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SPACELAB

PAYLOAD ACCOMMODATION HANDBOOK

MAIN VOLUME

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ISSUE No.: 1

DATE: JULY 1978

EUROPEAN SPACE AGENCY

SPACELAB

PAYLOAD ACCOMMODATION HANDBOOK

ISSUE No. : 1

DATE : 30 JUNE 1977

THIS IS A CONTROLLED DOCUMENT

For (T.J. LEE)
Manager, NASA MSFC
Spacelab Program Office

B. Pfeiffer.
(B. PFEIFFER)
ESA Spacelab
Project Manager

FOREWORD

With the advent of the Space Shuttle System, operations in earth-orbital space will become less complex and less costly. This system will make launching of payloads into earth orbits 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 issue of the Spacelab Payload Accommodation Handbook under contract to ESA, are acknowledged.

Any inquiries concerning the contents of this document should be addressed to:

In USA :

T. J. Lee
Manager, Spacelab Program
NATIONAL AERONAUTICS AND
SPACE ADMINISTRATION
George C. Marshall Space Flight Center
Huntsville, Alabama 35812
U. S. A.

In Europe :

D.E.Mullinger
Head of SPICE
(Spacelab Payload Integration and
Co-ordination in Europe)
c/o DFVLR
Linder Höhe
D 5000 Köln 90
West Germany

DCN 002

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Section 1 - INTRODUCTION

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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. Spacelab/experiment interfaces, Spacelab payload support systems and requirements that the experiments have to comply with are described to allow experiment design and development. The basic operational aspects are outlined as far as they have an impact on experiment design.

The general aspects of the Spacelab program and Spacelab utilization are covered in the "Spacelab User's Guide". The relationship of the "Spacelab Payload Accommodation Handbook" to Space Transportation System documentation is outlined in Section 9.2 . Data concerning the Space Shuttle System are briefly described in the Spacelab Payload Accommodation Handbook but more complete information is available in the documents "Space Shuttle Payload Accommodation" (JSC-07700, Vol. XIV) and "Shuttle Orbiter/Cargo Standard Interfaces" (ICD-2-19001). DCN 002

The Spacelab Payload Accommodation Handbook is jointly controlled by ESA / NASA. It reflects the Spacelab baseline as it is presently configured and contains the officially controlled data set concerning Spacelab capabilities and Spacelab / payload interfaces. It should be noted that, since the as-built capabilities are not generally available at this time, certain values may be specification values, analytical predictions, etc. DCN 002

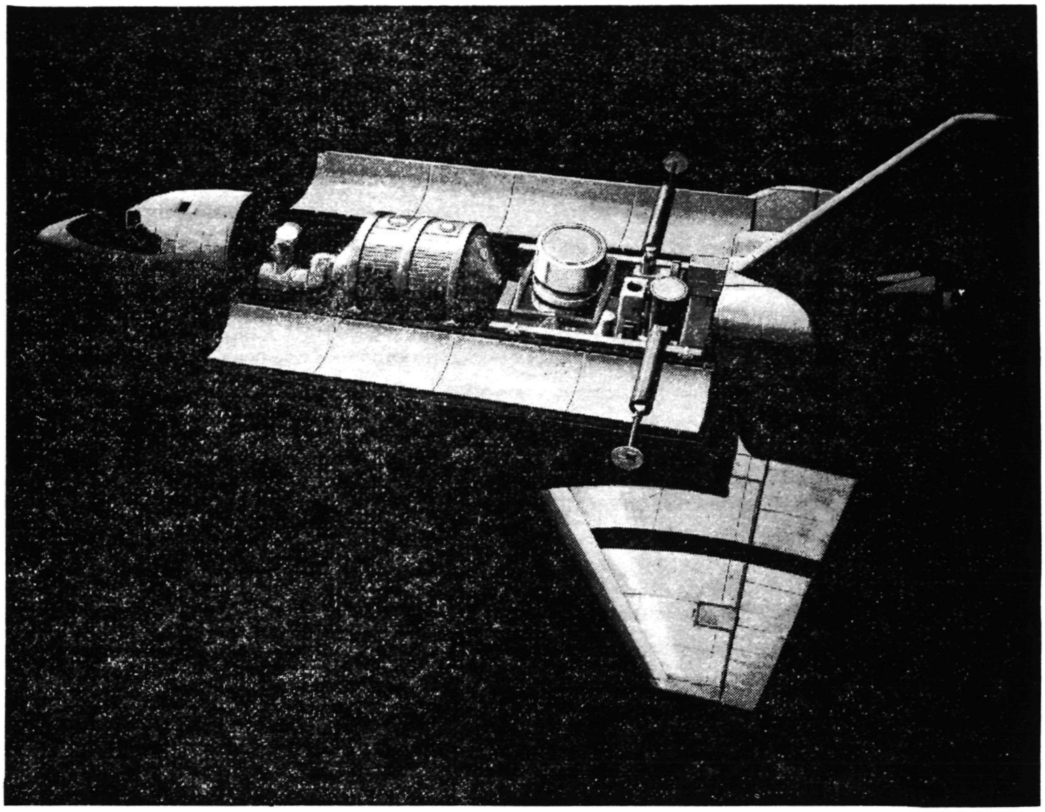

Figure 1 - 1: Conceptual View of Spacelab in Orbit

1.2 General Spacelab System Description

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

The modular design of the module (up to 2 segments) and the pallet (up to 5 segments) allows for a large variety of flight configurations.

These flight configurations can be grouped into three configuration types:

- 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, heat rejection 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, heat rejection 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 candidate concept of this operation.

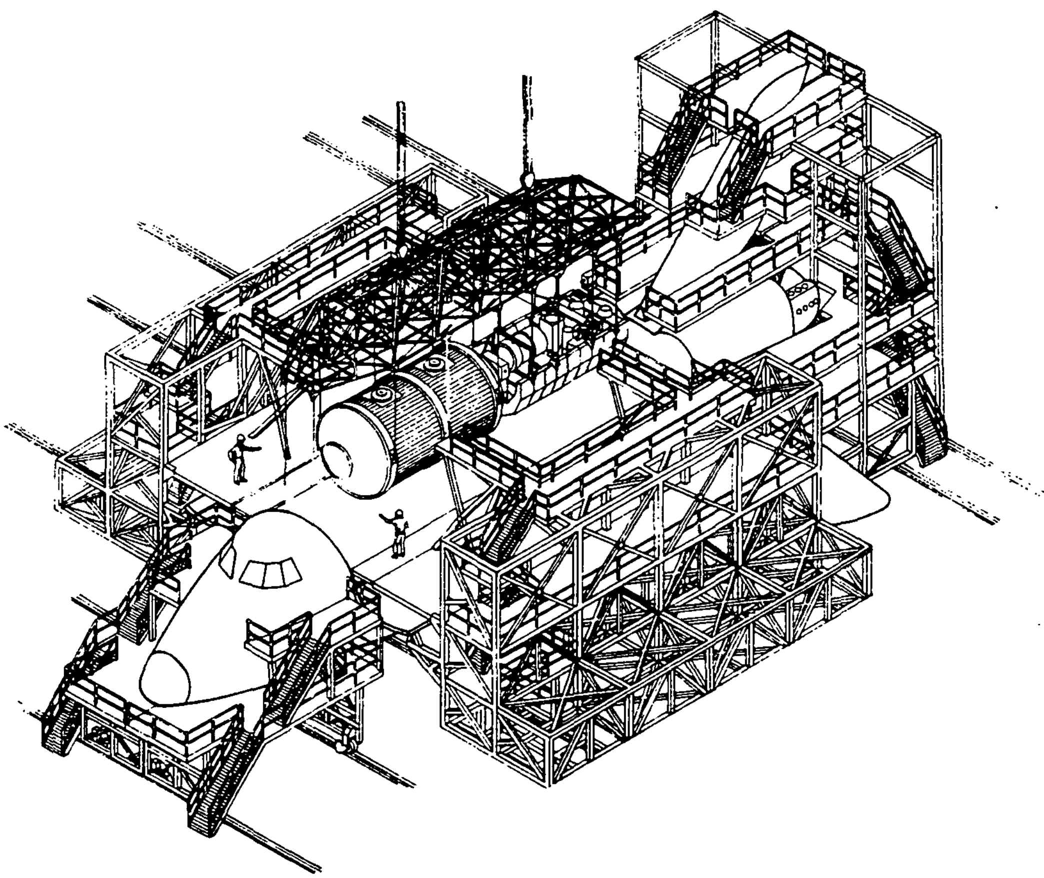

Figure 1-2 Possible Concept of Spacelab Removal From Orbiter

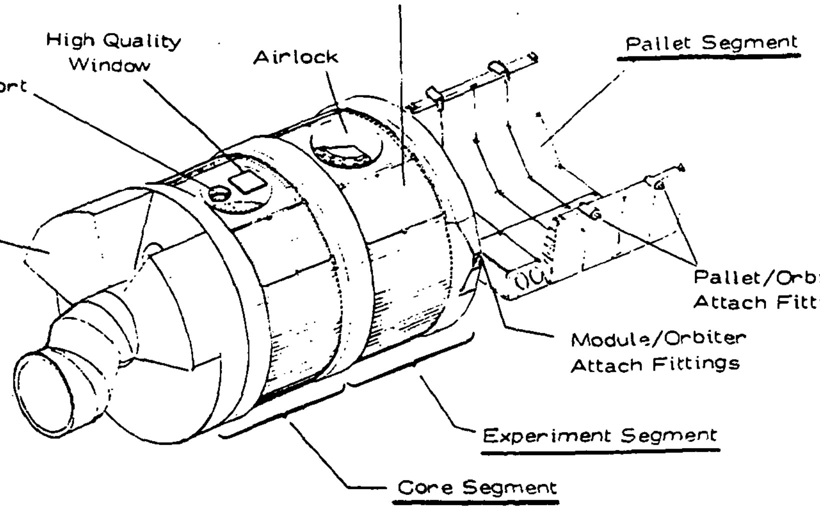

Figure 1-3: Spacelab External Configuration

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 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. 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 aerodynamic constraints during re-entry 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/EVA airlock combination is attached to the Orbiter forward bulkhead. Extra-vehicular activity (EVA) can be performed through the EVA airlock on the top of the tunnel adapter. The design of this unit is such that access to Spacelab from the Orbiter is not interrupted during EVA.

A viewport or a high quality window/viewport assembly can be mounted in the top opening of the core segment or experiment segment. The top opening in the experiment segment can also be used for mounting the airlock. If the airlock, the viewport or the high quality window/viewport assembly are not flown, the top openings are closed by coverplates. A second viewport is located in the aft end cone to give an unobstructed view of the pallet. The forward and aft end cones provide feedthrough panels for utility routing.

The U-shaped pallet segments are covered with aluminium 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 pallet segments may be flown on a single mission.

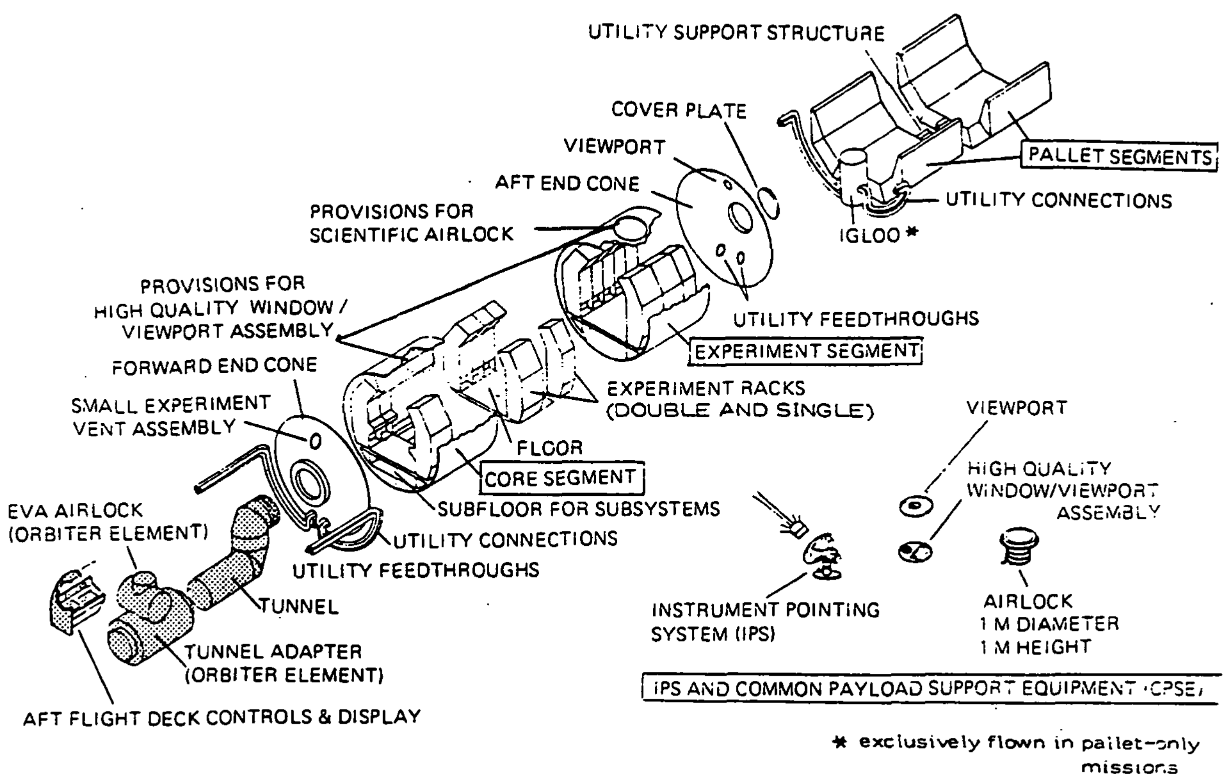

Figure 1 – 4: Major Spacelab Flight Elements

Figure 1 - 4 shows the major Spacelab flight elements. The EVA airlock and the tunnel adapter are part of the Shuttle program. The tunnel and the utility connections, Orbiter to module and Orbiter to igloo, are part of the US Spacelab program. All other elements are part of the European Spacelab program.

The interior design of the module 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 subsystem double rack assembly on each side. Other segments build up the floor for support of experiment racks for both the short and the long module. 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 racks are attached independently from each other 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. Signal, power and other utility lines to and from the Orbiter are routed from the forward end cone. In module plus pallet configurations, lines between module and pallet are routed via a utility support structure. For pallet configurations an "Igloo", a pressurized cylinder attached to a pallet, is provided for installation of certain subsystem hardware which is needed for pallet-only configurations. In pallet-only configurations, operations of subsystems and experiments will be performed from the Orbiter aft flight deck or from the ground. Signal and other utility lines to and from the Orbiter are routed from the igloo. Spaced pallet segments are connected via a utility support structure.

Spacelab equipment in the aft flight deck (AFD) of the Orbiter permits the control of Spacelab subsystems and experiments, and permits the display of data. This aft flight deck equipment is independent of the Spacelab configurations flown. In the aft flight deck there are also limited space and resources available for payload use.

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 subsystem equipment, which can be removed without affecting the basic operation of the Spacelab system is defined as "mission dependent" equipment.

The Spacelab flight hardware is divided into the following subsystems: Structure, environment control, electrical power and distribution, command and data management, common payload support equipment, and instrument pointing subsystem. DCN 002

The structure subsystem comprises essentially the mechanical parts of the module and pallet. Its prime function is to provide the required load carrying capability during ground and flight phases.

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 ECS includes a valve in the forward bulkhead by which experiment chambers etc. inside the module can be connected with the outside vacuum. This facility is referred to as the small experiment vent assembly.

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

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 (with dedicated data bus), one dedicated to subsystems (with a dedicated subsystem data bus), both with standard interface units (RAU's) and one back-up computer for either of the two dedicated computers. For high rate experiment data acquisition, a multiplexer is provided together with a recorder to bridge mission phases without data downlink. The CDMS subsystem is largely independent from the Orbiter. Communication with ground facilities, either via the Tracking and Data Relay Satellite System (TDRSS) or, as a back up, directly to a Spaceflight Tracking and Data Network (STDN) Station is provided through the Orbiter's communication system. System activation and monitoring (SAM) of Spacelab is performed through dedicated hardware and CDMS. DCN 001

Common payload support equipment (CPSE) consists of an airlock, a top cover plate with a viewport and a top cover plate with a high quality window and viewport.

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

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. DCN 002

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.

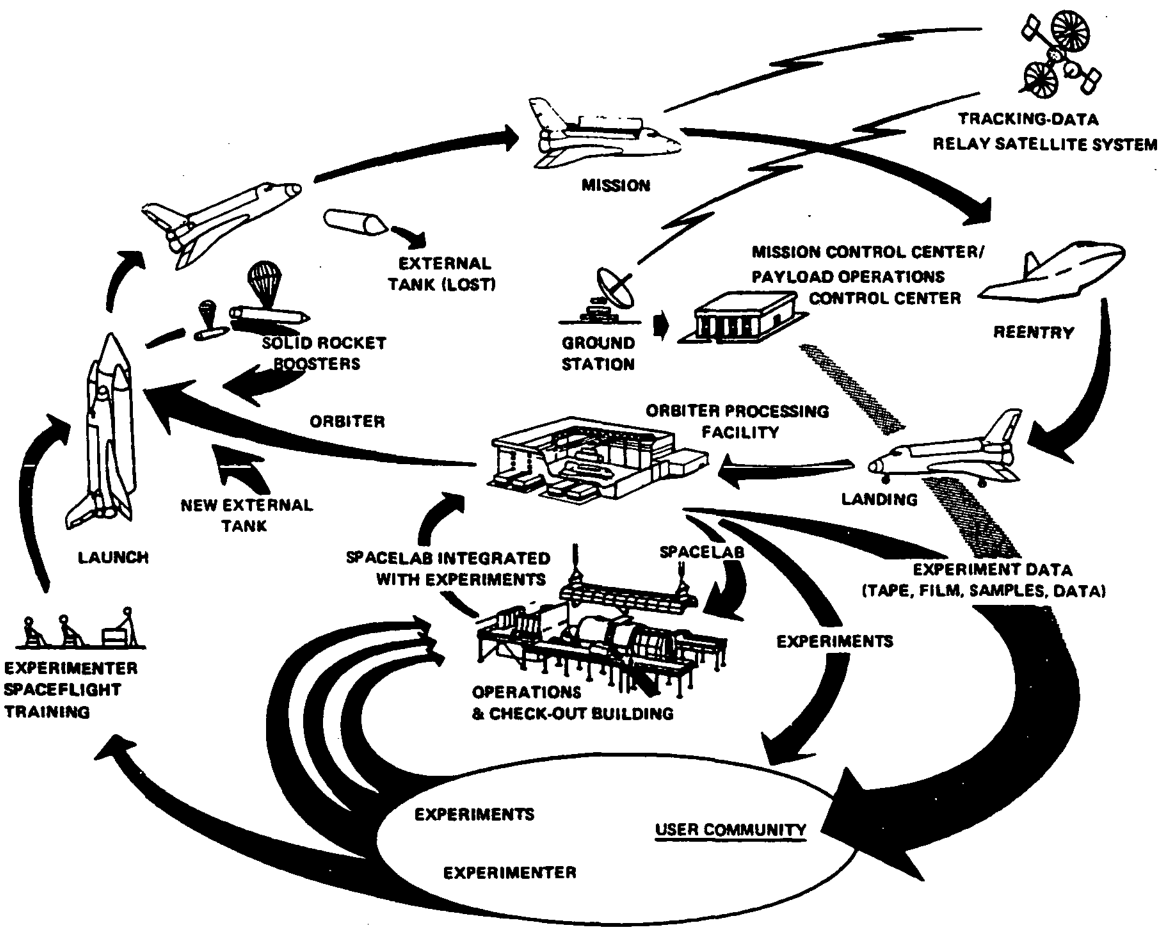

Figure 1 - 5 : Shuttle - Spacelab Operational Profile

Section 2 - ORBITER/SPACELAB PERFORMANCE AND CONSTRAINTS

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LIST OF TABLES

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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. It is for reference only and should not be considered as offical. The source of information on this subject is the Space Shuttle Interface Control Dokument "Shuttle Orbiter/Cargo Standard Interfaces", ICD-2-19001 and the NASA document "Space Shuttle System Payload Accommodation", JSC 07700, Vol. XIV, Revision E, June 17, 1977, including changes dated up to March 29, 78. This document will be referenced as "Vol. XIV, Rev. E".

DCN 009

2.1 Coordinate Systems

This section identifies those coordinate systems which are of prime importance for interfaces between the Orbiter and Spacelab.

2.1.1 Orbiter Structural Body Coordinate System (X_o, Y_o, Z_o)

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.

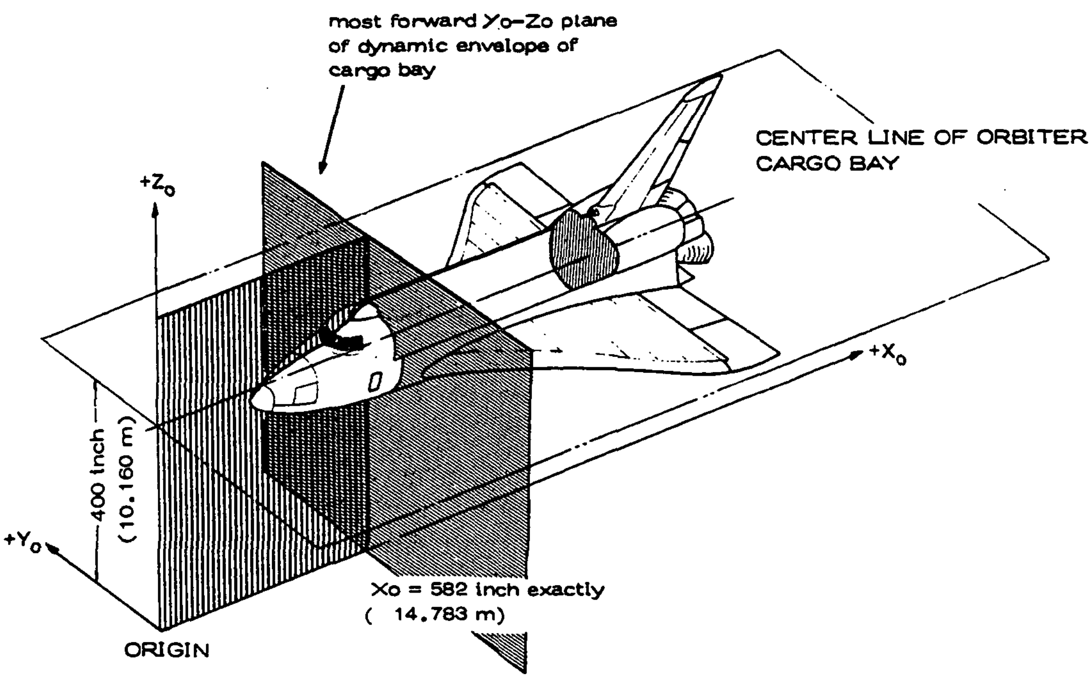

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Figure 2-1: Orbiter Structural Body Coordinate System
(Reference, Vol. 14, Rev. E, Fig. 2 - 5)

2.1.2 Structure Coordinate System (X, Y, Z) for Spacelab Mechanical Load Definition

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

Type: Orbiter fixed and related to location of center of gravity

Origin: Center of gravity

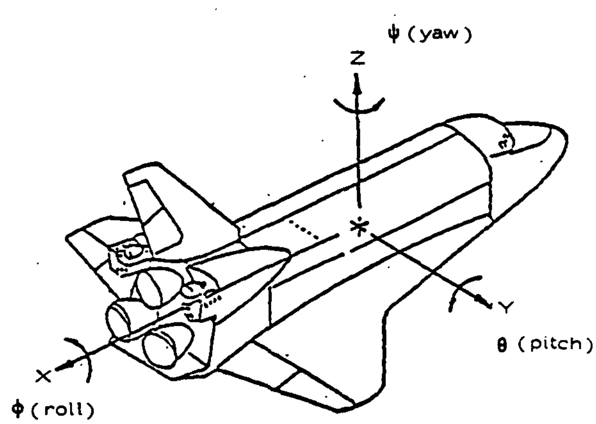
Orientation: X – axis is parallel to the Orbiter structural body X_0 axis; positive toward the tail.

Z – axis is parallel to the Orbiter plane of symmetry and is perpendicular to X, positive up with respect to the Orbiter fuselage. DCN 009

Y – axis completes the right-handed orthogonal system.

Characteristics: X, Y, Z: roll, pitch and yaw axes

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



DCN 011

Figure 2-2: Structure Coordinate System (for Spacelab Mechanical Load Definition) (Reference: ICD - 2 - 05101)

2.1.3 Spacelab System Coordinate System (X_L, Y_L, Z_L)

The Spacelab System Coordinate System derives from the Orbiter Structural Body Coordinate System by a transposition of the origin and conversion into the metric 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.

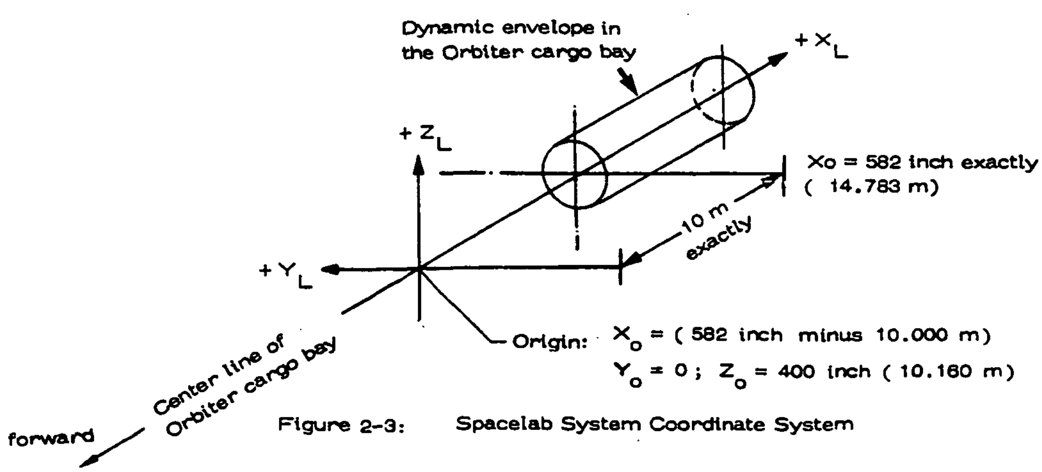


Figure 2-3: Spacelab System Coordinate System

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 6.

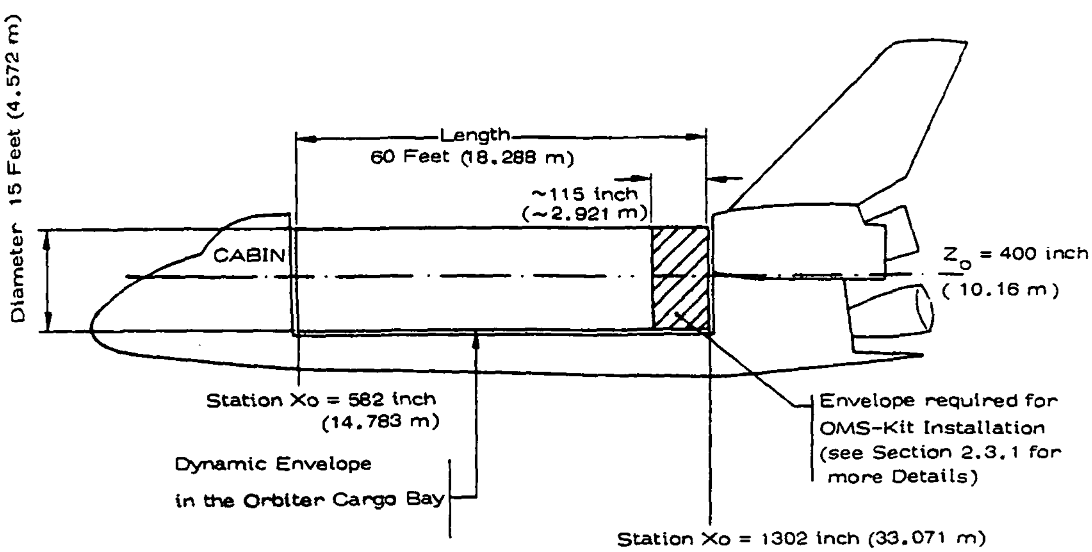


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Figure 2-4: Dynamic Envelope for Spacelab and its Payload in the Orbiter Cargo Bay (Reference: ICD - 2 19001)

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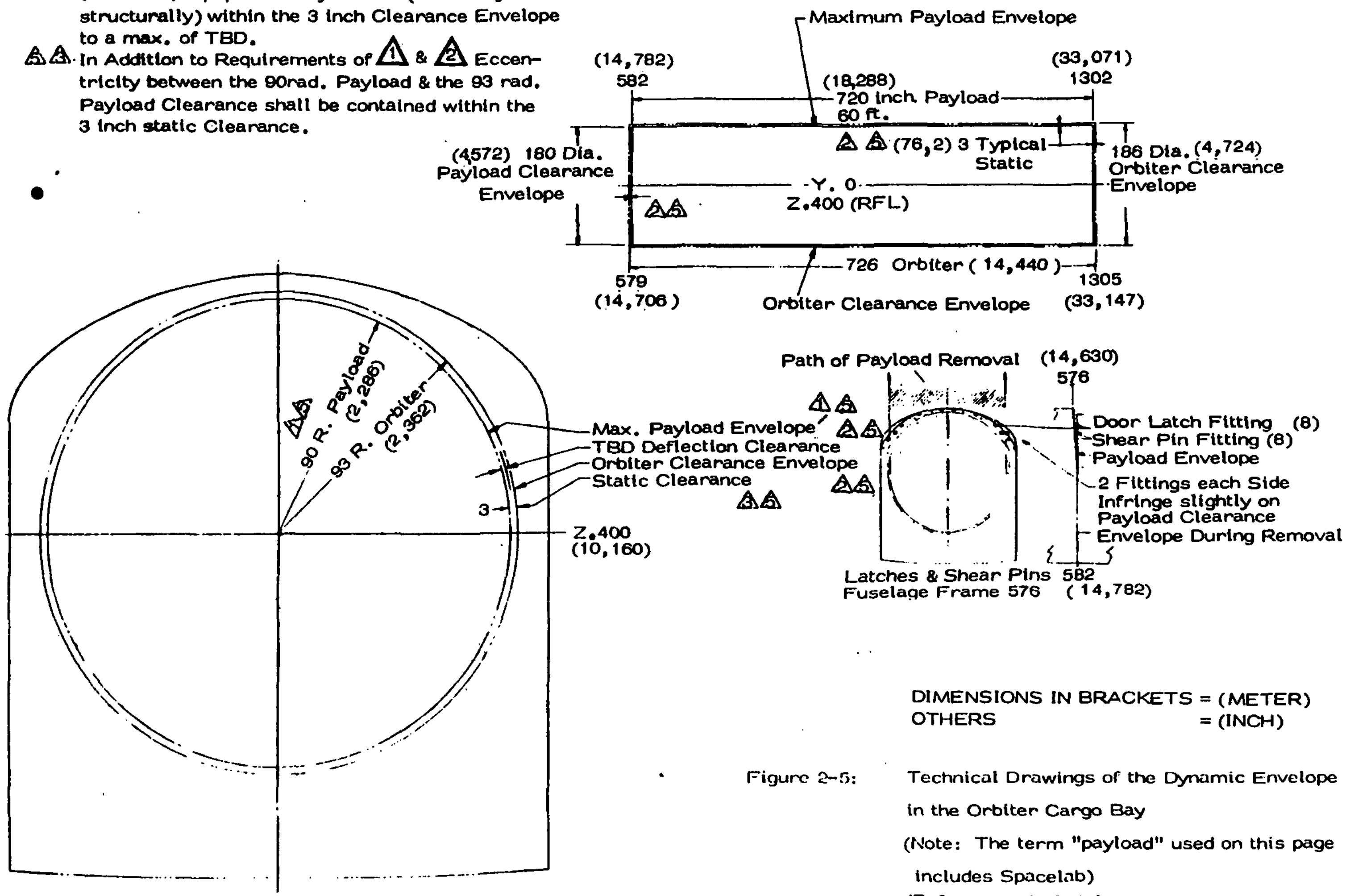
DCN 010

PAYLOAD ENVELOPE/CLEARANCE

1. Payload Thermal/Structure Deflections contained within max. Payload Envelope

2. 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.

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)
(Reference: Vol. XIV, Rev. E, p. C 14)

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 the 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 such as the remote manipulator and its supports, TV cameras, the door hinges etc.) for any point along the line Yo = 0, Zo = 427 (10845,8 mm) between Xo = 582 (14782.8 mm) and Xo = 1302 (33070.8 mm). The manipulator supports are not removed from the Orbiter, even if the remote manipulator is not flown. The location of the manipulator supports and the stored manipulator is given in Figure 2-24. 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.

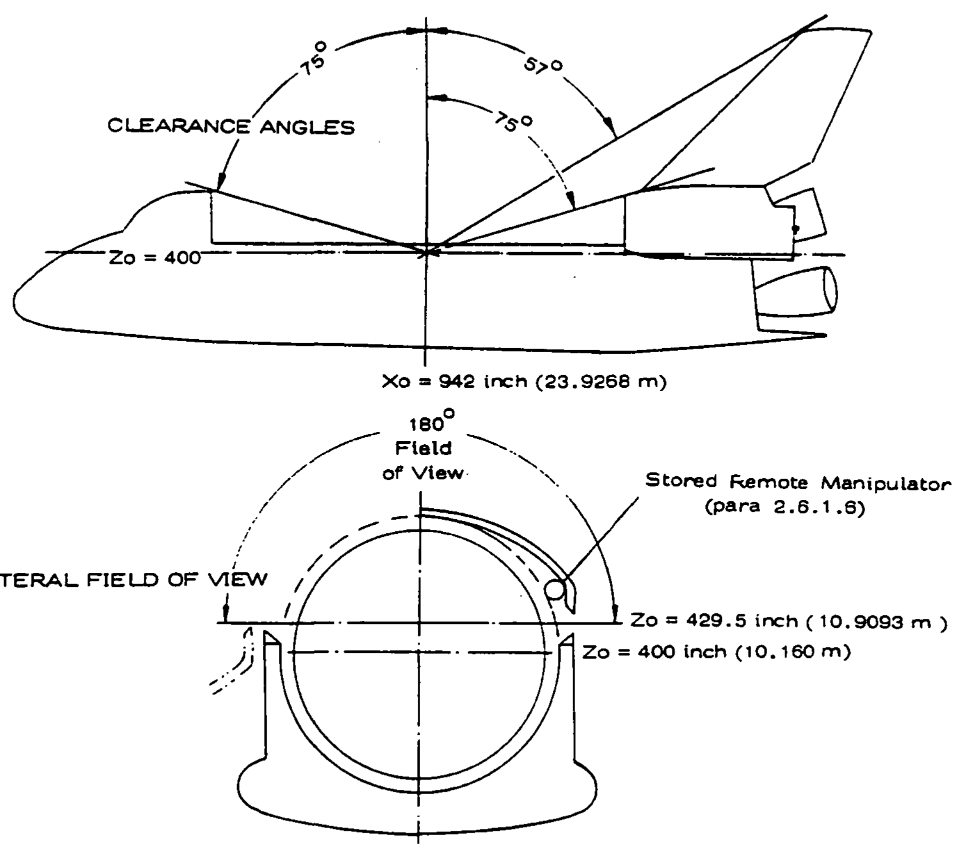
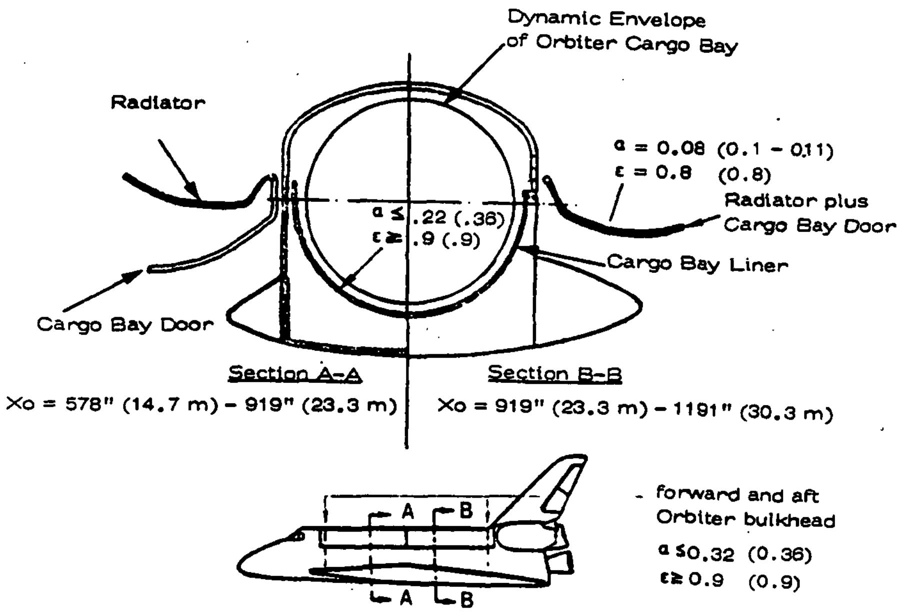


Figure 2-6: Field of View and Clearance Angles of Orbiter Cargo Bay (Reference: ICD - 2 - 05101, para 3.1.1.3.2)

2.2.3 Orbiter Cargo Bay Doors, Radiator and Thermo-Optical Properties of Surfaces

The doors of the Orbiter cargo bay serve primarily as a protection and heat shield during ascent and descent. They are opened on-orbit. Inside the doors there are thermal radiators which are exposed to space when the doors are open. The function of these radiators is to provide heat rejection for the Orbiter by cooling of heat transfer liquids of the Orbiter heat rejection system. The radiator consists of various radiator panels. When the doors are opened the forward radiator segments are set to a different deployment
DCN 009 angle than the rearward radiator segments. It is pointed out that there will be thermal interaction between large, deployed payload systems (e.g. large antennas) and the radiators. When such payload systems are defined, a thermal assessment will have to be made on an individual basis, taking into account the specific configuration and mission attitude requirement. Figure 2 - 7 gives a general outline of the Orbiter cargo bay doors and the absorptivity α of sunlight and emissivity ε of thermal radiation of major Orbiter elements to which Spacelab and its payload are exposed.



