operational option are as follows:

- Reinstallation of the Spacelab in the Orbiter is not required
- Payload Changeout Room (PCR) provides the capability to handle Spacelab from the payload bay as an integrated package and to position on transporter
- Spacelab is in deactivated status

COORDINATE FUNCTIONS	APPLICATION: Planning Phase	Processing Activities (Ref. Fig. 6–) 1	2	3	4	5	6	7	8	9	10	11	12
1. Experiment Proposal (per S/L users guide)	■												
2. Safety certification data	■				■								
3. Cleanliness/flight approval data	■				■								
4. Configuration requirements of experiment equipment													
• Weight, volume and mass properties	■	■		■	■								
• Dimensions	■	■	■	■	■								
• Layout drawings	■	■	■	■									
• Experiment peculiar drawings	■	■		■	■								
• Experiment peculiar fluid/gas line design	■	■	■	■	■	■							
• Interface requirements (physical & functional)	■	■	■	■	■	■	■	■	■	■	■	■	■
5. Operations and training data	■		■	■	■	■	■	■	■	■	■	■	■
6. Experiment processing documentation (prelaunch & post flight)													
• Preservation	■	■		■	■	■	■				■	■	
• Test requirements	■	■		■	■		■	■					
• Calibration, checkout & maintenance requirements	■	■		■	■		■	■			■	■	
• Mounting & alignment requirements	■	■	■	■	■	■	■	■					
• Servicing/deservicing	■	■		■	■	■	■	■	■	■	■	■	
• Stowage/de-stowage	■	■	■	■	■	■	■	■	■	■	■	■	
• Handling & transportation	■	■		■	■	■	■				■	■	
• Environmental control	■	■	■	■	■	■	■	■	■	■	■	■	
• Experiment removal & shipment												■	
• Experiment holdover requirements		■										■	
7. Personnel/skill to support ground operations		■	■	■	■	■	■	■	■	■	■	■	
8. Provide Hardware													
• Flight experiment hardware		■	■	■	■	■	■	■	■	■	■	■	
• Experiment peculiar GSE and associated plumbing/ cables		■		■	■	■	■	■				■	
• Experiment unique tools		■		■	■	■	■					■	
• Experiment shipping containers		■										■	
• Spare parts/units		■		■	■	■	■	■	■	■	■	■	■
9. Post flight processing data update								Post Flight					

■ Indicates the user supplied item would normally be required to accomplish the given ground processing activity.

Figure 6 – 4: User Involvement in Spacelab Ground Processing

6.1.2.5 Western Test Range (WTR) Launch/Landing Operations

Ground operations will provide the capability of supporting a Spacelab launch from WTR. Major requirements and constraints are as follows:

- The Spacelab and its integrated payload arrives pre-assembled at WTR in an integral pre-checked-out, pre-integrated condition.
- Necessary prelaunch and post launch operations will be accomplished in the Orbiter Maintenance Checkout Facility (OMCF) or Payload Preparation Room (PPR) and applicable.

Figure 6-5 is a flow diagram isolating the unique hardware processing activities and sequence necessary to support a WTR launch/landing option. This flow is the alternate processing flow for WTR and would substitute for items 7.1 thru 9.4 on Figure 6 – 3.

6.1.2.6 Secondary Landing Site Operations

Edwards, Air Force Base (EAFB), California has been designated as a secondary landing site for the Space Shuttle program. Existing facilities and support services will be utilized wherever possible.

After the Orbiter lands, it will be immediately deactivated and secured. The capability will be provided for safing and securing pyrotechnic systems, partially powering down electrical systems, safety-checking the vehicle, and providing for ground and interstitial cooling. Provisions will also be made for Orbiter towing, jacking and leveling the vehicle, maintaining clean and dry environment, draining and purging, Spacelab removal and placing in shipping container and mating the Orbiter to the carrier aircraft.

Operational use of the Edwards AFB secondary landing site will be oriented toward providing the minimum manpower and equipment on site and relying on the transporting of manpower and equipment from the primary landing site to provide the required functions (except for the capability of payload removal and the safing and purging provisions, which will be maintained throughout the operational phase).

6.1.2.7 Contingency Landing Site Operations

Hickam Air Force Base, Hawaii and Anderson Air Force Base, Guam have been defined as contingency landing sites. Only that capability required to tow the vehicle off the runway and safe the systems and maintain the Spacelab in a clean and dry environment is required at the contingency landing site. Capabilities for all other functions will be transported to the contingency sites should the use of these sites be necessary.

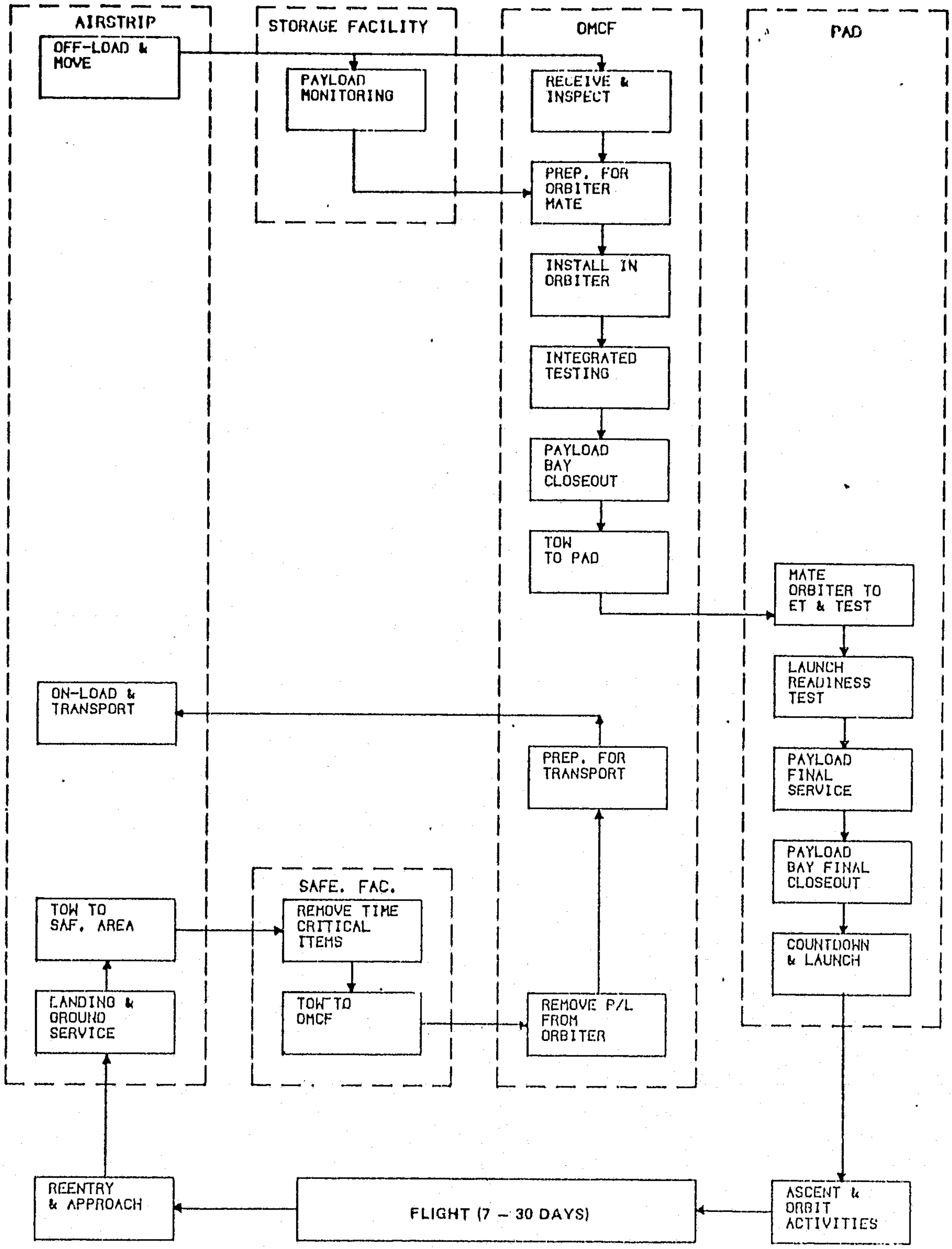


Figure 6 – 5: WTR Launch/Landing Processing Flow

6.1.3 Spacelab Provided Capabilities

This subsection defines those capabilities which are directly related tothe Spacelab ground processing top level functional flow diagram as shown on Figure 6 - 3 .

6.1.3.1 Ground Support Equipment

Ground Support Equipment (GSE) and special facilities will be provided to support the Spacelab ground test program. Items will be allocated to all Spacelab operational processing activities including transportation, receiving inspection, buildup (including experiment installation), Spacelab support for Orbiter servicing and test, subsystem and system tests associated with the Spacelab integration activities required for initial receipt of hardware and subsequent refurbishment activities. The quantity of each GSE item required, its use as well as supplying agency or source for each item along with a brief item description is included in NASA MSFC document 40A99005, Spacelab GSE Allocation and Requirements Plan, dated 27 February 1976.

6.1.3.2 Facility Interfaces (NASA)

The KSC and WTR will have facilities that are capable of supporting the STS payload that will be processed for Spacelab flights; however, the facilities and resources will differ significantly between the two launch sites due to the different processing concepts.

This subjection identifies the payload involvement at KSC, WTR and Level IV experiment integration sites to support planned NASA payload prelaunch and post mission ground processing operations.

6.1.3.3 Facility Resources Availability (NASA)

Subsequent to the installation of the payload in the Orbiter, facilities power, purge, fluid services and personnel access will be provided as follows:

- OPF – Power and environment purge will be provided continuously. Personnel access will be to the interior and exterior of the payload in the horizontal position and will be available during the time the Orbiter bay is open. Subsequent to closure of the Orbiter bay, limited access will be available through the Orbiter cabin.
- Orbiter Tow – Purge gas will be provided during Orbiter towing operations.
- VAB – Power and environment purge will be provided subsequent to the assembly of the Orbiter with other elements.

- Pad/PCR – Access to the exterior ot the interior of the Spacelab in the vertical position will be available via the Orbiter bay. Power and purge will be available until lift-off.

6.1.3.4 Facilities Characteristics (NASA)

Spacelab processing facilities will be compatible with the Spacelab program requirements from the standpoint of: Spacelab and payload power requirements; environmental and cleanliness requirements; ground operations and handling; data handling and processing; hardware interfaces; software; and safety.

The characteristics of the Spacelab NASA facilities are described below.

a) Spacelab Staging

The facility will be capable of maintaining the required temperature, humidity, and cleanliness level within the staging area at 70 ± 5^{o} F max., 50 % (max.) relative humidity, and 100K cleanliness classification. The system will be able to recover to the stated values after exposure to outside ambient conditions (such as transporting in, or out, of Spacelab elements or subassemblies) within three to five hours. If an experiment element requires more stringent temperature, humidity, or cleanliness controls than the facility is capable of providing, then the user will be responsible for providing the necessary environmental control equipment.

Fluids services provided for staging functions include GN_2, missile grade air, shop air, freon and water (potable and distilled). Fluids that are experiment peculiar (such as unique gases) shall be supplied by the user.

Power supplies provided at the work stations include a 28 VDC(regulated and non-regulated); 115 volt, 60 Hz, single phase a.c.; 110/208 volt, 60 Hz, three phase a.c.; and 110/220 volt, 60 Hz, single phase a.c.