Note: Figures for α and ε without and with brackets are for new and degraded surfaces respectively

DCN 010 Figure 2 - 7: Orbiter Cargo Bay Doors and Thermal Surfaces (Reference, Vol. XIV, Rev. E, Fig. 4.12 and ICD - 2 - 05201)

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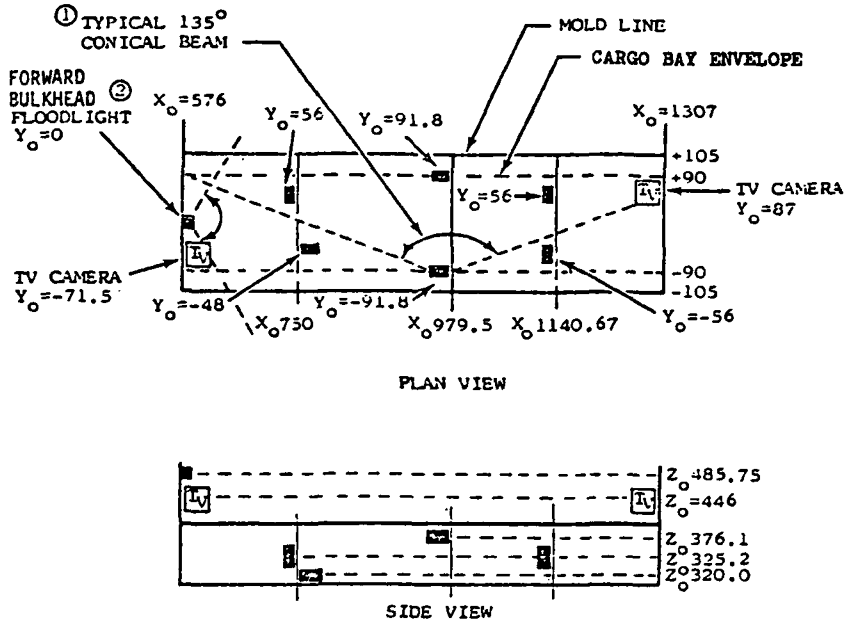

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NOTES:
1. SIX LIGHTS MOUNTED OUTSIDE CARGO BAY ENVELOPE. 135° CONICAL BEAM POINTED WITHIN APPROX 5° OF NORMAL TO CARGO BAY CENTERLINE
2. FORWARD BULKHEAD FLOODLIGHT -120° CONICAL BEAM POINTED IN +X_o DIRECTION
3. X_o DIMENSIONS IDENTIFY FRAMES OR BULKHEADS TO WHICH LIGHTS AND CAMERAS ARE ATTACHED
4. Y_o AND Z_o DIMENSIONS ARE APPROPRIATE DIMENSIONS AT CENTER OF LIGHT FIXTURE OR CAMERA
5. FIGURE NOT TO SCALE

Figure 2 - 7a: Cargo Bay Lighting and TV Camera Locations

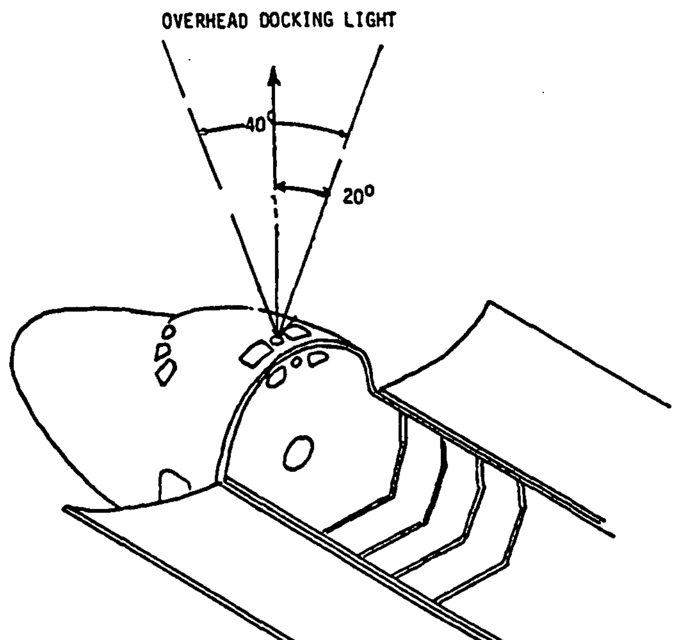

Figure 2 - 7b: Orbiter Overhead Docking Light

2.2.4 Orbiter Exterior Lighting

To facilitate payload handling and viewing of payloads, the Orbiter provides lamps inside the cargo bay, on the remote manipulator system (see 2.6.1.6) and above the aft flight deck between the overhead windows. Viewing is additionally facilitated by TV cameras in the cargo bay and or the remote manipulator system (RMS). Figures 2.7a and b show the location of the lights and TV cameras in the cargo bay and the location of the overhead docking light. Table 2 - 1a shows the lighting characteristics.

It should be realized that large items in the Orbiter cargo bay such as the Spacelab module or pallet segments will block the light from many of the outlined locations. Therefore little or no illumination for direct or television viewing of Spacelab payload in the payload areas will be available from these lights.

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Table 2 - 1a: Exterior Lighting Characteristics

STATION	LAMP TYPE	CHROMATICITY AND TEMPERATURE	CONE OF RADIATION	INTENSITY / BRIGHTNESS RATIO	FIXTURE	CONTROLS	LOCATION	NO.
PAYLOAD BAY	WIDE ANGLE FLOOD METAL HALIDE	WHITE 3400°K	120°	.5 FT. CD. CENT. LN. / 10:1	GROUND ADJUSTMENT	INDEP. FROM D AND C	3 EACH SIDE OF BAY	6
RMS LIGHT	NARROW ANGLE FLOOD INCANDESCENT	WHITE 2800°K	40°CONE	3 FT CD AT 30 FT.	PAN/TILT WITH CAMERA	INDEP. FROM D AND C	ABOVE CAMERA ON RMS ROLL JOINT	1
PAYLOAD BAY FORWARD BULKHEAD	WIDE ANGLE FLOOD METAL HALIDE	WHITE 3400°K	120°	5 FT. CD. AT 30 FT.	GROUND ADJUSTMENT	INDEP. FROM D AND C	576 BULKHEAD BETWEEN AFT WINDOWS	1
OVERHEAD DOCKING	NARROW ANGLE FLOOD INCANDESCENT	WHITE 2800°K	40°CONE	.002 FT CD AT 1000 FT.	GROUND ADJUSTMENT	INDEP. FROM D AND C	ON -Z AXIS BETWEEN OVERHEAD WINDOWS	1

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 29 484 kg (65 000 lbs) 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 lbs), but no mission should be planned with landing weights exceeding 14 515 kg (32 000 lbs). 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 specific constraints. The implications of these constraints 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. The mass properties include personnel and usable, unusable and residual fluids.

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}
89 108 (196 446)	2908 (1145)	1.3 (0.5)	980 (386)	1.17 (0.87)	9.1 (6.7)	9.5 (7.0)	~0 (~0)	0.4 (0.3)	~0 (~0)

(Reference: JSC - 08934 (Volume II) Rev. A - Shuttle Operational Data Book, Revision A, Sep 1975, Amendment 17, 15 Feb 1978)

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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 which is in this context Spacelab plus Spacelab payload

2.2.8.1 Orbiter Cargo Bay Design

The Orbiter cargo bay is designed to minimize contamination of payload and critical cargo bay surfaces. Orbiter elements which are not easily cleaned, e.g., internal ribbed structure, door actuators, etc., and elements which are sources of particulate, vapor, VCM (volatile condensible material), or othercontamination, are isolated from the payload and critical cargo bay surfaces. All nonmetallic materials exposed to the payload are selected for low outgassing characteristics. The Orbiter cargo bay is designed to protect critical payload and cargo bay surfaces from contamination by the external environment during any operational phase of the Space Shuttle System when the Orbiter cargo bay doors are closed.

Isolation of the payload and critical cargo bay surfaces from the lower sections of the unit fuselage is accomplished by a liner. This liner is designed of lightweight non-rigid material and will permit the flow of gases through filter areas in either direction during a scent depressurization and descent repressurization with a maximum allowable pressure buildup across the liner of 0.10 psi (689 N/M^2). DCN 009
The liner will prevent the transfer of particulates greater than 87 microns (equivalent to 35 microns GBR glass bead rating) from the lower mid-fuselage to the Orbiter cargo bay.

Critical surfaces such as Orbiter radiators, windows, optics, etc., within the Orbiter cargo bay and part of the Orbiter System must be protected in the same manner as payloads. That is, payloads must insure that their effluents and operations do not jeopardize the performance of these systems. Materials shall be selected for low outgassing characteristics.

2.2.8.2 Payload Loading and Checkout

Prior to payload loading the internal surfaces of the Orbiter cargo bay will be cleaned to a visibly clean level, as defined in JSC Specification SN-C-0005. This cleaning will be accomplished within a protective enclosure in order to isolate sources of contamination from critical regions. This enclosure will be continuously purged with nominally class 100, guaranteed class 5000 (HEPA filtered) air per FED-STD-209B and will contain less than 15 parts per million hydrocarbons, based on methane equivalent. The air within the enclosure shall be maintained at $70 \pm 5^{\circ}$ F ($21.1 \pm 2.8^{\circ}$C) and 50 % or less relative humidity. The payload loading operation will be accomplished so as to avoid contaminating the payload and Orbiter cargo bay by temperature, humidity, or particulates consistent with operational capabilities described in this section. More stringent particulate and relative humidity requirements may be implemented on particular missions pending technical justification of the requirements.

2.2.8.3 Contamination Control Subsequent To Payload Loading

Subsequent to payload loading, accumulation of visible particulate and film contamination on all surfaces within the Orbiter cargo bay will be prevented by controlled work discipline, cleanliness inspections, and effective cleaning as required. The Orbiter cargo bay will continue to be purged with air meeting the class, temperature and humidity requirements of Section 2.2.8.2 (see also Section 5.3.2.2).

2.2.8.4 Preparation For Close-Up of Orbiter Cargo Bay

Prior to final closure of the Orbiter cargo bay, inspection and cleaning as required will be conducted to verify that all accessible surfaces within the Orbiter cargo bay, including external surfaces of payloads, meet the visibly clean level capability stipulated in Section 2.2.8.2. When payload changeout in the vertical configuration is required, the purge gas class, temperature, and humidity capabilities of Section 2.2.8.2 will apply.

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2.2.8.5 Closed Orbiter Cargo Bay Operations

The Orbiter is designed for closed Cargo bay purging subsequent to cargo bay closure using dry nitrogen or air which has been HEPA filtered and contains 15 ppm or less hydrocarbons based on methane equivalent.

2.2.8.6 Launch Through Orbit Insertion

At launch and during ascent the Orbiter cargo bay 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 affect payloads) until earth orbit is established and the Orbiter cargo bay doors are opened. For several seconds after lift-off the cargo bay vents are closed, thus preventing ingestion of dust particles, and in particular, combustion products of the Orbiter cryogenic main engines and the solid propellent booster engines.

During ascent and until engine cutoff, the Auxiliary Power Unit (APU) and hydraulic subsystems are operating with the effluents of large quantities of steam from the water boilers and combustion products of hydrazine from the APU. 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 140.000 feet (42.7 km) until the Orbiter cargo bay doors are opened and the radiators put into operation.

Any purging, other than that provided by normal depressurization of the Orbiter cargo bay during this operational phase, is the responsibility of the Spacelab payload requiring this purging.

The level of cleanliness maintained at pre-flight on the payload and Orbiter cargo bay, will be retained through launch to orbital insertion including lift-off, SRB separation etc.

2.2.8.7 On-Orbit Phase

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

a) the RCS vernier firings which may be required by the payload for attitude stabilization and 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; DCN 009

c) the release of particulates and outgassed species. It is a design and operational goal to control in an instrument field of view particles of 5 microns in size to one event per orbit, to control induced water vapor column density to 10^{12} molecules/cm^2, or less, to control return flux to 10^{12} molecules/cm^2/sec., to control continous emissions or scattering to not exceed 20th magnitude/arc sec^2 in the UV range, and to control to 1 % the absorption of UV, visible, and IR radiation by condensibles on optical surfaces.

Reaction Control Subsystem (RCS)

Payloads may require either the primary or vernier thrusters for on-orbit operations (Section 2.4) 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. The main axes of the four aft thrusters lie along the -Zo or +Yo axes of the Orbiter. The two thrusters mounted in the forward RCS module also fire in the - Zo direction but are canted 37° outward toward the Yo axis. Although this orientation precludes direct impingement on Orbiter cargo bay surfaces, reflection of the thruster 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 presented in Table 2.2. These contributions to the contaminant environment are present only during firing. The contribution decays instantaneously after thruster cut-off. The combustion products are O, O_2, NO, CO_2, H, OH, H_2O, N_2, CO with roughly the following relative densities: 3500, 2800, 80, 30, 11, 6, 2, 1.3, 1.

If the contributions to the Orbiter environment due to vernier thruster firings are unacceptable for a given payload, the use of payload pointing and stabilization systems may be required or Orbiter flight attitudes must be selected which minimize thruster firings. The divergence from a given Orbiter attitude as a function of time with the RCS inhibited is mission dependent.

Flash Evaporator

During orbital operations, the flash evaporator (Section 2.6.1, 3c) will discharge approximately 311 lbs. (141 kg) of water overboard per day. The water is discharged in the vapor state through two nozzles, one on each side of the aft fuselage, located at Xo 1505.62 inch (38243 mm), Yo ± 127.12 inch (± 3229 mm), Zo 305.00 inch (7747 mm). The plume expands along the ± Yo axes of the Orbiter. This results in some wing reflection as with the reaction control thrusters, however, the contribution for this location is within design requirements. The contribution of the flash evaporator to contamination is shown in Table 2.2. Operational flexibilities permit manually deactivating this system for short periods of time (about 11.5 hours) with some possible concurrent heat rejection capability loss. The water produced during this period must be stored and subsequently dumped.

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Other Sources

During orbital operations release of particulates, outgassed species, and/or leaks may also contribute to the Orbiter external environment. General sloughing of particulates from Orbiter surfaces is minimized by 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. The contribution of air leakage from the Orbiter cabin (mainly O_2, CO_2, H_2O with the following relative intensities: 490, 140, 1) is given in Table 2.2.
To first approximation, the outgassing cloud in steady state (after desorption of mainly water) consists of mainly hydrocarbons. Typical outgassing figures are given in Table 2.2.
Outgassing of materials used inside the cargo bay and exterior surfaces of the Orbiter exposed to payloads is controlled by selection of materials for low outgassing properties. Exposure of the materials to space on a flight will result in reduced outgassing rates on subsequent missions.

2.2.8.8 Deorbit and Descent Phase

During descent the aft main RCS thrusters operate with the emission of propellant combustion products. At this time, the APU and hydraulic subsystems are also in operation. Down to about 100,000 feet (30.48 km), large quantities of water are emitted by the flash evaporator. Below this height heat is rejected by an ammonia boiler. To avoid ingestion of these subsystem effluents, attention has been given to locating 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 deorbit and remain closed until about 75.000 feet (22.86 km), at which time they must be opened to allow for repressurization of the Orbiter. The forward RCS module is deactivated prior to deorbit burn.

Effluents from the aft RCS operation below 75.000 feet (22.86 km) will be swept back into the wake and circulated within it. Similarly, the ammonia boiler vent, APU hydrazine exhaust vents, and water boiler vents are located aft of the midfuselage 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. The resultant contamination will be determined by experimental results from the first orbital flight test. To inhibit ingestion of particulates into the Orbiter cargo bay during descent, the vent ports leading into the Orbiter cargo bay will be covered with a 35 micron glass bead rated (GBR) filter. DCN 009

2.2.8.9 Landing and Post-Landing Phases (at Kennedy Space Center)

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 15 minutes after touchdown, ground support operations will provide HEPA filtered purge gas to the Orbiter cargo bay volume (Section 5.4.2).

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

(Reference : Vol. XIV, Rev. E, Table 4.4)

Parameter / Source	Number Column Density (NCD) (polar 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				Orbital Altitude		
				700 km	435 km	200 km
LOS 1 A	5.6×10^{12}			8.4×10^{8}	2.4×10^{10}	1.3×10^{12}
LOS 5 A	5.6×10^{12}			8.5×10^{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				Orbital Altitude		
				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, Line-of-sight in the +Zo direction originating at Xo = 1 107 inch (17.8 m)

** LOS 5 A, Line-of-sight 50° off of + Zo towards - Xo (forward) originating at Xo = 1 107 inch (17.8 m)

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.

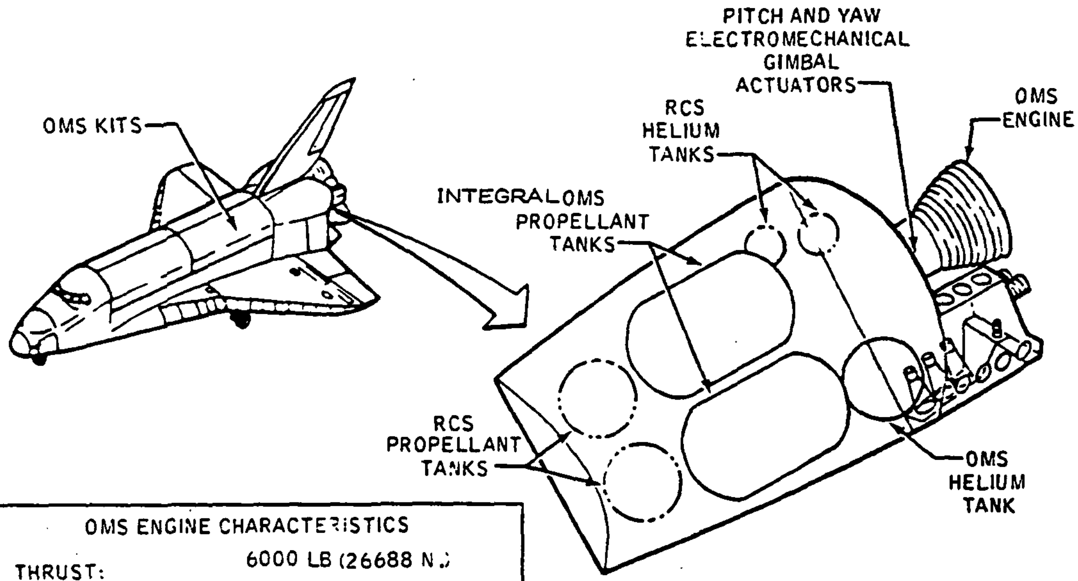
2.3.1 Orbital Maneuvering

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, rendezvous 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 (see also para. 2.4.1.4).

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 envelopes required for the installation of one, two and three OMS-kits are described in Figure 2-9. DCN 009

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 28.5 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 achievable inclination decrease below 28.5 degree is about 1 degree per OMS kit only because the inclination has to be restored to 28.5 degree prior to descent. The use of OMS-kits to obtain orbits with high altitude is shown in Figure 2 - 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. E, Figure 3 - 1)

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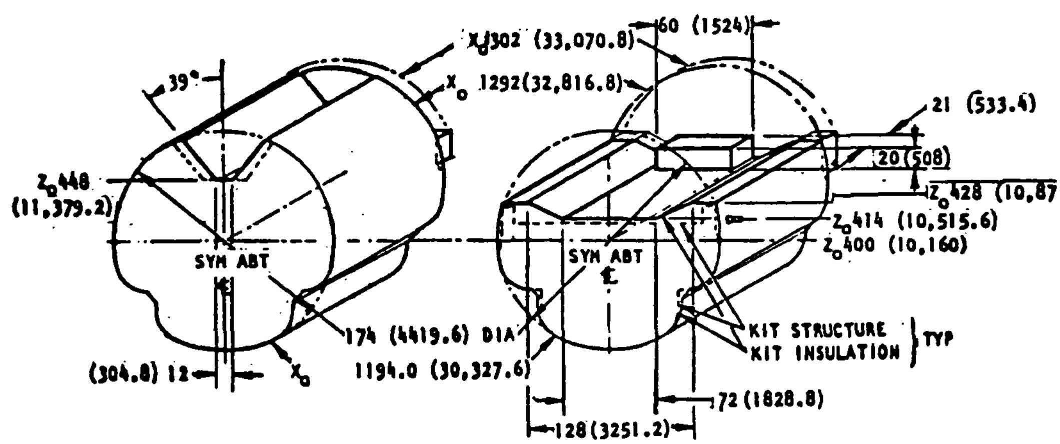

Triple Kit

Single and Double Kit

Concept – Potential Volume Above OMS Kits, No Scale, inch (m)

Insulate / Kit Clearance Shown

Figure 2-9: Envelope for OMS-Kits (Reference: ICD - 2 19001)

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Table 2-3 Weight and Velocity increments of OMS-Kits
(Reference, Vol. XIV, Rev. E, Table 3.9)

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*	Velocity increments m/sec (ft/sec) for 29 484 kg (65 000 lb) cargo weight*
1	TBD	6466 (14255)	~ 183 (~ 600)	152.4 (500)
2	TBD	12533 (27631)	~ 366 (~1200)	304.8 (1000)
3	TBD	18601 (41009)	~ 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 28.5° and 57° and from VAFB for inclinations between 56° and 104°.

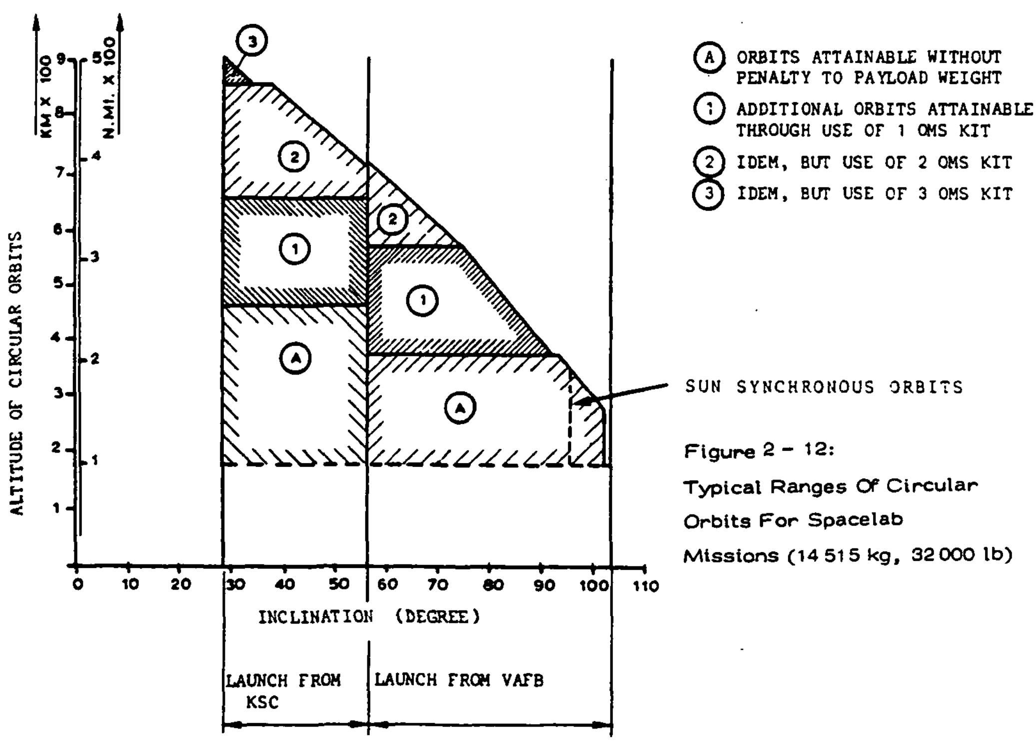

Figure 2 - 12:
Typical Ranges Of Circular Orbits For Spacelab Missions (14 515 kg, 32 000 lb)

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 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 numbers 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.

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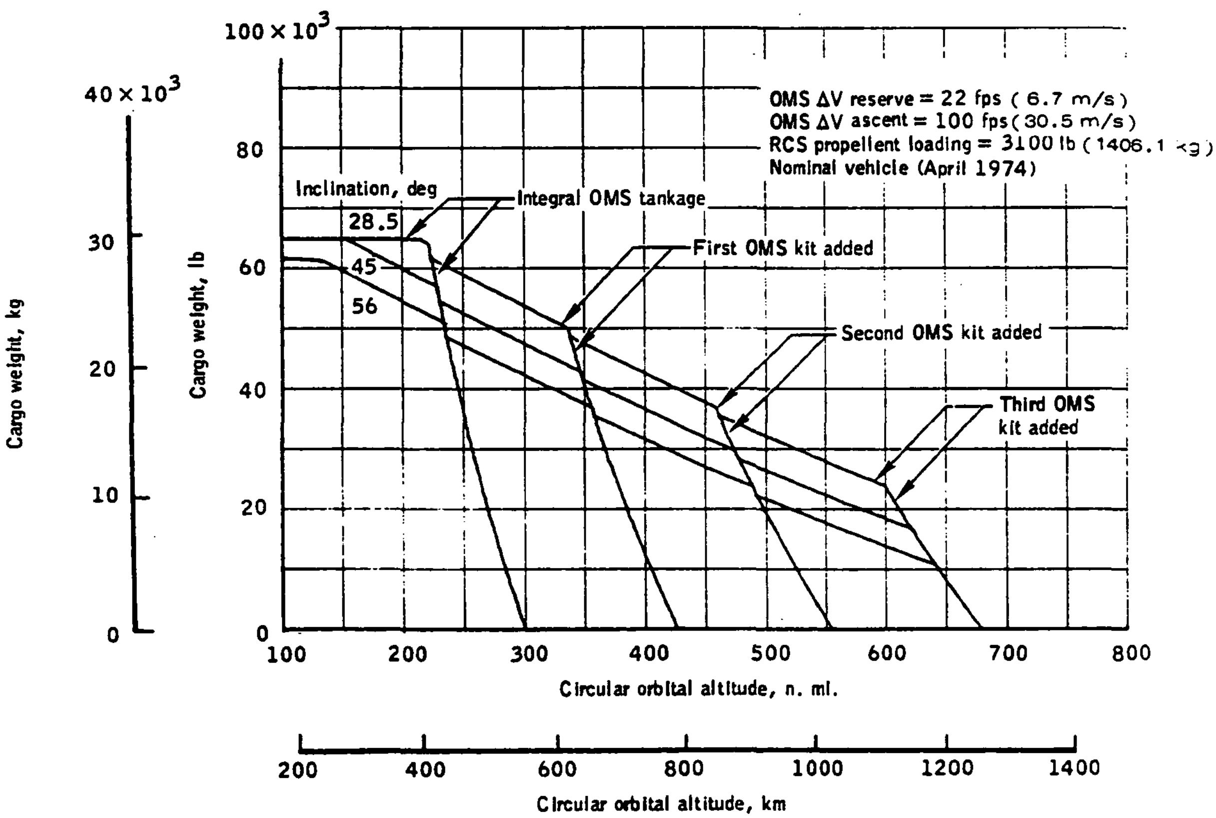

Figure 2-13: Cargo weight versus circular orbital altitude - KSC launch (Reference, Vo.. XIV, Rev. E, Fig. 3.3)

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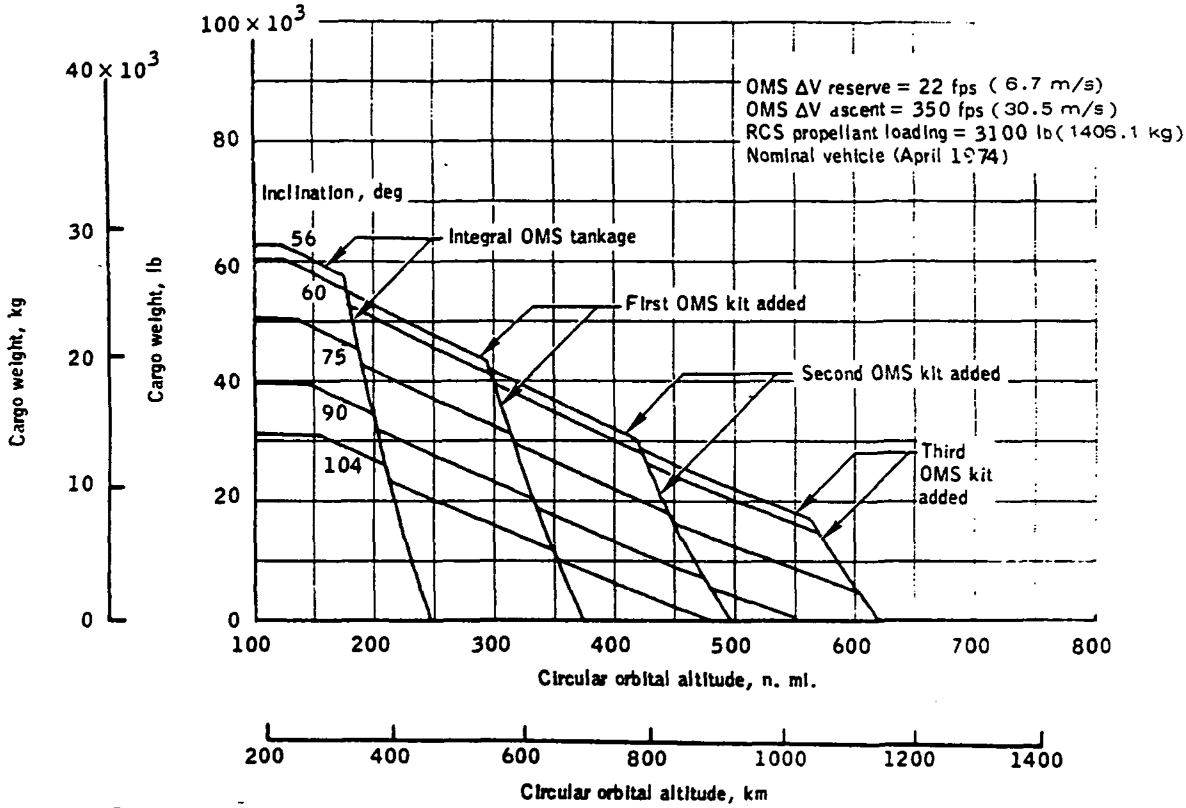

Figure 2-14: Cargo Weight Versus Circular Orbital Altitude - VAFB Launch (Reference, Vol. XIV, Rev. E, Fig. 3.4)

The Space Shuttle also has the capability to place Spacelab into elliptical orbits. This capability depends significantly on the de-orbit mode. Orbits with maximum eccentricity 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, ce 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 of covering the whole range of possible angles of right ascension of ascending nodes. Missions requiring specific angles of right ascension of ascending node have to be evaluated on an individual basis.

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

Inclination	direct de-orbit		indirect de-orbit	
	apogee in n.mi. (km)	number of OMS-kits	apogee in n.mi. (km)	number of OMS-kits
28.5°	1350 (2500)	3	620 (1150)	3
55°	1100 (2050)	2	510 (950)	2
104°	300 (550)	0	150 (280)	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: PRCBD 500781 R 1)

Launch Site	Position n.mi. (km)				Velocity ft/sec (m/sec)			
	Down-range $\pm x$	Cross-range $+ y$	Altitude $\pm z$	Total r	Down-range $\pm \dot{x}$	Cross-range $\pm \dot{y}$	Altitude $\pm \dot{z}$	Total v
KSC	8.3 (15.3)	1.1 (2.0)	0.5 (0.9)	8.4 (15.5)	9.4 (2.9)	22.5 (6.9)	13.7 (4.2)	28.0 (8.6)
VAFB	10.9 (20.2)	1.5 (2.8)	0.5 (0.9)	11.0 (20.4)	10.2 (3.1)	22.5 (6.9)	14.7 (4.5)	28.8 (8.8)

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 System (TDRSS), 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 TDRSS navigation accuracies are based on two tracking passes from a single TDRS. The data on position and velocity of the Orbiter are available to Spacelab and its payload (Section 2.4.2.1).

Table 2 - 6: Expected On-Orbit Navigation Accuracies (3 Sigma) for 100 Nautical Miles (185 km) Orbital Altitude

(Reference: Vol. XIV, Rev. E : Table 3.1) DCN 010

Navigation System	Position, Feet (Meters)				Velocity, Feet/Sec (Meters/Sec)			
	Altitude	Down-track	Cross-track	Total	Altitude	Down-track	Cross-track	Total
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)	2070 (630)	1.6 (0.5)	0.35 (0.11)	0.5 (0.15)	1.7 (0.5)
After one revolution	300 (90)	2010 (610)	1 520 (460)	2400 (740)	2.4 (0.7)	0.3 (0.1)	0.5 (0.15)	2.5 (0.7)

DCN 009

The time required to perform an orbit correction maneuver and 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 to these activities have to be assessed on an individual basis. DCN 009

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 Figure 2 - 15.

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

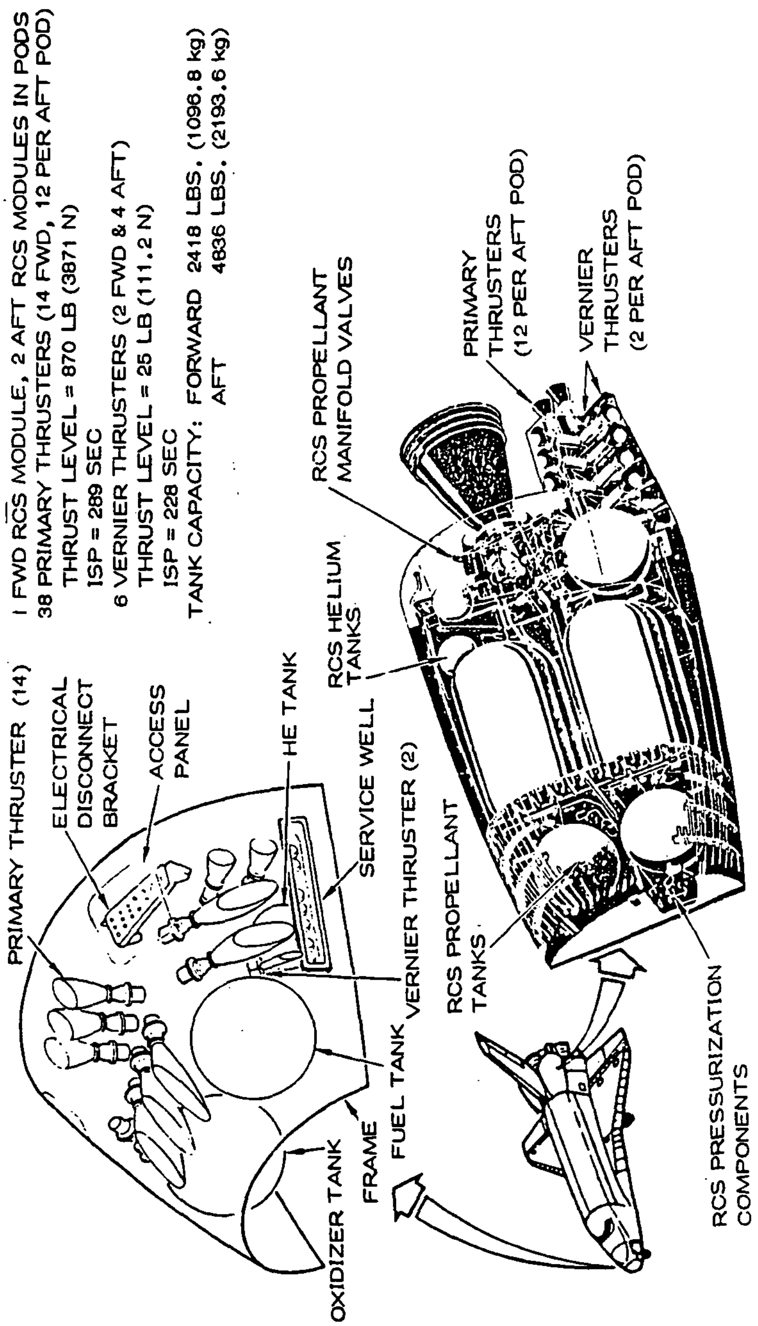

Figure 2 - 15: Orbiter Reaction Control Subsystem (RCS)
(Tank and Thruster Arrangement)
(Reference, Vol. XIV, Rev. E, Figure 3.2)

DCN 010

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 (para. 3.6.2).

2.4.1.2 Primary Thrusters

The propellant consumption of the RCS system for attitude maneuvers, for example,pointing from one inertial direction to another, is described below for the primary thrusters. Table 2 - 8 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). The propellant consumption for initiation and termination of the maneuvers around each axis is included in the figures given in Table 2 - 8.

2.4.1.3 Vernier Thrusters

The vernier RCS propellant usage for various orbital altitudes and Orbiter orientation modes is presented in Table 2 - 9 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 - 9. 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 include the aerodynamic and gravity gradient disturbances. Disturbing torques due to venting are not included in the data and could be significant.

Table 2 - 7: Typical RCS Propellant Budget

(Reference, Vol. XIV, Rev. E, 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

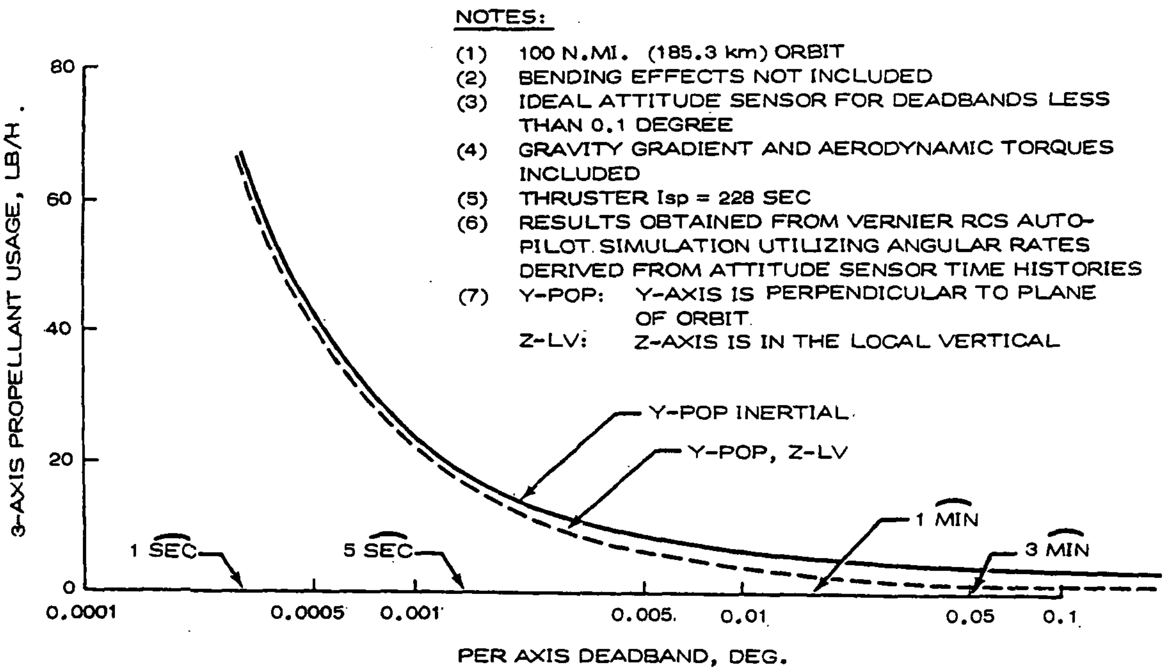

Figure 2 - 16: Effect of Attitude Deadband on Vernier Thruster Propellant for Typical Local Vertical and Inertial Orientation

(Reference, Vol. XIV, Rev. E, Fig. 3 - 14)

Table 2 - 8: Typical RCS Propellant Usage of Primary Thrusters (Reference, Vol. XIV, Rev. E, Table 3 - 5)

ROTATION RATE DEG/SEC	PRIMARY THRUSTER PROPELLANT USAGE, LBS (KG)			
	ROLL	PITCH	YAW	TOTAL
0.25	3.9 (1.77)	5.9 (2.68)	7.2 (3.26)	17.0 (7.71)
.50	7.0 (3.17)	10.3 (4.67)	12.9 (5.85)	30.2 (13.70)
.75	10.5 (4.76)	14.5 (6.58)	19.7 (8.93)	44.7 (20.27)
1.00	14.0 (6.35)	20.3 (9.21)	26.3 (11.93)	60.6 (27.49)

Notes:

- Utilizing primary thruster configuration (38 primary thrusters, I_{sp} = 289 sec, minimum on time = 40 ms).
- Compensated for rotational cross-coupling.
- Includes propellant for initiation and termination of each maneuver.
- Attitude hold (± 0.5 deg/axis) maintained in other axes during maneuver about each axis.
- X, Y. Z are standard airplane axes (see Figure 2 - 2).
- Roll, pitch, yaw are rotations about standard airplane X, Y and Z - Axes, respectively.
- Includes impingement effects of thrusters pointing into the direction - Z.
- Typical mass properties of Orbiter including a weight of Spacelab plus its payload of 32 000 lbs (14 515 kg).

Table 2-9: Typical Propellant Usage of Vernier Thrusters for Various Altitudes and Orientation

(Reference, Vol. XIV, Rev. E', 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[d]	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 vernier thruster configuration (six 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 (one sigma) noise included.

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

Gravity gradient and aerodynamic torques included, atmospheric density based on the 1962 US standard atmosphere for the 100 n mi (185.2 km) orbit.

Deadband per axis: $\pm$ 0.1 deg.

Venting effects neglected

b) X, Y, Z axes are standard airplane axes (see Figure 2 - 2).

c) Perpendicular to orbit plane (POP)

d) Local vertical (LV)

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 (after rendezvous) are typically 1580 lbs (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 lbs (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 thermally critical Orbiter attitudes are TBD.

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

Translation 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)

Notes:

- Utilizing primary thruster configuration (38 primary thrusters, I_{sp} = 289 sec, minimum on-time = 40 m s).
- Includes compensation for rotational and translational cross-coupling.
- X, Y, and Z are standard airplane axes (see Figure 2 - 2).
- Includes impingement effects on -Z thrusters.
- Weight of Orbiter is 202 000 lbs (91 627 kg) including a weight of Spacelab plus its payload of 32 000 lbs (14 515 kg).
- Attitude hold (± 0.5 deg/axis) maintained during maneuvers.

Table 2 - 11: Rendezvous Limits
(Reference, Vol. XIV, Rev. E, Table 3-11)

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Parameter	Type of Orbiting Payload: Cooperative (1)	Type of Orbiting Payload: 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 mradians/sec	$\pm$ 4 mradians/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 at 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. A typical propellant consumption for a three-hour period of thermal conditioning, a Y-POP inertial Orbiter attitude, a 100 nmi (185.2 km) circular orbit (see Table 2 - 9) and a roll rate of 5 revolutions per hour (see Table 2 - 8) is 8 kg (18 lb). This figure includes the propellant required for initiating and terminating the roll about the Xo-axis. Furthermore, 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. In order to describe the pointing performance the terms "accuracy"and "stability" are used. These terms are defined in Figure 2 - 17. Pointing accuracy for inertial or earth referenced directions is within a $\pm$ 0.5 degree (3-sigma) half cone angle. The pointing error for continuous pointing will increase with time due to drift of the IMU. Also, the duration of continuous pointing is limited by the thermal constraints mentioned in para. 2.4.1.6.

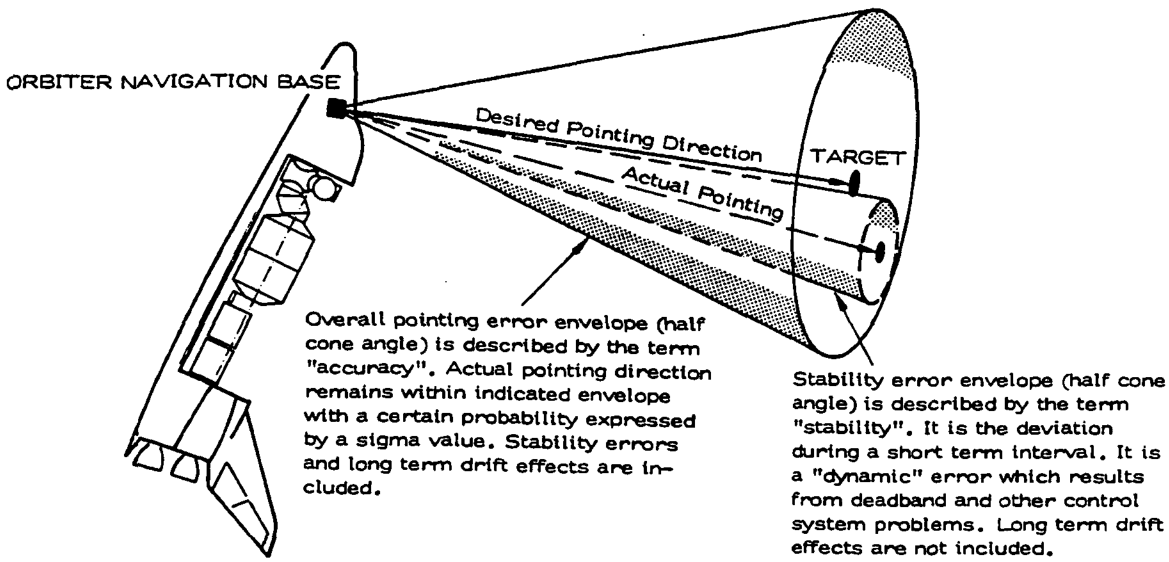

Figure 2 - 17: Definition of Orbiter Pointing Accuracy and Stability

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. The Orbiter Guidance, Navigation and Control computer will receive and process the attitude error signals from such a sensor. In order to meet the pointing accuracy this sensor information must be updated to the Orbiter Guidance, Navigation and Control computer at rates compatible with sample rates of the general purpose computer and consistent with the chosen method for determining angular rates and accelerations during payload pointing. The combined effect of quantization and noise on sensor readout must be no greater than 30 arc seconds (1 sigma) per axis. Details of the interfaces between the Spacelab payload supplied sensor, Spacelab 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. The rate of change of this pointing accuracy will now also depend upon the drift characteristics of the Spacelab payload sensor. DCN 009

The Orbiter Guidance, Navigation and Control computer will be able to provide the following initialization or ephemeris data, also to Spacelab and its payload:

a) position and velocity of Orbiter

b) attitude orientation angles and attitude rate

c) time

The specific frame, data format etc. is TBD.

2.4.2.2 Pointing Stability

For Spacelab payload pointing utilizing the Vernier Thruster, the Orbiter Flight Control System provides a stability of $\pm$ 0.1 deg/axis. This figure is in essence identical to the dead band of the Flight Control System. The maximum stability rate is $\pm$ 0.01 deg/sec/axis for the limit cycle of the control system when no Vernier Thrusters have failed. When using the primary thrusters, the Orbiter Flight Control System is capable of providing a stability of $\pm$ 0.1 deg/axis and a stability rate of $\pm$ 0.1 deg/sec/axis. For pointing and/or stability requirements beyond the capability of the Orbiter, the Orbiter is capable of accepting compatible commands from a Spacelab payload supplied and Spacelab mounted stabilization and control system such as the Instrument Pointing Subsystem (reference Section 4.8). However the details of such an interface have not yet been determined. DCN 009

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 pointing the Orbiter into an inertial direction utilizing the Orbiter IMU includes

- errors due to the deadband ($\pm$ 0.1 deg/axis) of the Flight Control System
- errors due to the IMU alignment uncertainty of $\pm$ 0.133 deg/axis (3 sigma)
- read-out errors of the IMU ($\pm$ 0.073 deg/axis, 3 sigma)
- drift rate of IMU ($\pm$ 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 $\pm$ 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 is 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° 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 in pointing the Orbiter along the local vertical or to a target on the earth surface includes the inertial pointing errors 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 TDRSS 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 inflight 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

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

Type of Pointing	Pointing Accuracy (3 Sigma) (Half-Cone Angle)	IMU-Drift 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. E, 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/Sun line 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 Xo - upward 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 view 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: commander, pilot, mission specialist and 4 payload specialists.

For Spacelab flights for a continuous 24 hour operation a minimum 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 may 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 up-dates, checklist reviews etc. The concept of crew timelining is presented in Figure 2 - 18.

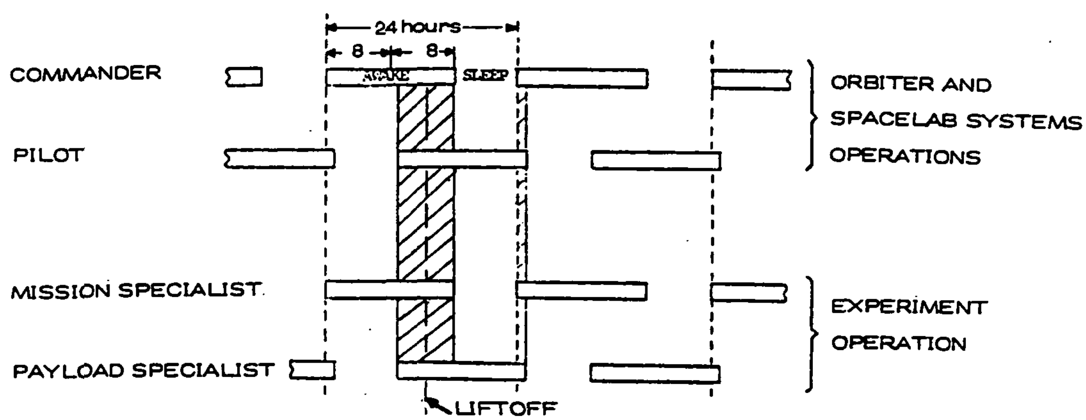

Figure 2 - 18: Concept of Crew Timelining for Continuous 24-Hours-Operation, Minimum Total Crew of 4

2.5.2 Crew Compartment and Accommodation

The Orbiter crew compartment is a two-level cabin consisting of the flight deck and the mid-deck as shown in Figures 2 - 19 a and b .

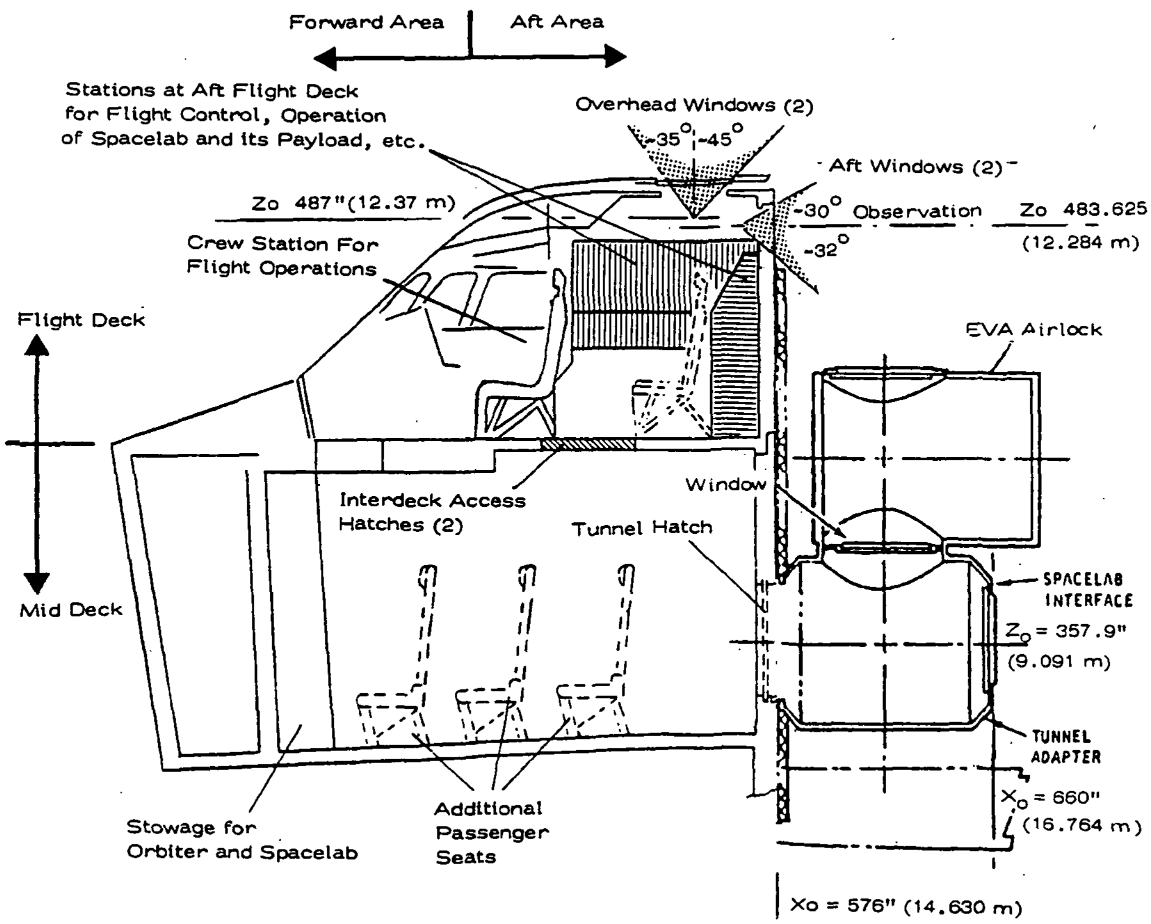

Figure 2 - 19a: Crew Compartment Concept - Looking sidewards (References: ICD - 2 - 19001, Vol. XIV, Rev. E, Fig. 11-1, 13.3 A Passenger seats and view angles according to Vol. XIV, Appendix C, p.C8)

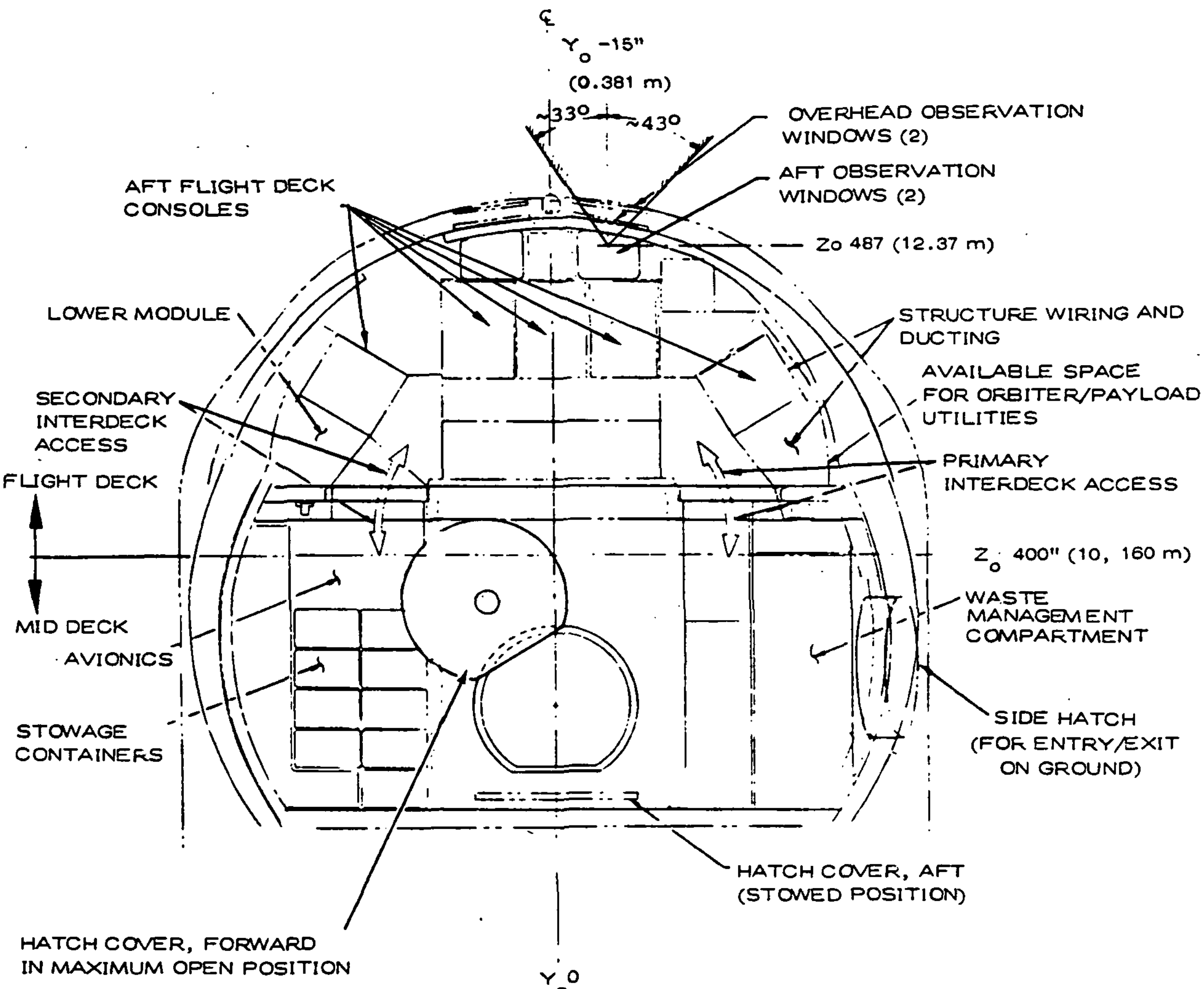

Figure 2 – 19 b: Crew Compartment Concept – Looking Aft

(Reference, Vol. XIV, Rev. E, Fig. 11 – 3, C10)

DCN 010

The forward area of the flight deck is dedicated primarily to Orbiter operations during ascent and descent. It is equipped with displays, controls and two seats for the commander and pilot. These seats are not removable.

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

The seat arrangement during ascent and descent is shown in Figure 2 - 20.

The mid-deck area includes sleeping stations, galley, waste management and loose equipment stowage provisions. Three additional payload specialists can be accommodated on this deck during ascent and descent.

The layout of the aft flight deck is shown in Figure 2 - 21: Three work stations, namely the "mission station", the "on-orbit station" and the "payload station" can be distinguished. Furthermore, attention is drawn to the panel R 7 which is the only panel which can be accessed by the mission specialist during ascent and descent. Figure 2-21 indicates the panels which are available for Spacelab and payload. All consoles in the aft flight deck (with the exception of the R7 panel) support 19 inch MIL-Standard 189 equipment. Increased panel surface area available on inboard surfaces of consoles L13, L14, R15 may be utilized as display and control panels allowing 6" (0.1524 m) for depth. The additional area would be 5.5 ft^2 (0.51 m^2). These panels are shear panels and are not compatible with MIL-standard 189. The physical accommodation for Spacelab and its payload is summarized in Table 2-14. The entire R12, L10 and L12 panels, together with volumes associated with all these panel areas are allocated to Spacelab payloads. If the instrument pointing subsystem (IPS) is flown, an IPS panel will be located in the on-orbit-station. The panels R7 and L11 are fully dedicated to Spacelab hardware.

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As indicated on Figure 2-21 the consoles in the mission and payload station are removable from the Orbiter in order to permit equipment integration off-line from the turn-around cycle of the Orbiter. It is envisaged that the Spacelab subsystem hardware in the aft flight deck(with the exception of the IPS panel) is independent of the Spacelab configuration to be flown.

In the present operational concept for Spacelab it is foreseen that the Spacelab subsystems can be operated from the mission station using the Orbiter display and keyboard located in the panel R11 and from the Spacelab integrated control panels located in panel R 7. This panel contains also provisions for switches and indicators for payload safing function required during ascent and descent. The payload station and part of the on-orbit station are dedicated to experiment operation.

DCN 009

The power for Spacelab and its payload at the aft flight deck is defined in para 3.6.3. The available heat rejection is compatible with the quoted power figure. Details will be presented in connection with the overall Spacelab power and heat rejection budget (Sections 3.6.3 and 3.6.4).

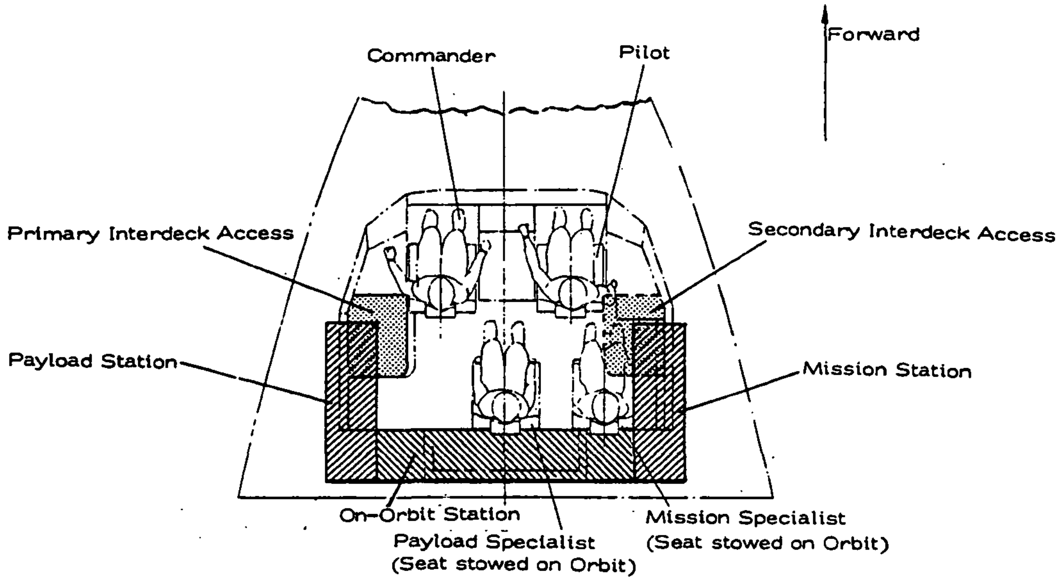
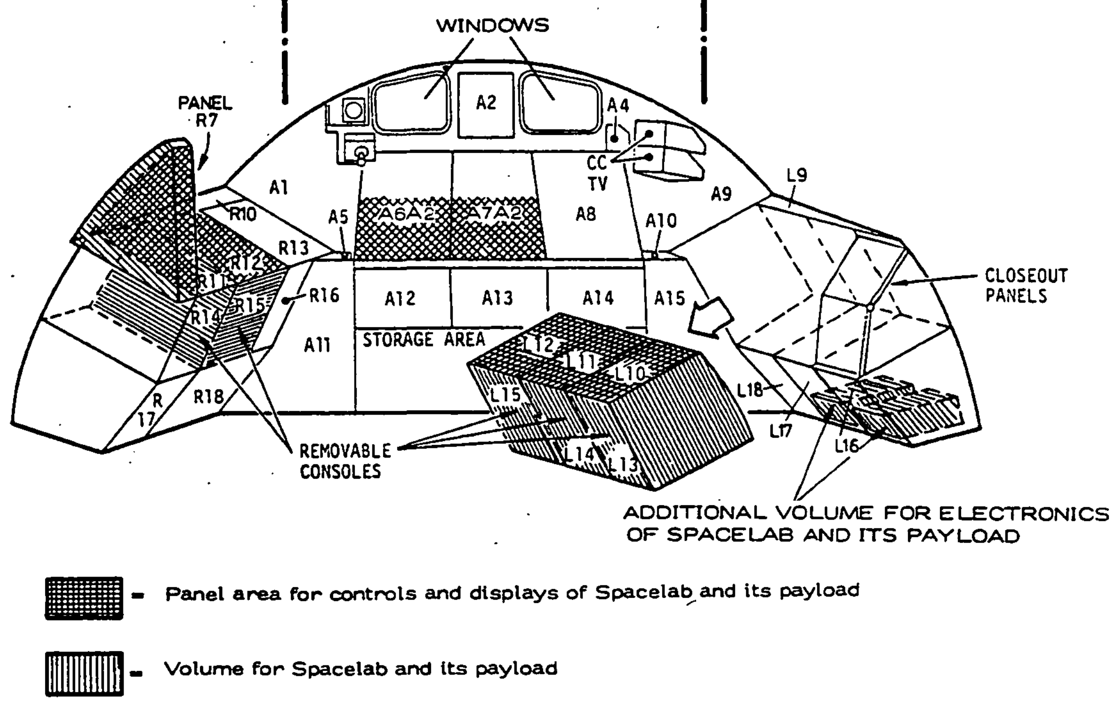


Figure 2 - 20: Crew Accommodation on the Flight Deck during Ascent/Descent
(Reference, Vol. XIV, Rev. E, p. C - 9)

DCN 010



DCN 009

Figure 2 - 21: Layout of Aft Flight Deck (View Looking Aft)
(Reference: ICD 2 - 05101)

DCN 009

Table 2 – 14: Physical Accommodations on the Aft Flight Deck for Spacelab and its Payload
(Reference: ICD – 2 – 0510)

Location at Aft Flight Deck	Usage	Panel Depth	Panel Area	Volume	Mass 1)
		inch (m)	ft^2 (m^2)	ft^3 (m^3)	lbs (kg)
• Panel R 7	Spacelab)2	8 (0.203)	2.3 (0.21)	1.5 (0.042)	45 (20.4)
• Mission Station R 12	Payload	20 (0.508)	2.8 (0.26)	4.6 (0.130)	138 (62.6)
• On-orbit station A6A2 & A7A2	partially by Spacelab)3	5–10 (0.127 – 0.254)	3.7 (0.34)	2.4 (0.068)	72 (32.6)
• Payload Station	L10, L12 avail' to payload; L11 used by Spacelab	20 (0.508)	8.3 (0.77)	13.8 (0.391)	414 (187.8)
• Additional Volume for electronics (L 16, L 17)	partially by Spacelab)5	–	–	1.3 (0.036)	39 (17.7)
Total	–	–	17.1 (1.59)	23.6 (0.668)	708 (321.1)

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NB: The masses quoted in the following notes are design masses for the AFD

Note 1 : Maximum loading based on 30 lbs/ft^3 (480 kg/m^3)
Note 2 : Integrated Monitoring and Control Panel, 19.8 lb (9 kg)
Note 3 : Panel for IPS (Instrument Pointing Subsystem)
Note 4 : Data Display Unit, 70.4 lb (32 kg); keyboard, 13.2 lb (6 kg); three data bus interconnection stations, 1.1 lb (0.5 kg); AFD power distribution box behind L11, L14, 4.4 lb (2 kg)
Note 5 : Subsystem RAU, 22 lb (10 kg)

A total volume of approximately 150 cubic feet (4.24 m^3) will be provided in the crew compartment for loose equipment stowage for the Orbiter, Spacelab and Spacelab payload. Approximately 95 percent of this volume will be on the mid-deck. Loose equipment includes those items which are not permanently mounted in the crew compartment. The orbiter supporting structure for stowage containers has an average rated capacity of 20 lbs/ft^3 (320 kg/m^3) with a rating of 30 lbs/ft^3 (481 kg/m^3) for each attachment point. The allocation of stowage containers for loose equipment of Spacelab and for Spacelab payload 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, LiOH-canisters for CO_2 removal from the crew compartment, food 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, 28 mandays of expendables for normal operations plus 16 mandays of expendables for rescue operations for four men and for a duration of 4 days. In addition, volume (chargeable to the Orbiter) can be provided for expendables for up to 42 mandays, but the weight of these expendables is charged to the Spacelab payload. Weight and volume above the outlined provisions will be charged to the Spacelab payload. Table 2 – 17 gives a survey of the items and services charged to the Spacelab payload.

The crew compartment is connected with the Spacelab module through a tunnel. For this purpose a tunnel adapter must be attached to the rear end of the mid-deck. A hatch (Figure 2 - 19) separates the crew compartment from the tunnel adapter, tunnel and module. On top of the tunnel adapter the "EVA airlock" is attached permitting to perform Extra Vehicular Activities (EVA) without interrupting the connection between the crew compartment and the module. Figure 2 - 19 shows the arrangement of tunnel adapter and EVA airlock used for all module and module plus pallet mission. For pallet-only missions the tunnel and tunnel adapter are not required. For these missions the EVA airlock can be attached to the opening at the rear end of the mid-deck at Xo = 576" (14.630 m) and be placed either inside or outside of the mid-deck.

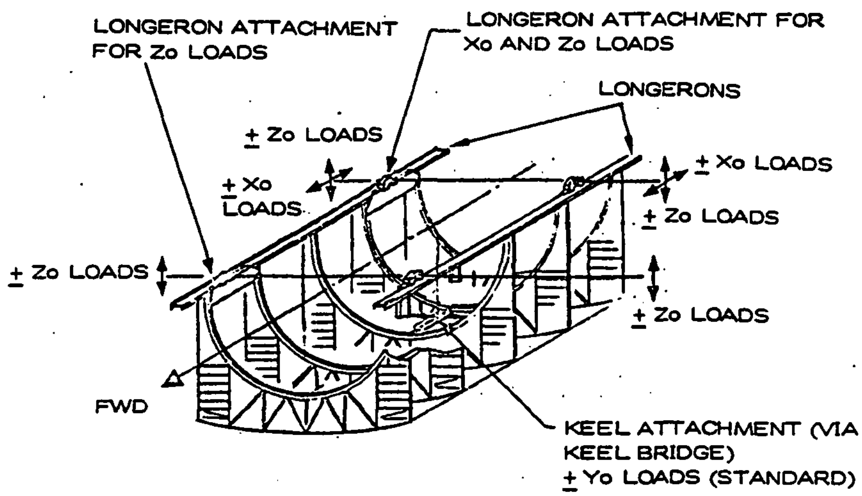
2.6 Orbiter Support to Spacelab

2.6.1 Summary of Support

In addition to orbital maneuvering, attitude control and crew accommodation described in the previous sections,the Orbiter provides a number of services and items which are available for use by Spacelab and its payload.

2.6.1.1 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 attachment points. The structural attachment system is designed to facilitate statically determined attachment. There are mounting provisions of attachment points at the bottom (keel) of the Orbiter cargo bay and on longerons on both sides of the Orbiter cargo bay at Zo = 414 inches (10.516 m) and Yo = $\pm$ 94 inches (2.388 m). The keel attachment points are designed to take loads in the Yo-direction only. There are two types of attachment points on the longerons. One type is able to take primarily loads in the Xo- and Zo-directions, the other type is able to take primarily loads in the Zo-direction only. Figure 2 - 22 illustrates the attachment system. Table 2 - 15 lists the location of attachment points available to Spacelab. DCN 009



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

DCN 010

Figure 2 - 22: Orbiter Attachment System
(Reference, Vol. XIV, Rev. E, C 13)

Table 2 – 15: Attachment Points – Longeron and Keel
(Orbiter Structural Coordinate System Locations)

X_o – Location

Attach Pt #	X0 Location	Attach Pt #	X0 Location	Attach Pt #	X0 Location	Attach Pt #	X0 Location
154K	608.80	197	777.93	240	947.07	282	1112.27
155K	612.73	198	781.87	241	951.00	283	1116.20
156*	616.67	199	785.80	242	954.93	284	1120.13
157	620.60	200	789.73	243	958.97	285	1124.07
158	624.53	201	793.67	244K	962.80	286	1128.00
159L	628.47	202	797.60	245K	966.73	287	1131.93
160L	632.40	203L	801.53	246K	970.67	288L	1135.87
163L	644.20	204L	805.47	247L	974.60	289L*	1139.80
164L	648.13	207L	817.27	250L	986.40	292L	1151.60
165	652.07	208	821.20	251L	990.33	293	1155.53
166	656.00	209	825.13	252	994.27	294	1159.47
167	659.93	210	829.07	253	998.20	295	1163.40
168K	663.87	211	833.00	254	1002.13	296	1167.33
169K	667.80	212	836.93	255	1006.07	297	1171.27
170K	671.73	213	840.87	256	1010.00	298	1175.20
171K	675.67	214	844.80	257	1013.93	299*	1179.13
172K	679.60	215	848.73	258	1017.87		
173K	683.53	216	852.67	259	1021.80	304*	1198.80
177L	**699.27	217	856.60	260	1025.73	305	1202.73
178L	703.20	218L	860.53	261	1029.67	306	1206.67
179	707.13	221L	872.33	262L	1033.60	307	1210.60
180	711.07	222	876.27	263L	1037.53	308	1214.53
181	715.00	223	880.20	266L	1049.33	309	1218.47
182	718.93	224	884.13	267	1053.27	310	1222.40
183	722.87	225	888.07	268	1057.20	311	1226.33
184	726.80	226	892.00	269	1061.13	312	1230.27
185	730.73	227	895.93	270	1065.07	313	1234.20
186K	734.67	228*	899.87	271	1069.00	314L	1238.13
187K	738.60	229K	903.80	272	1072.93	315L	1242.07
188L	742.53	230K	907.73	273	1076.87	316L	1246.00
189L	746.47	234L	923.47	274	1080.80	317K	1249.93
192L	758.27	235L	927.40	277L	1092.60	322L	1269.60
193L	762.20	236L	931.33	278L	1096.53	323L	1273.53
194	766.13	237	935.27	279L	1100.47	324L	1277.47
195	770.07	238	939.20	280	1104.40	325L	1281.40
196	774.00	239	943.13	281	1108.33	330L	1303.00

Y_o and Z_o Locations

Longeron: $Y_o \pm$ 94.0, Z_o 414 except for att. pt. 330L $Y_o \pm$91.4, Z_o 409.0
Keel: Forward of attach pt. 305: Y_o=0, Z_o=305.0

(Reference: ICD 2 – 05101)

Notes:

All attach points normally available are included in this list.

Attach points designated "L" (e.g. number 169L) are available only at the longerons.

Attach points designated "K" (e.g. number 168K) are available only at the keel.

* The longeron attach points at these locations may be used only for primary (bolt-down) attachment.

** Usage as a keel point made possible by special keel bridge fitting furnished by Spacelab tunnel.

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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 in the Orbiter baseline configuration. 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 weight 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 plus any residual or unused fuels and the wet weight of additional energy kits will be charged to the landing and launch weight of the Spacelab payload, respectively (para. 3.6.2). The weight of the fuel and the energy kits has to be 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 Orbiter Environment Control and Life Support

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

a) Atmospheric Revitalization in the Orbiter crew compartment
b) Food, Water, and Waste Management Service
c) Active Thermal Control in the Orbiter
d) Heat transfer from Spacelab and its payload to space
e) Fire Suppression in the Orbiter

The Orbiter provided expendables are outlined at the end of para 2.6.2 and in Table 2 - 17.

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 2.6.1.2); 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 via radiators on the doors of the Orbiter cargo bay (Section 2.2.3). The heat rejection can be supplemented by the operation of a flash evaporator when the Orbiter attitude is thermally unfavourable. Vaporized water produced by the hydrogen/oxygen fuel cell is expelled overboard through the flash evaporator. Heat generated in the Orbiter is rejected by this process. 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 and payload equipment located in the Orbiter cabin must be compatible with the use of Freon 1301. This applies also to payload equipment which is temporarily located in the Orbiter cabin, e.g. during ascent/descent.

DCN 009

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 or payloads for either planned or unscheduled EVA operations, the third operation is for rescue. There is no weight credit to Spacelab or payloads if no EVA is planned for a flight. 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.

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 hours of the three-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.

The prime mode of rescue is EVA. The Orbiter provides the equipment to support the rescue operation for 4 persons; 2 persons use the above mentioned 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. For any persons above four, the Spacelab payload will be charged for the weight of required kits of the personnel rescue system and expendables including contingencies to support these persons.

2.6.1.5 Orbiter Avionics

The Orbiter avionics provides for:

a) Reception, transmission and distribution of voice

b) Transmission of operational telemetry

c) Reception 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

i) Recording (payload recorder)

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

a) Tracking and Data Relay Satellite (TDRS) operating in KU-band and S-band

b) Space Tracking and Data Network (STDN) during ascent and descent

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 in its stored position 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 are made for the effect on the Orbiter center of gravity. The weight credited to payload if the Remote Manipulator System is not flown, is TBD.

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 series, not simultaneously. However, it is possible to hold or lock one manipulator arm while operating the other one.

Several end effector devices are currently undergoing investigation to identify a standard inventory of devices which will be provided by the Orbiter. The design will provide the capability for change-out of the end effectors on the ground. The standard end effector will have provisions for interfacing on orbit with a special purpose end effector. The weight of this special purpose end effector will be charged to the Spacelab payload. The capability for ground change-out of end effectors will permit the use of end effectors tailored to perform specific functions. Any weight excess of such a special end effector over that of the standard end effector will be charged to the Spacelab payload.

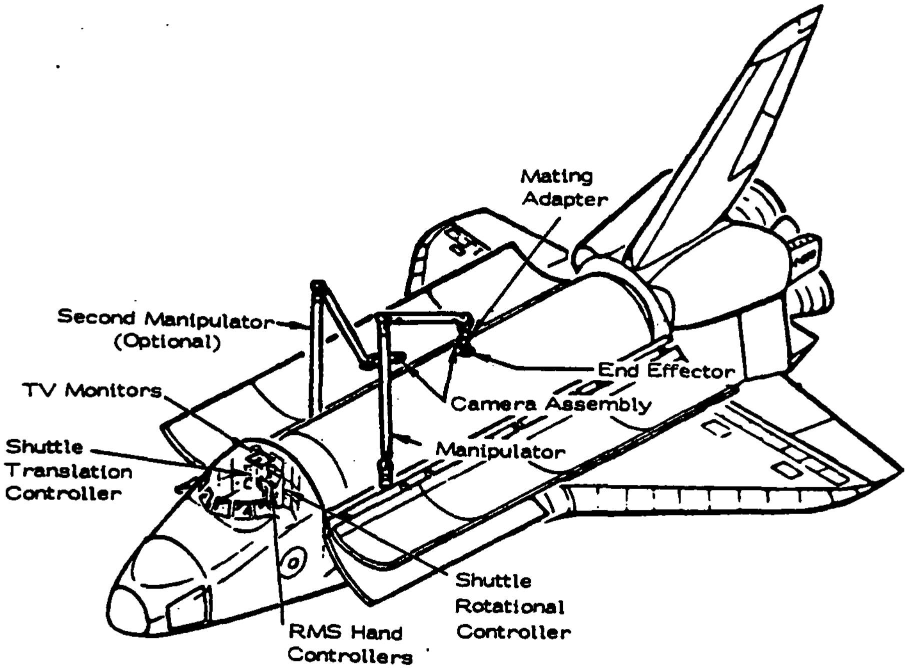

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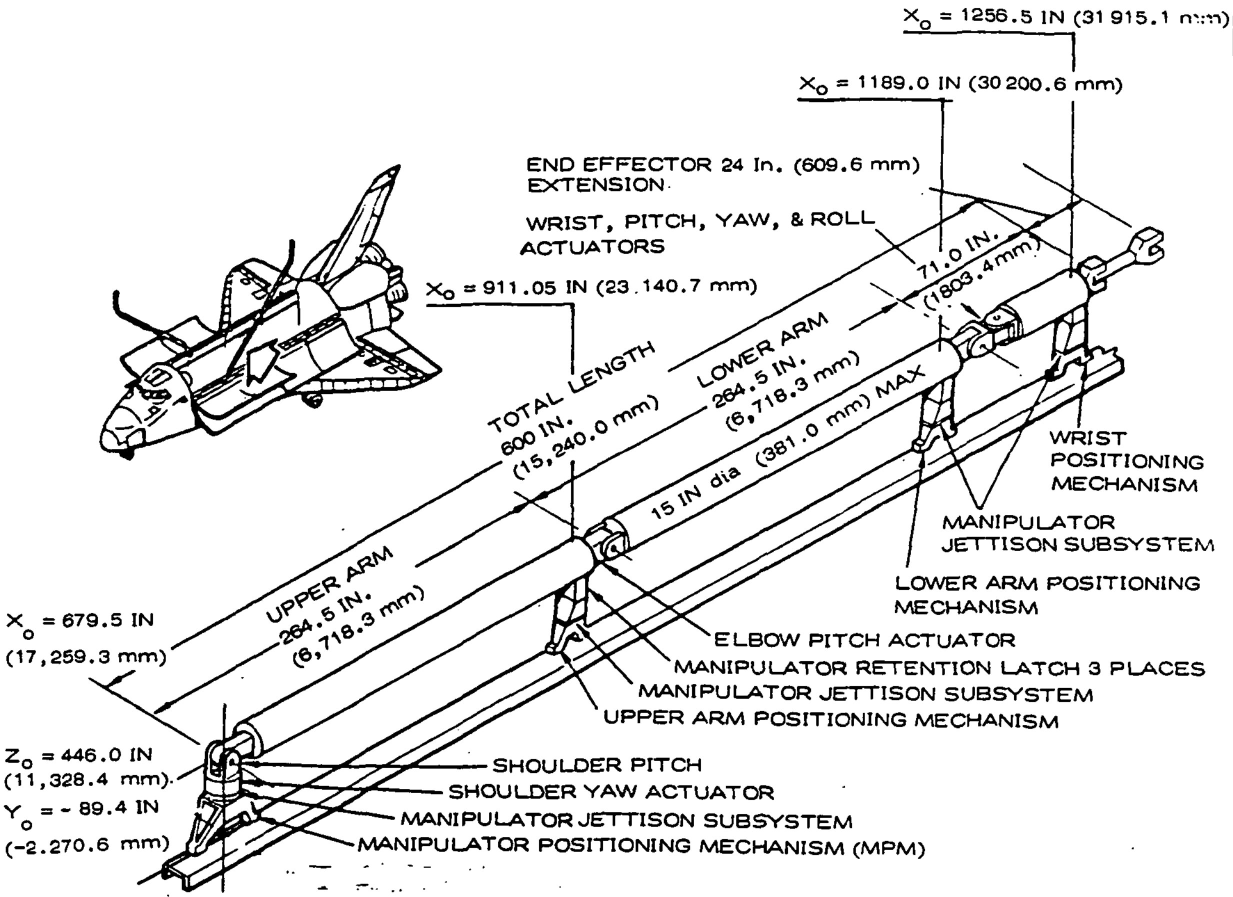
Figure 2 - 23: Orbiter Remote Manipulator System (Reference, Vol. XIV, Rev. E, Figure 8.3)



DCN 011

Figure 2 - 24: Manipulator Arm of the Remote Manipulator System
(Reference, Vol. XIV, Rev. E, Figure 8.1)

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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.