The facility will provide overhead cranes of sufficient capacity to move an integrated Spacelab as well as portable cranes for handling Spacelab components and subassemblies. Height for overhead cranes considers GSE handling equipment (such as slings and transportation dollies), and the height of the Spacelab.

b) Level IV Integration

The facility environment will satisfy the cleanliness requirements for at least standard class 100K according to Federal Standard 209 B, April 24, 1974, Clean Room and Work Station Requirements for Controlled Environments.

c) Payload Buildup and Experiment Integration
(Level III Integration)

The facility will be capable of maintaining the required temperature, humidity, and cleanliness level within the Spacelab processing areas at $70 \pm 5^{\circ}$ F max., 50 % (max.) relative humidity, and 100K cleanliness classification (the predominant payload cleanliness requirement). The system will be able to recover to the stated values after exposure to outside ambient conditions (such as during the transporting in, or out, of Spacelab elements) within three to five hours.

Separate environmental controls will be provided that will maintain the required temperature, humidity, and cleanliness level for designated areas requiring better than 100K cleanliness.

The facility air conditioning and humidity controls will maintain proper temperature and relative humidity for the computer facilities and data processing rooms. Heating and air conditioning will maintain the non-Spacelab processing areas within temperature limits that are normal for office buildings.

Fluids services provided for Spacelab processing include GN_2 GO_2, GHe, missile grade air, shop air, freon and water. Fluids that are payload peculiar will be supplied by the user. Industrial water will be required for the wash area.

Power supplies provided at the work stations include: a 28 volt d.c. (regulated and non-regulated); 115 volt, 60 Hz, single phase a.c.; 110/208 volt, 60 Hz, three phase a.c.; and 110/220 volt, 60 Hz, single phase a.c.

The facility will provide overhead cranes of sufficient capacity to move a completed or partially assembled Spacelab as well as portable cranes for handling Spacelab components. Height for overhead cranes must consider the GSE handling equipment (such as the vertical hoisting kit), the height of the Spacelab element, and the GSE transport dolly.

d) Spacelab Assembly and Integration with Payload Assembly
(Level II Integration)

The facility will be capable of maintaining the required temperature, humidity, and cleanliness level within the Spacelab processing areas at $70 \pm 5^{\circ}$ F max., 50 % (max.) relative humidity, and 100K cleanliness classification (the predominant payload cleanliness requirement) as per Federal Standard 209 B. The system will be able to recover to the stated values after exposure to outside ambient conditions within three to five hours.

The facility environmental controls will maintain the required temperature, humidity, and cleanliness level for designated areas requiring better than 100K.

The facility heating and air conditioning will maintain the non-Spacelab processing areas within temperature limits that are normal for office buildings.

Fluid services provided for Spacelab processing include GN_2, GO_2, GHe, missile grade air, shop air, freon and water. Fluids that are payload peculiar (such as unique gases) must be supplied by the user.

Power supplies provided at the work stations include: 28 volt d.c. (regulated and non-regulated); 115 volt, 60 Hz, single phase a.c.; 110/208 volt, 60 Hz, three phase a.c.; and 110/220 volt, 60 Hz, single phase a.c.

Handling equipment includes overhead cranes of sufficient capacity to move a completed or partially assembled Spacelab assembly, as well as portable cranes for handling Spacelab components. Height for overhead cranes must consider the GSE handling equipment (such as the vertical hoisting kit), the height of the Spacelab element, and the GSE transport dolly.

e) Level I Integration

The Level I integration facilities will be capable of maintaining the required temperature, humidity, and cleanliness level in and around the Spacelab at $70 \pm 5^{\circ}$ F max., 50 % (max.) relative humidity, and 100K cleanliness classification.

Fluids services will be provided for Spacelab through umbilical connections.

Note: 1. Fluids that are payload peculiar (such as unique gases) must be supplied by the user.
2. Launch facility will provide capability of topping off fluids to operational levels and pressures (payload service).

6.2 Flight Operations

Flight Operations encompasses those activities, both onboard and on the ground, from ascent to return, that are necessary to accomplish the mission objective, enhance scientific return, and insure crew and vehicle safety. This execution phase, however, must be preceded by operations planning and preparation/ training phases. Operations planning is the analyses to determine the "realtime" activities, and the personnel, equipment, and capabilities required for the optimum execution of these activities. The preparation/training phase consists of the equipment configuration, detailed procedure formulation, and the equipment operation and procedural training of the personnel.

The purpose of this section is to provide potential users with a description of the Spacelab flight operations concept identifying planned services and capabilities as well as inherent requirements and constraints.

Subsection 6.2.1 is a synopsis of the planned Spacelab operations/capabilities and services.

Subsection 6.2.2 consists of the Groundrules and Guidelines under which Spacelab Flight Operations has been developed.

Subsection 6.2.3 summarizes the concept for the User's flight operations utilization of the Spacelab.

6.2.1 Operational Capabilities Description

6.2.1.1 Flight Modes

a) Pallet Only

The physical description of the pallet-only mode configuration variations is given in section three (3) of this document.

In this mode, all scientific equipment will be mounted on the pallet segments in an open space environment and will be operated remotely through the Spacelab Experiment Computer from the Orbiter Aft Flight Deck (AFD) or from a Payload Operations Control Center (POCC) on the ground. The requirements and constraints of individual flights will determine the degree of AFD or POCC control activity. More will be said later in this section about the POCC capabilities.

b) Module Only and Module/Pallet Mode

The various physical configurations involving a pressurized,shirt-sleeves environment module is also given in section three (3).

Since the module makes available more control and display (C&D) capability and more work station space for a large complement of Payload Specialists, primary control of scientific equipment in module configurations will be from the module itself, regardless of whether the equipment is located there or on the pallet segment(s).

The POCC will function in more of a support and advisory capacity to the onboard activity.

6.2.1.2 Flight Phases

a) Ascent/Descent

During ascent, launch through orbital insertion, and descent, re-entry through landing, the module remains pressurized, but the Spacelab, pallet segment, and module, is otherwise inactive, except for equipment necessary to monitor its status and to perform holding functions required by experiments. This holding function will be limited to power and cooling only, within the constraints of the Orbiter resources and the partially active Spacelab subsystem consumption (this capability is not yet baselined).

This holding function will be limited to power and cooling only, within the constraints of the Orbiter resources and the partially active Spacelab subsystem consumption (this capability is not yet baselined).

Interaction with the Spacelab or its payload (scientific equipment) such as initiation, termination, or reconfiguration of equipment operation will not be possible during these phases.

b) Orbital

The Spacelab scientific payload will be operated, including activation and deactivation, during the orbital phase from approximately two hours after launch to two hours before re-entry.

During this period, the Spacelab and Orbiter are fully activated and configured to support payload operations. These physical and resource support characteristics are described in the other sections of this document.

6.2.1.3 Flight Manning

a) On-board Flight Crew

The Orbiter crew consists of the commander and pilot who are always required to operate and manage the Orbiter. In addition, the basic Orbiter crew - commander and pilot - will be responsible for operation of the Spacelab systems. The systems management function will be performed from the mission station of the Orbiter aft flight deck for all Spacelab configurations.

The remainder of the crew - a mission specialist and up to 4 payload specialists - will be primarily responsible for operation of the Spacelab payload. The duties of the crew are:

Commander: The commander will be in command of the flight and will be responsible for the overall space vehicle operations, personnel, and vehicle safety. He will be proficient in all phases of vehicle flight as well as Orbiter and Spacelab systems. He will be responsible for the on-orbit operation and management of Spacelab and Orbiter systems. He may support/perform specific Spacelab payload operations if appropriate and at the discretion of the payload sponsor.

Pilot: The pilot will be in command of overall space vehicle operation and will be equivalent to the commander in proficiency and knowledge of the vehicle and Spacelab systems and operations. He will be responsible for on-orbit operations and management of Spacelab and Orbiter systems. He will normally perform operations with the Remote Manipulator system, and will be the second crewman for EVA operations. He may support/perform specific Spacelab payload operations if appropriate and at the discretion of the payload sponsor.

Mission Specialist: The Mission Specialist will be proficient in Spacelab payload operations. He will have a detailed knowledge of the payload operations, requirements, objectives, and supporting equipment. He will be the prime crew man for EVA operations. He will be responsible for the coordination of overall Orbiter operations in the area of flight planning, consumable usage, and other activities affecting payload operations. He may perform special payload handling or maintenance operations via the Remote Manipulator System. At the discretion of the payload sponsor, he may assist in the management of payload operation, and may in specific cases serve as the payload specialist. Because of training requirements and mission responsibilities, he will be selected by NASA on a career basis.

Payload Specialist: The Payload Specialist will be responsible for the attainment of the payload objectives. The Payload Specialist will be an expert, proficient in payload operations. He will have detailed knowledge of the payload instruments (and their systems), operations, requirements, objectives, and supporting equipment. He will be responsible for the management of payload operations and for the detailed operations of particular instruments or experiments. He must be knowledgeable of certain Orbiter and Spacelab systems, e.g., accommodations, life support, hatches, tunnels, caution and warning systems.

Detailed responsibilities of the Mission Specialist and Payload Specialist(s) will be tailored to meet the requirements of each individual mission. The crew size will be a function of the mission complexity and duration, but the maximum crew, including commander and pilot, is seven persons. The maximum number of Payload Specialists who may be Principal Investigators and may have minimal astronaut training is four.

The module, can accommodate up to three Payload Specialists working for one shift of 12 hours followed by one Payload Specialist for the second 12 hours followed by one Payload Specialist for the second 12 hour shift. For shift overlap up to four Payload Specialists can be accommodated for one hour.

In the pallet-only mode, the work station for the Payload Specialist(s) is the aft flight deck of the Orbiter. A maximum of two Payload Specialists can work simultaneously at the aft flight deck. However, the space available at the aft flight deck may be sufficient for only one Payload Specialist to work in comfort. The layout and mode of operation of the aft flight deck are under definition.

The Orbiter will provide habitability accommodations for all crew members, including food, waste management, sleeping and personal hygiene.

b) Ground-based Flight Support

The primary Spacelab Program objective is to provide the scientific commity easy, economical access to space. Consistent with this objective is the direct involvement of ground-based scientific personnel in flight operations. The NASA is currently structuring itself to facilitate as much direct realtime User participation as possible in flight support activities.

For Spacelab flights, NASA will provide a Payload Operations Control Center (POCC) which will function in conjunction with the Space Transportation System Mission Control Center (SMCC) to conduct and support Orbiter/Spacelab/Payload operations.

POCC implementation details not available at this time will be provided in subsequent documentation.

The level and type functions of this facility will vary according to the requirements and constraints of a particular flight as mentioned previously concerning the composition and duties of the onboard flight crew.

Generally, however, the following functions will be provided for in the two indicated major flight modes:

- Module Mode (sufficient Payload Specialists on board)
  - Permit ground-based personnel to interface with onboard payload specialists.
  - Provide ground-based user personnel with mission data for evaluation and mission input.
  - Manage onboard payload functions wasteful of payload specialist time.
  - Perform contingency analysis
  - Optimize realtime payload activities rescheduling.
- Pallet-Only Mode (limited Payload Specialists on board)
  - Allow ground-based user personnel to interact with their experiment.
  - Provide ground-based user personnel with mission data for evaluation and mission input.
  - Manage Payload Operations
  - Perform contingency analysis
  - Perform realtime payload activities rescheduling.

6.2.2 Flight Operations Groundrules and Guidelines

The following groundrules and guidelines have been used in developing the concept for Spacelab Flight Operations presented in the following subsection 6.2.3.

- Payload mission requirements will be in the responsibility of the user(s).
- Payload mission/flight planning will be done by the NASA center assigned to sponsor a particular payload in conjunction with the user(s).
- Final flight plan integration will be done by the NASA Space Transportation System Operator.
- Facilities, capabilities, and support will be provided by NASA for direct user participation in actual onboard flight operations and ground based-flight support operations.
- Preparation and training for participating user personnel in the operation of his specific equipment is the user's responsibility.
- Preparation and training for operating the Spacelab support equipment, related payload equipment, STS support facilities, and ground-based flight support facilities will be provided by NASA.

6.2.3 Flight Operations Concept

For each NASA Spacelab Mission, NASA will identify a Payload Mission Manager whose office will provide a single, consistent focal point for the many activities and elements involved from payload definition through analytical and physical integration, mission planning, preparation and execution.

The following subsections will identify the primary user involvement/participation foreseen for the three phases of Spacelab flight operations.

6.2.3.1 Operations Planning Phase

After a Spacelab payload has been selected a lead NASA center has been designated, and the Mission Manager identified, operations planning will begin.

The individual user(s) will identify their mission objectives; resource, orbit and timeline requirements and constraints; specific payload equipment operating characteristics and particular onboard skill requirements. NASA will then perform an Operations Requirements Analysis (ORA) for the complete payload based on these inputs and the defined Spacelab configuration/capabilities and recommend assignment of certain tasks to onboard Payload Specialists or the POCC with identification of the associated requirements for Payload Specialists number and skill mix, POCC configuration and personnel and the flight timeline that most optimumly satisfies the mission objectives.

These recommendations will then be iterated with the involved user(s) until agreement by all concerned elements is reached and the next operations phase can begin.

6.2.3.2 Operations Preparation/Training Phase

This phase consists of the implementation of facility configuration requirements, development of specific Payload Specialists and flight support operating procedures, and selection and training of both Payload Specialists and flight support personnel.

NASA will configure the facilities to the previously identified requirements and train the flight support personnel in facility operation and the Payload Specialists in required Spacelab systems operation, Orbiter habitability familiarity and procedures that interface otherwise with the STS. The users will provide the previously agreed upon personnel for flight support and are responsible for the training of this payload team as well as the Payload Specialist(s) in the specific payload equipment operation.

The Payload Specialists may come from any element of the Scientific community (i.e., industrial, scientific, adacemic, governmental). They will not necessarily be either astronauts or career NASA employees, although these personnel are included in the spectrum. Responsibility for nomination, selection, and flight assignment rests with the users on a given flight, and will be accomplished through the mission management structure established for the flight. Candidate payload specialists must meet minimum NASA STS physiological criteria for safety and tolerance of space flight, to be established by NASA. Procedure development will be a joint User/NASA activity.

6.2.3.3 Operations Execution Phase

During this realtime activity, the user flight support teams, which have been designated and trained in facility operation and protocol, will man the POCC, which, as mentioned, has been configured to facilitate their interaction with the specific flight.

The user teams will be assisted by a trained NASA control center cadre and will interface with the STS and the payload through the same Mission Management team described at the beginning of this section.

7. DESIGN REQUIREMENTS FOR EXPERIMENTS

7.1 Purpose of Design Requirements

This section defines design requirements imposed on all experiment equipment carried aboard Spacelab. The purpose of these requirements is to ensure physical and functional compatibility between the experiment and the Spacelab/Orbiter during all phases of a Spacelab mission and to minimize the risk of damage and/or hazardous conditions which would affect the safety of personnel or equipment. It is not the purpose of these requirements to ensure that the experiments will meet their scientific and functional objectives. The experiment objectives, design, development and performance will be the responsibility of the experiment developer and /or user (subject to mission peculiar constraints which may be imposed for any particular mission).

7.2 Mechanical Design Requirements

7.2.1 Experiment Mass and Volume

Experiment equipment shall be compatible with the mass and volume constraints described in Sections 3 and 4.

7.2.2 Experiment Mounting Interfaces

Experiment equipment shall be designed to utilize the standard Spacelab/Orbiter attachment points and storage provisions described in Sections 3 and 4.

7.2.3 Experiment Integrity

All experiment equipment shall be designed so that it can withstand the launch, operational and re-entry dynamic environment defined in Section 5 without failures, leaking hazardous fluids, or releasing equipment, loose debris and particles which could damage the Spacelab/Orbiter or cause injury to the crew.

Experiment equipment shall be designed so that the package integrity and load carrying capability of structural mounting provisions have the following minimum factors of safety in lieu of performing static load structural tests:

Yield factor of safety	=	2.0
Ultimate factor of safety	=	3.0

Experiment equipment shall be designed so that, when subjected to the crash landing environment specified in Table 5 – 3, there shall be no hazard to personnel or prevention of egress from a crashed vehicle.

7.2.4 Extension, Ejection, Deployment and Capture

7.2.4.1 Emergency Retraction and Ejection

Experiment equipment (except that which is mounted on the IPS or on the top airlock extension platform) which may be extended beyond the Orbiter cargo bay envelope shall be designed with a capability for emergency back-up retraction and/or ejection.

The design of all such emergency capabilities shall allow their initiation from inside the Orbiter and/or the module. Residual material following emergency retraction or ejection shall not interfere with the closure of Orbiter cargo bay doors, airlock hatches, window covers etc.

7.2.4.2 Routine Ejection, Deployment and Capture

Experiment equipment such as subsatellites, canisters etc. which is intended for deployment as free flyers outside the Orbiter payload bay envelope with or without subsequent recapture shall be designed to comply with TBD requirements.

Equipment which is designed for deployment and/or recapture using the Orbiter Remote Manipulator System shall comply with the specific requirements of TBD.

7.2.5 Crew Interface

7.2.5.1 Loose Equipment Restraint

Means shall be provided for convenient temporary containment or restraint of all loose experiment equipment that cannot be contained or restrained by Spacelab provisions. This includes items which become loose as a result of disassembly or activation of equipment on orbit. All fasteners, latches, retainers, etc. that are handled by the crew on orbit should be made captive.

7.2.5.2 Handholds and Handrails

Specific handholds and handrails shall be provided where sufficient capability is not inherent in basic equipment and/or where Spacelab equipment with handholds (e.g. experiment racks) is replaced with experiment equipment. The minimum clearance between handholds/handrails and structure shall be 57 mm and the minimum straight grasping length shall be 150 mm.

Handhold and handrail stand-offs shall not form finger traps. Ends of handholds and handrails shall have a minimum radius of 25 mm and have a maximum gap of 5 mm from the surface.

7.2.5.3 Equipment Transfer On Orbit

The following requirements apply to experiment equipment which has to be relocated on orbit. Equipment which has a mass greater than 45 kg shall have a handle or equivalent grasping surface. Equipment which has a mass greater than 95 kg shall have 2 handles or equivalent grasping surfaces. Equipment which is larger than 0.03 m^3 shall have a handle or equivalent grasping surface. Equipment which is more than 0.2 m^3 or 110 kg shall have provisions for 2 crew members to handle it.

7.2.5.4 Corners, Edges and Protrusions

The following requirements apply to experiment equipment which is accessible to crew members:

<u>Exposed Edges (see Figure 7-1)</u>

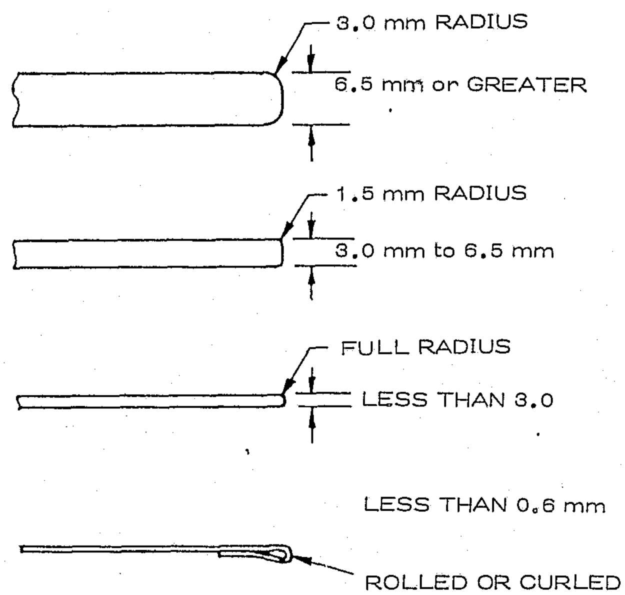


Figure 7-1: Exposed Edges Design Criteria

Exposed edges 6.5 mm thickness or greater shall be rounded to a minimum radius of 3.0 mm.
Exposed edges 3 to 6.5 mm thickness shall be rounded to a minimum radius of 1.5 mm .
Exposed edges less than 3 mm thickness shall be rounded to the full radius.
Exposed edges less than 0.6 mm thickness shall be cu. led or bent.

Exposed Corners (see Figure 7-2)

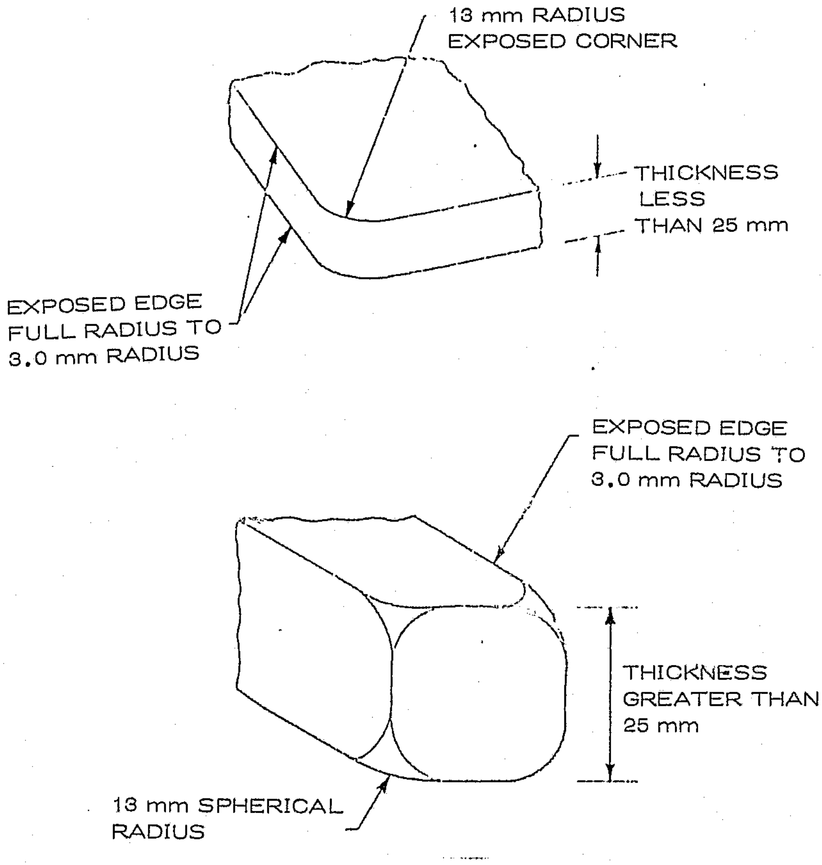


Figure 7-2: Exposed Corners Design Criteria

Exposed corners (material less than 25 mm thickness) shall be rounded to a minimum radius of 13 mm.

Exposed corners which cannot be rounded to 13 mm shall be rounded to a dimension which approximates 13 mm as closely as possible.

Exposed corners (material greater than 25 mm thickness) shall be rounded to 13 mm spherical radius.

Protrusions

Protrusions shall have all sharp edges and exposed corners removed or eliminated in accordance with Figures 7-1 and 7-2. Protrusions which (for operational reasons) cannot be made safe shall be covered with a protective device.

Loose Equipment

Loose equipment with hard surfaced exposed corners, edges and protrusions shall have minimum corner and edge radii or spherical radii as specified in Table 7-1.

Table 7-1 Minimum Corner and Edge Radii for Loose Equipment

Mass (kg)			Edge Radius (mm)	Corner Radius (mm)
0	to	0.25	0.3	0.5
0.25	to	0.5	0.8	1.5
0.5	to	3	1.5	3.5
3	to	15	3.5	7
15	to	50	3.5	13

7.3 Thermal Requirements

Experiment Equipment shall be designed for compatibility with the Spacelab/Orbiter ECS capabilities and interfaces described in Sections 3 and 4.