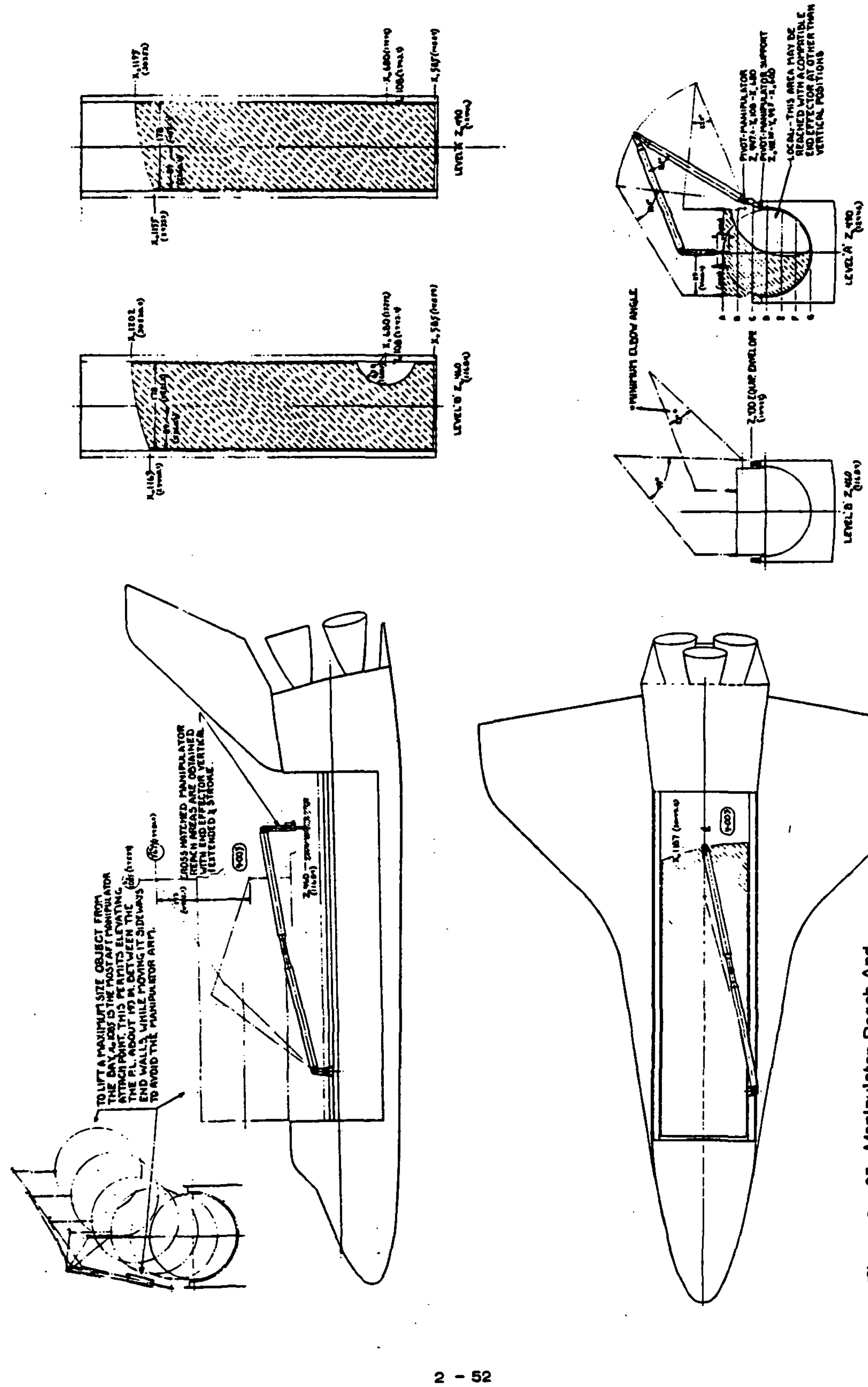

Figure 2 - 25: Manipulator Reach And Viewing Capabilities

(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, volumetric 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 - 17. The first column of Table 2 - 17 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.

DCN 010

Table 2 – 17: Summary of Orbiter Services Available to Spacelab Payloads (Reference, Vol. XIV, Rev. E, Table 3.13)

DCN 009

Items/Service	Orbiter provided/charged	Chargeable to Spacelab Payload	to launch weight lb (kg)	to landing weight lb (kg)	in volume (ft^3)	Center of Gravity (Inches) Xo	Yo	Zo	Reference Section
• Crew	4 men	above 4 (maximum 7 total)	200 (90.7)/man		possible locations:	490 490 490	48 28 8	349 349 349	2.5.1
• Crew Related Crew expendables (oxygen, food, LIOH, hygiene etc.)	28 MD 1) supplies + 16 MD contingency	above 28 MD	9.5 (4.3)/ MD		—		TBD		2.6.1.3
Seat & Restraints	4 men	above 4 (max. 7 total)	54 (24.5)/man		possible locations:	494 494 494	48 28 8	340 340 340	
Stowage 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			435 475	- 10 + 5	370 315	2.5.2
Crew Equipment (helmet, garment, food tray etc.)	for 4 men	above 4 (max. 7 total)	43.2 (19.6)/man		Vol. XIV				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	71 (32.2)/tank (165 lb capacity/tank)						
• 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)	202.9(92)/hardware set +15(6.8) for spare battery		TBD		TBD		2.6.1.4
		Expendables above Orbiter provision	9.6 (4.4)/EVA and man 2.7 (1.2)/EVA		TBD		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 (19.7)/man		TBD		TBD		2.6.1.4
Contingency of Expendables for Rescue Operations	4 men for 4 days	above 4 men for 4 days	8.7 (4)/MD 3 (1.4)/man		TBD				2.6.1.4
Survival Kit		above 6th crew man	42(19)		TBD				
• Tunnel Adapter For module and module plus pallet missions only	tunnel adapter in cargo bay plus EVA airlock on top of it	--	--	--	--	--	--	--	2.5.2
• EVA Airlock For pallet-only missions	EVA airlock inside of mid-deck	EVA airlock outside of mid-deck	123 (55.8) (additional structure)		--		TBD		2.5.2

Table 2 - 17: Summary of Orbiter Services Available to Spacelab Payloads (Reference, Vol. XIV, Rev. E, Table 3.13) (cont.)

DCN 010

Items/Service	Orbiter provided/charged	Chargeable to Spacelab Payload							Reference Section
			to launch weight lb (kg)	to landing weight lb (kg)	in volume ft^3	Center of Gravity (inches) Xo	Yo	Zo	
• Structural Support Set of fittings for attach points for statically determined suspension, including bridging hardware		Spacelab payload dedicated support	450 (204.1)			application dependent			2.6.1.1
Single fitting for attachment point		Spacelab payload dedicated support	51 (23.1)			application dependent			2.6.1.1
• Orbiter Service Spacelab Atmosphere Revitalization	TBD	TBD	TBD		TBD		TBD		4.3
• Heat Rejection	Nominal 8.5 kW heat rejection during orbital operations - kit increasing heat rejection from 6.3 kW to 8.5 kW 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 (3.2)/day	--	TBD		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 additional kits)	1632 (740.3) per kit	759 (344.2) per kit + TBD unusable and unused fuel			TBD		2.6.1.2
		above 3 kits, 873 lbs fuel per kit	TBD	TBD					
• Remote Manipulator System	One manipulator on left side of the Orbiter	second manipulator on right side of the Orbiter	867 (393.3)			935.8	84.0	436.1	2.6.1.6
• Propulsion Reaction Control System (RCS)	Tankage capacity	Propellant for on-orbit attitude control:	3993 (1811) max., limited by existing tankage capacity	0	--				2.4.1
• Orbit Maneuvering System (OMS)	Integral Tank	One OMS kit	14255 (6466)	TBD	defined in Figures 2-9 to 2-11	1246.5 TBD	4.4 TBD	377.9 TBD	
		Two OMS kits	27631 (12533)	TBD		1256.4 TBD	2.0 TBD	368.9 TBD	
		Three OMS kits	41009 (18601)	TBD		1244.4 TBD	1.6 TBD	389.7 TBD	
• Avionics wide band communication	Single steerable high-gain antenna	Scar provisions for second antenna available only	290 (131.5)			590	-90	480	2.6.1.5
Recording	--	payload recorder kit	60 (27.2)/unit						2.6.1.5

Note 1): MD = Man Day

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Section 3 - SPACELAB SYSTEM CAPABILITY

LIST OF TABLES

LIST OF FIGURES

3. SPACELAB SYSTEM CAPABILITIES

The Spacelab system provides versatile services to payloads as depicted in Figure 3 - 1. The overall system capabilities and resources will be described in this section. A detailed description of the design aspects of the various subsystems of interest to the user is provided in Section 4 - Subsystems. The Spacelab side of the interface between Spacelab subsystem equipment units and experiments will be described in Interface Definition Appendices attached to the main volume of SPAH.

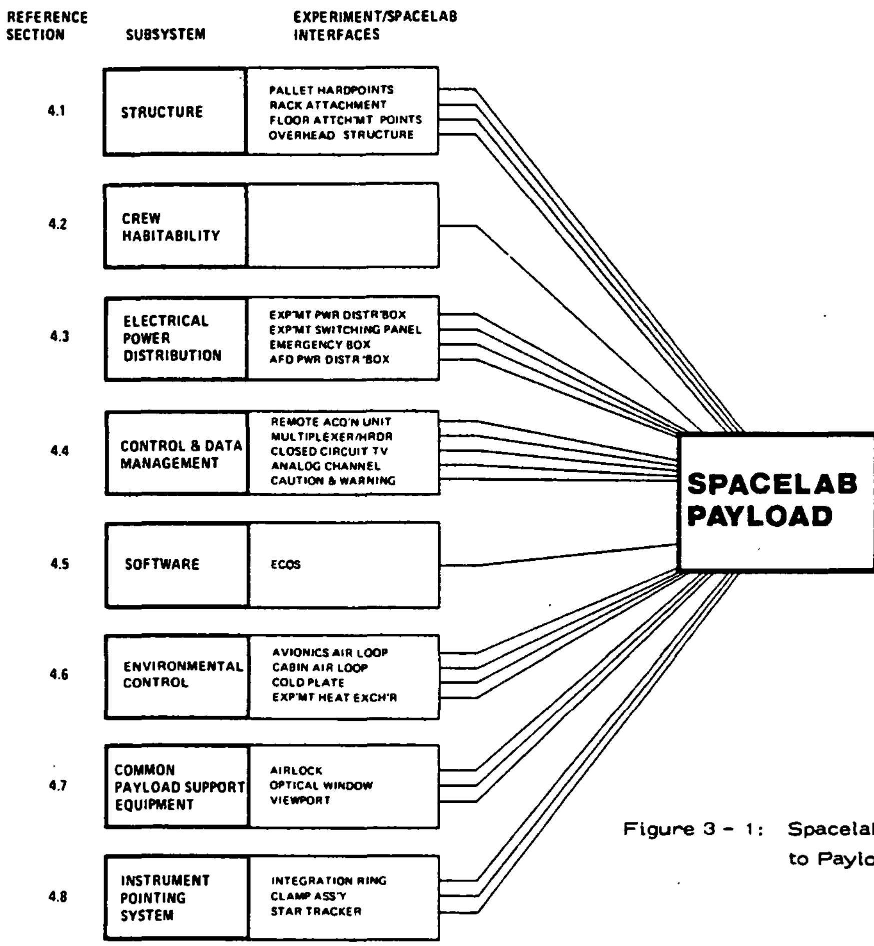


Figure 3 - 1: Spacelab Services to Payloads

This section describes the overall capabilities of the Spacelab system. Eight basic flight configurations and their center of gravity constraints are shown. An overview is given on the modular elements of Spacelab including module, pallet, igloo, and the crew transfer tunnel. The basic Spacelab subsystem elements, as well as the mission dependent equipment, are presented in tabular form. Utility services provided and utility routing are described. Payload Accommodation capabilities are tabulated and described in a comprehensive form, comprising geometric data, mass and power budget, heat rejection capabilities and command and data handling resources. Extended flight capabilities are briefly discussed.

3.1 Flight Configurations

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

Eight basic flight configurations are presented in detail with respect to their physical accommodation capabilities. While other configurations are basically possible, only these eight configurations are under configuration control with respect to the mechanical interfaces to the Orbiter. In addition, of these eight 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/9 m pallet (3 pallet segments)
- Long Module
- 15 m pallet (5 pallet segments)
- 9 m pallet (3 independently suspended pallet segments)

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

3.1.1 Long Module Configuration (Baseline Design Configuration)

The long module configuration is shown in Figure 3 - 2. It consists of the core and experiment segment and provides the largest pressurized volume for Spacelab payloads. It is accessible from the Orbiter cabin through the transfer tunnel. 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, since this volume is reserved for the EVA airlock and tunnel adapter.

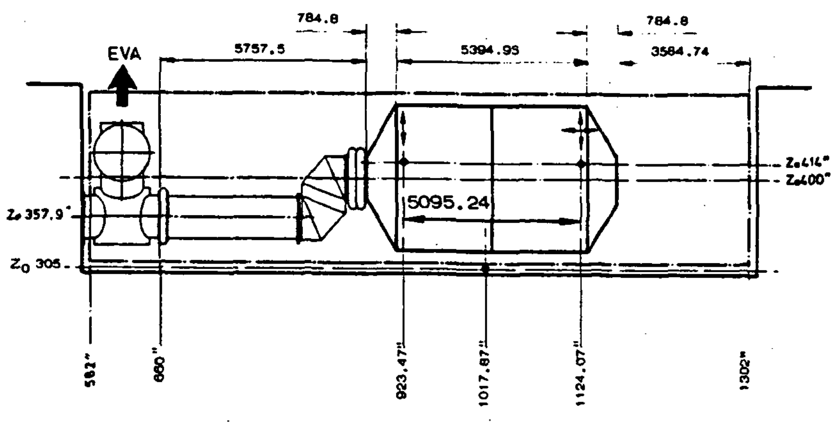


Figure 3 - 2: Long Module

These signs indicate the load carrying direction of the attach fittings

3.1.2 Long Module plus 3 m Pallet Configuration

Figure 3 - 3 depicts the long module/3 m pallet configuration. This configuration provides both pressurized volume for payloads and pallet mounting area for experiments requiring exposure to space environment. Utility services to the pallet are routed from module aft end cone feedthrough plates to the pallet as described in para. 3.5.2.

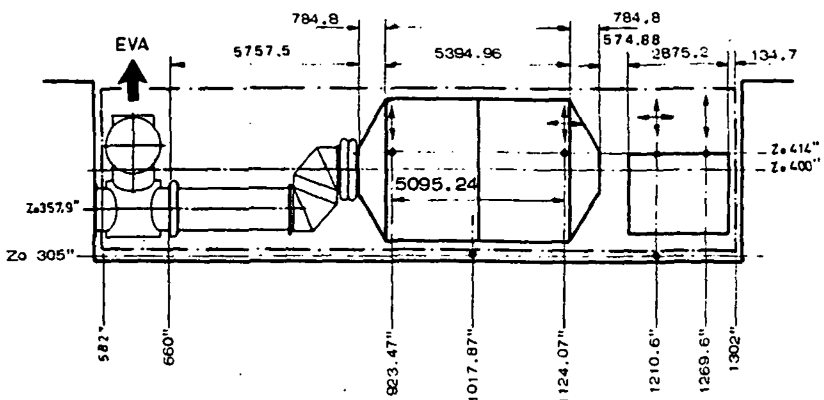


Figure 3 - 3: Long Module Plus 3 m Pallet Configuration

3.1.3 Long Module plus 6 m Pallet Configuration

This configuration (Figure 3 - 4) increases the pallet mounting area by connecting two pallet segments to form a pallet train. Utility routing is the same as for the long module plus 3 m-pallet-configuration.

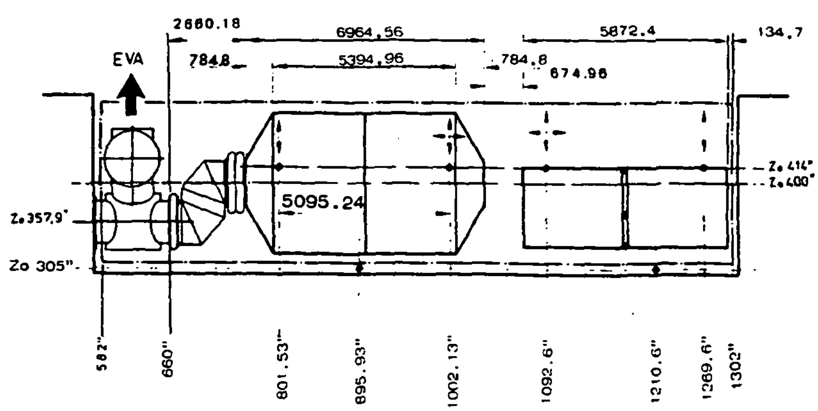


Figure 3 - 4: Long Module Plus 6 m Pallet Configuration

3.1.4 Short Module plus 6 m Pallet Configuration

A short module may be used in place of the long module to provide the configuration shown in Figure 3 - 5.

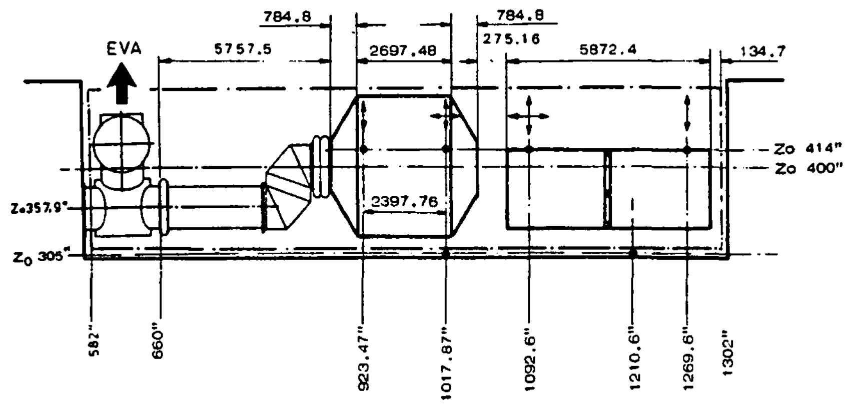


Figure 3 - 5: Short Module Plus 6 m Pallet

3.1.5 Short Module plus 9 m Pallet Configuration (Baseline Design Configuration)

This configuration offers the largest pallet mounting area which may be used in a module/pallet configuration, as shown in Figure 3-6. The three pallet segments are rigidly attached to form a single pallet train.

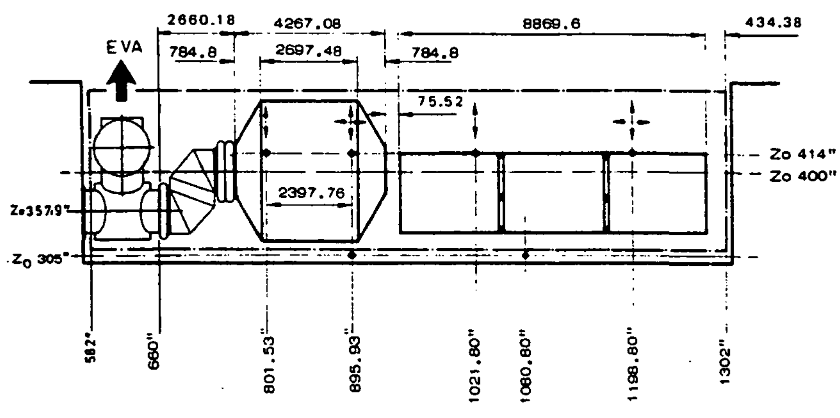


Figure 3 - 6: Short Module Plus 9 m Pallet Configuration

3.1.6 9 Meter Pallet Configuration (Baseline Design Configuration)

As shown in Figure 3 – 7, this configuration consists of three independently suspended pallet segments. The pallet segments are spaced along the length of the cargo bay. Utility routing between pallet segments is described in para. 3.5.3. The "Igloo" at the forward pallet provides a controlled pressurized environment for a set of Spacelab subsystem equipment similar to that located in the core segment of the module (see para. 3.2.2.2). Utility services are routed directly from the Orbiter to the Igloo/first pallet segment. DCN 002

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

* Note: If the Orbiter airlock is installed outside the mid-deck in the cargo bay, the clearance for EVA must be increased from Station 627" to Station 691".

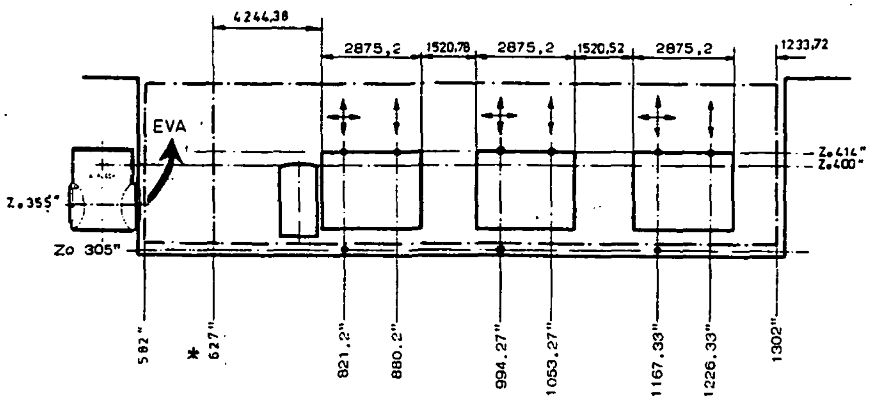


Figure 3 – 7: 9 Meter Pallet Configuration

3.1.7 12 Meter Pallet Configuration

A potentially well suited configuration for a number of astronomy missions is depicted in Figure 3 – 8, consisting of two independently suspended pallet trains composed of two pallet segments each. 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.

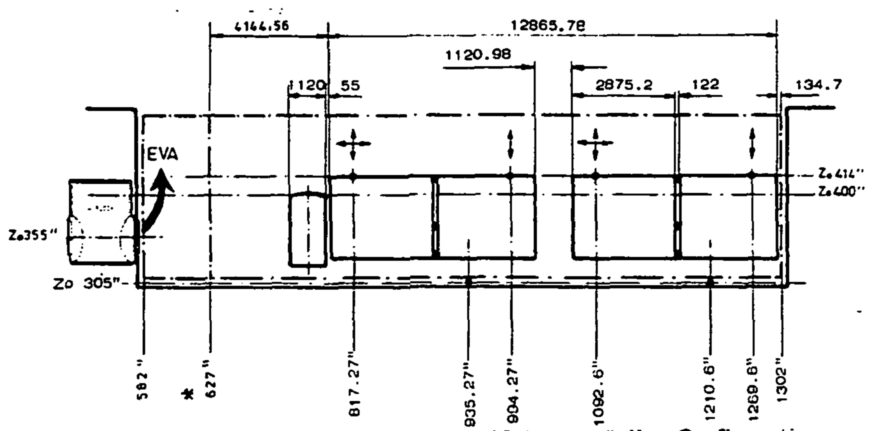


Figure 3 – 8: 12 Meter Pallet Configuration

3.1.8 15 Meter Pallet Configuration (Baseline Design Configuration)

This configuration provides the longest possible experiment platform for Spacelab payloads requiring exposure to the space environment. The configuration shown in Figure 3 - 9 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.

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.

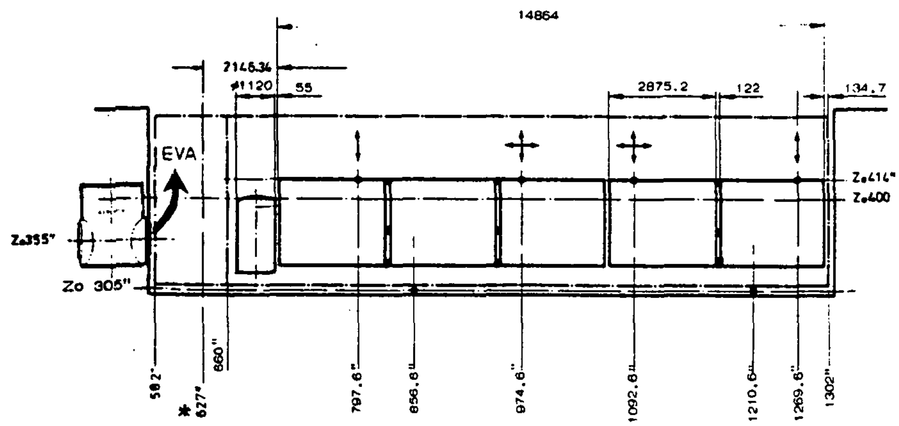


Figure 3 - 9: 15 Meter Pallet Configuration

3.2 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 conditions. Table 3 - 1 shows, for the 8 basic configurations, the center of gravity locations of the basic Spacelab (i.e. without mission dependent equipment and without payload).

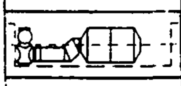
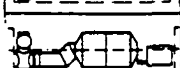
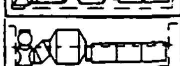
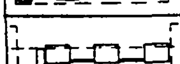
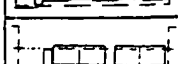
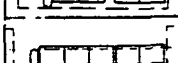
The c.g. locations are superimposed on the allowable c.g. location envelopes imposed by the Orbiter for each of the three ref. axes in Figures 3 - 10, 3 - 11 and 3 - 12. The shaded areas of each envelope indicate the constraints on the launch condition when the overall Orbiter payload mass is greater 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 defined in Figure 2 - 3.

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.

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 eight basic configurations the reference locations of the module and pallet are indicated in Figures 3 - 2 thru -9.

DCN 003

Table 3 - 1: Center of Gravity for Basic Spacelab at Landing Condition

Configuration	Total Mission Independent Mass (kg)	Center of Gravity (m)			Moment of Inertia ($\times 10^3$ kgm^2)		
		X_L	Y_L	Z_L	Ixx	Iyy	Izz
	6768	19.137	-.112	-.510	23.29	98.37	108.03
	7908	20.314	-.100	-.431	29.33	163.48	169.36
	8438	18.247	-.071	-.490	29.95	166.94	172.95
	7478	19.974	-.088	-.590	30.31	186.37	195.24
	8008	18.309	-.065	-.617	28.54	175.50	183.85
	4549	19.816	-.057	-.729	22.88	107.06	116.10
	4834	21.599	-.042	- 1.016	22.13	120.89	131.10
	5434	19.515	-.011	- 1.014	21.37	162.07	173.86

DCN 004

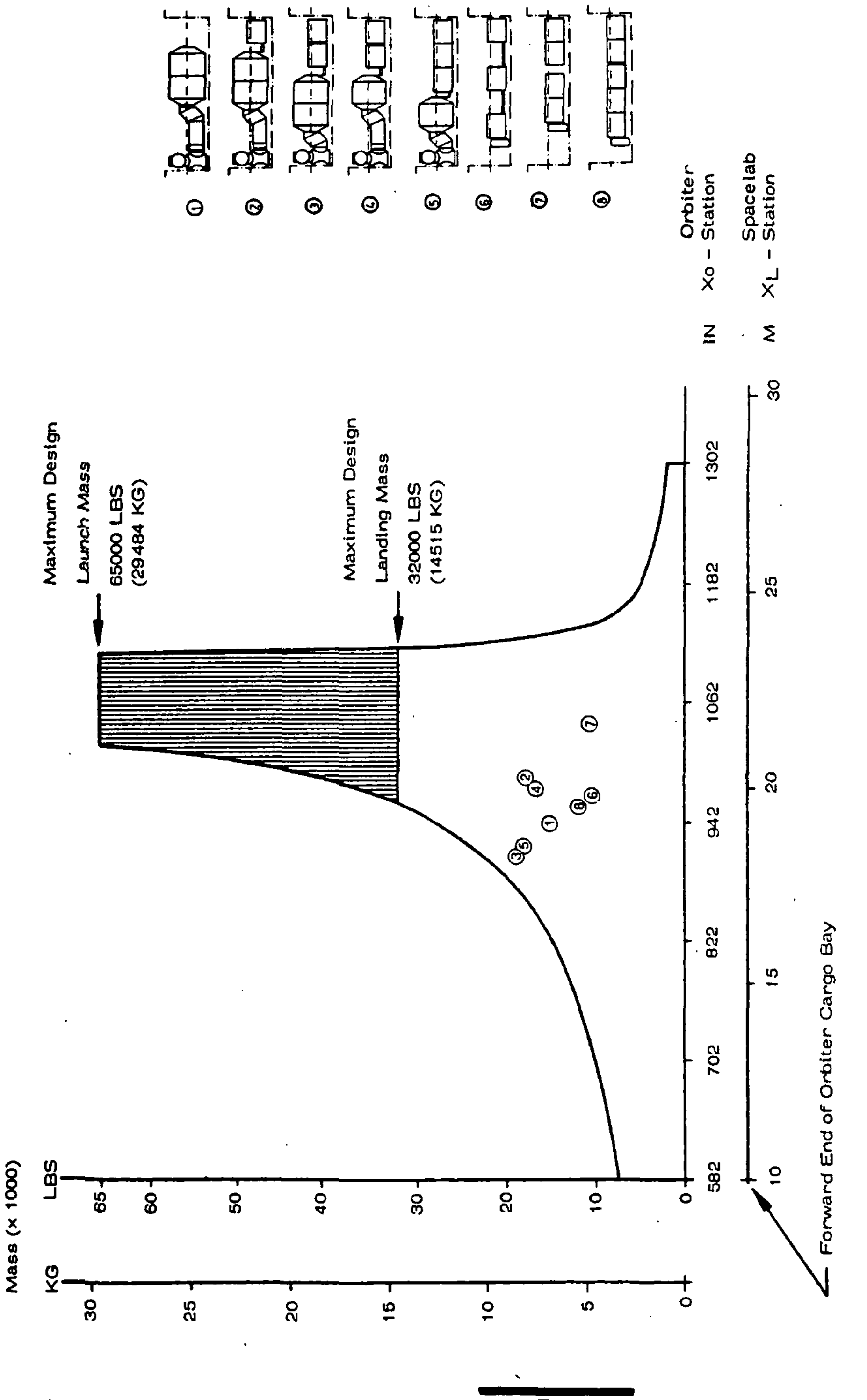


Figure 3 - 10 X-Axis CG Location of Basic Spacelab Configurations

DCN 004

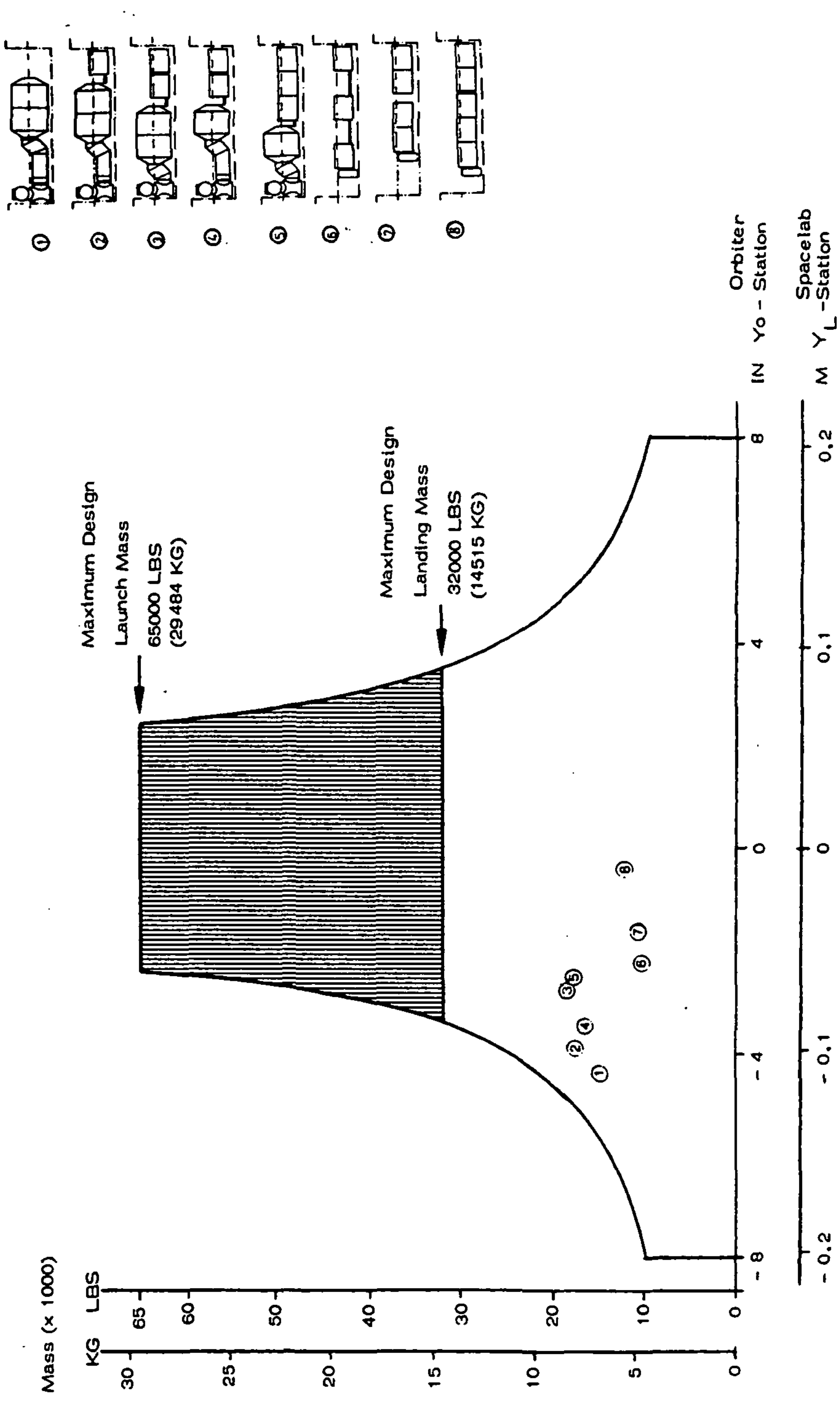


Figure 3 - 11 : Y-Axis CG Location of Basic Spacelab Configurations

DCN
004

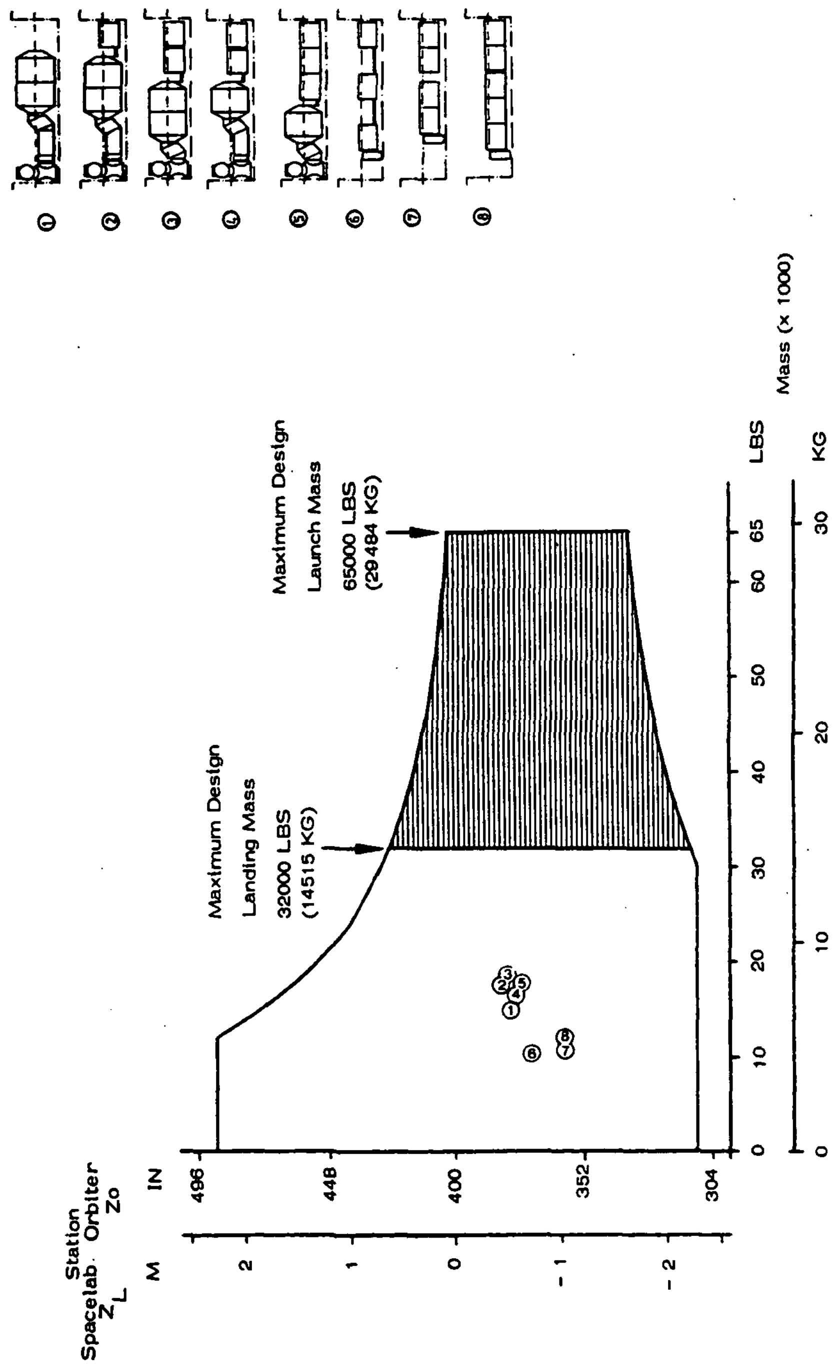


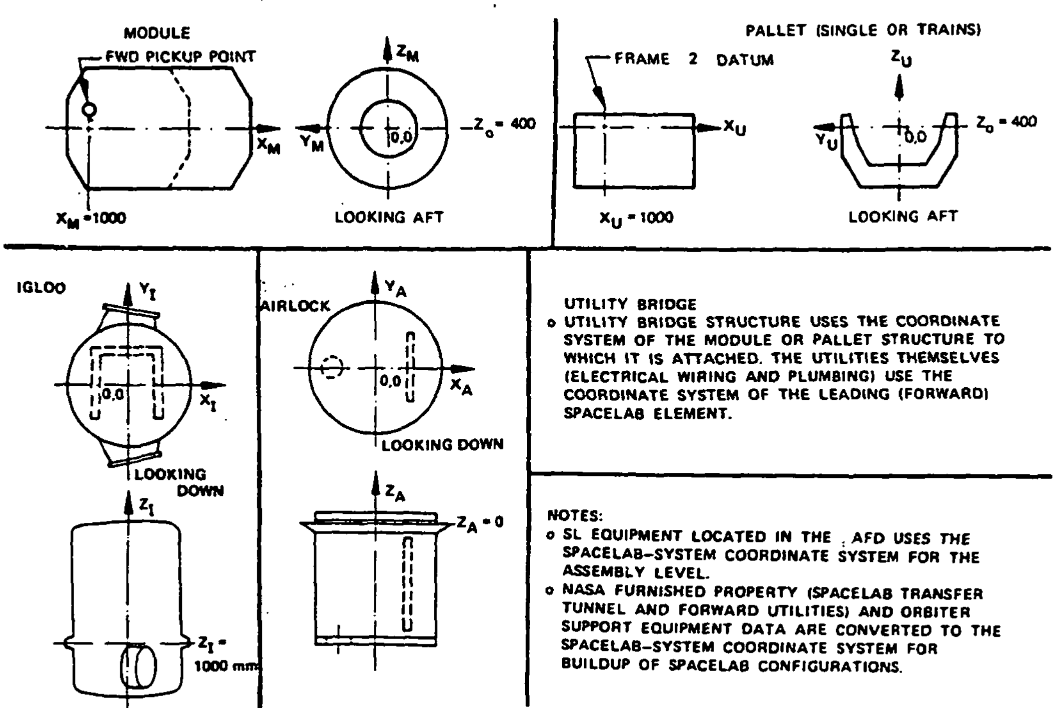
Figure 3-12 : Z-Axis CG Location of Basic Spacelab Configurations

DCN 004

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- 2 shows 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 para. 3.6.2) which are considered as part of the Spacelab payload mass. The mass and c.g. data for these items are described in Section 2 (Table 2 - 17) for the Orbiter related items, and in para. 4.8 for the IPS.