The temperature of any experiment equipment surface which is accessible to the crew inside the module or inside the Orbiter shall not exceed 45° C.

The temperature of any experiment equipment surface which is intended to be accessible to the crew during EVA shall be maintained in the range from -70° C to 93° C.

7.4 Electrical Power Requirements

Experiment equipment shall be designed for compatibility with the Spacelab power and energy capabilities described in Sections 3 and 4. Experiment equipment of Spacelab shall interface with the EPDS only via the experiment power distribution boxes, the experiment power switching panels, the top and aft airlock power connectors and the IPS power connector defined in Section 4. Experiment equipment in the Orbiter shall interface with TBD.

7.5 Command and Data Handling Requirements

Experiment equipment shall be designed for compatibility with the Spacelab CDMS capabilities and characteristics described in Sections 3 and 4 and shall interface with the CDMS only at the standard interfaces defined in Section 4.

7.6 GSE Requirements

7.6.1 Spacelab/Orbiter/Integration Center/Launch Site Provided GSE

The requirements for experiment equipment which interfaces with GSE provided by the Spacelab, Orbiter, Integration or Launch Site are TBD.

7.6.2 Experiment Provided GSE

The requirements for experiment-provided GSE which interfaces with the Spacelab and/or the Orbiter, or which interfaces with GSE provided by the Spacelab, Orbiter, Integration Center, or Launch Site are TBD.

7.7 Environmental Requirements

7.7.1 Natural and Induced Environment

Experiment equipment shall be designed to be compatible with the natural and induced environmental levels specified in Section 5. The design shall be such that any damage or malfunction of experiment due to the natural and induced environment shall not adversely affect the Spacelab, Orbiter or crew.

7.7.2 Experiment Induced Environment

7.7.2.1 Acoustic Environment

The audible noise generated by any experiment inside the module shall not exceed the noise rating curve NR 40 of ISO-R-1996 or the noise rating curve NC 40 of the United States Noise Standard.

7.7.2.2 Electromagnetic Environment

7.7.2.2.1 Bonding and Shielding Requirements

Experiment equipment shall be effectively enclosed and shielded such that the equipment is compatible with the electromagnetic environment specified in Section 5 and complies with the requirements regarding AC magnetic field emissions in para 7.7.2.2.3.

The external cases of experiment equipment shall ultimately be grounded to the Spacelab structure via the equipment mounting points with the following exceptions:

a) Boxes that have to be thermally isolated from the Spacelab structure or mounted on special shock mounts shall be fitted with a bond strap on the case for connection to Spacelab structure.

b) Experiment equipment housings that need to be electrically isolated from Spacelab structure shall have the case connected to the experiment secondary power return inside the unit.

All bonds shall be sized to carry the maximum credible fault current for the particular unit until actuation of protective devices can occur.

7.7.2.2.2 Isolation and Grounding Requirements

Isolation Requirements

Experiment equipment which utilizes Spacelab DC or AC power shall maintain DC isolation of at least 1 Megohm in parallel with a stray capacitance of less than 10 n F between the power positive and return and experiment chassis for DC power and between the three power and one return lines and experiment chassis for AC power.

All command input circuits interfaces with a RAU shall maintain a DC isolation of at least 1 Megohm between each command positive or return line and chassis. This applies to both powered and unpowered status.

Grounding Requirements

The Spacelab structure shall not be used as an intentional power or signal return line. Within an individual experiment the single point ground principle shall be applied i.e. all electrical references shall be grounded (if required) to Spacelab structure via a single external bond strap.

7.7.2.2.3 Conducted Noise Emission on Power and Signal Lines

7.7.2.2.3.1 Differential Conducted Emissions

These are the emissions appearing between any positive line and its corresponding return.

(1) Experiment DC Power Bus

Within the band 30 Hz - 50 MHz the differential noise appearing on positive or return DC power lines (at the outlets of the experiment power distribution boxes, switching panels, top or aft airlock connectors, or IPS connectors, when loaded with an impedance of the characteristics shown in Figure 7-3, shall be within the following limits:

a) narrowland spectral conducted current: as shown in Figure 7-4. Measurement bandwidths, β, as defined on the Figure.

b) broadband spectral conducted current: as shown in Figure 7-4.

c) time domain conducted current ripple and spikes: not required.

d) time domain conducted voltage ripple and spikes shall be less than 2 Vpp.

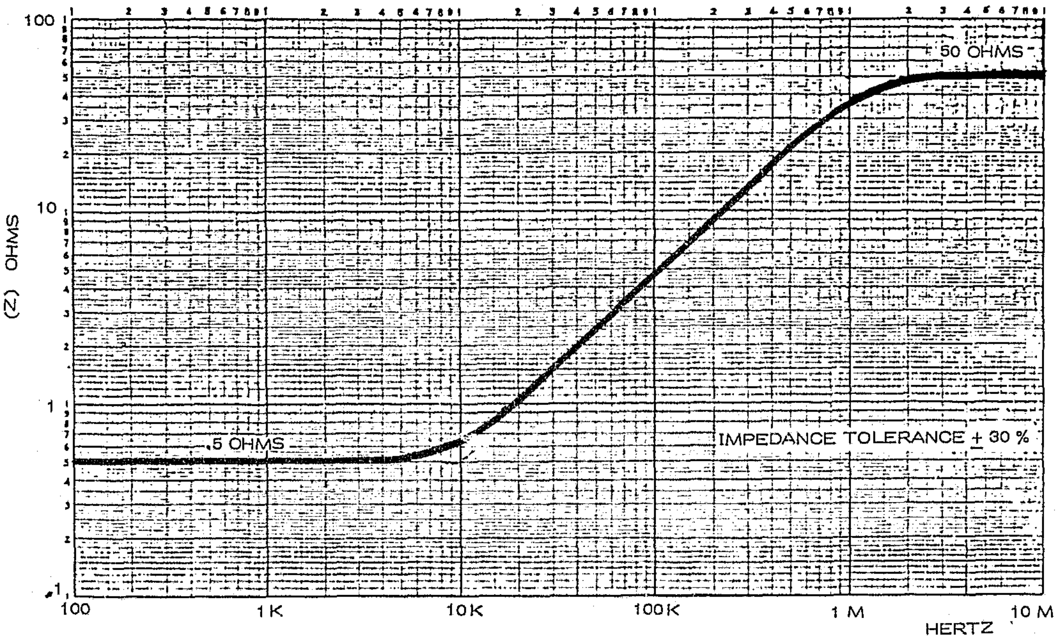


Figure 7-3: 400 Hz and 28 V DC Bus Impedance Characteristic

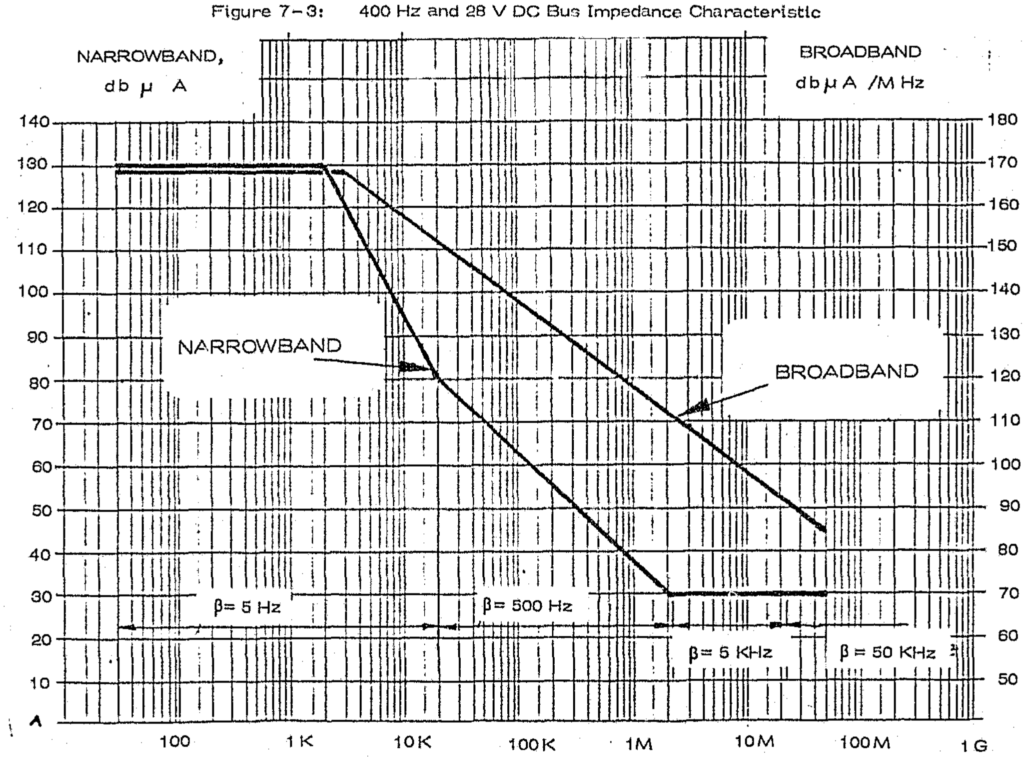


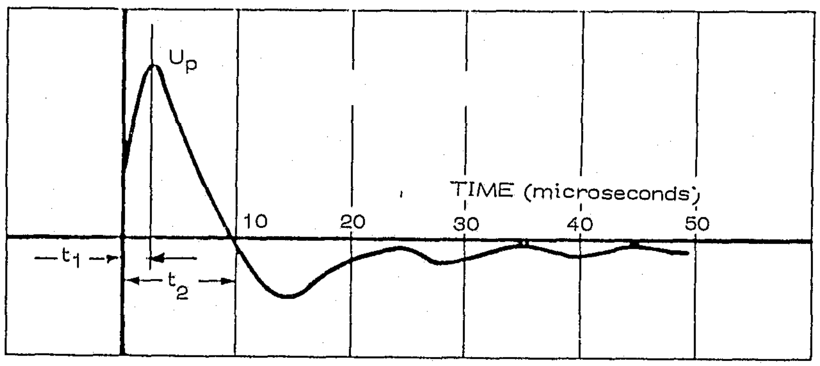
Figure 7-4: Conducted Noise Limits for DC Power Lines

e) inrush transients shall be less than:

$\pm$ 28 Vp in amplitude

10 μ sec in duration

shape as defined in Figure 7 - 5



U_p = 28 V

t_1 = 2 microseconds

t_2 = 10 microseconds

Figure 7 - 5: Conducted Inrush Transient Waveform

(2) Experiment 400 Hz Power Bus

The experiment AC power bus is used only for applying experiment loads and, in principle, the maximum allowable conducted emission levels are those above which the Spacelab inverter would suffer physical damage. Since these levels are unrealistically high for normal equipment the levels specified below are equivalent to those specified for Spacelab subsystem equipment.

Within the band 30 Hz – 50 MHz, the differential noise appearing on positive or return 400 Hz power lines (at the outlets of the experiment power distribution boxes, switching panels, top or aft airlock connectors, or IPS connectors, when loaded with an impedance of the characteristic shown in Figure 7-3, shall be within the following limits:

a) narrowband spectral conducted current: as shown in Figure 7-6.

b) broadband spectral conducted current: as shown in Figure 7-6.

c) time domain conducted current ripple and spikes: not required.

d) time domain conducted voltage ripple and spikes shall be less than 50 Vpp

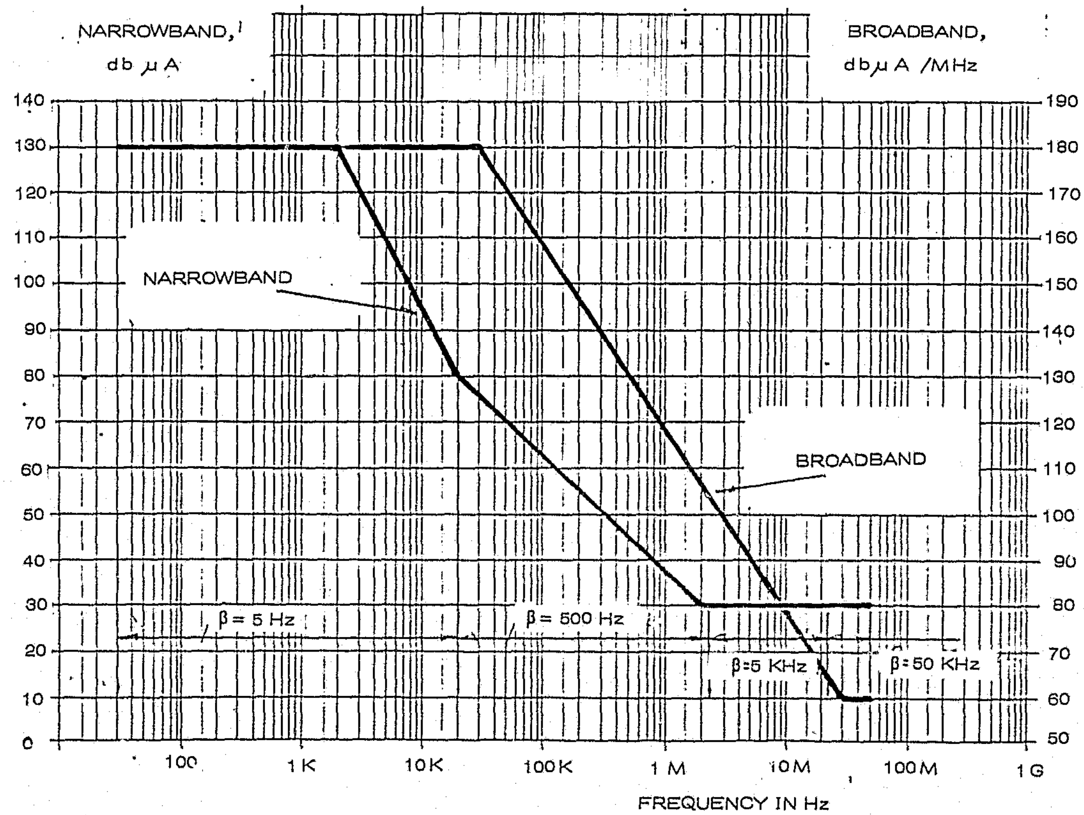


Figure 7-6: Conducted Noise Limits for AC Power Lines

e) inrush transients shall be less than:

± 60 Vp in amplitude

10 μ s in duration

shape as defined in Figure 7-5

(3) RAU Output Lines (On/Off Command Lines, Serial Data and Clock Lines and User Clock Output Lines)

Within the band 30 Hz - 50 MHz, the differential noise appearing (at the RAU connector) on the positive or return lines of the on/off command outputs, the serial data outputs (including clocks) and the user clock outputs when loaded with 2 500 pF in parallel with 100 KΩ shall be within the following limits.

a) narrow band spectral conducted current: not required

b) broad band spectral conducted current: as shown in Figure 7-7

c) time domain conducted current ripple and spikes: not required

d) time domain conducted voltage ripple and spikes:

LOGIC '1' shall be + 4 V ± 1.5 V

LOGIC '0' shall be + 0.2 V ± 0.2 V

Both cases shall include the sume of DC plus AC ripple and spikes.

These levels shall apply for digital lines in both the logic "1" and "C" state.

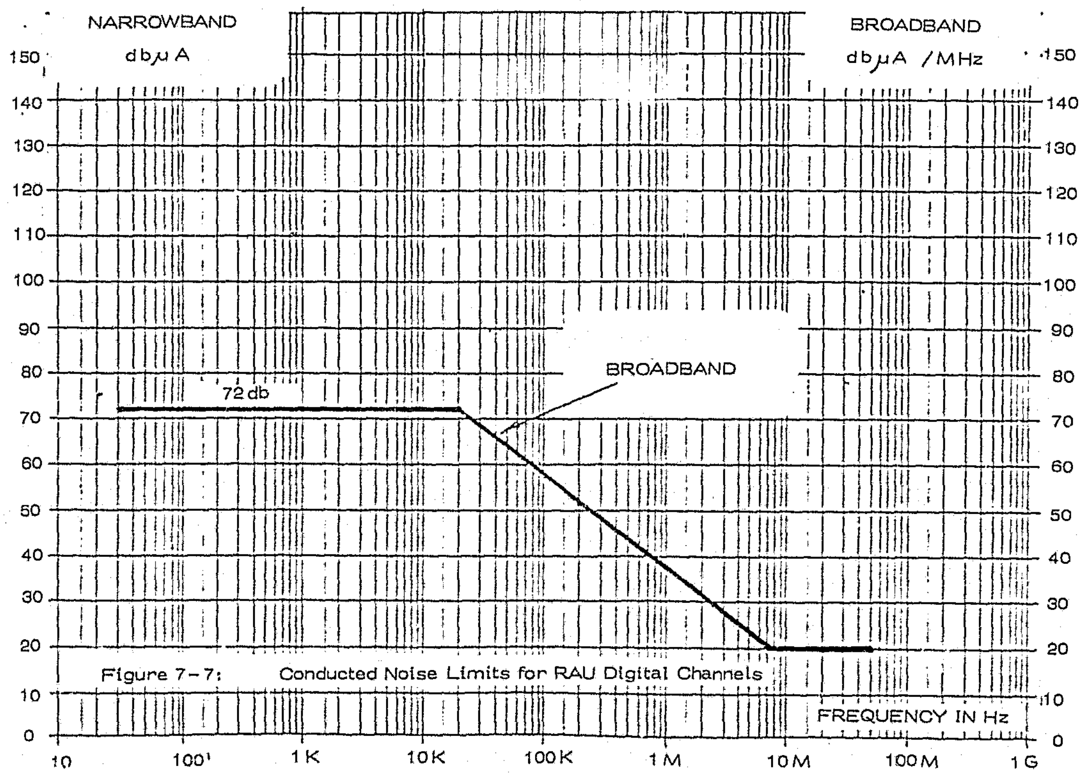


Figure 7-7: Conducted Noise Limits for RAU Digital Channels

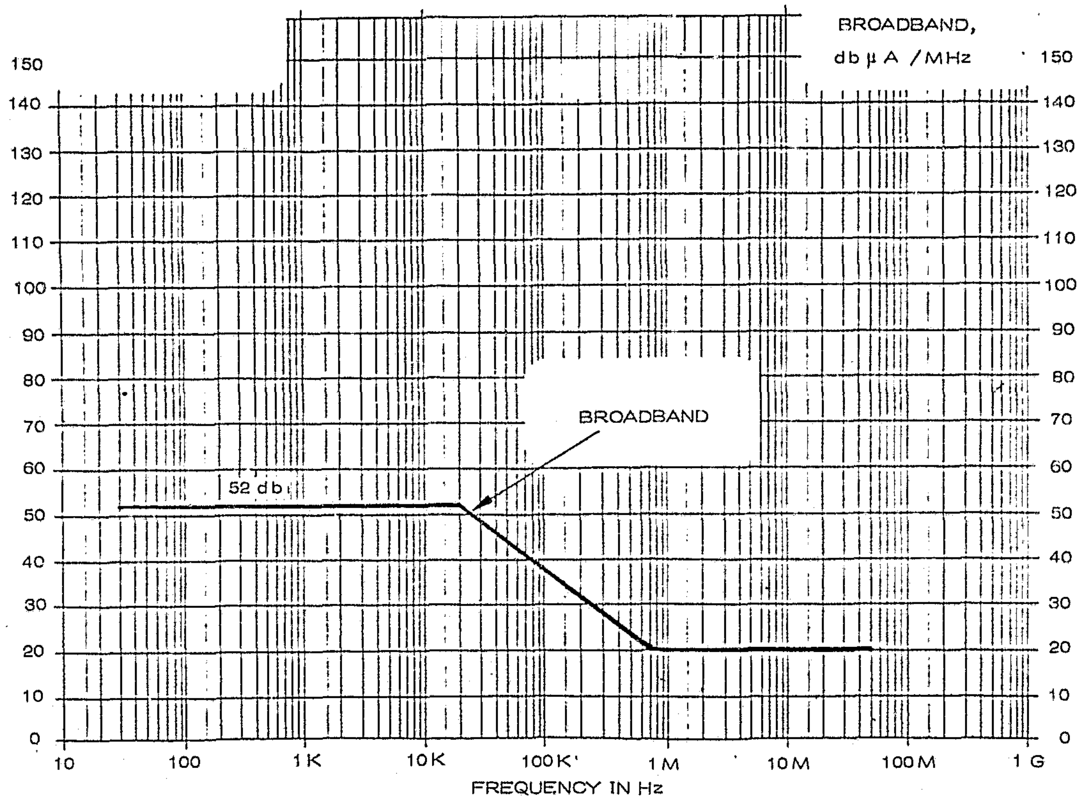


Figure 7-8: Conducted Noise Limits for RAU Flexible (Analog/Discrete) Inputs

(4) RAU Flexible (Analog/Discrete) Input Lines

Within the band 30 Hz - 50 MHz, the differential noise appearing (at the RAU connector) on the positive or return flexible (analog/discrete) input lines, when loaded with 2 500 pF in parallel with 500 KΩ, shall be within the following limits:

a) narrow band spectral conducted current: not required

b) broad band spectral conducted current: as shown in Figure 7-8

c) time domain conducted current ripple and spikes: not required

d) time domein conducted coltage ripple and spikes shall be less than 40 m Vpp

(5) Video Lines and 4.2 MHz Analog Lines: TBD

(6) Multiplexer Input Lines: TBD

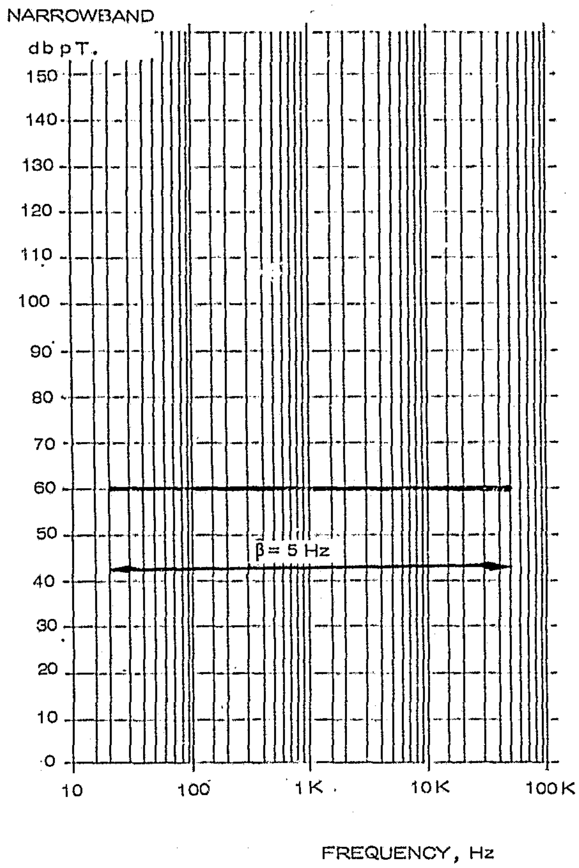


Figure 7-11: AC Magnetic Field Emission Limits

(5) AC Magnetic Field Emissions

Within the band 20 Hz to 50 KHz, at any point 1 meter from the outer edge of experiment equipment (including cable harnesses), the radiated AC magnetic field shall be within the following limits:

a) narrowband spectral: as shown in Figure 7-11 measurement bandwidths, β, as defined on the Figure

b) broadband spectral: not required.