The reference coordinate systems for the individual major Spacelab assemblies (module, pallet and igloo) are shown in Figure 3- 13. DCN 003
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.

*Reference: RP-ER-0005



Transformation of Igloo Coord. to Pallet Coord.

$X_U = X_I - 303.3$

$Y_U = Y_I$

$Z_U = Z_I - 2417.4$

Transformation of A/L Coord. to Module Coord.

$X_M = X_A + 4896.4$

$Y_M = Y_A$

$Z_M = Z_A + 2063.0$

DCN 032

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

Table 3 – 2: Center of Gravity Locations of Major Mission Dependent Equipment (para. 3.4.3)

DCN 002

Note: For masses see Table 3–4 .

DCN 004

ITEM			LOCATION IN SPACELAB	CENTERS OF GRAVITY		
				X (MM)	Y (MM)	Z (MM)
STRUCTURE	PORT	STARBOARD				
SINGLE RACK	NO 5	6	MODULE	3263	± 1361	201
SINGLE RACK	NO 11	12	MODULE	5961	± 1361	201
DOUBLE RACK	NO 3	4	MODULE	2436	± 1336	135
DOUBLE RACK	NO 7	8	MODULE	4070	± 1338	135
DOUBLE RACK	NO 9	10	MODULE	5132	± 1336	135
ATCS						
THERMAL CAPACITOR ASSY			MODULE	405	– 1090	–1090
EXP. HEAT EXCHANGER			MODULE	3090	405	–1770
EPDS						
400 HZ INVERTER			MODULE	1161	– 1445	– 350
400 HZ INVERTER			PALLET	–100	– 1688	– 95
CDMS						
EXPERIMENT COMPUTER			MODULE	1377	1475	–604
EXPERIMENT COMPUTER			IGLOO	–140	0	1988
EXPERIMENT I/O UNIT			MODULE	1170	1445	–168
EXPERIMENT I/O UNIT			IGLOO	140	–25	1273
DATA DISPLAY UNIT AND KEYBOARD			MODULE	•	•	•
HIGH RATE DATA RECORDER			+ AFD	1140	1142	138
HIGH RATE MULTIPLEXER			MODULE	1152	1483	0
CREW HABITABILITY						
OVERHEAD STOWAGE CONTAINER NO 1			MODULE	1097	0	1594
OVERHEAD STOWAGE CONTAINER NO 2			MODULE	1467	0	
OVERHEAD STOWAGE CONTAINER NO 3			MODULE	1837	0	
OVERHEAD STOWAGE CONTAINER NO 4			MODULE	2207	0	
OVERHEAD STOWAGE CONTAINER NO 5			MODULE	2577	0	
OVERHEAD STOWAGE CONTAINER NO 6			MODULE	2947	0	
OVERHEAD STOWAGE CONTAINER NO 7			MODULE	3317	0	
OVERHEAD STOWAGE CONTAINER NO 8			MODULE	3787	0	
OVERHEAD STOWAGE CONTAINER NO 9			MODULE	4157	0	
OVERHEAD STOWAGE CONTAINER NO 10			MODULE	4527	0	
OVERHEAD STOWAGE CONTAINER NO 11			MODULE	4897	0	
OVERHEAD STOWAGE CONTAINER NO 12			MODULE	5267	0	
OVERHEAD STOWAGE CONTAINER NO 13			MODULE	5637	0	
OVERHEAD STOWAGE CONTAINER NO 14			MODULE	6007	0	1594
CPSE						
AIRLOCK			MODULE	4992	0	1757
HIGH QUALITY WINDOW/ VIEWPORT ASSEMBLY			MODULE	2308	0	1994

+ AFT FLIGHT DECK (AFD) LOCATED EQUIPMENT USES SPACELAB SYSTEM COORDINATE SYSTEM (FIGURE 2–3)

• DDU & KEYBOARD CAN BE MOUNTED IN A NUMBER OF DIFFERENT RACKS

3.3 Modular Elements of Spacelab

3.3.1 Module

3.3.1.1 Overall Configuration

The pressurized module consists of a combination of either one or two 4-m-diameter cylindrical segments of 2.7 m length. The module end closures are conical sections of equal 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 closed by a cover plate.

One segment, the Core Segment, and the forward and aft end cones comprise the Short Module. Two segments, Core Segment and Experiment Segment, together with the end cones comprise the Long Module. The module exterior is covered with high-performance insulation. EVA mobility aids are also located at the exterior.

Each segment is equipped with a flange ring of 1.3 m internal diameter on the top to provide accommodation for the following Common Payload Support Equipment (CPSE):

- top airlock (experiment segment only)

or

- optical window/viewport assembly

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

Planned and/or contingency access constraints during ground operations (late access in vertical position) do not allow the use of the top airlock in the CPSE opening of the core segment.

3.3.1.2 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 primarily located forward in the core segment. It is installed in the first double rack on each side and on the subfloor extending the whole length of the core segment.

Experiment equipment can be accommodated in the remaining 60 % of the core segment and in the experiment segment as shown in Figure 3 - 14.

The main floor consists of segments and is designed to carry the racks with their equipment. Racks and floor are interconnected at the integration site. The floor segments allow adaptability of the secondary structure to both module sizes. The main floor itself consists of a load-carrying beam structure and is covered by panels on the main walking surface providing also for noise attenuation from the subfloor area. The floor also contains openings equipped with debris traps to allow cabin air return flow. Except for the center floor panels, all panels are hinged to allow underfloor access in orbit and on the ground, as can be seen in Figure 3 - 14. 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.

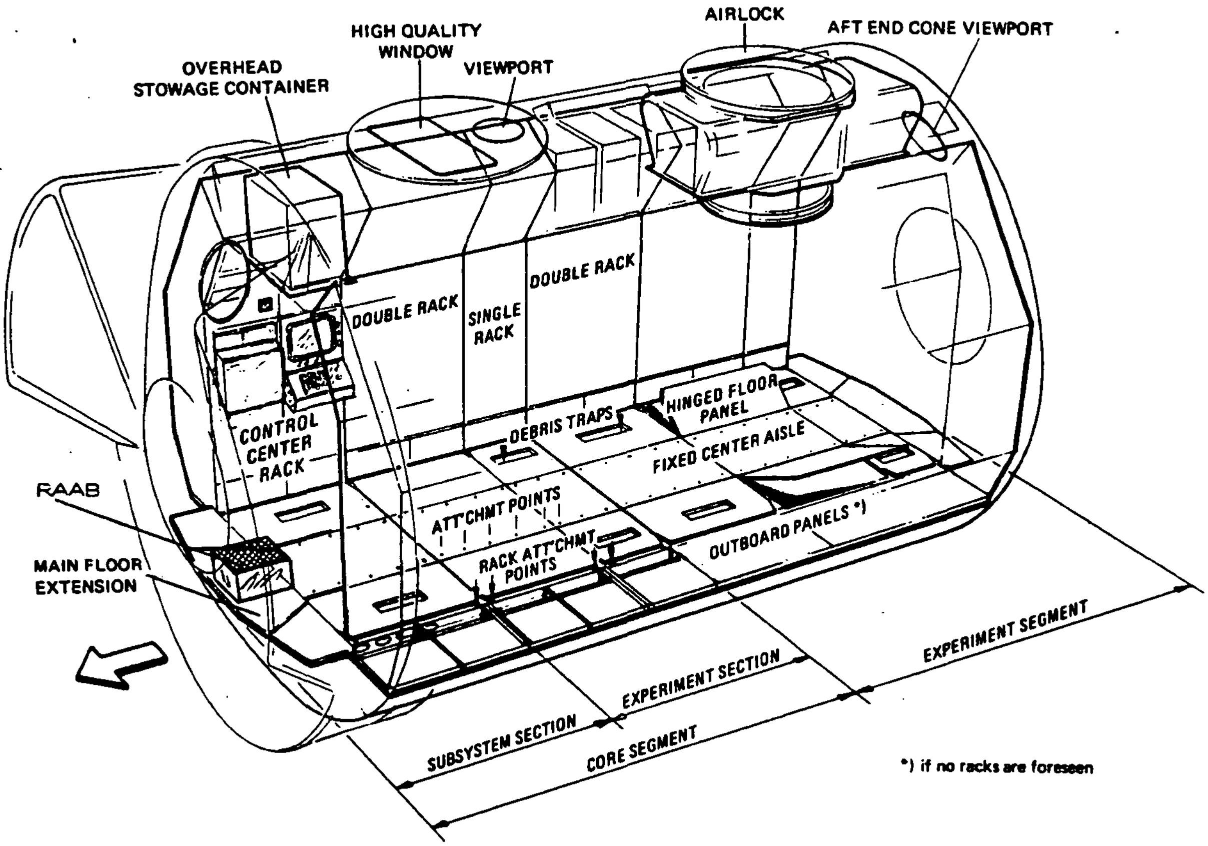


Figure 3 - 14: Internal Accommodation Layout

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 racks in the core segment and four double and two single racks in the experiment segment.

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

There is only a single interface plane between the subsystem equipment remaining inside the module and the experiment train for electrical and avionics cooling-loop connections after roll-in and before roll-out of the experiment train.

Figure 3 - 15 shows a frontal view of the module. The left and right hand sections through the module are shown in Figure 3 - 16, illustrating the subsystem arrangement, the airlock and the rack numbering scheme. The control center and the work bench rack contain subsystem equipment only. The experiment racks are shown with the location of the experiment power switching panels and intercom remote stations. Subsystem equipment in the underfloor space of the core segment is mounted on a 2.7m subfloor attached to the primary module structure.

While an underfloor space for experiment is available in the experiment segment, only attachment points in the primary structure are provided but no subfloor.

Overhead stowage containers marked with asterisk may have to be installed on-orbit to allow for on-ground late access through the core segment CPSE opening (detailed late access provisions are currently under investigation). Access and operational requirements of experiments mounted to the High Quality Window/ Viewport Assembly may necessitate removal of further overhead stowage containers. DCN 002

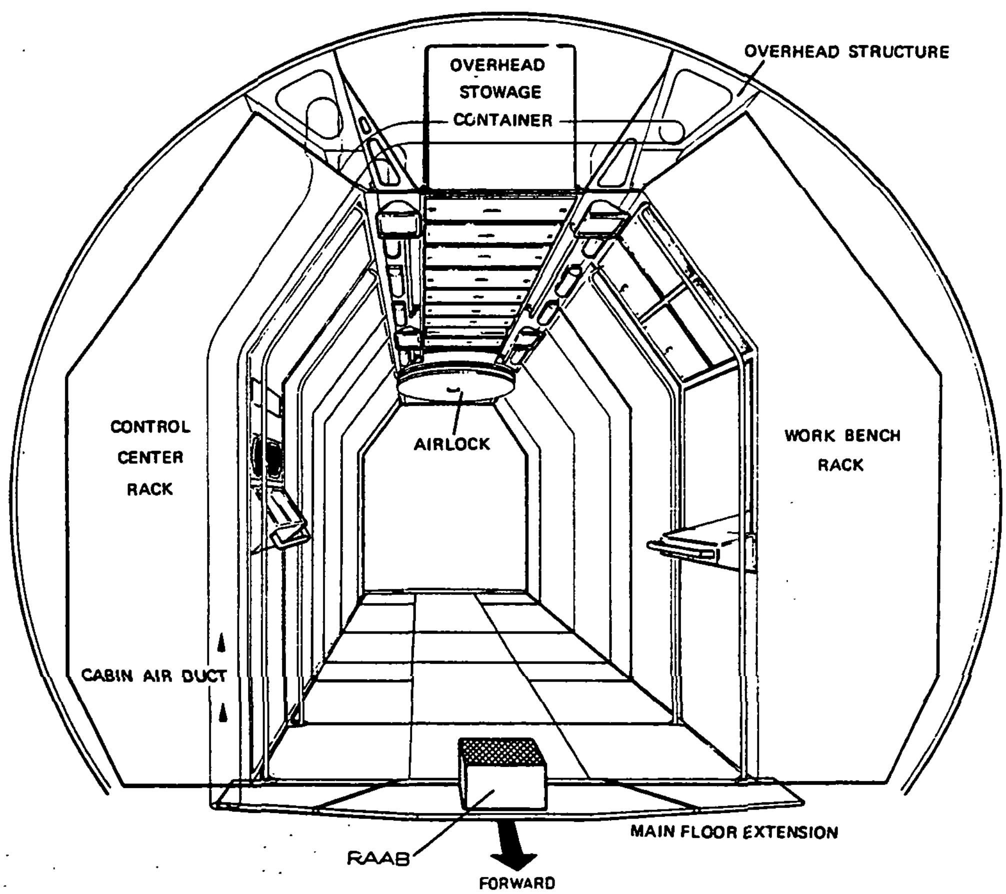
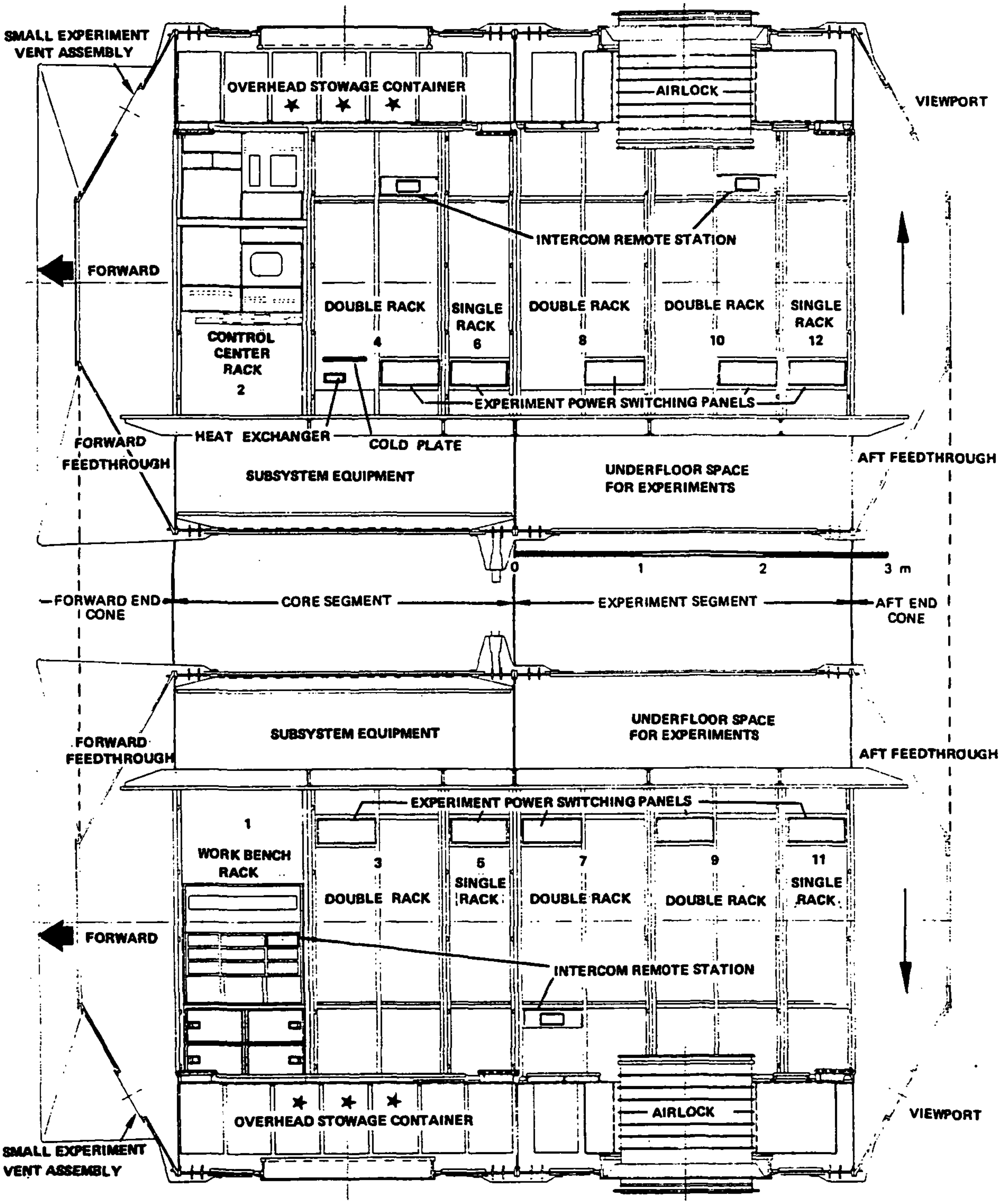


Figure 3 - 15 : Module Frontal View - Endcone Removed

DCN 004



DCN 004

Figure 3 - 16 : Sectional View - Module

3.3.2 Pallet

3.3.2.1 Pallet Segment

The pallet cross-section is U-shaped providing hard points for mounting heavy experiments and a large panel surface area to accommodate lighter payload elements. Pallet segments are of 3 m length and 4 m width and can be flown independently or interconnected. As many as three pallets can be interconnected to form one pallet train supported by one set of attach fittings; whereas pallet-configurations may consist of one to five pallet segments.

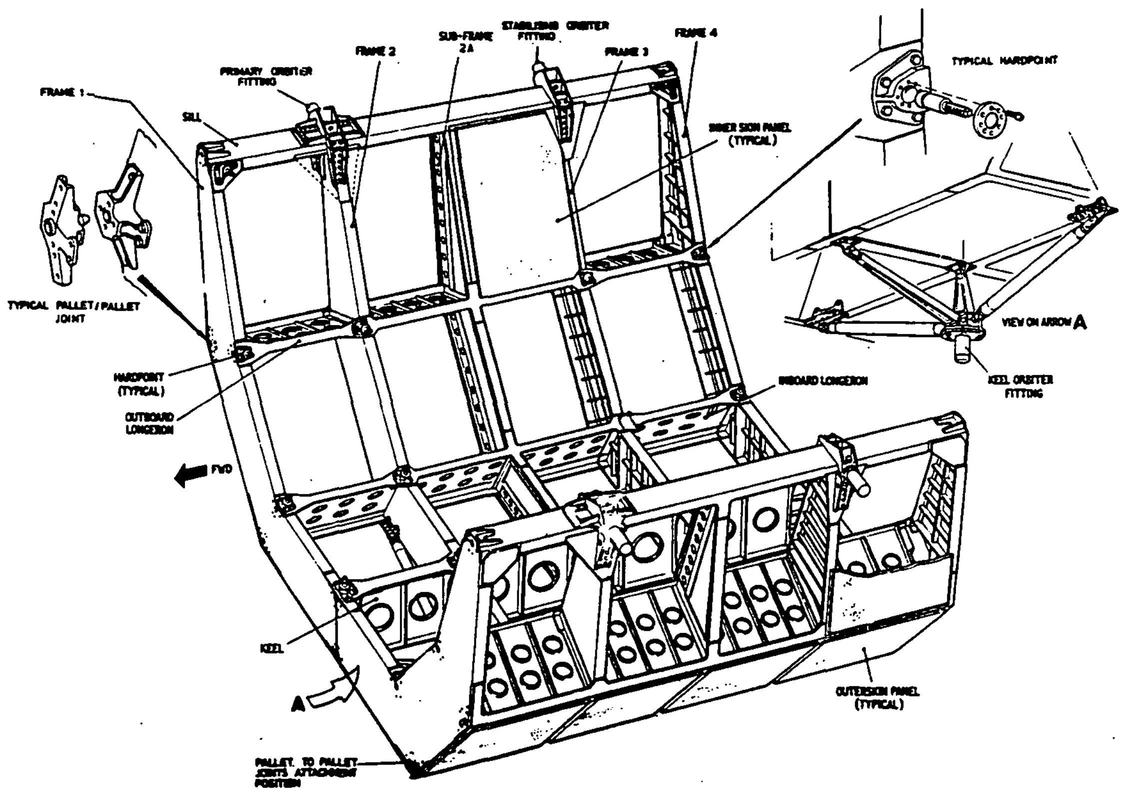


Figure 3-17 : Pallet Segment

DCN 004

Figure 3 - 17 shows a basic pallet segment with hardpoints and typical honeycomb sandwich skin panels with inserts (not shown here). Each segment consists of the basic structure and subsystem equipment which includes:

Mission Independent Equipment:

- subsystem and experiment electrical power busses
- subsystem and experiment data busses
- a subsystem equipment package consisting of:
  - 1 experiment power distribution box
  - 1 subsystem RAU
  - 1 subsystem interconnection station
  - 2 experiment interconnection stations

DCN 001

Mission Dependent Equipment:

- up to 4 experiment RAU's
- cold plates and thermal capacitors
- plumbing

DCN 002 DCN 003

The mission independent subsystem equipment package will be mounted on an experiment cold plate. It has to be pointed out, however, that it might not be required to always fly a complete basic subsystem package on each pallet segment, depending on specific experiment requirements.

While the experiment RAU's can also be mounted to an experiment cold plate, it is the experimenter's/payload integrator's responsibility where to mount the RAU. He is also responsible for the thermal control of the RAU, he has to assure that the environmental limits for the RAU are not exceeded (RAU environmental limits are stated in Section 4.4), and he has to provide the cable harness necessary for control and data transfer via the RAU.

3.3.2.2 Igloo/Pallet Front Frame

In pallet-only configurations, subsystem equipment necessary for the operation of Spacelab is located in the "Igloo" which is mounted to the front frame of the first pallet segment. The Igloo, as shown in Figure 3 - 18, is a pressurized cylinder equipped with a removable bulkhead providing full access to the interior. The weight of an equipped Igloo is about 640 kg, the volume provided for subsystems is 2.2 m^3. Thermal control of subsystem equipment is achieved by cold plates which are connected to the pallet freon cooling loop.

A set of Spacelab subsystem equipment, similar to a set which in module only and module/pallet configuration is integrated within the module, is installed within the Igloo in the pallet-only configuration. The only operator interface for the pallet-only configuration is through the Spacelab equipment in the Orbiter aft flight deck, e.g. DDU, keyboard and control panels. This equipment is installed for all Spacelab configurations.

The following is the list of the equipment (basic and mission dependent) which is located in the Igloo:

- 3 computers (subsystem, experiment and back-up computer)
- 2 I/O units (subsystem and experiment I/O)
- 1 mass memory
- 2 subsystem RAU's
- 1 subsystem interconnecting station
- 1 emergency box
- 1 power control box
- 1 subsystem power distribution box
- 1 remote amplifier and advisory box (RAAB)
- 1 high rate multiplexer (HRM)

In addition to the Igloo the following major subsystem equipment is also mounted to the front frame of the first pallet segment

- 1 subsystem 400 Hz inverter (only in pallet-only configurations)
- 1 experiment 400 Hz inverter (only in pallet-only configurations)
- freon cooling loop components

Thermal control of the 400 Hz inverters is also achieved by cold plates connected to the pallet freon cooling loop.

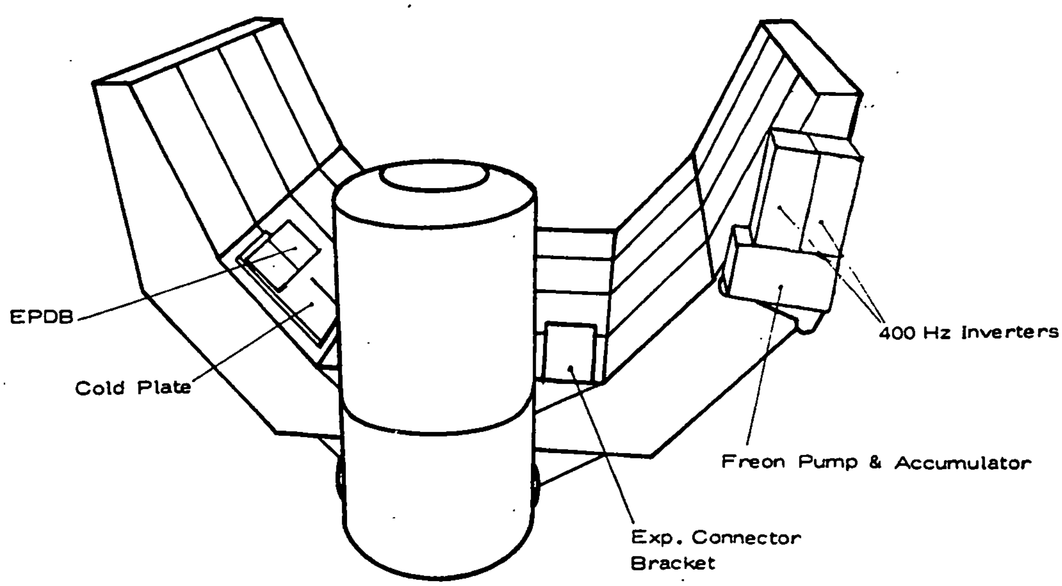


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Figure 3 - 18 : Igloo and Front Frame

3.3.3 Transfer Tunnel

Note: *The transfer tunnel is NASA furnished equipment*

The transfer tunnel enables 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 has an internal clear diameter of nominally 1 m and allows the passage of a crew member with a rectangular package of size 0.56 x 0.56 x 1.27 m or a spherical package of diameter 0.91 m.

There are two different tunnel configurations the selection of which is dependent on the location of the forward end of the module in the cargo bay (see Figures 3-2, 3-3, 3-4, 3-5 and 3-6).
Figure 3-19 shows, in a simplified form, the longer of the two tunnel configurations. This consists of a forward extension which mates with the tunnel adapter, a cylindrical section, a "joggle" section which provides the transition between the tunnel adapter axis and the module axis, and an aft adapter which interfaces with the forward end cone of the module. Additional flexible sections enable load decoupling and allow for deflections. The tunnel is mounted to the Orbiter attach fittings via struts (not shown in Figure 3-19). The cylindrical section is removed to obtain the short tunnel configuration.

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Orbiter supplied air together with a fan inside the tunnel provide atmosphere conditions similar to those in the module. Also located in the tunnel is an atmospheric scrubber which removes contaminants from both the module and tunnel air. Internal lighting and handrails are provided. External handrails facilitate EVA.

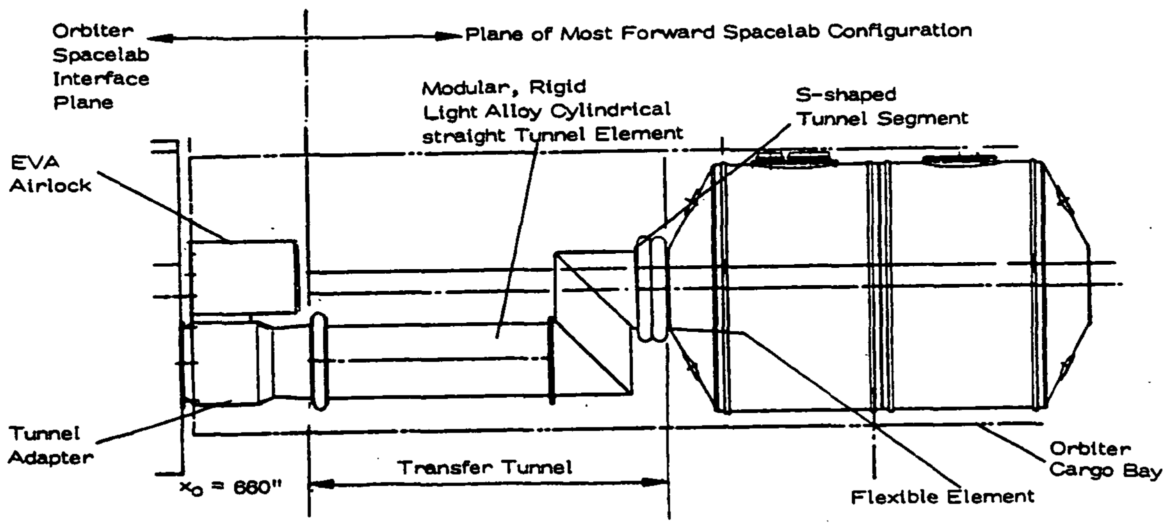

Figure 3 - 19: Transfer Tunnel

3.4 Spacelab Subsystem Equipment

3.4.1 Basic Spacelab Equipment

Table 3 - 3 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 are included in the mass and power budget of the basic Spacelab.

The table indicates the amount of units of equipment to be delivered by the Spacelab Program Baseline and their respective dedications to the modular elements. For a particular configuration selected, summing up the equipment thus dedicated to the modular elements will yield the content of basic equipment of this configuration.

3.4.2 Mission Dependent Equipment

The Spacelab Program provides, in addition to the basic (i.e. mission independent) hardware, a spectrum of mission dependent equipment (Table 3 - 4), which can be selected and flown according to the requirements of a particular mission. However, because of operational or other constraints, some mission dependent equipment may not be removed from Spacelab as indicated in Table 3 - 4.

If particular payload requirements would make it desirable to fly Spacelab without these normally non-removable mission dependent items - which can, of course, be removed on the ground for maintenance and reconfiguration purposes - then it may be possible to do so if the experimenter/payload integrator provides the necessary analysis and possible extra hardware/software items not provided by the Spacelab program.

Table 3 - 4 lists the mission dependent equipment together with the mass and power requirement 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 and functionally accommodated by each Spacelab configuration. The last five columns of Table 3 - 4 indicate how the mission dependent units are typically distributed between the different modular elements of Spacelab and the Orbiter Aft Flight Deck. Again it should be noted that, while for some items, such as the 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.

Power consumption values are quoted only for maximum power conditions. For some items, the actual power may be between these two values depending on the operating mode. More details are provided in the relevant subsystem description sections. DCN 002

Table 3 – 3: Major Basic Spacelab Equipment

MAJOR BASIC SPACELAB EQUIPMENT	NUMBER OF UNITS DE–LIVERED BY S/L PROGR. (BASELINE)	NUMBER OF UNITS FORESEEN PER CORE SEGMENT	EXPERIMENT SEGMENT	PALLET SEGMENT	AFD	IGLOO/ PALLET FRONT FRAME
STRUCTURE:						
PRIMARY STRUCTURE/MODULE:						
2.7 M CYLINDER ASSEMBLY	2	1	1	–	–	–
FORWARD END CONE ASSEMBLY	1	1	–	–	–	–
AFT END CONE ASSEMBLY	1	– (1)	1	–	–	–
1.3 M Ø CLOSE-OUT PLATE	3	1	2	–	–	–
FITTING ASSEMBLY	1	x	x	x	–	–
SECONDARY STRUCTURE/MODULE:						
2.7 M OVERHEAD SUPPORT STRUCTURE	2	1	1	–	–	–
2.7 M FLOOR SUPPORT STRUCTURE	2	1	1	–	–	–
1.1 M FLOOR ASSEMBLY	2	1	1	–	–	–
1.6 M FLOOR ASSEMBLY	2	1	1	–	–	–
FLOOR END EXTENSION	2	1 (2)	1	–	–	–
SUBSYSTEM SUBFLOOR	1	1	–	–	–	–
AFT VIEWPORT	1	– (1)	1	–	–	–
DOUBLE RACK ASSEMBLY FOR SUBSYSTEMS	2	2	–	–	–	–
PALLET STRUCTURE:						
2.9 M PALLET SEGMENTS	5	–	–	1	–	–
INNER–PANELS	120	–	–	24	–	–
OUTER–PANELS	120	–	–	24	–	–
UTILITY ROUTING:						
MODULE UTILITY SUPPORT STRUCTURE	1	– (1)	1	–	–	–
PALLET UTILITY SUPPORT STRUCTURE						
750 mm	1	–	–	1	–	–
650 mm	2	–	–	2	–	–
570 mm	2	–	–	2	–	–
350 mm	1	–	–	1	–	–
160 mm	1	–	–	1	–	–
CREW HABITABILITY:						
TOOLS AND MAINTENANCE SET	1	1	–	–	–	–
CEILING RAILS	12	6	6	–	–	–
CONSOLE VERTICAL RAILS	4	4	–	–	–	–
CONSOLE HORIZONTAL RAILS	2	2	–	–	–	–
RACK FOOT RESTRAINTS	3	2	1	–	–	–
WORK BENCH	1	1	–	–	–	–
WORK BENCH CONTAINERS	4	4	–	–	–	–
EVA MOBILITY AID	18	10	8	6	–	–
PORTABLE FIRE EXTINGUISHER	2	1	1	–	–	–
PORTABLE O_2 SYSTEM	4	2	2	–	–	–
EPDS						
EPDS MONITOR AND CONTROL PANEL	1	1	–	–	–	–
POWER CONTROL BOX	2	1	–	–	–	–
EMERGENCY BOX	2	1	–	–	–	–
SUBSYSTEM POWER DISTRIBUTION BOX	2	1	–	–	–	1
AFD POWER DISTRIBUTION BOX	1	–	–	–	1	–
EXPERIMENT POWER DISTRIBUTION BOX	6	1	2	1	–	–
400 HZ SUBSYSTEM INVERTER	2 *	1	–	–	–	1
HARNESS (SIGNAL/POWER)	x	x	x	x	x	x
LIGHTS	15	9	6	–	–	–

( ) NUMBERS IN PARANTHESIS APPLY FOR SHORT MODULE ONLY
x HARDWARE NOT EASILY QUOTABLE IN DISCRETE UNITS
* ADDITIONAL UNITS LISTED AS MISSION DEPENDENT EQUIPMENT (TABLE 3–4)

Table 3 – 3 (cont'd): Major Basic Spacelab Equipment

MAJOR BASIC SPACELAB EQUIPMENT	NUMBER OF UNITS DE-LIVERED BY S/L PROGR. (BASELINE)	NUMBER OF UNITS FORESEEN PER				
		CORE SEGMENT	EXPERIMENT SEGMENT	PALLET SEGMENT	AFD	IGLOO/ PALLET FRONT FRAME
CDMS						
KEYBOARD	2 *	1	–	–	1	–
DATA DISPLAY UNIT	2 *	1	–	–	1	–
SUBSYSTEM COMPUTER	2	1	–	–	–	1
BACK-UP COMPUTER	2	1	–	–	–	1
SUBSYSTEM I/O UNIT	2	1	–	–	–	1
MASS MEMORY	2	1	–	–	–	1
EXPERIMENT RAU INTERCONN'STATION	9	2	7	1	–	–
SUBSYSTEM RAU	12	4	1	2	1	2
INTERCOM MASTER STATION	1	1	–	–	–	–
INTERCOM REMOTE STATION	1 *	1	1	–	–	–
INT'GR MONITOR & CTRL. PANEL (R7)	–	–	–	–	1	–
C & W FIRE SUPPRESSION PANEL	1	1	–	–	–	–
ECS						
ATMOSPHERE STORAGE AND CONTROL SYSTEM (ASCS):						
O_2/N_2 CONTROL PANEL	1	1	–	–	–	–
N_2 TANK ASSEMBLY	1	1	–	–	–	–
OXYGEN SUPPLY ASSEMBLY	1	1	–	–	–	–
NITROGEN SUPPLY ASSEMBLY	1	1	–	–	–	–
ATM.PRESS. CONTROL ASSEMBLY	1	1	–	–	–	–
ATM SUPPLY CONTROL ASSEMBLY	1	1	–	–	–	–
AIRLOCK SUPPLY ASSEMBLY	1	1	–	–	–	–
CABIN PRESSURE RELIEF ASSEMBLY	1	1	–	–	–	–
EXPERIMENT VENT ASSEMBLY	1	1	–	–	–	–
ATMOSPHERE REVITALIZATION SYSTEM (ARS):						
ECS MONITOR AND CONTROL PANEL	1	1	–	–	–	–
CABIN FAN ASSEMBLY	1	1	–	–	–	–
CABIN LOOP DISTRIBUTION SYSTEM	x	x	x	–	–	–
CONDENSING HEAT EXCHANGER	1	1	–	–	–	–
CO_2 AND TEMP. CONTROL ASSEMBLY	1	1	–	–	–	–
WATER SEPARATOR ASSEMBLY	1	1	–	–	–	–
OVERBOARD DUMPING ASSEMBLY	1	1	–	–	–	–
AVIONICS FAN ASSEMBLY	1	1	–	–	–	–
AVIONICS LOOP DISTRIBUTION SYSTEM	x	x	x	–	–	–
RACK COOLING ASSEMBLY	2 *	2	–	–	–	–
AVIONICS HEAT EXCHANGER	1	1	–	–	–	–
CABIN SENSOR ASSEMBLY (TEMP./PRESS)	1	1	–	–	–	–
ACTIVE THERMAL CONTROL:						
WATER LOOP PLUMBING	1	1	–	–	–	–
WATER PUMP PACKAGE	1	1	–	–	–	–
SUBSYSTEM COLDPLATES	20	9	–	–	–	11
FREON LOOP PLUMBING	x	–	–	x	–	x
FREON PUMP PACKAGE	1	1	–	–	–	1
INTERLOOP HEAT EXCHANGER	1	1	–	–	–	–
PASSIVE THERMAL CONTROL:						
HIGH PERFORMANCE INSULATION	x	x	x	–	–	–
THERMAL COATING	x	x	x	x	–	x
FIRE PROTECTION SYSTEM:						
FIRE SUPPRESSION ASSEMBLY	5 *	4	1	–	–	–
PYRO CONTROL BOX	1	1	–	–	–	–
FIRE/SMOKE DETECTION ASSY (REDUNDANT)	3	3	–	–	–	–

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Table 3 – 4: Mission Dependent Spacelab Subsystem Equipment

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MAJOR MISSION DEPENDENT SPACELAB SUBSYSTEM EQUIPMENT		UNIT MASS (KG)	UNIT POWER CONSUMPTION (W) STANDBY	MAX CONT	UNIT SIZE (MM)	NUMBER OF UNITS DE-LIVERED BY S/L PROGR. (BASELINE)	NUMBER OF UNITS FORESEEN PER: CORE SEGMENT	EXPERIMENT SEGMENT	PALLET SEGMENT	AFD	IGLOO/ PALLET FRONT FRAME
SINGLE RACK		41.5	–	–	2753 x 564 x 760	4	2	2	–	–	–
COOLING ASSEMBLY		4.6	–	–	–	4	2	2	–	–	–
DOUBLE RACK		59.2	–	–	2753 x 1052 x 760	6	2	4	–	–	–
COOLING ASSEMBLY		9.2	–	–	–	6	2	4	–	–	–
PALLET HARDPOINTS		1.4	–	–	TBD x 105 x 65	120	–	–	24	–	–
EXP. HEAT EXCHANGER		9.7	–	–	380 x 200 x 90	1	1	–	–	–	–
COLDPLATE (PALLET)		6.2	–	–	750 x 500 x 6	8	–	–	8	–	–
COLDPLATE (MODULE)		14.4	–	–	500 x 389 x 4.4	1	1	–	–	–	–
COLDPLATE SUPPORT STRUCTURE		11.6	–	–	TBD	8	–	–	8	–	–
WATER PLUMBING		x				x	x	–	–	–	–
THERMAL CAPACITOR		14.8	–	–	750 x 500 x 28	4	4 +	–	4	–	–
THERMAL CAPACITOR ASSY		70.7	–	–	750 x 500 x 125	1	1	–	–	–	–
FREON LINE SYSTEM		x	–	–	–	x	–	–	x	–	–
PALLET THERMAL COVER SET		8.0	–	–	TBD	5	–	–	1	–	–
FIRE SUPPRESSION ASSEMBLY		4.5	–	–	–	10	4	6	–	–	–
400 HZ INVERTER	NR	34.1	3.0 (OFF)	160 +.12P	446 x 165 x 540	1 *	–	–	1	–	1
EXP. SWITCH PANEL		4.5	–	–	483 x 177 x 207	10	4	6	–	–	–
POWER HARNESS		x	–	.05 P	–	x	–	–	–	–	–
SIGNAL HARNESS		x	–	–	–	x	–	–	–	–	–
EXP. COMPUTER	NR	30.9	385	385	1 ATR LONG	1	1	–	–	–	1
EXP. I/O UNIT	NR	30.33	140	140	193.5 x 391 x 550	1	1	–	–	–	1
EXP. RAU		8.8	20.3	31.8 S	180 x 170 x 408	8	4	6	4	–	–
KEYBOARD		32.82	45.6	186	182 x 442 x TBD	1	1	1	–	–	–
DATA DISPLAY UNIT					314 x 442 x 520	1 *	1	1	–	–	–
HIGH DATA RATE RECORDER (HDRR)		49.1	19	138.1	537 x 442 x 152.6	1	1	–	–	–	–
HIGH RATE MULTIPLEXER (HRM)		22.9	112	112	400 x 498 x 192	1	1	–	–	–	1
INTERCOM REMOTE STATION		0.9	–	–	133 x 240 x 139	3 *	1	2	–	–	–
CCTV MONITOR (NFE)		10.0	TBD	50	254 x 178 x 318	1	1	1	–	1	–
OVERHEAD STORAGE CONTAINERS		17.6	–	–	565 x 560 x 326	8	4	7	–	–	–
FILM STORAGE KITS IN OVERHEAD STORAGE CONTAINER		5.8	–	–	–	4	4	4	–	–	–
RACK STORAGE CONTAINER		9.8	–	–	438 x 513 x 323	4	4	4	–	–	–
FILM STORAGE KIT IN RACK CONTAINER		3.1	–	–	–	4	4	4	–	–	–
RAILS CONSOLE (VERTICAL)		1.2	–	–	CROSS SECTION 19 x 33.5	10	4	6	–	–	–
RACK FOOT RESTRAINT		3.4	–	–	33.5 x TBD x 1000	4	4	4	–	–	–
AIRLOCK		140.2	0	238	1000 Ø x 1000	1	–	1	–	–	–
VIEWPORT ASSEMBLY		36.1	0	20.0	1300 Ø	1	1	1	–	–	–
HIGH QUALITY WINDOW/ VIEWPORT ASSEMBLY (NFE)		120.0	0	250	1300 Ø	1	1	1	–	–	–
IPS		750.	**	**		1	–	–	1	–	–

x MASS AND QUANTITY CONFIGURATION DEPENDENT
NR THIS EQUIPMENT CANNOT BE REMOVED
* ADDITIONAL UNITS LISTED AS BASIC SPACELAB EQUIPMENT (TABLE 3–3)
** DEPENDING ON MISSION REQUIREMENTS
S SCANNING MODE

P POWER OUTPUT
\+ MODULE–ONLY
NFE NASA FURNISHED EQUIPMENT

THE MASS VALUE QUOTED FOR EACH ITEM IS THE ACTUAL HARDWARE MASS OF THE ITEM PLUS/MINUS THE INSTALLATION IMPACT MASS (E.G. FOR THE AIRLOCK, THE MASS QUOTED IS THE MASS OF THE AIRLOCK HARDWARE MINUS THE MASS OF THE COVER PLATE WHICH IS USED WHEN THE AIRLOCK IS NOT FLOWN)

3.5 Utility Routing

The routing of Spacelab utilities (signal, power and fluid lines), the main utility interfaces and the provisions for additional experiment provided utilities are described in the following sections.