The levels shall apply for both vertical and horizontal polarization.

(6) AC Electric Field Emissions

Within the band 14 KHz to 10 GHz, at any point 1 meter from the outer edge of experiment equipment (including cable harnesses), the radiated AC electric field shall be within the following limits:

a) narrowband spectral: as shown in Figure 7-12 measurement bandwidths, β, as defined on the Figure

b) broadband spectral: as shown in Figure 7-12

The levels shall apply for both vertical and horizontal polarization.

7.7.2.3 Ionizing Radiation

TBD

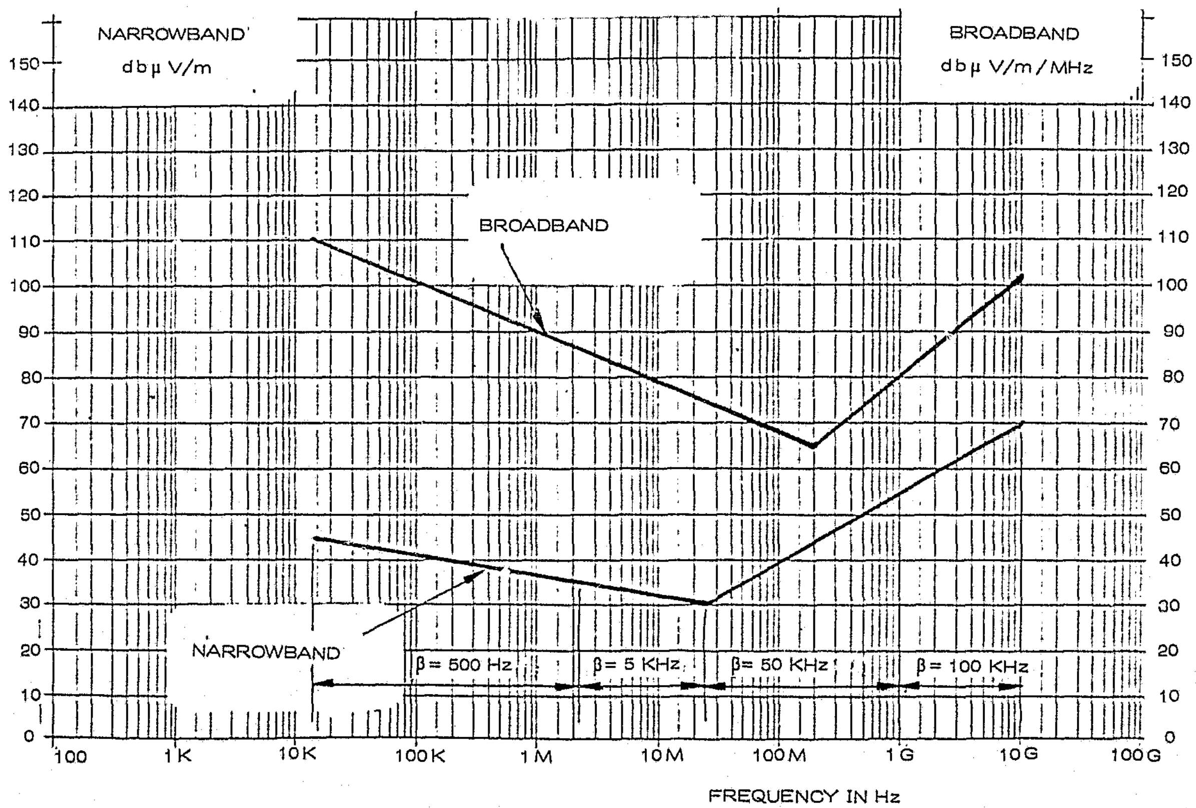


Figure 7-12: AC Electric Field Emission Limits

7.8 Test and Integration

7.8.1 Test Requirements

The requirements for testing experiment equipment prior to integration with Spacelab are TBD.

7.8.2 Integration and Checkout Requirements

The requirements for integration of experiment equipment with the Spacelab and the requirements for experiment checkout during and subsequent to integration are TBD.

7.9 Operational Requirements

The unique requirements and guidelines for the various levels of operations are not established yet. However, some general requirements are listed in the following:

(1) Experiment peculiar ground support equipment for integration and test shall be provided by the experimenter.

(2) Experiment peculiar protection for experiment equipment shall be provided by the experimenter.

(3) Experiments should minimize operation on the ground except to verify interfaces with Spacelab or to satisfy launch site safety and compatibility requirements.

(4) Experiment to Spacelab compatibility testing should be planned to address only unique requirements.

(5) Launch site ground checkout requirements for experiments should be included in design and test of experiment software and checkout procedures.

(6) When experiments require Level III integration interface verification at Level I integration, then they must provide reverification method and reverification capability.

(7) When experiments require some operation during all integration levels until lift-off, they must provide capability to be remotely controllable via CDMS. The use of CDMS for functions other than on/off during these phases shall be minimized.

(8) When experiments require some operation during ascent and re-entry, they must not require any command action from the ground or the Shuttle crew during the powered flight phase except safeing commands.

(9) Experiment requiring remote control from the ground must interface with the Spacelab CDMS.

(10) For module and module/pallet missions the primary control station for experiment operation should be in the module. The need for experiment peculiar control equipment on the Orbiter AFD shall be minimized.

(11) Experiments shall require activation on orbit not earlier than TBD minutes after lift-off.

(12) Experiment design shall allow de-activation TBD hours before touch down.

(13) The experiment should provide the capability to allow easy verification of equipment status and experiment activity to the operator.

(14) Experiments shall be designed to require no physical access on the launch pad unless it is absolutely necessary to achieve experiment objectives.

(15) Experiments shall be designed to require physical access not earlier than TBD hours after landing, unless required to achieve experiment objectives.

7.10 Material Control Requirements

Editorial note: Material Control Requirements are still under discussion between ESA and NASA. This section represents only the ESA proposal.

7.10.1 Purpose of Material Control for Experiments

Some requirements must be imposed on specific properties of materials and controls exercised on materials being used in experiments to avoid hazards to personnel and detrimental effects on Spacelab and Orbiter equipment. Control must be exercised on the following specific material properties:

(1) off-gassing of possibly toxic or odourous trace contaminants from materials used inside the habitable area of Spacelab or the Orbiter.

(2) flammability of materials which can result in fire hazards inside the Orbiter or Spacelab or on the pallets.

(3) outgassing products from materials exposed to vacuum, which may interfere with the correct function of other equipment.

C-5

(4) corrosion or material incompatability which may affect the correct operation of other equipment.

(5) specific properties of "Forbidden Materials" or "Restricted Materials", which are listed in para. 7.10.9.

7.10.2 Experiment Location and Associated Requirements on Specific Material Characteristics

For the purpose of defining relevant material requirements, the following areas in which experiments may be flown, shall be distinguished:

7.10.2.1 Orbiter Flight Deck

Materials exposed to the atmosphere of the habitable area of the Orbiter

(1) shall not offgass toxic or odourous products at the expected worst case temperatures (test and acceptance criteria, see para. 7.10.4).

(2) shall be non-flammable in an atmosphere of 23.8 % O_2 and 1 atmosphere pressure (test and acceptance criteria see para. 7.10.5).

The material selection and control program shall be carried out according to para 7.10.3 a).

7.10.2.2 Spacelab Habitable Area

Materials exposed to the atmosphere of the Spacelab module (including those in the airlocks)

(1) shall not off-gass toxic or odourous products at the expected worst case temperature (test and acceptance criteria see para. 7.10.4).

(2) shall be non-flammable in an atmosphere of 23.8 O_2 and 1 atmosphere of pressure (test and acceptance criteria, see para. 7.10.5).

The material control program shall be carried out according to para. 7.10.3 a) or b).

7.10.2.3 Spacelab Pallets

Materials used in pallet-mounted experiments

(1) shall be non-flammable in normal air (test and acceptance criteria, see para. 7.10.5).

(2) shall have low-outgassing properties in vacuum (test and acceptance criteria, see para. 7.10.6).

7.10.2.4 Airlocks

Materials used in the airlocks shall meet the requirements on off-gassing of toxic and odourous products and flammability as defined for the Spacelab habitable area and on outgassing under vacuum as defined for pallet-mounted equipment.

7.10.2.5 Sealed Containers

The requirements on off-gassing, flammability and outgassing do not apply for materials used inside sealed containers, if such containers do not rupture and emit gases or flames under expected worst case conditions, including internal ignition.

7.10.2.6 Requirements Independent of Experiments Location

The requirements on corrosion, material compatibility, Forbidden Materials and Restricted Materials as defined under paragraph 7.10.8 and 7.10.9 are applicable independent of experiment location.

7.10.3 Material Control Program

The most suitable and effective method and the necessary degree of material control to achieve the required safety for personnel and undisturbed operation of other equipment will depend on the construction of the experiment and its location in Spacelab or the Orbiter.

One of the following approaches or a combination thereof shall be applied:

a) Control of all individual materials being used in an experiment by complete listing and verification of acceptability of each material by material sample testing as applicable according to para. 7.10.4 to para. 7.10.6 (Lists of materials which have been found acceptable with respect to those specific properties can be obtained from ESA and NASA).

b) "Black-Box" testing and analysis to verify acceptability of a completed experiment.

An outline for an off-gassing test on black-box-level to verify that an experiment would not emit untolerable contaminants into the habitable area of the spacecraft is given in para. 7.10.8. In order to decrease the risk of rejection, possibly shortly before the intended flight, a materials list should be prescreened and material application reviewed by a materials expert before the test to eliminate known offenders to material requirements. Also the off-gassing of materials can sometimes be reduced considerably by a bake-out at elevated temperature or application of barrier-layers with suitable coatings.

The fire-risk of an experiment may be evaluated by a fire-hazard analysis, which may require only a very limited amount of flammability testing.

The approach lined out here shall not be applied for hardware to be mounted in the Orbiter Aft Flight Deck, see para. 7.10.2a).

The experimenter shall propose a Material Control Program which is tailored to the specific equipment being used and which meets the basic requirements defined in this section and serves the purpose of avoiding hazards and detrimental effects on other equipment. The proposed material control program will be subject to approval by ESA or NASA.

ESA and NASA are prepared to provide assistance for analysis and testing and will provide, upon request, examples of materials control programes which were used and approved before.

7.10.4 Test and Acceptance Criteria for Off-Gassing from Materials (Toxic and Odourous Products)

Materials shall meet the test and acceptance criteria of para. 407 and para. 406 of NASA-specification NHB 8060.1 A.

7.10.5 Test and Acceptance Criteria for Flammability Characteristics of Materials

Materials shall be non-flammable or selfextinguishing if tested according to para. 401 of NHB 8060.1 A. For specific material applications (e.g. wire insulation) evaluation shall be performed according to other relevant test methods specified in NHB 8060.1 A.

7.10.6 Test and Acceptability Criteria for Outgassing of Materials under Vacuum

Materials are acceptable if their outgassing rate is TWL $\leq$ 1 % (total weight loss) and VCM $\leq$ 0.1 % (volatile condensible materials) if tested according to ESA-specification QRM-02 or NASA-specification JSC SPR-0022.

7.10.7 Corrosion and Material Compatibility

Materials used in experiments shall be compatible with materials of other equipment with which they come into contact and shall not form corrosion products which could affect the correct function or future use of other equipment.

7.10.8 Outline for an Off-Gassing Test on "Black-Box" Level

For the performance of this test a completed (and cleaned) experiment or parts of it shall be enclosed in a suitable test chamber at ambient pressure and operated at its maximum expected usage temperature. After 24, 48 and 72 hours gas probes shall be taken from the chamber and analyzed for possibly unacceptable toxic or odourous trace contaminants. Details of the test procedure and acceptance criteria are TBD. (Further preliminary information may be found in NHB 8060.1 A, para. 412, "Guidelines for Total Spacecraft Offgassing Test".)
Assistance for this test and analysis is foreseen by ESA and NASA.