3.5.1 Basic Spacelab Utility Routing

Basic Spacelab utility routing and interfaces are as follows:

a) Module Configurations (Module-Only and Module-Pallet)

All electrical utilities are routed from the Orbiter cargo bay cable trays to two feedthrough plates in the forward end cone of the module. From there they are distributed inside the module and routed to feedthrough plates in the aft end cone, from where they are routed to the pallet via a utility support structure which is mounted to the front frame of the first pallet segment, and distributed further on the pallet,

Water lines for the Spacelab environmental control system are routed from the Orbiter fluid line support to thermal control components mounted to the forward end cone. In module-pallet configurations, fluid lines to the pallet are routed outside of the module from the forward end cone to thermal control components on the front frame of the first pallet segment, via the same utility support structure.

Orbiter provided oxygen is routed from the Orbiter gas line support to a feedthrough connector in one of the forward end cone feedthrough plates (nitrogen is provided from N_2 tanks mounted to the module).

b) Pallet-Only Configurations

Electrical utilities are routed from the Orbiter cargo bay cable trays to Igloo interface connectors and electrical interface brackets mounted to the front frame of the first pallet segment.

Freon lines are routed from the Orbiter fluid line support in the cargo bay to thermal control components mounted to the front frame of the first pallet segment.

Utilities between independently suspended pallet segments (9 m pallet configuration) and independently suspended pallet trains (12 m pallet configuration) are routed via pallet utility support structures, mounted to the opposing pallet segments.

3.5.2 Routing Provisions for Experiment Provided Utilities

3.5.2.1 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, experiment dedicated lines are provided to allow the connection of experiment equipment in the module or on the pallet with experiment equipment in the AFD.

These experiment dedicated lines are

27 TP (twisted pair)	for switching < 1 Amp
54 TSP (twisted shielded pair)	for low level medium frequency signals
7 TSP	for high frequency signals
5 coax cables 75	for high frequency signals

The routing is

- for Module configurations

DCN 001 from AFD Payload Station distribution panel (see Fig 2 - 21) to Module CB 5 (see Fig. 3-20 a)

- for pallet-only configurations

  from AFD Payload Station distribution panel to pallet front frame CB 57 (see Fig . 3-23)

Further routing to experiment equipment has to be done by experiment provided cabling, making use of the Spacelab provisions for experiment utilities (see 3.5.2.2 and 3.5.2.4)

3.5.2.2 Utility Routing Inside the Module

Under the Module mainfloor envelopes are reserved for the routing of experiment provided utilities such as signal lines, dedicated power lines and liquid lines. Figure 3-20 showing the provisions for experiment utilities gives the location of these reserved envelopes.

Using these provisions experiments can interface with

- experiment equipment located in experiment racks, via the experiment dedicated connector brackets at the bottom of the racks (see Figure 4.1-9)
- experiment equipment located on the center aisle, via experiment dedicated connector brackets in the center aisle (see Figure 4.1-5)
- experiment equipment located in the Airlock, via an experiment rack and the connector bracket CBS 25 at the overhead structure (see 4.7)
- experiment equipment located on the pallet, via the aft end feed through and further using provisions for utility routing module - pallet (see 3.5.2.3) and utility routing on the pallet (see 3.5.2.4)
- experiment equipment located in the AFD, via the connector bracket CB 5 (see 3.5.2.1)
- the High Rate Multiplexer at the connector bracket CB 5
- the experiment heat exchanger via the experiment dedicated connector bracket at the bottom of rack No. 4

The cabling or plumbing and any necessary support provisions such as cable trays, clamps and fasteners have to be provided by the experiment/payload integrator

The transverse beams of the module main floor provide a standard hole pattern and attachment points for cable trays, clamps, etc. (see Figure 3-20).

In addition to the mainfloor provisions it is shown that experiment cabling can be routed directly from rack to rack through cut-outs in the lower part of the rack (see Figure 4.1 - 9). Further cut-outs in the upper part of the rack, increase the flexibility for rack to rack cable routing (see Figure 4.1 - 9). Cables routed there are subject to fire and smoke detection via the avionics air loop. This might result in some constraints because it is not allowed to route a "hot" cable through a rack with the air flow shut off and thus with fire and smoke detection not operative.

Installation of user provided cable harness under the mainfloor will generally be carried out during Level III integration (see para 6.4). Rack cabling and mainfloor cabling will be mated after the racks have been mounted to the mainfloor. At this point the direct rack to rack cabling can also be mated/installed.

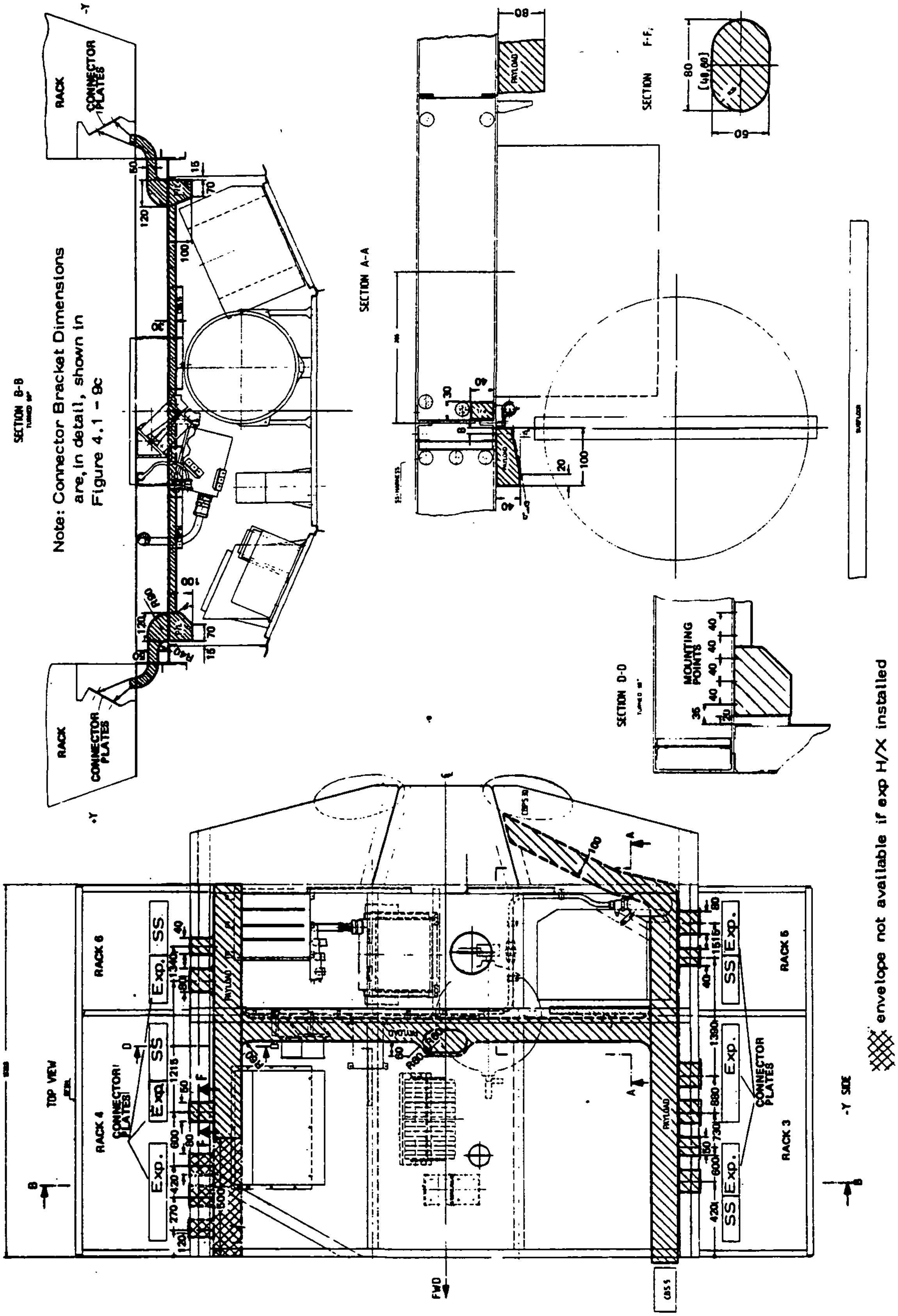

Figure 3 - 20 a: Provisions for Experiment Utility Routing in Core Segment

Editorial Note: Connector Bracket Pin Allocation For SS/Exp Under Revision.

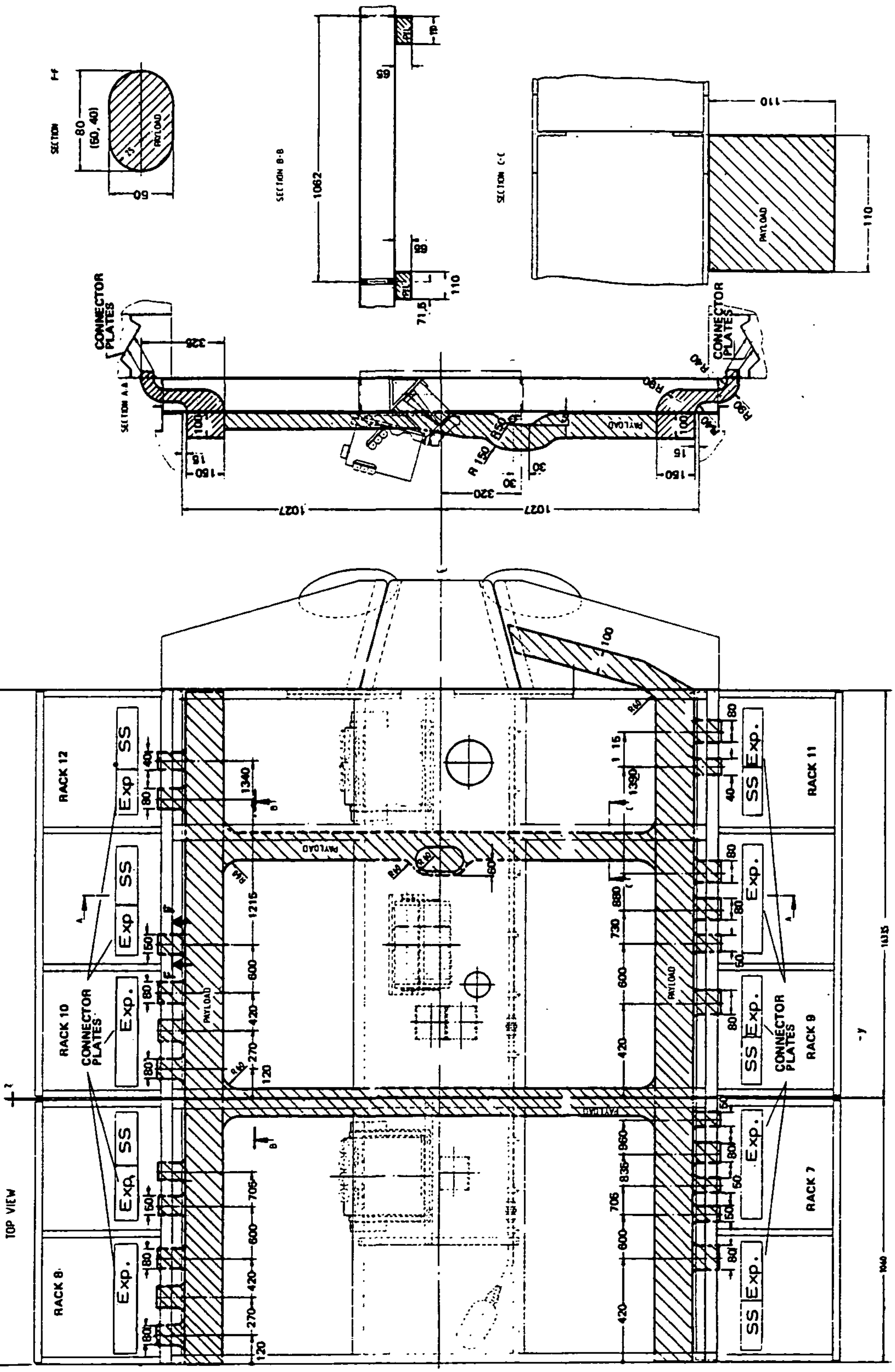

Figure 3 - 20 b: Provisions for Experiment Utility Routing In Experiment Segment

3.5.2.3 Module to Pallet Utility Routing

Spacelab subsystem and additional experiment provided utility lines (electrical and fluid lines) are routed from the feedthrough connectors in aft end cone feedthrough plates to the pallet via a module utility support structure mounted to the aft end cone and a pallet utility support structure mounted to the front frame of the first pallet segment as shown in Figure 3 - 21: Two feedthrough plates are provided in the aft end cone, one for Spacelab subsystem utilities and one for experiment provided utilities.

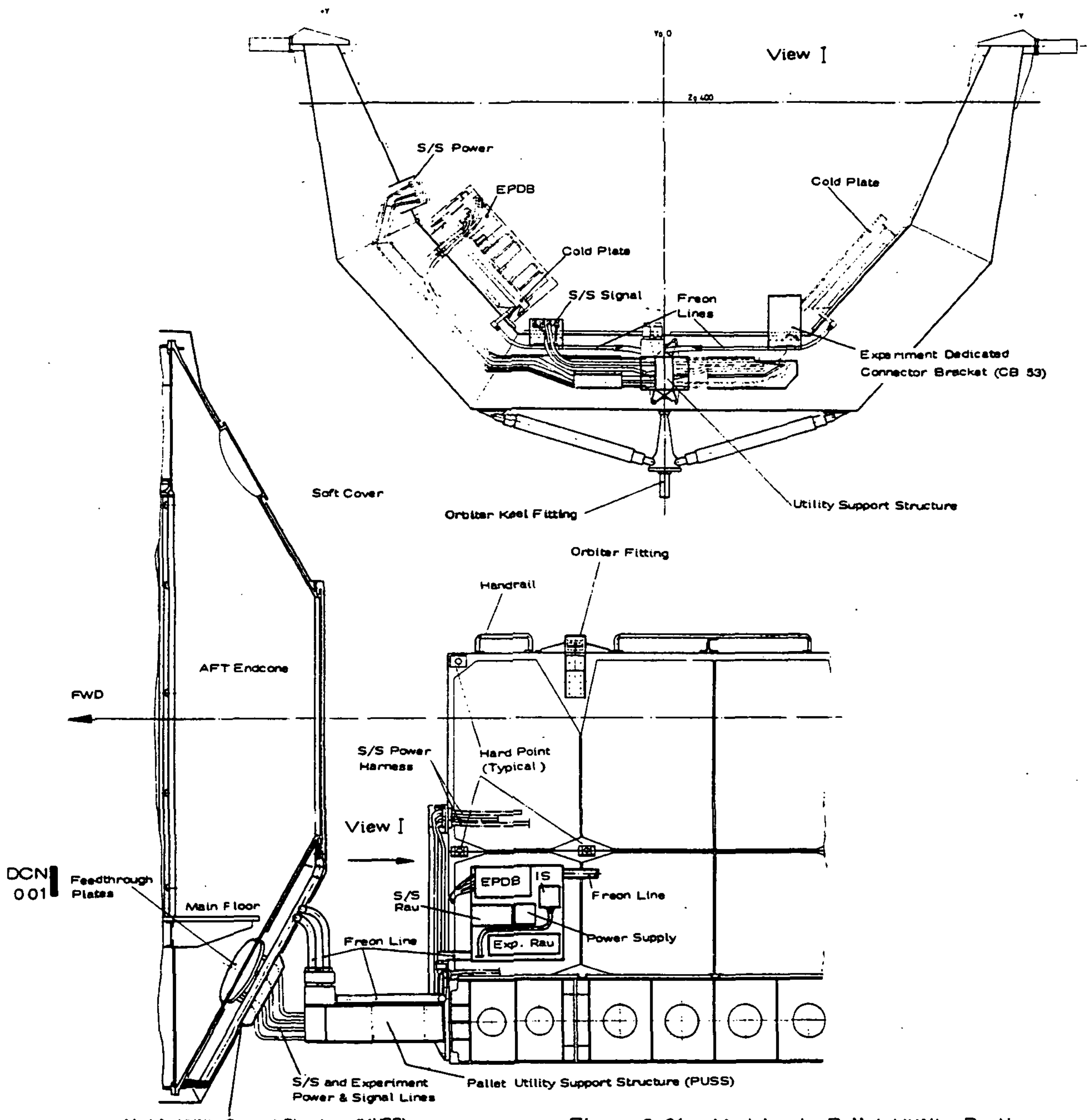

Figure 3-21: Module -to-Pallet Utility Routing

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 provided by the experiment/payload integrator. The utility support structure is designed to accommodate the lines listed above in addition to the permanently installed subsystem lines.

A section through the pallet utility support structure is shown in Figure 3 - 22 , indicating the separate routing of subsystem and experiment lines and of power and signal lines in two parallel ducts to minimize electromagnetic interference. Fluid lines will be routed on top of the structure. The pallet utility support structures are provided in different lengths to fit the various Spacelab configurations.

On the pallet, Spacelab will offer a blank interface connector on the front frame of the first pallet segment for experiment provided utilities.

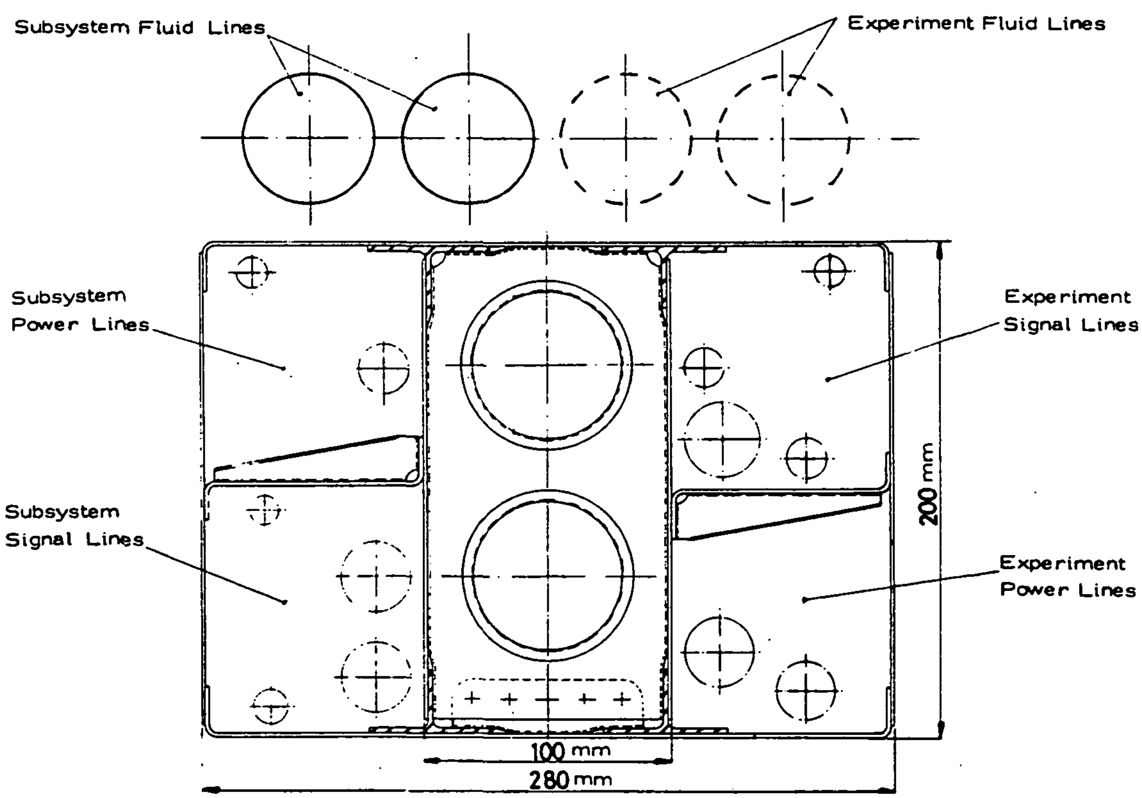

Figure 3 - 22 : Pallet Utility Support Structure

3.5.2.4 Utility Routing on the Pallet

Routing of experiment provided utilities on the pallet is the responsibility of the experiment/payload integrator. Figure 3- 23 identifies possible experiment routing paths. From the interface connector bracket on the front frame of the first pallet segment (CB 57) and from the Igloo feedthrough (CB 42) shown in Figure 3 - 25, utilities can be distributed by using the pallet panel inserts for mounting cable trays, clamps or any other cable harness and fluid line supports. The panel inserts can also be used to mount additional interface brackets at both ends of a pallet segment.

For closely spaced pallet segments and on pallet trains, utilities can be routed directly across pallet segments. For independently suspended, widely spaced pallet segments or pallet trains, the experiment provided utilities will also be routed across the pallet utility support structures (see Figure 3 - 24) by using experiment dedicated connector brackets (CB 56 and 53). These are mounted to the front or end frame, respectively, of opposing pallet segments. Both, CB 56 and CB 53 are provided as blank plates.

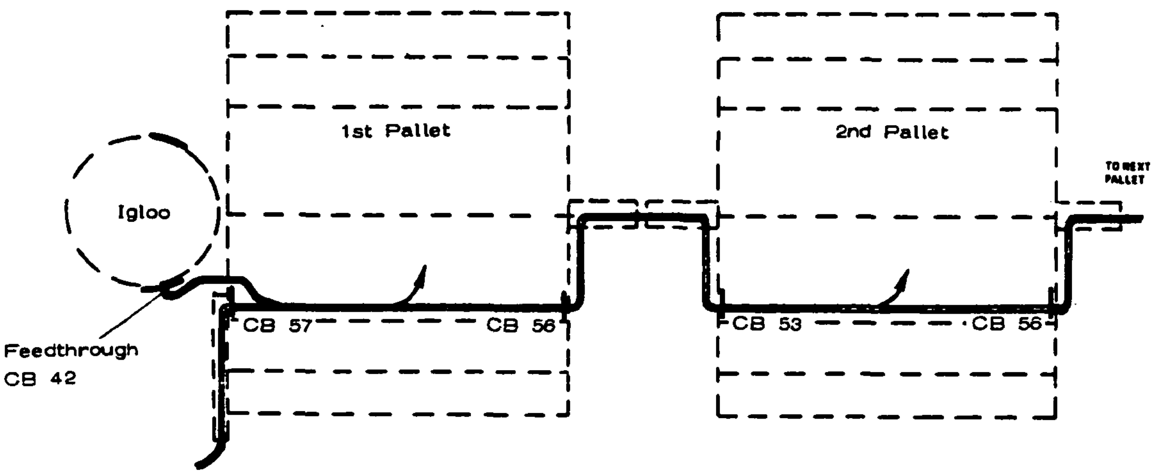

Figure 3 - 23: Provisions for Experimenter Cabling

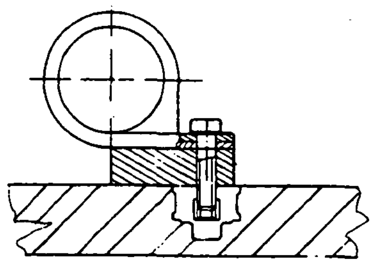

Figure 3 - 23 a : Example for Experiment Cable Fastening on Pallet Panel Inserts

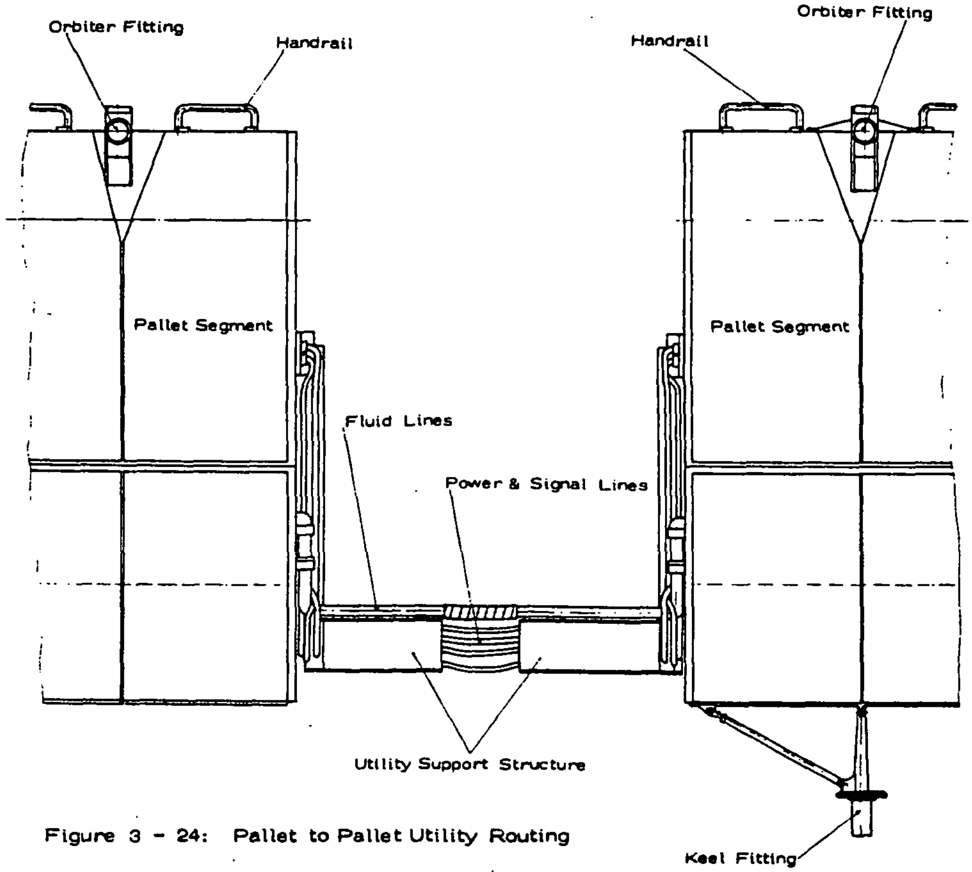

Figure 3 - 24: Pallet to Pallet Utility Routing

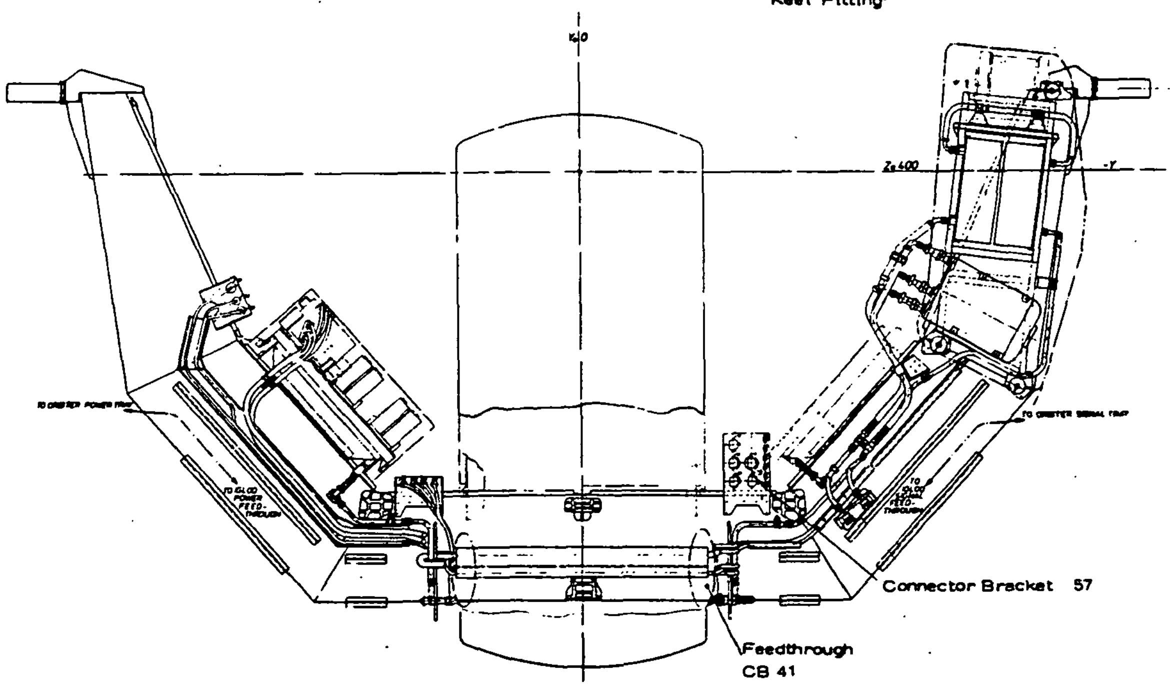

Figure 3 - 25: Utility Routing on Front Frame (Pallet-Only Mode)

3.6 Payload Accommodation Capabilities

3.6.1 Volume and Mounting Area Available to Spacelab Payloads

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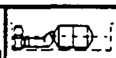
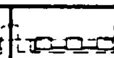
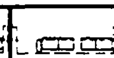
It should be noted that the values quoted in this paragraph are to provide an approximate summary of the gross volumes available for payloads. More precise data are in Appendix B - Structure Interface Definition. The overall volume and mounting area available for payload equipment is shown in Table 3 -5 for the eight 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.

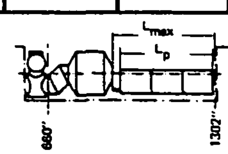
For the pallets, the areas shown represent the maximum available mounting surface. The maximum length indicates the space available for payload lengths including payload overhang. Provisions available in the Orbiter aft flight deck are mentioned in para. 2.5.2. A more detailed breakdown of the available volume is shown in Table 3 - 6. For the racks the quoted values are for the available volume inside the racks. The overhead volume comprises the volume inside the mission dependent stowage containers (of which 8 are provided in the baseline).

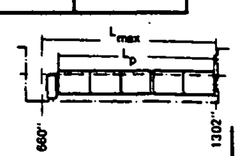
Table 3 - 5: Volume And Area Available to Spacelab Payloads Plus Mission Dependent Equipment For the Basic Spacelab Configurations.

PARAMETER \ CONFIGURATION		DIM								
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	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	—	—	—
	PANEL MOUNTING AREA[1]	m^2	16.56	16.56	16.56	6.196	6.196	—	—	—
PALLET DIMENSIONS[2]	TOTAL LENGTH AVAILABLE FOR PAYLOAD – NO OVERHANG – WITH OVERHANG	m	—	 2.88 3.65	 5.87 6.65	 6.87 6.25	 8.87 9.38	 8.63 16.31	 11.74 16.31	 14.86 16.31
MOUNTING AREA	TOTAL MOUNTING AREA FOR PAYLOAD PROJECTED ON X–Y PLANE – NO OVERHANG – WITH OVERHANG	m^2	—	 11.37 14.02	 23.19 26.27	 23.19 26.27	 35.04 37.05	 34.09 64.42	 46.37 64.42	 58.70 64.42
	TOTAL MOUNTING AREA FOR PAYLOAD ON INNER SKIN PANELS		—	17.1	34.2	34.2	51.3	51.3	68.4	85.5
VOLUME	TOTAL VOLUME AVAILABLE FOR PAYLOADS – NO OVERHANG – WITH OVERHANG	m^3	—	 32.4 39.9	 66.1 74.8	 66.1 70.3	 99.8 105.5	 97.1 183.48	 132.2 183.48	 167.4 183.48
ORBITER AFT FLIGHT DECK	TOTAL VOLUME AVAILABLE FOR SPACELAB PAYLOAD AND SUBSYSTEMS	m^3	0.67	0.67	0.67	0.67	0.67	0.67	0.67	0.67
	TOTAL PANEL AREA AVAILABLE FOR SPACELAB PAYLOAD AND SUBSYSTEMS	m^2	2.102	2.102	2.102	2.102	2.102	2.102	2.102	2.102

[1] TOTAL AVAILABLE IN THE EXPERIMENT RACKS FOR PAYLOADS AND MISSION DEPENDENT EQUIPMENT

[2] 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 FOR PALLET-ONLY MISSIONS.





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.

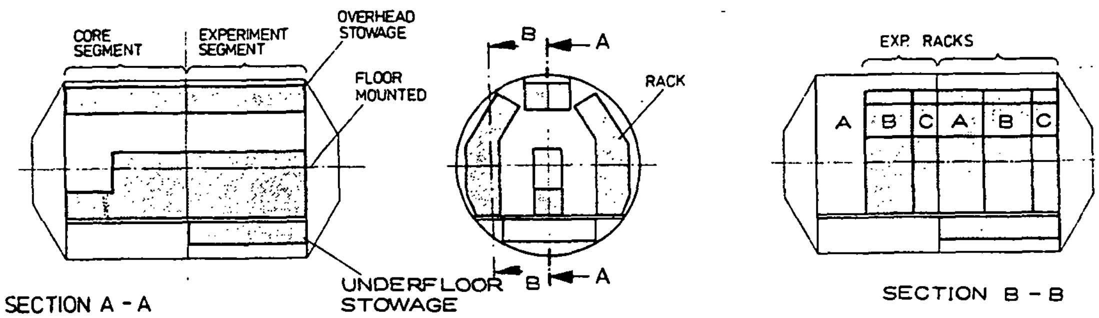
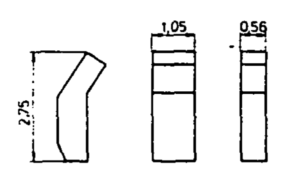
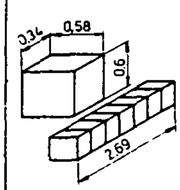
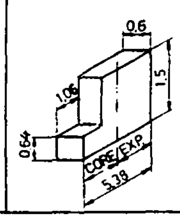
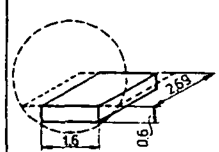


Figure 3 - 26: Module Payload Volume Allocation

A subfloor for experiments in the underfloor area is not part of the current baseline. However, mounting provisions for a subfloor are available.

Figure 3-26 shows the approximate volume available for payload equipment in experiment racks, overhead and underfloor area and in the center aisle. Table 3-6 depicts the gross volume available in those areas.

Table 3 - 6: Volume Breakdown (Gross Volume)

	Volume inside Racks m^3 left side			right side			Ceiling (m^3)	Center Aisle (m^3)	Underfloor (m^3)	Total (m^3)
	A	B	C	A	B	C				
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
										

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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.

3.6.2 Payload Mass and Nominal Load Carrying Capabilities

Spacelab offers a wide range of payload mass capabilities because of the variety of possible Spacelab configurations which can be flown according to specific mission requirements. The total mass of payloads for any given configuration of Spacelab and Orbiter hardware will generally be limited by

- the launch/landing mass capabilties of the Shuttle
- 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 limit mass for his equipment which cannot be exceeded.

The launch/landing mass capabilities are described in para 2.2.5 and 2.3.2. For a large range of orbits the launch capability exceeds the landing capability. The maximum landing mass carried by the Orbiter is 14 515 kg (32 000 lbs). This figure includes the mass of Spacelab hardware, payload equipment and all other items charged to Spacelab or its payload but which are located elsewhere in the Orbiter. Because the launch capability of the Shuttle exceeds the landing capability for a large range of orbits the available mass to Spacelab payloads is generally constrained by the maximum landing mass.

For comprehensive mission planning in terms of masses, it is planned to verify that:

- the maximum Orbiter landing mass is not exceeded.
- the center of gravity constraints (Section 3.2) are satisfied. For certain areas of the permitted center of gravity envelope the maximum landing mass cannot be attained. If this is the case, the total mass available to payloads is reduced accordingly.
- the launch capability for the selected orbit is consistent with the Spacelab plus payload mass at launch. Payload hardware, which is released on orbit or consumables (provided or charged to Spacelab or payloads) which are dumped on-orbit, have to be taken into account in determining the launch mass.