7.10.9 Forbidden and Restricted Materials

The following materials shall not be used and compliance to this requirement shall be certified by the experimenter

- Mercury
- Cadmium and Cadmium plating
- Zinc
- Polyvenyl chloride (PVC, e.g. wire insulation, wrapping)
- Shatterable or flaking materials except if suitable protection is provided
- Known Carcinogens

The use of the following materials shall be restricted as far as possible. If their application cannot be avoided, they may be used only, if suitable protection is provided and if formally approved for each individual application by ESA or NASA.

- Radioactive materials
- Beryllium and beryllium alloys

The use of magnetic materials shall be minimized as far as possible. If their use cannot be avoided then the type quantitiy and locating of such materials shall be clearly identified and formally approved by ESA or NASA for each individual application (no formal waiver requests are required for application which are normal in electronic circuits).

7.10.10 Waivers and Deviations

If full compliance with the requirements specified above is impossible or highly impractical (e.g. because of specific application or small amount of material with marginal outgassing rate) waiver or deviation requests can be approved by ESA or NASA if it can be shown that safety of personnel or correct operation of other equipment will not be affected.

Applicable Documents

(Referenced in Section 7.10 for the definition of test and acceptance criteria for off-gassing, flammability and outgassing).

- ESA-specification PSS 09/QRM-02 T
  "A screening test method employing thermal vacuum for the selection of materials to be used in space"
- NASA-specification JSC-SP-R-0022
  "Vacuum stability requirements of polymeric materials for spacecraft application."
- NASA-specification NHB 8060.1 A
  "Flammability, Odor and Offgassing Requirements and Test Procedures for Materials in Environment that support combustion."

8.3.3 Rapid Evacuation

Experiment requiring the presence of personnel in the Payload Bay while on the ground shall not preclude rapid evacuation from the Payload Bay in the event of an emergency.

8.3.4 Hazardous Material Storage

Toxic, corrosive, and/or flammable materials shall be stored and used such that failure of the primary container will not release the material into the cabin atmosphere.

Provision shall be made for the safe collection and storage of used or spent materials, considering also their possible chemical or physical interaction.

8.3.5 Fluid Release

Hazardous fluids shall not be released into the Payload Bay. Hazardous fluid containment shall be designed to remain intact under crash loads with assurance provided that tank integrity will not be violated by other equipment due to impact as a result of crash loads. Release of inert gases into the Payload Bay may be permitted under some conditons.

8.3.6 Hazardous Material Isolation

Toxic materials and other materials determined by analysis and/or test to be hazardous must be isolated from the crew and cabin system, and suitable measures for neutralization provided in case of hazard.

8.3.7 Mercury

Experiment materials and equipment shall be free of mercury or mercury contamination.

8.3.8 Material Incompatibility

Where hazards can occur due to the presence or contact of mutually-incompatible materials, components at electrical differences or of chemically-incompatible substances shall be separated to the maximum practical extent.

8.3.9 Pressure Vessels

Pressure vessels* shall be in accordance with NASA Aerospace Pressure Vessel Safety Standard NSS HP 1 740.1 or in accordance with ASME Boiler and Pressure Vessel Code, Section III, Division 1 and 2. If pressure vessels are used which are not in accordance with NSS HP 1 740.1, then these pressure vessels must be tested to demonstrate fluid compatibility of the vessel with the contained fluid per NSS HP 1 740.1.

8.3.10 Pressure Vessel Location

Pressure vessels shall normally be installed exterior to the Spacelab cabin, and suitable regulation, pressure relief, and flow restriction provided so that flow into the cabin is limited to the capability of the Spacelab vent system. Small pressure vessels may be permitted inside the cabin provided they do not have a credible explosive failure mode and their failure will not expose the crew or vehicle to hazard.

8.3.11 Cryogenic Storage

Cryogenic materials must be stored external to the Spacelab Module shell in containers with adequate safety margins and venting provisions for flight and ground operations.

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*For the purpose of system safety analysis and safety criteria, the following definition of a pressure vessel is provided:

A pressure vessel is a vessel containing a compressible fluid with a stored energy greater than 19,310 J (14,240 ft lb), equivalent to 4.536 g (0.01 lb) TNT and having a credible explosive failure mode, that is, failure based on explosive fracture of the vessel and not merely on localized yielding or leakage.

Stored energy is the energy relative to the local environment based on adiabatic expansion of a perfect gas and may be calculated according to the formula:

$$W = \frac{P_1 V_1}{k-1}\left(1 - \left(\frac{P_2}{P_1}\right)^{\frac{k-1}{k}}\right)$$

Where W = Energy (J)

P_1 = Vessel internal pressure (N/m^2)

P_2 = Ambient external pressure (N/m^2)

V_1 = Gas volume or ullage in the vessel m^3

k = Specific heat ratio Cp/Cv for the gas under pressure

8.3.12 High Temperature Processes

High temperature processes must be carried out in suitable process chambers. Emergency switch-off in the case of cooling interruption or other control failure must be provided. Exterior surfaces must be kept below 45° C or suitable guards provided to prevent contact.

8.3.13 Vacuum and Process Chamber Venting

The user shall provide vent lines from experiment vacuum chambers and process chambers to the Spacelab experiment vent. Vacuum chambers and process chambers shall not vent into the Module.

8.3.14 Rotating Machinery

Rotating machinery must be protected by suitable guards. Where machinery is highly stressed, containment for possible failure must be provided.

8.3.15 Corners and Protrusions

Exposed sharp corners, edges and protrusions shall be avoided (see 7.2.5.1).

8.3.16 Material Shattering

Material which can shatter shall not be used in the module unless positive protection is provided to prevent fragments from entering the cabin environment. Photographic and optical equipment which cannot comply with this standard must be protected by suitable covers when not in use.

8.3.17 Stored Mechanical Energy

Mechanical devices such as springs, springloaded levers and torsion bars which are capable of storing energy should be avoided in experiment design. Where stored mechanical energy devices are absolutely necessary, safety features such as locks, protective devices and warning placards shall be provided.

8.3.18 Equipment Movement

Means for the control of movement of equipment which is not easily hand manipulated shall be provided for ground and orbital operations where applicable. Adequate handles, hoisting and ground support equipment attachment hardpoints shall be included in the design.

8.3.19 High Voltage

High voltage systems shall be suitable insulated, isolated and provided with circuit breakers. Provisions for automatic cut-off of high voltage is required when access to high voltage equipment for adjustment, maintenance or repair is needed.

8.3.20 Experiment Grounding

Experiment grounding shall be such as to preclude electrical discharge hazards and shocks.

8.3.21 Accidental Swich Actuation

All switches shall be recessed or otherwise protected against accidental actuation.

8.3.22 Emergency Switch Off

A rapid means of switching off power under emergency conditions shall be provided.

8.3.23 Lightning Strikes

Safety critical experiment equipment shall be designed, or protection provided to preclude hazards to the ground and flight crews in case of lightning strikes.

8.3.24 Pyrotechnics

Explosive devices capable of producing fragments or significant environment overpressure shall not be used.

All experiment pyrotechnic devices shall meet the requirements of JSC document number 08060 "Space Shuttle System Pyrotechnic Specification" or MIL-STD-1512 "Electro Explosive Subsystems, Electrically Initiated, Design Requirements and Test Methods".

8.3.25 Radiation Sources

Experiments that contain radioactive materials or contain equipment that generated ionizing radiation shall be identified and approval obtained for their use. The initial description shall state source type, strength quantity, containment/shielding and chemical/physical form. Review will be implemented through the ESA or NASA center responsible for development review and will be implemented by the Safety Office of that organization. Major radioactive sources require approval by the Interagency Aerospace Nuclear Safety Review Panel through the NASA coordinator for the Panel. Reference to be used for isotope SNAP devices is NASA/AEC Interangency Agreement 1052.72 A

8.3.26 Biological Specimen

The presence of biological specimens such as primates, mice, bacteria, etc., may constitute hazard sources to flight and ground personnel. These hazard sources must receive careful consideration from the experiment developer who has the detailed knowledge regarding the special risks that his specimen(s) imposes and how these hazards are most appropriately controlled.

Hazardous and/or toxic micro-biological materials must be isolated from the crew and cabin systems. The extent and method of isolating hazardous micro-biological materials will be considered on an individual case basis and appropriate procedure developed.

8.3.27 Microwave Experiment

Experiments which include microwave (200 MHz to 25.4 GHz) sources shall be designed to preclude crew exposure to greater than 10 mW/cm^2.

8.3.28 Free Flyer Launch

Free-flying experiments or experiment equipment shall have command and control circuitry associated with their launch/propulsion systems which are designed to preclude inadvertent launch or firing in case of hardware failure.

8.3.29 Free Flyer Hazard Sources

Free-flying experiments which contain hazard sources (e.g. explosive devices) shall be designed such that their function is positively inhibited until at a safe distance from the Orbiter.

8.3.30 Free Flyer Retrieve

Retrievable free-flying experiments shall include provisions to permit pre-retrieval safing which can be verified prior to the performance of retrieval operations.

APPENDIX A - DEFINITIONS

The Spacelab project has created a new set of terms that are descriptive but may have various interpretations. Some of the more basic terms used in this document are provided here as a means of developing common usage in working with this handbook.

Airlock - An intermediate volume between vacuum and a pressurized volume which can be pressurized or depressurized on command.

Cargo - Everything contained in the Shuttle cargo bay plus other equipment located elsewhere in the orbiter which is user unique and not carried in the standard baseline orbiter weight budget.

Caution - A condition where a hazard to crew safety could develop if no remedial action is taken.

Checkout - A sequence of activities and processes to examine the performance of a unit, subsystem, or system under various operating conditions.

Common Payload Support Equipment (CPSE) - a special set of Spacelab provided mission dependent equipment .

Component - An article composed of a group of assembled parts which is a self-contained element of a complete operating unit and performs a function necessary to the operation of the assembly. Meters, valves, actuators, etc., are examples of components.

Core Segment - a pressurized section of the Spacelab Module which houses subsystem equipment and experiments.

Dedicated Spacelab - An experiment module and/or pallet devoted to a single discipline which may fly more than once a year for several years and which may be assigned to a payload development center.

Docking Module - A removable module that can be installed in the forward end of the Orbiter cargo bay to provide shirt-sleeve transfer to another Orbiter or space vehicle equipped with a compatible docking device.

Emergency - A condition where an immediate hazard exists threatening crew safety.

Experiment Installation - The physical installation of experiments on Spacelab experiment racks or rack sets, sections, or pallet sections.

Experiment Integration- Those Spacelab project activities that are performed to assure physical and functional compatibility of experiments with the Spacelab, with other experiments, with GSE, with the test, operational and storage environments, and with ground and flight personnel.

Experiment Segment - A pressurized section of the Spacelab Module which houses experiments and sensors

Extra Vehicular Activity (EVA) - Crewman activities conducted in free space.

Fail-Safe - The ability to sustain a failure and retain the capability of terminating a flight without injury to personnel or unacceptable damage to vital Spacelab or Orbiter systems.

Flight - That portion of a mission encompassing the period from launch to landing or launch to termination of the active life of a spacecraft. The term Shuttle "flight" means a single Shuttle round trip - its launch, orbital activity and return. One flight might deliver more than one payload. More than one flight might be required to accomplish one mission.

Flight Unit - A unit that comprises all system constituents necessary to assemble any flight configuration and has met all qualification and acceptance test requirements.

Ground Operations - Operations concerned with receiving flight ready experiments and processing them to a launch ready condition; and after return from space, preparing them for reuse of disposition.