3.6.2.1 Spacelab Load Carrying Capability

The nominal load carrying capability for payloads and mission dependent equipment within the module and for various combinations of pallet segments is summarized in Table 3 - 7. Within the module, the load carrying capabilities for the various locations are only additive to the extent that the total load carrying capability of the module is not exceeded.

Table 3 - 7a: Nominal Load Carrying Capability for Payload and Mission Dependent Equipment Inside the Module

		SHORT MODULE	LONG MODULE
MODULE	- ALONG SIDE WALLS (RACK LOCATION)	634 kg/m PER SIDE	634 kg/m PER SIDE
	- AT OVERHEAD STRUCTURE	100 kg/m PER SIDE	100 kg/m PER SIDE
	- AT CENTER AISLE	300 kg/m	300 kg/m
	- AT AFT-END CONE	298 kg	298 kg
	- AT SUBFLOOR *	—	500 kg
	TOTAL	2900 kg	6380 kg

*) Subfloor not provided as baseline hardware

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Table 3 - 7b: Nominal Load Carrying Capability for Payload on Various Pallet Trains

		WITHOUT IGLOO	WITH IGLOO
PALLET	- SINGLE PALLET SEGMENT	3110 kg	2880 kg
	- two segment train	5000 kg	5000 kg
	- three segment train	5000 kg	5000 kg

3.6.2.2 Spacelab Mass Breakdown

The mass which must always be flown for a given Spacelab configuration, referred to as a "Total Mission Independent Spacelab Mass", is presented in Table 3 - 8, Column 4 for eight Spacelab configurations. This mass consists of the basic Spacelab subsystems furnished by the European Spacelab program, NASA furnished equipment and Orbiter support equipment. All figures quoted in Table 3 - 8 refer to masses at landing. The mass at launch is approximately 400 kg (883 lbs) greater than the mass of landing, mainly due to the fuel contained in the energy kit (Section 2.6.1.2 and Table 2 - 17). In addition, the total mass of all baseline provided mission dependent Spacelab equipment that can be flown on a given Spacelab configuration is presented in Column 5 of Table 3 - 8.

The masses given in Table 3 - 8 are based on a formally agreed upon ESA/NASA control masses applicable to Spacelab. They should not be confused with the current Spacelab mass status.

Column 1 - "Basic Spacelab Subsystems"
This consists of equipment that is essential for the operation of Spacelab and will always be flown (Table 3 - 3).

Column 2 - "NASA Furnished Equipment" (Mission Independent)
This equipment comprises:

- transfer tunnel } for module and module/
- trace gas analyser } pallet configuration only
- forward utilities (connections between Orbiter and Spacelab)

Column 3 - "Orbiter Support Equipment" (Mission Independent)
This category comprises Orbiter equipment which is necessary for Spacelab operations and which is considered as part of Spacelab from a mass chargeability point of view. The following items comprise this category (see Section 2):

- Tunnel adapter and airduct between module and Orbiter crew compartment (ARS-duct) } module and module/ pallet configuration only
- Attach fittings
- Heat rejection kit to increase Orbiter heat rejection from 6.3 kW to 8.5 kW
- One energy kit to increase the energy available to Spacelab from 50 kWh to 890 kWh.

Column 4 - "Total Mission Independent Spacelab Mass"
This represents the sum of the masses in Columns 1, 2 and 3.

Column 5 - "Mass of 100 % Mission Dependent Equipment"

This category comprises the maximum amount of mission dependent Spacelab equipment, defined in Section 3.4.2, that can be accommodated on a given Spacelab configuration. The mass of an actual set of mission dependent equipment for a specific payload can be derived from Table 3 - 4. It should be noted that, whereas the masses quoted in Table 3 - 8 are control masses, the masses of the individual mission dependent items in Table 3 - 4 are current masses.

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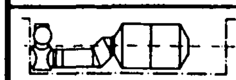
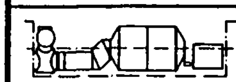
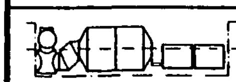
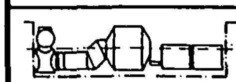
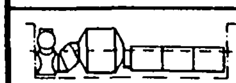
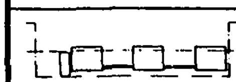
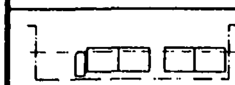
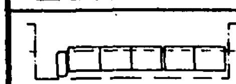
Table 3 - 8: Total Mission Independent Spacelab Mass at Landing

COLUMN	1	2	3	4	5
CONFIGURATION	MISSION INDEPENDENT EQUIPMENT: BASIC SPACELAB SUBSYSTEMS kg	NASA FURNISHED SPACELAB EQUIPMENT kg	ORBITER SUPPORT EQUIPMENT kg	TOTAL MISSION INDEPENDENT SPACELAB MASS, COLUMN 1+2+3 kg	MASS OF 100% MISSION DEPENDENT EQUIPMENT kg
	4780	1082	906	6768	1415
	5665	1082	1161	7908	1610
	6295	982	1161	8438	1660
	5235	1082	1161	7478	1015
	5865	982	1161	8008	1065
	3200	300	1049	4549	575
	3740	300	794	4834	610
	4340	300	794	5434	660

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3.6.2.3 Mass and Load Carrying Capability for Payloads

DCN 001 Table 3-9 lists the Mass (Column 2) and the load carrying capability (Column 3) for payload. Whichever value is lower will constrain the total amount of payload for a particular configuration. The mass capability takes into account a program mass reserve.

Column 1

DCN 034 The program reserves are composed of margins for mass growth. These margins have not yet been allocated to a definite purpose and payloads should not make use of these reserves unless they are subsequently released by the Spacelab Program for payloads.

Column 2

DCN 034 The mass available to payloads and mission dependent equipment is obtained by subtracting the "Total Mission Independent Spacelab Mass" (Table 3 - 8, Column 4) and the program reserves from the maximum landing mass of 14515 kg (32000 lbs). The payload and mission dependent equipment comprise the following categories:

a) All payload hardware (user provided) located in Spacelab and/or the Orbiter (e.g. aft flight deck).

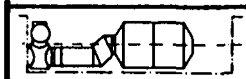
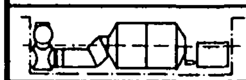
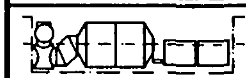
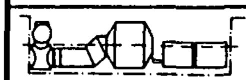
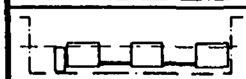
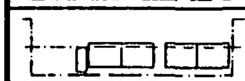
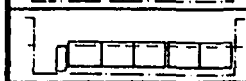
b) All items(Orbiter or Spacelab provided) which have been selected for a specific mission , but are chargeable to payload mass. These items consist of:

- Mission Dependent Orbiter Support Equipment. (see Table 2 - 17)
- Mission Dependent Spacelab Equipment. (see Table 3 - 4)
- Instrument Pointing System (IPS) described in Section 4.8.

Column 3

It is emphasized that the given load carrying capabilities are nominal values. In particular it is pointed out that the load carrying capability of a pallet is dependent on the layout of the payload and the method of attachment. Therefore, the actual load carrying capability of a pallet can substantially differ (in both directions) from the nominal value. The figures given describe the load carrying capability on pallets and in the module and do no include payload equipment stored in the Orbiter on the Aft Flight Deck (Section 2.5.2), where approximately 160 kg (352 lbs) of payload equipment can be carried in the locations designated by R 12, A 6, A 7, L 10.

Table 3 - 9: Mass Available to Payloads and Mission Dependent Equipment

COLUMN	1	2	3
CONFIGURATION	NOT RELEASED PROGRAMME RESERVES kg	MASS AVAILABLE TO PAYLOAD AND MISSION DEPENDENT EQUIPMENT ** kg	NOMINAL LOAD CARRYING CAPABILITY OF SPACELAB FOR PAYLOAD AND MISSION DEPENDENT EQUIPMENT kg
	1532	5965	5965
	547*	6060	9510
	517*	5560	11 420
	522*	6515	7940
	492	6015	7960
	591	9375	9375
	571*	9110	10 200
	771	8310	10 220

*) PROVISIONAL

**) 14516 kg (32000 lb) MINUS COLUMN 4, TABLE 3-8, MINUS COLUMN,1, TABLE 3-9.

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3.6.3 Power and Energy

3.6.3.1 Power and Energy Resources

The electrical power delivered to Spacelab at the Orbiter/Spacelab interface in the cargo bay and to the AFD at the AFD power distribution box input depends on the operational phase as shown in Table 3 – 10. The term "maximum continuous power" used herein is defined as power available with no time limitations except those due to the energy available.

During the operational phases ascent and descent, the maximum continuous power to the module and/or pallet may be increased by as much as the corresponding AFD power, provided there is an equivalent decrease in power to the AFD. A change is under consideration to have the same capability for the on-orbit phase.

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A timeline diagram depicting the operational phases in more detail is given in Figure 3-27. The basic electrical energy available from the Orbiter for Spacelab is 890 kWh. Additional energy is available with the addition of extra energy kits, the mass of which is chargeable to the payload.

Table 3 – 10: Power Resources Available to Spacelab and Payload for Various Operational Phases

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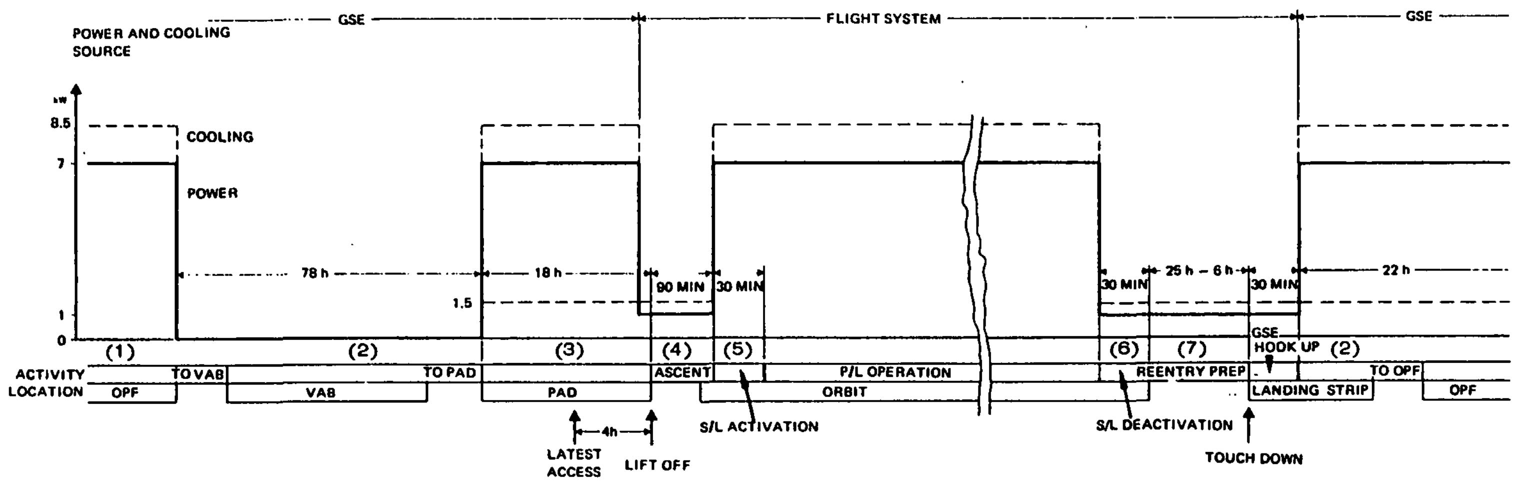
		Ground Phases (GSE prov.)	Prelaunch/ Ascent Orbiter Provided	On–Orbit	On–Orbit (degraded mode)	Descent
Maximum Continuous Power	**Cargo Bay**					
	Prim. DC	0 – 7 kW	1 kW	7 kW	5 kW	1 kW
	Aux. DC	400 W	400 W	400 W	400 W	400 W
	Total	7 kW	1 kW	7 kW	5 kW	1 kW
	AFD					
	DC	750 W	350 W	750 W	750 W	350 W
	AC (each of 2)	650 VA	0 VA	690 VA	690 VA	350 VA
	Total	750 W	350 W	750 W	750 W	350 W
Peak Power	**Cargo Bay**					
	Prim. DC	12 kW 15min/3h	1.5 kW 2min	12 kW 15min/3h	8 kW 15min/3h	1.5 kW 2min
	Aux. DC	25 A 2s	25 A 2s	25 A 2s	25 A 2s	25 A 2s
	Total	12 kW	1.5 kW	12 kW	8 kW	1.5 kW
	AFD					
	DC	1 kW 15min/3h	420 W 2min	1 kW 15min/3h	1 kW 15min/3h	420 W 2min
	AC (each of 2)	1 kVA 2min/3h	0	1kVA 2min/2h	1kVA 2min/3h	420 VA 2min
	Total	1 kW	420 W	1 kW	1 kW	420 W



(1) Power actually available for Spacelab depends on Orbiter operation. It may vary from 0 to 7 kW (0 to 8.5 kW cooling).

(2) During this period power may be made available, but there are no-power intervals with a maximum length of 26 working hours, based on a 2 shift, 5 days a week work schedule.

(3) Power actually available for Spacelab depends on Orbiter operation. 7 kW (8.5 kW cooling) is the maximum power when the Orbiter is powered down. 1 kW (1.5 kW cooling) is the minimum power during this period.

(4) For experiment design purposes, a 2nd contingency orbit doubling the period to 3 hours should be taken into account.

(5) Partial payload activation may be possible if required before full Spacelab activation (mission dependent).

(6) Duration of Spacelab de-activation is mission dependent.

(7) Re-entry preparation is mission dependent, due to thermal conditioning of the Orbiter.

Figure 3 - 27: Timeline of Power Available to Spacelab (Cargo Bay)

DCN 001

3.6.3.2 Power and Energy Available for Payload and Mission Dependent Equipment

The power available to experiments depends on the power consumption of the basic Spacelab subsystems and, similarly to the weight budget, is also a function of the use of mission dependent equipment.

The power budget differs from the mass 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 budgeting has to be taken into account.

3.6.3.2.1 Power and Energy Available On-Orbit

Spacelab

DCN 034 The allocation of the electrical power available in Spacelab (7 kW max. continuous, 12 kW peak minus wire loss between Orbiter and Spacelab) to basic Spacelab equipment, mission dependent equipment and payload is presented in Table 3 - 11 for different configurations. The values are projected values for planning purposes and include wire losses between the Orbiter and Spacelab.

The approach used for power budgeting is outlined graphically in Figure 3 - 28. The columns of Table 3 - 11 are explained in detail

DCN 001

DCN 003

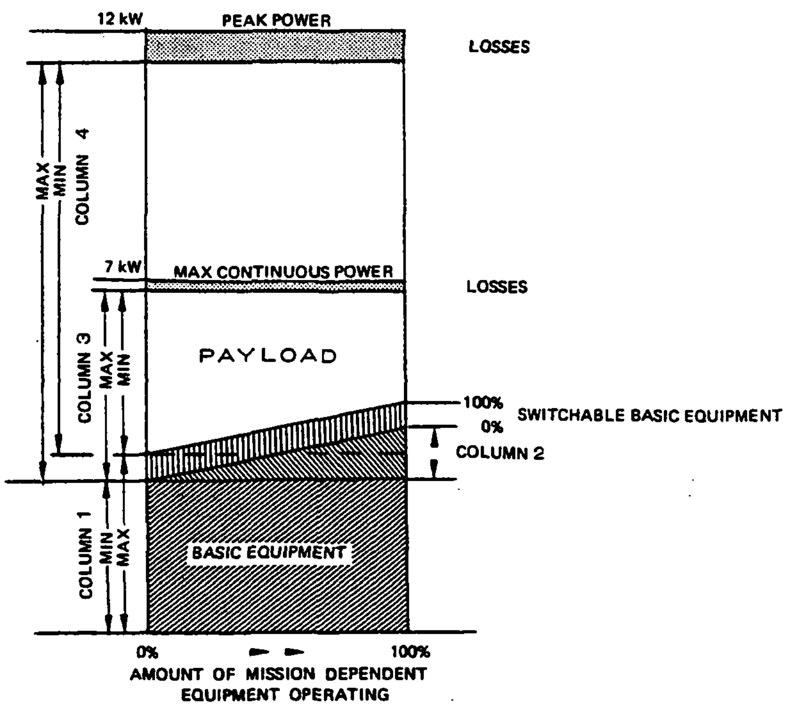
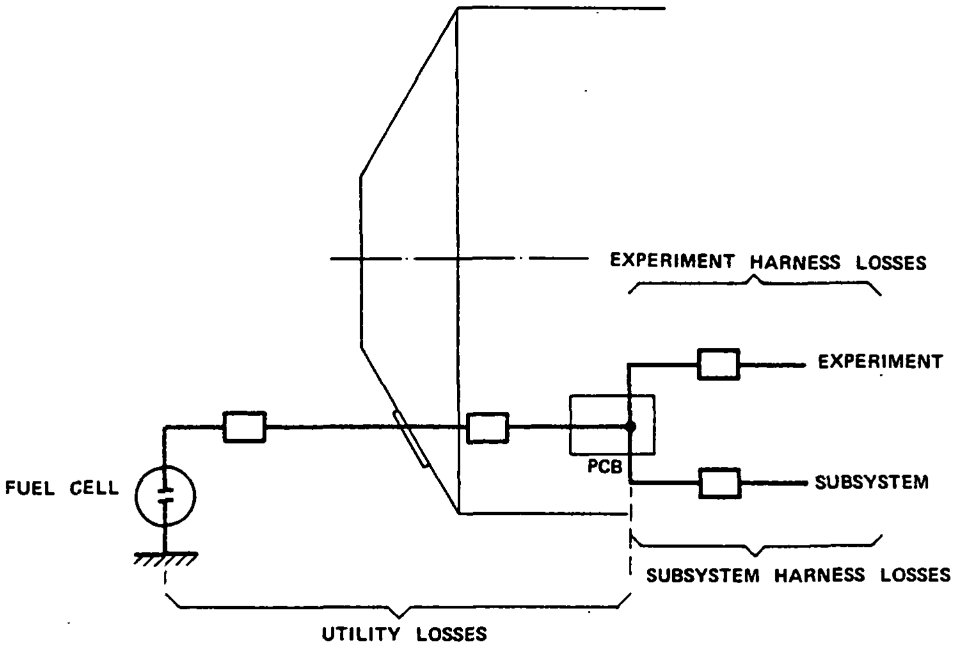


Figure 3 - 28: Graphic Presentation of Power Budgeting Approach



DCN 049

Figure 3 - 28a: Harness Power Losses

The utility losses are the wire losses between the Orbiter fuel cell and the branching point of the subsystem bus and the experiment bus in the power control box (PCB).
These utility losses are deducted from the maximum continuous power (7 kW to 6.34 kW) and from the peak power (12 kW to 10.94 kW) allocated to Spacelab (see Figure 3 - 28).

The subsystem harness losses are the wire losses between the PCB branching point and the various subsystem end items. These subsystem harness losses are included in the power figures for basic Spacelab and mission dependent equipment (see Table 3.11, Column 1 and 2).

The experiment harness losses are the wire losses between the PCB branching point and the various experiment end items. These experiment harness losses have to be taken into account for the experiment power budgeting. The experiment harness losses have to be calculated for each payload based on the data in Appendix A, Section 3.9 - EPDS Impedance Characteristics.

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Column 1

The minimum value given represents the maximum continuous power consumption of the permanently switched-on basic Spacelab equipment. The maximum value encompasses in addition the full power of all additional switchable equipment of the basic Spacelab, such as ceiling normal lighting, tunnel lighting, work bench lighting, tunnel fan, heaters and valves.

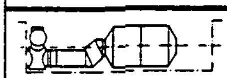
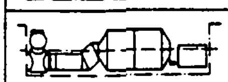
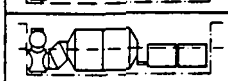
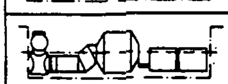
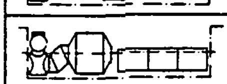
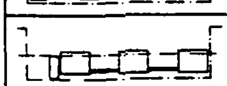
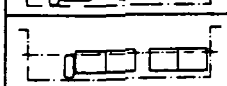
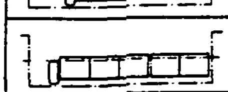
Column 2

This column represents an upper boundary value which is derived by summing the maximum power consumption of all mission dependent equipment (IPS not included) that can be accommodated in the indicated configuration. Actual values can only be derived by analyses of individual missions which have a specific set of mission dependent equipment with a specific operations timeline. The individual power consumption of each unit is given in Table 3 - 4.

Column 3

In view of the fact that, without individual mission analyses, no specific power requirements can be defined for mission dependent equipment, the available maximum continuous power for payload plus mission dependent equipment is given in Column 3. The minimum and maximum boundaries reflect the variance from Column 1. The power outlined in Column 3 includes the auxiliary power available to payload via the experiment essential power bus. This is approximately 300 W for pallet-only configurations and varies between approximately 40 W and 200 W for module configurations. The lower figure represents the continuous power available during emergencies.

Table 3 - 11: Power and Energy On-Orbit

CONFIGURATION	1 POWER REQUIRED BY BASIC SPACELAB (kW)		2 MAX POWER REQ'D FOR 100% OF MISSION DEPENDENT EQUIP-MENT (kW)	3 POWER AVAILABLE FOR PAYLOAD AND MISSION DEPENDENT EQ'MT (kW) MAX CONT' POWER		4 PEAK POWER		5 TOTAL ENERGY AVAILABLE TO PAYLOAD AND MISSION DEPENDENT EQUIPMENT (INCL. AFD, AND COMPRISING ALL MISSION PHASES) (kWh)	
	MIN.	MAX.		MIN.	MAX.	MIN.	MAX.	MIN	MAX
	2.9	3.4	1.5	3.24	3.74	7.54	8.04	240	320
	3.3	3.8	1.5	2.84	3.34	7.14	7.64	190	250
	3.3	3.8	1.5	2.84	3.34	7.14	7.64	190	250
	3.3	3.7	1.2	2.94	3.34	7.24	7.64	190	250
	3.3	3.7	1.2	2.94	3.34	7.24	7.64	190	250
	1.3		1.0	5.34		9.64		570	
	1.3		1.0	5.34		9.64		570	
	1.3		1.0	5.34		9.64		570	
AFT FLIGHT DECK	0.3		-	0.45		0.7			

Column 4

Same as in Column 3 with reference to the peak power available to payload mission dependent equipment. The peak power availability is limited to a duration of 15 min. once in a 3-hours period. In addition, peak power usage is limited by Orbiter thermal constraints.

Column 5

The energy available to payload and mission dependent equipment, for a normal 7 days, mission comprising the operational phases prelaunch/ascent on-orbit and descent, is given in Column 5. The values given result from basic Spacelab equipment, including AFD equipment being fully activated for 156 hours.

Use of mission dependent equipment will reduce the energy available to payload according to the specific mission timelines. Contrary to power, the energy available 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 Table 2 - 17.

Power Saving Mode

During operation of the avionics fan in the low speed mode (see para 4.6.3.2.3), an extra 550 W (added to power values in Columns 3 and 4 of Table 3 - 11) is available for payloads in all configurations with a module.

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3.6.3.2.2 Power Available During Other Operational Phases

During ascent/descent phases of the mission a partial activation of Spacelab is possible and limited power can then be made available for experiments. The power available for experiments in configurations using a module is approximately 670 W from the experiment DC bus (the experiment AC inverter will not be powered up). For pallet only configurations the provision of power for payloads is not foreseen since the pallet freon loop will not be operative and satisfactory cooling for partially activated subsystem equipment cannot be guaranteed.

3.6.4 Heat Rejection

3.6.4.1 Heat Rejection for Spacelab and Payload

The Orbiter heat rejection capability offered to Spacelab and its payload depends on the operational phase, in the same manner as the available power.

Basically the Orbiter heat rejection capability matches the power offered to Spacelab and in addition takes metabolic heat loads from the crew and some heat leaks into account.

Table 3 - 12 shows the available cooling as a function of the operational phase. The interrelationship between the cooling available in the aft flight deck and the cooling available to Spacelab and payload in the cargo bay is also shown. The timelining of cooling available to Spacelab and its payload in the Orbiter cargo bay is shown in Figure 3 - 27, along with the available power timelines.

Table 3 - 12: Heat Rejection Capabilities

	Cargo Bay		Orbiter AFD	
	average	peak	max. cont.	peak
Prelaunch/Ascent	1.52 kW	TBD	0.35 kW	0.42 kW, 2 min.
On-Orbit 1)				
max. cargo bay/ min.AFD cooling	8.5 kW	12.4 kW 15min/3h 2)	0.35 kW	1.0 kW 15min/3h 3)
min. cargo bay/ max.AFD cooling	8.1 kW	12.4 kW 15min/3h	0.75 kW	1.0 kW 15min/3h
Descent/Postlanding	1.52 kW	TBD	0.35 kW	0.42 kW, 2 min.

1) A total of 8.85 kW cooling capability is available to Spacelab (AFD and cargo bay). The maximum and minimum values stated are boundaries for reducing and increasing the cooling capability in the cargo bay and AFD concurrently.

2) The peak heat load is accommodated by the Spacelab ECS and,therefore, affects the average cooling capability.

3) The peak heat load in the AFD reduces the cooling capability in the cargo bay by an equivalent amount.

3.6.4.2 Heat Rejection for Experiments on Orbit

The primary cooling mode inside the module for experiments is air-cooling via the avionics and cabin cooling loops.

Liquid cooling for payloads in the module is also available via the experiment heat exchanger and an experiment dedicated cold plate.

On the pallet Spacelab offers cold plate cooling to experiments via the pallet freon cooling loop. The freon loop also cools subsystem equipment on the pallet and in the Igloo.

The hardware limitations for overall heat transfer (for basic, mission dependent subsystem equipment plus payload) at the design point are:

- avionics loop 4.5 kW
- cabin loop 2.7 kW
- experiment heat exchanger 4 kW
- experiment cold plate 1 kW

The water and the freon loop have the design capability to transfer the total 8.5 kW to the Orbiter under nominal conditions, i.e. within the specified temperature ranges.

Table 3 - 13 presents a breakdown of the nominal heat transfer capability for experiments and mission dependent equipment for the particular Spacelab cooling loops. It must be noted that the values given for each loop are not additive. They merely give the upper boundary in each loop for the heat rejection into that particular loop. The overall distribution of heat over the various loops has to take into account both these boundaries and the total heat rejection capability of Spacelab. Data given are based on 7.5 °C water loop inlet temperature and a cabin temperature of 19.5 °C. The numbers were derived by subtracting the respective basic subsystem equipment heat load, metabolic heat, and heat leaks between cooling loops from the design point of each loop or, in the case of the AFD, from the maximum allocated overall heat transfer. The basic subsystem equipment heat load taken is compatible with the power consumption given in 3.6.3.2.

The heat transfer capability given is nominal, i.e. with no heat leaks from / to space. These heat leaks, which depend on Orbiter attitude (see fig 4.6-7) mainly influence the cabin loop. They decrease / increase the heat transfer capability available for experiments and mission dependent equipment and, in extreme cases, limit the range of selectable temperatures of the cabin air.

The peak heat transfer capability is given in each column as the positive delta to be added to the average values listed. The peak again is defined with a duration of 15 minutes every 3 hours.

The recovery heat transfer that has to follow a peak heat transfer period to stay within the average heat transfer capability is given as a negative delta.

Column 1

The minimum and maximum available heat transfer values for payloads and mission dependent equipment refer to cases when the switchable basic Spacelab subsystem equipment (see 3.6.3.2) is ON or OFF, respectively. The minimum case assumes three crew members, causing a total metabolic load of 0.6 kW, the maximum case assumes no crew members in Spacelab.

Column 2

This column presents the integral heat transfer capability for aircooled payload and mission dependent equipment in the experiment racks. The aircooling capability in a particular rack depends on the airflow distribution for all Spacelab racks (see 4.6.3.2).

Column 3

For the pallet only configurations the freon loop heat transfer for payloads and mission dependent equipment represents the subtotal for Spacelab without AFD. For module configurations the heat transfer capability of the freon loop, which exceeds the total Spacelab capability, is given for completeness only.

Column 4

For Module configurations the water loop represents the subtotal of heat transfer capability for payloads and mission dependent equipment for Spacelab without AFD based on 8.5 kW overall heat transfer. Because the cabin loop, avionics loop and the freon loop are subloops of the water loop the values given are not the sum of the capabilities given in column 1, 2, and 3, but represent another boundary condition. The water loop also contains the experiment heat exchanger and the experiment cold plate in the Module.

The maximum and minimum values reflect the variance of the heat loads in column 1, cabin loop.

Column 5

The subtotal of heat transfer capability for payloads in the AFD is based on the maximum heat transfer of 750 W allocated to the AFD.

Column 6

The total heat transfer for payloads and mission dependent equipment in Spacelab including the AFD is based on 8.85 kW overall heat transfer from Spacelab including AFD to Orbiter.

The values given are not the sum of the subtotal for Spacelab without AFD, column 3/4 and the subtotal for the AFD, column 5, but represent another boundary condition. This is because the maximum overall heat transfer for Spacelab and the maximum overall heat transfer for the AFD are not additive (see Table 3-12).

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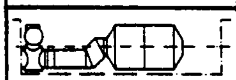
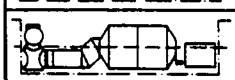
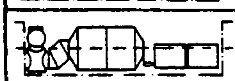
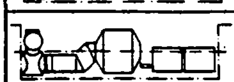
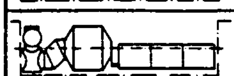
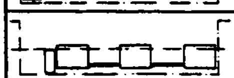
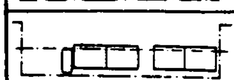
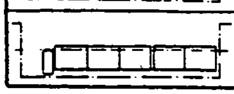
Low-speed Mode of Avionics Fan

When the avionics fan is switched to the low-speed mode (see para 4.6.3.2.3) the heat rejection available for payloads in the avionics loop (column 2) is approximately 800 W.

Table 3 - 13: Nominal Heat Rejection Available to Payloads and Mission Dependent Equipment (All units given in kW)

Column	1		2	3	4		5	6	
Configuration	CABIN LOOP		AVION. LOOP	FREON LOOP	WATER LOOP		AFD	TOTAL HEAT REJECTION	
	peak +2kW recover -TBD		peak +5.4kW recover -TBD	peak +3.9kW recover -1.1kW	peak +3.9kW recover -1.1kW		peak +0.25kW recover 0kW	peak +3.9 kW recover -1.1kW	
	MIN.	MAX.			MIN.	MAX.		MIN.	MAX.
	0.4	1.5	3.4	-	4.1	5.2	0.47	4.1[7]	5.2[7]
	0.4	1.5	3.4	8.0	3.7	4.8	0.47	3.7[7]	4.8[7]
	0.4	1.5	3.4	8.0	3.7	4.8	0.47	3.7[7]	4.8[7]
	0.6	1.6	3.3	8.0	3.8	4.8	0.47	3.8[7]	4.8[7]
	0.6	1.6	3.3	8.0	3.8	4.8	0.47	3.8[7]	4.8[7]
	-	-	-	7.0	-	-	0.47	7.0[7]	
	-	-	-	7.0	-	-	0.47	7.0[7]	
	-	-	-	7.0	-	-	0.47	7.0[7]	

The heat transfer capabilities have been given for payloads plus mission dependent equipment in view of the fact that, without individual mission analyses no specific heat loads of mission dependent equipment can be defined. The tools to budget the heat loads of mission dependent equipment for a given mission are

- Table 3-4, giving the heat loads (power consumption) for mission dependent equipment
- Table 3-14, showing the location of mission dependent equipment in the cooling loops.

For actual missions, thermal budgeting has to take heat loads from/to space into account (positive and negative for hot case and cold case, respectively), which have to be accommodated by Spacelab ECS. More precise details will be provided in Appendix C - Thermal Interface Definition. DCN 002

Table 3 - 14: Cooling Loops for Mission Dependent Subsystem Equipment

Equipment	Avionics Loop	Cabin Loop	Water Loop	Freon Loop
Experiment Inverter	-	-	x	x 1)
Experiment Computer	-	-	x	x 1)
Experiment I/O Unit	-	-	x	x 1)
Experiment RAU	x 2)	x 3)	-	x 4)
DDU/Keyboard	x	-	-	-
HRDR (Recorder)	x	-	-	-
HRM	-	-	x	x 1)
Airlock		x		
Viewport		x		

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Notes:
1) in pallet-only configurations
2) in rack location
3) in center aisle and airlock location
4) in pallet location

3.6.4.3 Heat Rejection for Experiments During other Operational Phases

The heat rejection available to experiments during GSE hookup, pre-launch/ascent, and descent depends on specific experiment requirements which, in turn, will influence to which extent Spacelab subsystems have to be activated and cooled. A partial activation of Spacelab subsystem equipment in the module is possible using the module water loop and the avionics air loop with the avionics fan switched to the low-speed mode (see para 4.6.3.2.3) while the cabin air fan is switched off. Under these conditions the total heat rejection capability for experiments (via experiment heat exchanger, cold plate or rack air cooling) is approximately 900 W. DCN 034

3.6.5 Command and Data Handling Resources

Table 3 – 15 summarizes the command and data management resources available to experiments. A more detailed description is given in Section 4.4 "Command and Data Management System".

Table 3 – 15: 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 length including one parity bit	17 bit
max basic sampling rate (for all inputs)	100 Hz
data rate of transfer RAU / computer (incl. overhead)	1 Mb/s
Wide Band Scientific Data	
number of experiment channels of the High Rate Multiplexer (HRM)	16
max data rate of HRM input channels	16 Mb/s
number of CCTV video input channels	3
number of 4.5MHz 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 Data Rate Recorder (HDRR) bridging TDRSS non-coverage periods	32 Mb/s
storage capability of HDRR (HDRR not available in pallet-only mode)	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 one parity bit	17 bit
Payload Data Processing and Displays	
payload data processing :	
word length	16 bit
speed (Gibson Mix)	350 kop/s
floating point arithmetic	32 (24+8) bit
mass memory	131 Mbit
payload data display: alphanumerical display screen	12 inch diagonal Tri-color

3.7 Extended Flight Capability

The nominal on-orbit stay time of Orbiter and Spacelab is seven days. Extended flight durations of up to 30 days can be achieved by the provision of additional hardware and consumables which are charged to the Spacelab payload.

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

3.7.1 Orbiter Related Items

Hardware and consumables which can be supplied by the Orbiter for extended flight durations are listed below:

Energy kits for electrical energy covering the needs of
- the Orbiter for flight durations above 7 days
- Spacelab and its payload above 890 kWh

Crew and crew support above Orbiter baseline
Compensation for air leakage from the Orbiter cabin
RCS propellant above Orbiter baseline

Table 2 - 17 provides quantitative data on the items listed above.

3.7.2 Spacelab Related Items

As regards Spacelab the following items have to be considered in the case of extended flight duration:

Additional nitrogen tanks above 12 day flight duration (24 kg for additional 12 days)
Compensation for igloo leakage
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 hardware needed for extended flight durations but does not include this hardware as deliverable items. With regard to condensate storage, periodic dumping of 47 kg of water approximately every 8 days is assumed

3.7.3 Mass Penalty for Various Flight Durations

Figure 3-29 presents the tentatively assessed mass penalty to Spacelab payloads due to extended flight duration assuming various crew sizes. This figure is based on a power consumption of 12.5 kW for the Orbiter and 7 kW for Spacelab and Spacelab payloads.

The most significant penalty results from the additional energy kits. Energy kits 1 through 3 are located outside the dynamic envelope of the Orbiter cargo bay. Further additional energy kits, however, must be located within the dynamic envelope of the Orbiter cargo bay resulting in a significant volume penalty which may also result in a restriction of the flyable Spacelab configuration.

The mass penalty given in Figure 3 - 29 has to be deducted from the "mass available to payload and mission dependent equipment" (Table 3-9). In this context it is pointed out that part or possibly all "programme mass reserves" given in Table 3-9 may be released to the Spacelab payload at a later stage of the Spacelab programme. In cases where the mass available to payload exceeds the payload load carrying capability of Spacelab (Table 3-9), the mission planner should be aware of the possibility to use any excess mass available for an extension of the flight duration.