Ground Support Equipment - Includes all specific Spacelab equipment and software required for ground handling, testing, transportation, reconfiguration, integration, refurbishment, checkout, prelaunch, and post-landing operations. Simulators needed for verification and checkout of interfaces are included.

Hatch - Door for ingress and egress.

Integration - A combination of activities and processes required to assemble components, subsystems, and system elements into a desired configuration and to verify compatibility among the constituents of the assembly.

Interface - The common physical and operational boundary between two constituents of a system. The major Spacelab interfaces are:

Spacelab/Orbiter interface
Spacelab/payload interface
Spacelab/ground support interface

Spacelab subsystem interfaces

Spacelab segment interfaces

Intravehicular Activities (IVA) – Those activities accomplished by the crewmen within the pressurized module.

Launch Site – Either Kennedy Space Center (KSC) or the Vandenberg Air-Force Base (VAFB). Current Spacelab operational processing studies have been based upon utilization of KSC as the prime launch site with VAFB being considered for later activation and use.

Line Replaceable Unit (LRU) – The assembly level at which replacement of Spacelab units takes place during maintenance operations.

Maintenance – The actions taken to retain an item in a specified condition by systematic inspection and servicing, or those taken to restore an item to such a condition. Maintenance activities include fault detection, repair, item replacement, and verification.

Mission – A Shuttle/Spacelab flight from launch to landing.

Mission Specialist – The crew member on a Shuttle flight who is responsible for the coordination of Orbiter operations affecting payload operations. Additional tasks are handling Remote Manipulator System and assisting payload operations.

Mission Success – A mission is considered to be successful if Spacelab, its subsystems, and the experiment support equipment provided to the users, but not necessarily the experiments themselves have functioned properly.

Mockup – A full-scale mockup, dimensionally correct, with all equipment simulated in actual size, use for internal arrangement studies.

Module – A pressurized manned laboratory suitable for conducting science, applications, and technology activities on Space Shuttle flights.

Orbiter – The orbital flight vehicle of the Shuttle System.

Orbiter Cargo Bay – The volume provided within the Orbiter for accommodation of Spacelab or other Shuttle payloads.

Orbital Maneuvering System Kit (OMS Kit) – An auxiliary tank and pressurization system that can be installed in the Orbiter payload bay in incremental kits to increase the on-orbit maneuvering capability of the Shuttle Orbiter.

Orbiter Manipulator System - A manipulator system providing one manipulator arm stowed inside the payload bay but outside the Spacelab dynamic envelope. A second arm can be provided, if required, but will be weight-chargeable to the Spacelab payload.

Orbiter Turnaround - The time between landing and launch of the same Orbiter.

Pallet - An external, unpressurized platform for mounting telescopes, antennas, and other instruments and equipment requiring direct space exposure for conducting research and applications activities on Space Shuttle missions. The pallet experiments will be operated automatically or remotely from the Spacelab module or the Orbiter cabin, or directly from the ground. The pallet is composed of segments.

Pallet Element Mode - A flight configuration in which one or more pallet segments, carrying self-contained experiments, may be mounted at any suitable location within the Orbiter cargo bay and which require no other Orbiter support. Pallet elements may be flown taking advantage of available volume and weight of any Shuttle mission.

Pallet-Only Mode - A mission mode utilizing only the pallet, with subsystem support from the Orbiter and/or from pallet-mounted subsystem equipment. Pallet-only missions shall have the same Orbiter resource interfaces as the module/pallet missions.

Payload Specialist - The crew member on a Shuttle flight who is responsible for the management of payload operations and for the detailed operations of particular instruments or experiments.

Racks - Removable/reusable assemblies that provide structural mounting and connections to supporting subsystems (power, thermal control, data management, etc.) and experiment equipment which is housed in the pressurized module.

Residual Hazard - Hazard for which safety or warning devices and/or special means or procedures to counteract the hazards have not been developed or provided for.

Shirt-Sleeve Environment - An atmospheric environment habitable for men without protective pressure suits.

Spacelab - A laboratory designed for space operations composed of module and pallet suitable for accommodating instrumentation for conducting research and applications activities on Shuttle sortie missions. On a given mission, the Spacelab configuration can be comprised of a module only, a pallet only, or module and pallet combinations.

Spacelab Payload - All Spacelab associated experiments, experiment support equipment, and experiment-required consumables carried by the Spacelab or elsewhere in the Orbiter for science or application missions.

Spacelab Refurbishment - A ground operation including maintenance and/or reconstitution of the Spacelab, all common payload support equipment, and all subsystems.

Spacelab Subsystem - A Spacelab subsystem is an integral part of the overall system dedicated to a particular support function. Descriptions of the six Spacelab subsystem follows:

1. Structure - The primary module and pallet structures including bulkheads and mounting provisions, the secondary structure providing mechanical support for subsystem and payload equipment, and all necessary structural support hardware.

2. Electrical Power and Distribution Subsystem (EPDS) - All equipment used to convert, regulate, and distribute primary power and secondary power, including secondary power sources.

3. Command and Data Management Subsystem (CDMS) - All equipment and software required to perform all data handling tasks such as status and performance monitoring, control of subsystems and payload equipment, onboard checkout, data acquisition, data control and storage, and command and communications handling.

4. Environmental Control Subsystem (ECS) - All equipment and consumables required for thermal control of subsystem and payload equipment, and for thermal and atmospheric control of the internal environment of the module.

5. Instrument Pointing Subsystem (IPS) - All equipment and software required to provide precision pointing and stability for experiment equipment.

6. Common Payload Support Equipment (CPSE) - A special set of mission dependent equipment.

Software - All nonhardware items necessary to operate a computerized system, such as programs, instructions, subroutines, etc..

Spacecraft Tracking and Data Network (STDN) - Existing NASA standard network.

Testing - A sequence of activities and processes used to determine under real or simulated conditions the capabilities, limitations, reactions, effectiveness, reliability, or suitability of materials, parts, components, subsystems and systems.

Tracking and Data Relay Satellite System (TDRSS) – A data link system consisting of two relay satellits and one ground station at White Sands, New Mexico, which will be available for Spacelab missions.

User – The organization or individuals having responsibility for experiment payloads installed in a Spacelab.

Warning – A condition where a hazard to crew safety will develop unless immediate remedial actions are taken.

APPENDIX B – ABBREVIATIONS AND ACRONYMS

AC	Alternating Current
ACCU	Adio Central Control Unit
ADC	Analog Digital Converter
AFB	Air Force Base
AFD	Orbiter Aft Flight Deck
APU	Auxiliary Power Unit
ARS	Atmospheric Revitalization Section
ASCS	Atmosphere Storage and Control Section
ATCS	Active Thermal Control Section
ATP	Authority to Proceed
ATE	Automatic Test Equipment
B.I.T.E.	Built in Test Equipment
BTU	British Thermal Unit
CAM	Commercial Aviation and Military
CCS	Central Control Section
CCTV	Closed Circuit Television
C&D	Control and Display
CDMS	Command and Data Management Subsystem
CDR	Critical Design Review
CG	Center of Gravity
CHA	Channel
CHX	Cabin Heat Exchanger
CMD	Command
C/O	Check Out
CPSE	Common Payload Support Equipment
CPU	Central Processing Unit
CRT	Cathode Ray Tube
CSS	Core Segment Simulator
CTL	Control
C&W	Caution and Warning
CWEA	Caution and Warning Electronic Assembly
DC	Direct Current
DDP	Design Development Plan
DDU	Data Display Unit
DMA	Direct Memory Access
EAFB	Edwards Air Force Base
ECF	Electro–Conductive Film
ECLS	Environmental Control and Life Support
ECLSS	Environmental Control and Life Support Subsystem
ECOS	Experiment Computer Operating System
ECP	Engineering Change Proposal
ECS	Environmental Control Subsystem
EDL	Linkage Editor
EGSE	Electrical Ground Support Equipment
EIS	Electrical Integration System
EM	Engineering Model
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EMS	Electromagnetic Susceptibility
EOM	End of Message
EPDB	Electrical Power Distribution Box
EPDS	Electrical Power Distribution Subsystem
ESI	Electrical System Integration
ESP	Experiment Power Switching Panel
ET	External Tank
ETR	Eastern Test Range
EVA	Extra Vehicular Activity
EXP.	Experiment
FAR	Flight Acceptance Review
FLAP	Spacelab Flight Application Software
FM	Frequency Modulation
FMECA	Failure Mode Effects and Criticality Analysis
FOV	Field of View
FWD	Forward
FWW	Food, Water and Waste Management Subsystem
GMT	Greenwhich Mean Time
GPC	Orbiter General Purpose Computer
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
HOL	High Order Language
HPI	High Performance Insulation
HRDR	High Rate Digital Recorder
HRM	High Rate Multiplexer
H/W	Hardware
HX	Heat Exchanger
ICS	Interpretive Computer Simulator
ICD	Interface Control Document
IECS	Igloo Environmental Control Subsystem
IMU	Inertial Measurement Unit
INSTR.WD	Instruction Word
INV	Inverter
I/O	Input/Output
IOBPS	Input Output Box and Peripheral Simulator
IOU	Input Output Unit

IPS	Instrument Pointing Subsystem
IS	Interconnecting Station
IVA	Intra–Vehicular Activity
JPP	Joint Program Plan
JSC	Johnson Space Center
JSLWG	Joint Spacelab Working Group
JURG	Joint User Requirements Group
KB	Keyboard
KSC	Kennedy Space Center
LED	Light Emitting Diode
LRU	Line Replaceable Unit
LOS	Line of Sight
LV	Local Vertical
MAS	Macro Assembler
MD	Man Day
MDM	Orbiter Multiplexer/Demultiplexer
MET	Mission Elapsed Time
MGSE	Mechanical Ground Support Equipment
MMU	Mass Memory Unit
MSFC	Marshall Space Flight Center
MTU	Master Time Unit
NASA	National Aeronautics and Space Administration
N/C	Numerical Control
NCD	Number Column Density
O&C	Operation and Checkout
OIA	Orbiter Interface Adapter
OMCF	Orbiter Maintenance Checkout Facility
OMS	Orbiter Maneuvering Subsystem
OP Amp	Operational Amplifier
OPF	Orbiter Processing Facility
OPS	Operations
ORA	Operations Requirements Analysis
OSE	Orbiter Support Equipment
PCM	Pulse Code Modulation
PCMMU	PCM Master Unit
PCR	Payload Changeout Room
PLSS	Portable Live Support Subsystem
POCC	Payload Operations Control Center
POP	Perpendicular to Orbiter Plane
PPR	Payload Preparation Room
PSA	Pressure Switch Assembly
PSA	Pressure Switch Assembly
PSS	Payload Specialist Station
PTCS	Passiv Thermal Control Subsystem
P&M	Process and Materials
QDR	Qualification Design Review
RAU	Remote Acquisition Unit
RAAB	Remote Amplifier and Advisory Box
RCS	Reaction Control System
RF	Return Flux
RF	Radio Frequency
RH	Relative Humidity
RIG	Rate Integration Gyros
RMS	Root Mean Square
SCOS	Subsystem Computer Operating System
SGP	Single Ground Point
SL	Spacelab
S/L	Spacelab
SMCC	System Mission Control Center
SRA	Support Requirements Analysis
SRB	Solid Rocket Booster
S/S	Subsystem
STDN	Spaceflight Tracking and Data Network
SW	Single Wires
S/W	Software
TBD	To be determined
TCS	Thermal Control Subsystem
TDRS	Tracking Data Relay Satellite
TDRSS	Tracking Data and Relay Satellite System
TSP	Twisted Shielded Pairs
TWL	Total Weight Loss
UTC	User Time Clock
VAB	Vandenberg Air Base
VAFB	Vandenberg Air Force Base
VCM	Volatile Condensible Materials
VDC	Volts Direct Current
WTR	Western Test Range