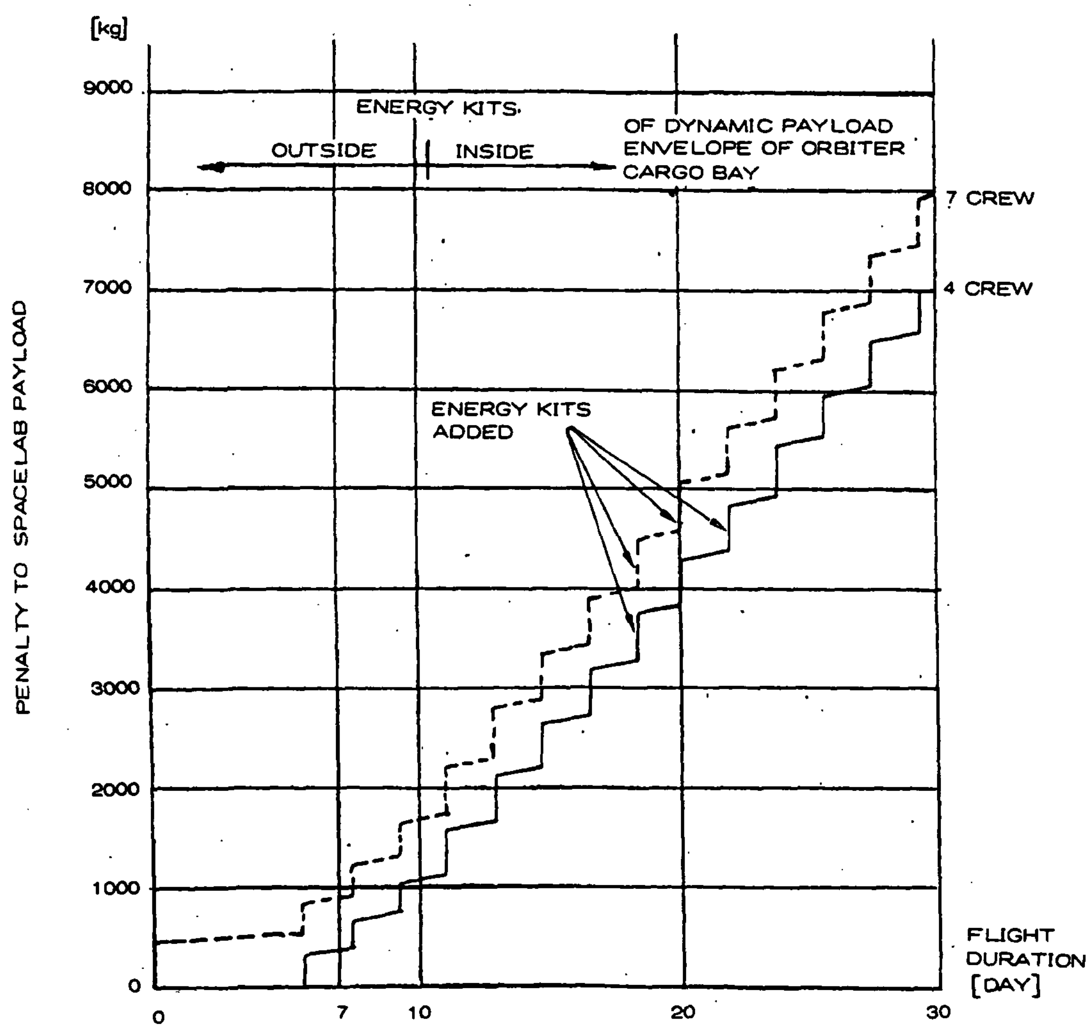

Figure 3-29: Mass Penalties due to Extended Flight Durations

Section 4 - SPACELAB SUBSYSTEMS

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LIST OF FIGURES (cont'd)

4. SPACELAB SUBSYSTEMS

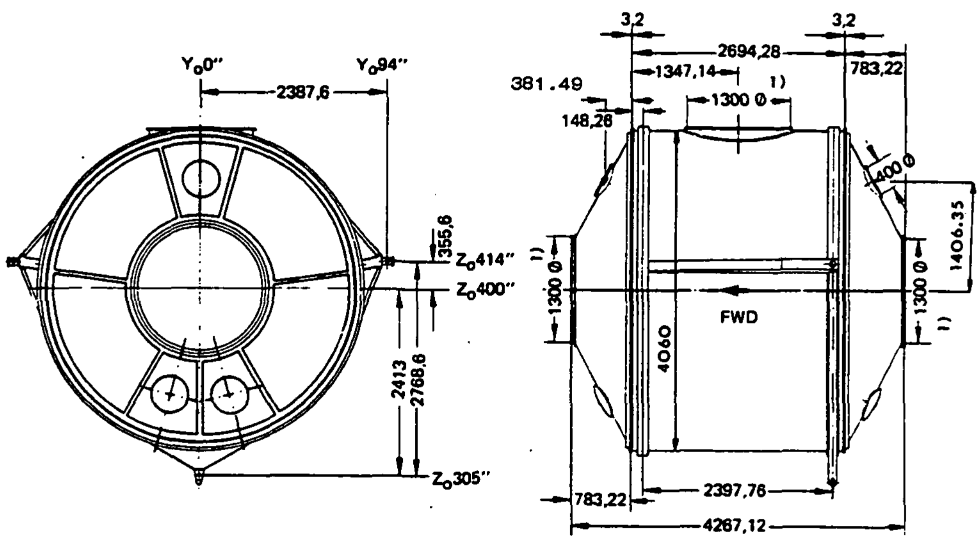
4.1 Structure

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
- Overhead Structure
- Racks



1) Aperture Clearance

Figure 4.1-1a): Short Module, Physical Dimensions

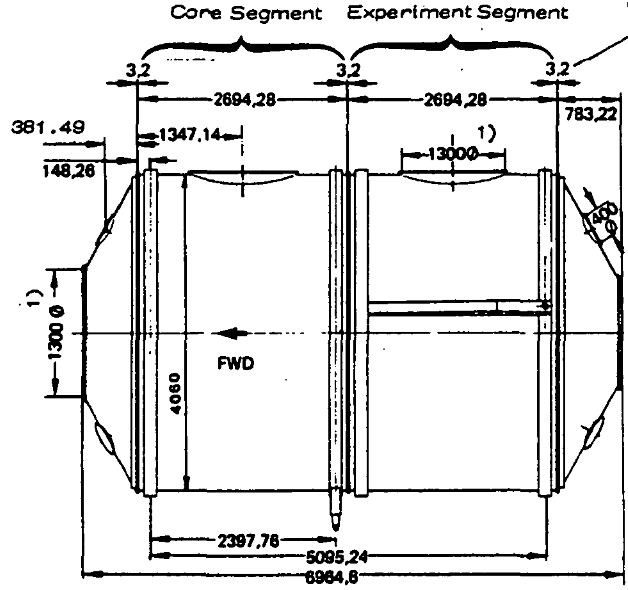


Figure 4.1-1 b: Long Module Physical Dimensions

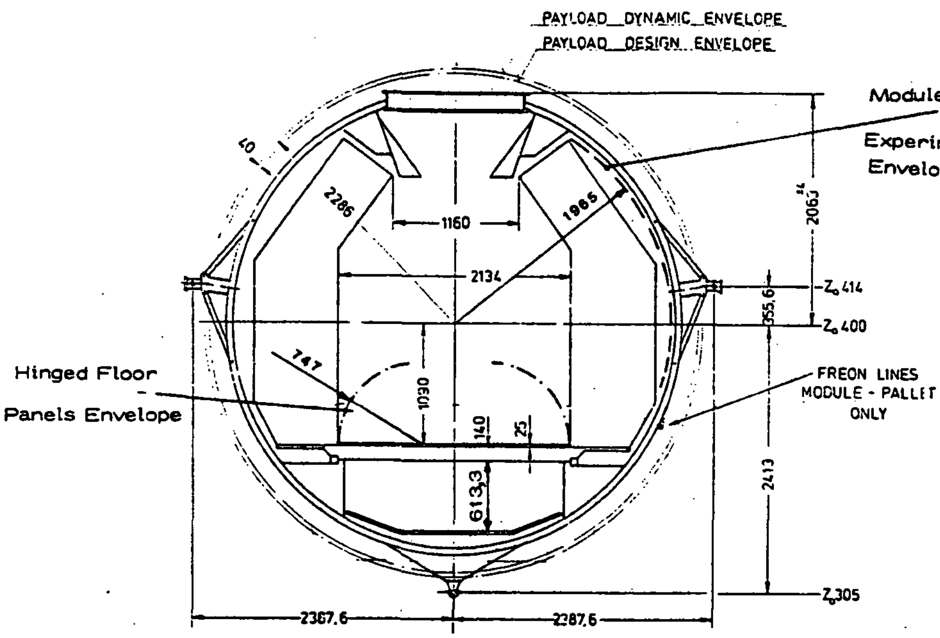


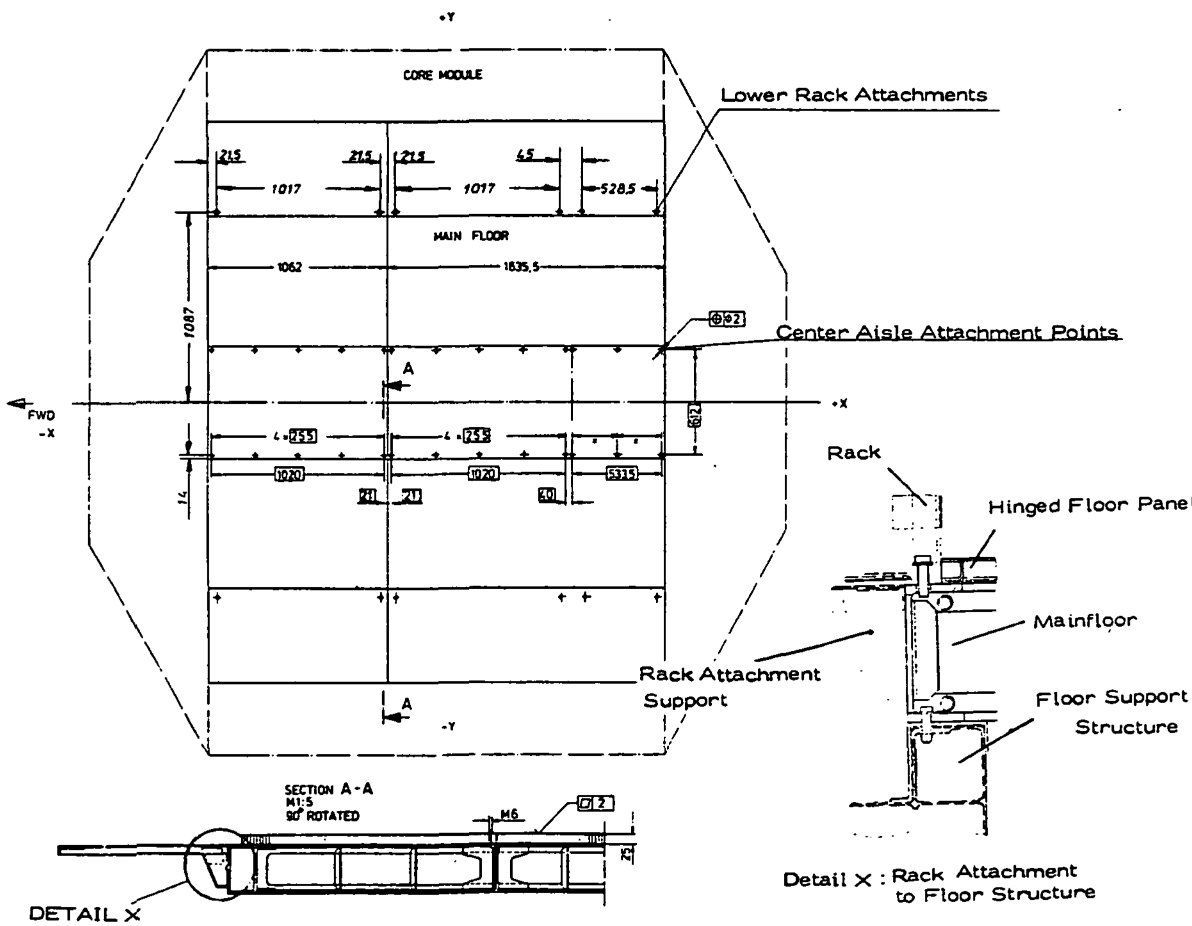
Figure 4.1-2: Characteristic Dimensions

4.1.1.1 Main Floor

The floor in each module segment (core and experiment segment) consists of two individual sections, 1062 and 1635.5 mm long, respectively. Each section consists of a framework of beams (Figure 4.1 – 3) providing support and mounting provisions for standard experiment racks and/or experiment equipment. If required, the two floor sections can be bolted together to provide a single continuous floor during integration and flight.

The outer part of the floor is usually covered by the racks and also includes the lower rack attachments, as shown in Figure 4.1 – 3. When the outer portion of the floor is not covered by racks or other experiment equipment, close-out honeycomb panels are available. The floor panels adjacent to the racks are hinged for accessibility to the subfloor area.

The center portion of the floor (center aisle) is covered by fixed honeycomb panels and can accommodate experiment equipment mounted to the attachment points, shown in Figure 4.1 – 3.



4.1 – 3: Floor Structure Interfaces

4.1.1.1.1 Load Carrying Capability

- Experiment Located on Center Floor:

An experiment mass up to 300 kg/m of length over the 0.6 m wide center floor may be accommodated.

DCN 026

- Experiment Replacing Racks

Experiments located at the outer portion of the floor may be mounted only at the lower rack attachments. It must be noted that such attachments points occur only along a single line and must, therefore, be used in conjunction with the rack attachment points at the overhead structure. The mass of experiments replacing racks shall be consistent with the max. allowed masses within racks plus the rack mass (see para. 4.1.1.3.2 and 3.4.2)

4.1.1.1.2 Provisions for Center Aisle Mounted Experiment

Structural provisions in the main floor allow the use of Spacelab resources for center aisle mounted experiments, as shown in Figure 4.1 - 5.

- ECS Provisions

Experiment equipment mounted on the center aisle may be cooled by:

(a) Cabin air which is drawn through the experiments by user provided fans. The only provisions of Spacelab are two round cut-outs in the center aisle floor, one in the experiment segment, one in the core segment. Warm air from experiments may be blown through these cut-outs into the underfloor space. The cut-outs have to be fitted with debris traps and are closed by cover plates if not used.

(b) A user provided liquid cooling loop connected to the mission dependent experiment heat exchanger. Hence cut-outs and a floor mounted bracket for fluid connectors are provided for experimenter provided liquid lines. A coverplate is provided if the cut-out is not used.

- EPDS and CDMS Provisions

Two cut-outs, one in the core segment and one in the experiment segment each, give access to a connector bracket providing DC and AC power for experiments. The same connector bracket provides the interface for one experiment RAU.

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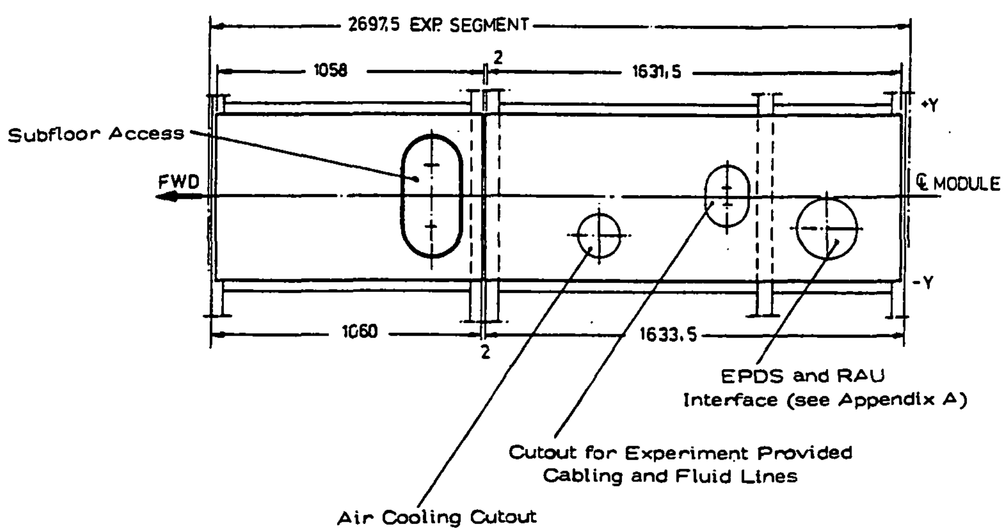


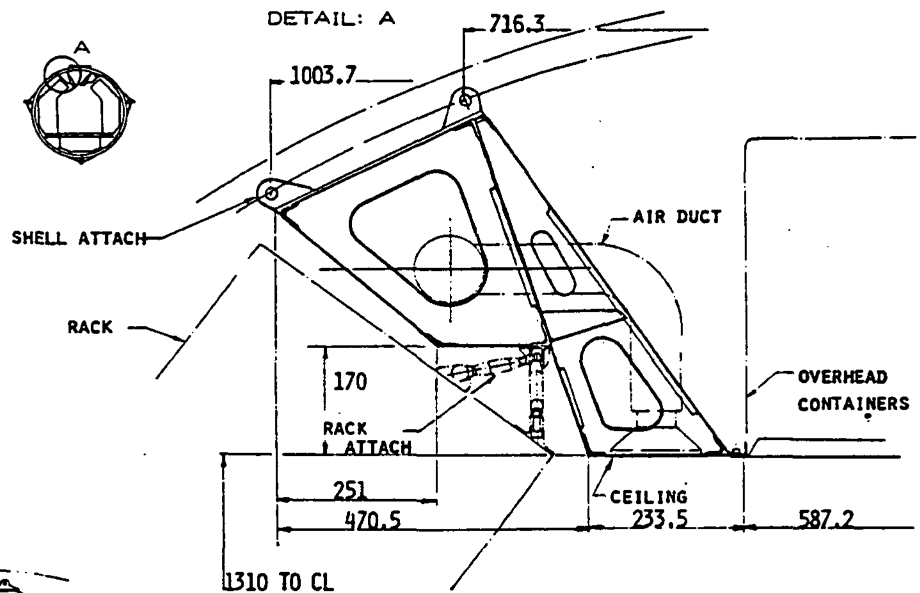
Figure 4.1 - 5: Provisions for Center Aisle Mounted Experiments in the Core Segment (Provisions in the Experiment Segment are Similar)

4.1.1.2 Overhead Structure

In the overhead structure, lighting, air ducts and horizontal handrails are installed. If top airlock installation is required, lighting fixtures and mounting brackets in the respective section, as shown in Figure 4.1 - 6, will be removed. The same is necessary if the high quality window/viewport assembly is installed.

The overhead structure, as shown in Figure 4.1 - 6, provides channels for the accommodation of stowage containers, as described under para. 4.2.2.

The upper attachment points for the racks are also located in the overhead structure, as shown in Figures 4.1 - 6 and 4.1 - 7. The interface with experiments replacing racks may be directly at the overhead structure fittings, or the rack attachment struts may be used.



Overhead Structure Interfaces

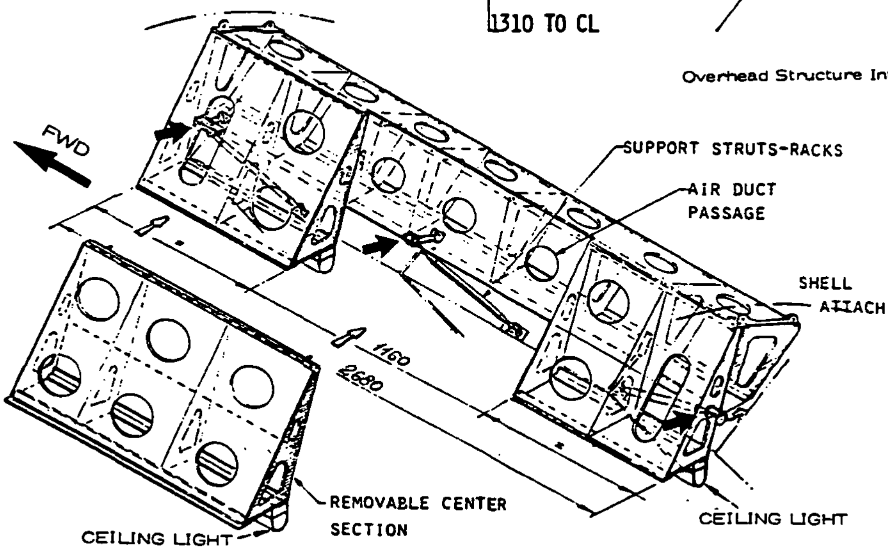


Figure 4.1 - 6: Rack Attachment Points at the Overhead Structure

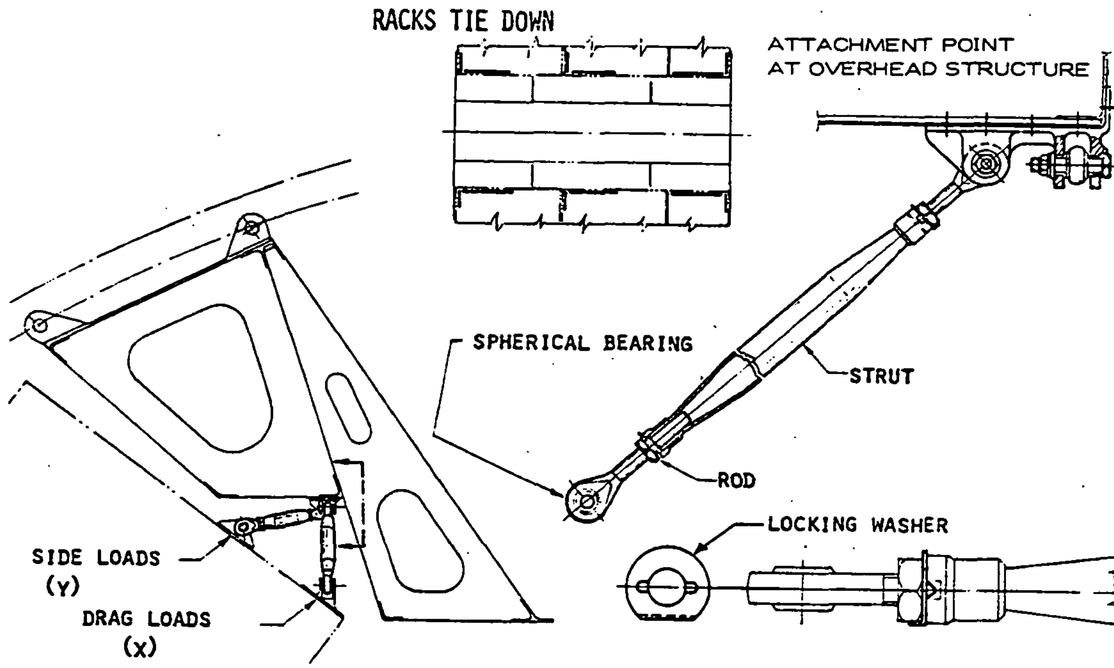


Figure 4.1-7: Rack Attachment - Details

4.1.1.3 Standard Experiment Racks

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 - 3 and 4.1 - 7.

Location and arrangement of the racks inside the module are as indicated in Figures 3 - 14, 3 - 15 and 3 - 16. Two types of racks are available:

Single racks with an overall width of 563.5 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, however, single and double racks are not interchangeable. Figure 4.1-8 shows the main features of a double rack.

The experiment racks are available for experiments and the following Spacelab mission dependent subsystem equipment:

- Experiment power switching panels; one in each rack is foreseen (see Figure 4.1 -9c).
- Remote Acquisition Units; one in each rack may be accommodated (see Figure 4.1 -9b).
- Experiment heat exchanger and one experiment dedicated cold plate; both located only in the experiment rack no.4 adjacent to the control center rack.(see Figure 4.1 -9d)
- Remote Intercom Stations ,located only in the racks no. 4, 7 and 10 shown in Figure 3-16 (see also Figure 4.1 - 9a).

The double rack is designed to accommodate two stacks of side-by-side mounted standard 19 inch pieces of equipment, while the single rack can accommodate a single stack 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 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; behind this panel are interface connector panels to the interface rack cabling with main floor cabling.Details are shown in Figure 4.1 -9c.

The rack is provided as a structural item with removable back panels, open at the front, closed sides and a bottom panel with cooling duct cut-outs. A partially removable frame is also provided which, when installed, divides the double rack into two sections.This frame is an open truss structure 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 middle frame may be removed in the lower part of the rack. All removable panels, except the front access panel at the bottom of the rack, can be removed and replaced only on the ground.

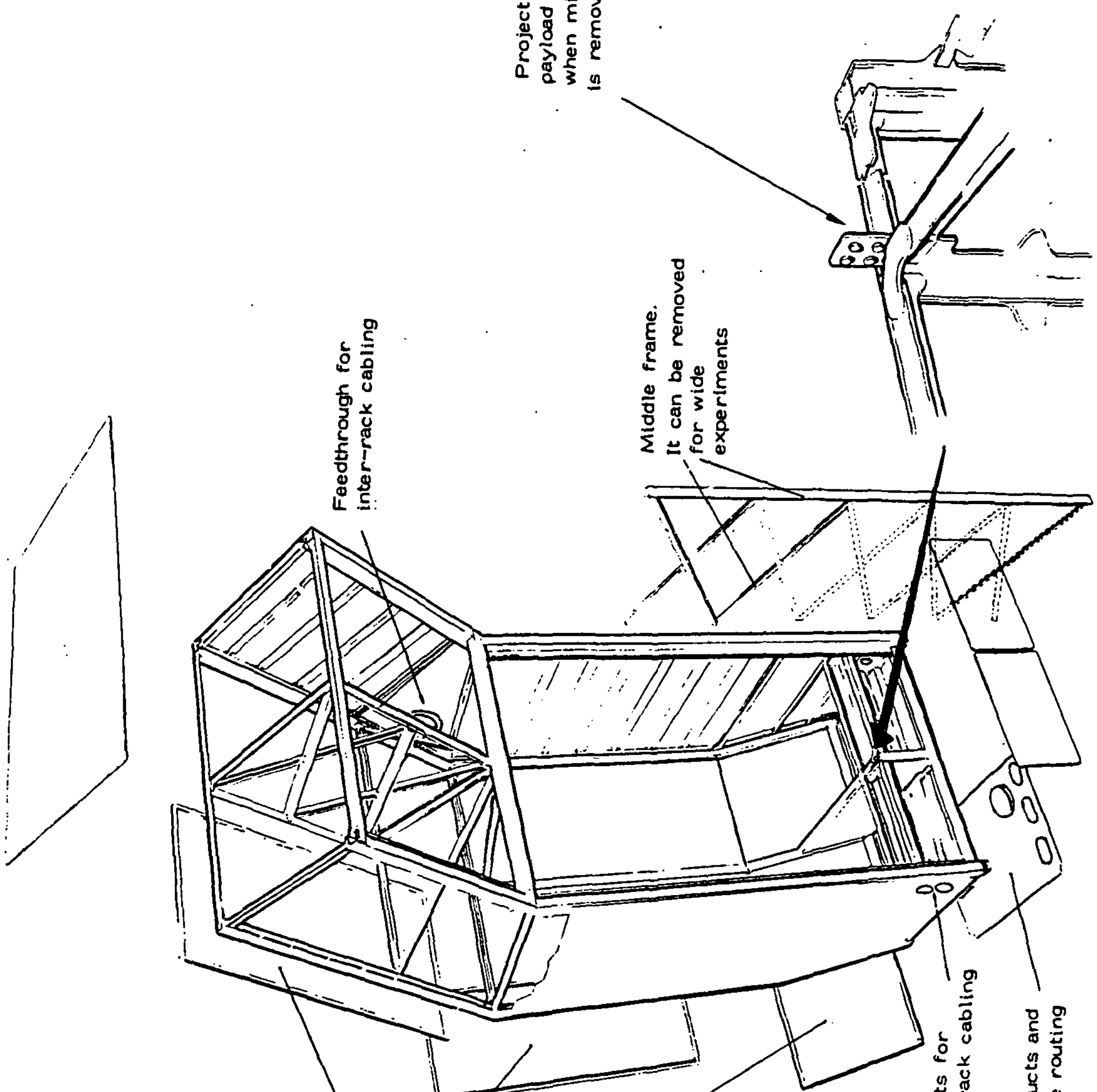


Figure 4.1-8 Exploded View of a Double Rack

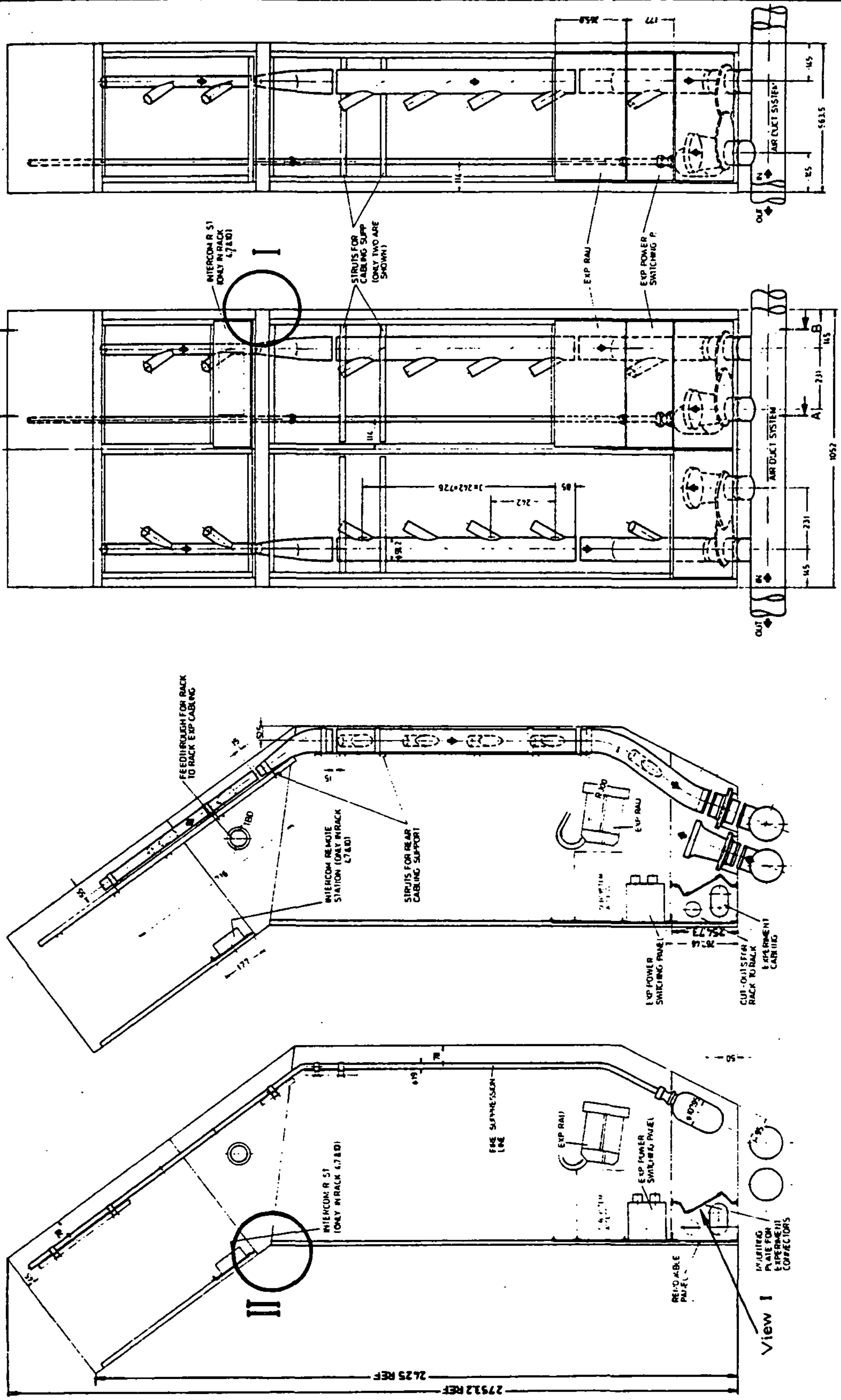


Figure 4.1 - 9a : Payload services in Standard Experiment Rack

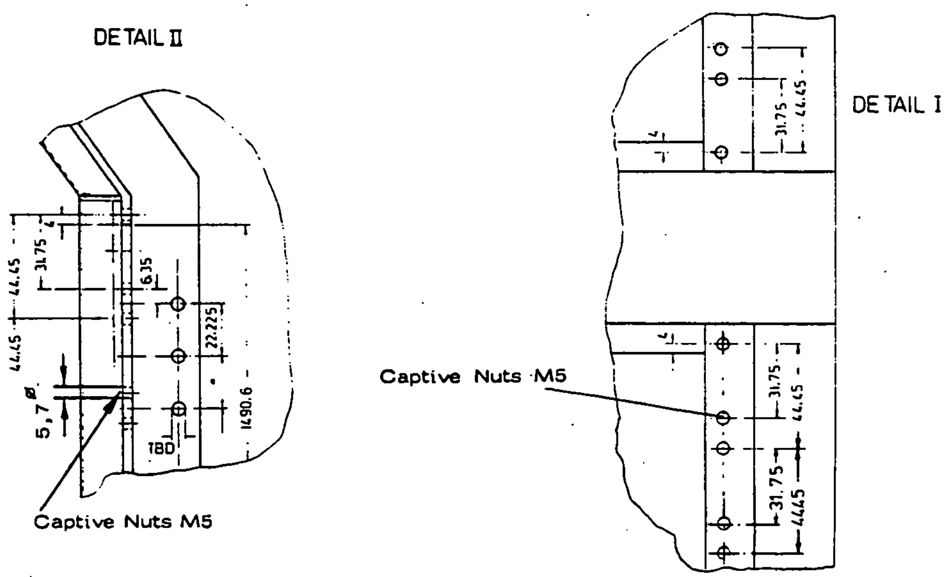


Figure 4.1 - 9b : Details of Experiment Provisions

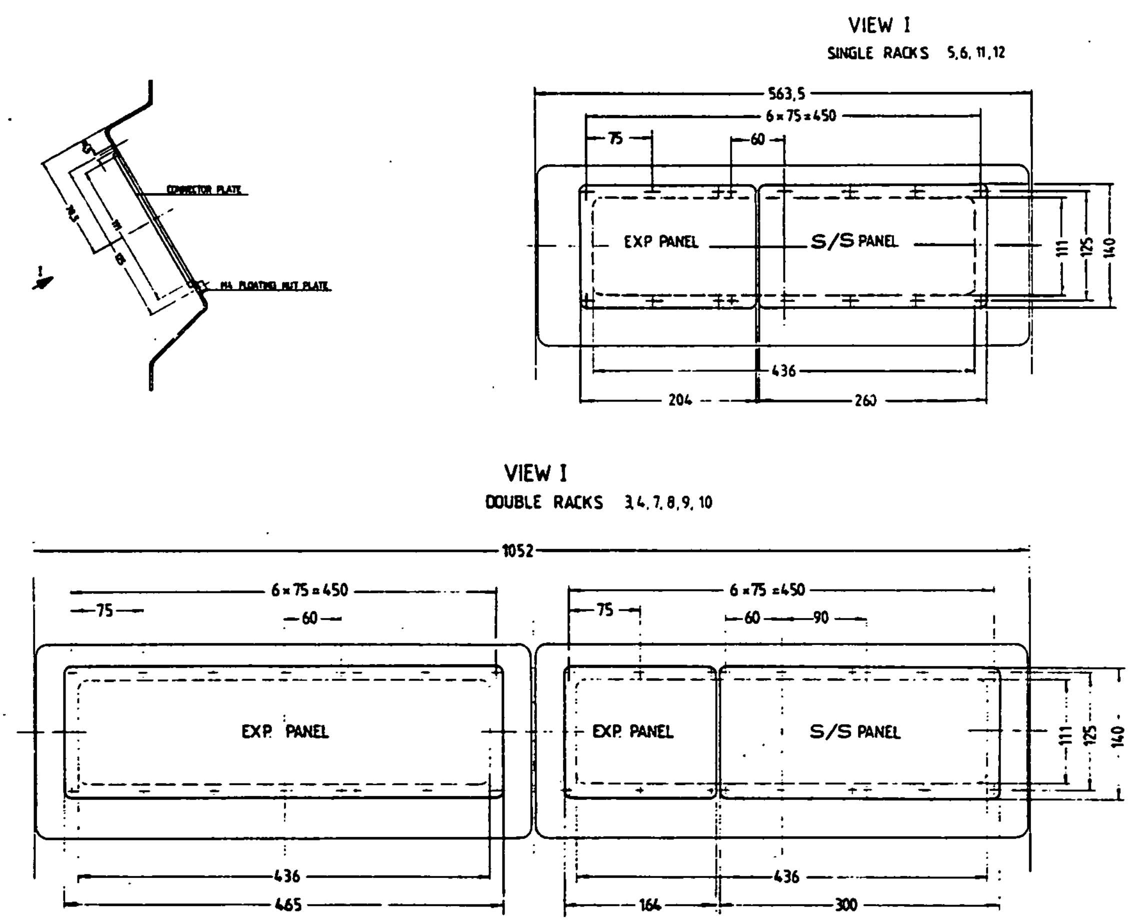


Figure 4.1 - 9c : Connector Area Available for Payloads

4.1.1.3.1 Services to Payloads within Racks

ECS Services

- Standard ECS air suction ducts in the rear part of experiment racks provide a total of five ports in the lower part and three ports in the upper (oblique) part of a single rack, and on both sides of a double rack.

  The interface to the experiments is at the stubs branching from the main suction ducts, as shown in Figure 4.1 - 9a.

  The ducting from the stubs to experiment units, including any quick disconnects as necessary to allow experiment extraction, has to be provided by the user.

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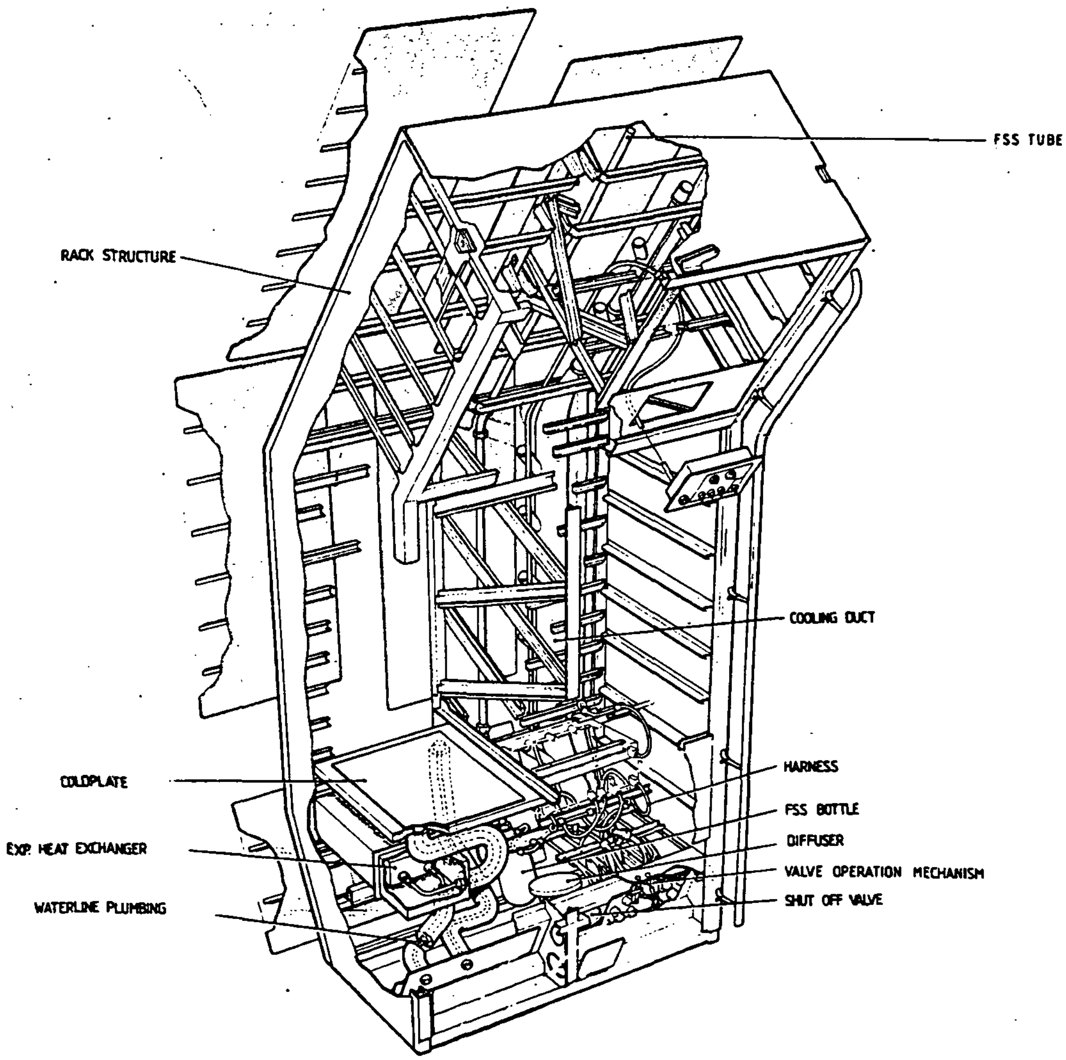


Figure 4.1 - 9d : Experiment Dedicated HX and Cold Plate in Rack No. 4 (Left Side)

Mounting of payload in the racks shall permit complete closure of the front rack aperture. Payload front panels, although not leak tight, must therefore be appropriately sealed to permit satisfactory avionic loop cooling operation. DCN 026

The capabilities of the rack air cooling system are described in Section 4.6.

- The experiment rack adjacent to the control center rack also accommodates the experiment dedicated liquid cooling heat exchanger and one experiment dedicated cold plate. The Experiment dedicated HX and the Cold Plate are mission dependent. If installed, no air duct can be mounted in the same side of the rack.
- Each Standard Experiment Rack accommodates also a fire extinguishing system, as shown in Figure 4.1 - 9a.

- EPDS Services

The interface with the EPDS is at the Experiment Power Switching Panel (EPSP); one per rack, as shown in Figure 4.1 -9c. The power outputs available are detailed in Section 4.3.

- CDMS Services

To interface with CDMS, there are provisions in each experiment rack to accommodate one experiment Remote Acquisition Unit (RAU). The location is just above the EPSP in the single racks. Figure 4.1 - 9 shows the RAU in a double experiment racks.

4.1.1.3.2 Experiment Cabling Provisions

Experiment Cabling inside Experiment Racks

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 -9a. The struts are also used to support the air cooling ducts and the fire suppression line. The number and vertical location of these struts may be varied according to payloads needs. DCN 026

Inter-Racks Experiment Cabling

Inter-rack Experiment Cable Routing

a) At the bottom of a rack through user provided connectors and lateral cut-outs as shown in Figure 4.1 - 9c.

b) At high location at the rack through inter-rack feedthroughs. These feedthroughs, shown in Figure 4.1 - 9, however, are not foreseen for connector mounting, but only for cabling transfer; when not used they are closed by a cover plate. The actual feedthrough closure will be user provided and suited to his specific needs. DCN 026

4.1.1.3.3 Standard Experiment Racks – Allowable Envelope

DCN 019

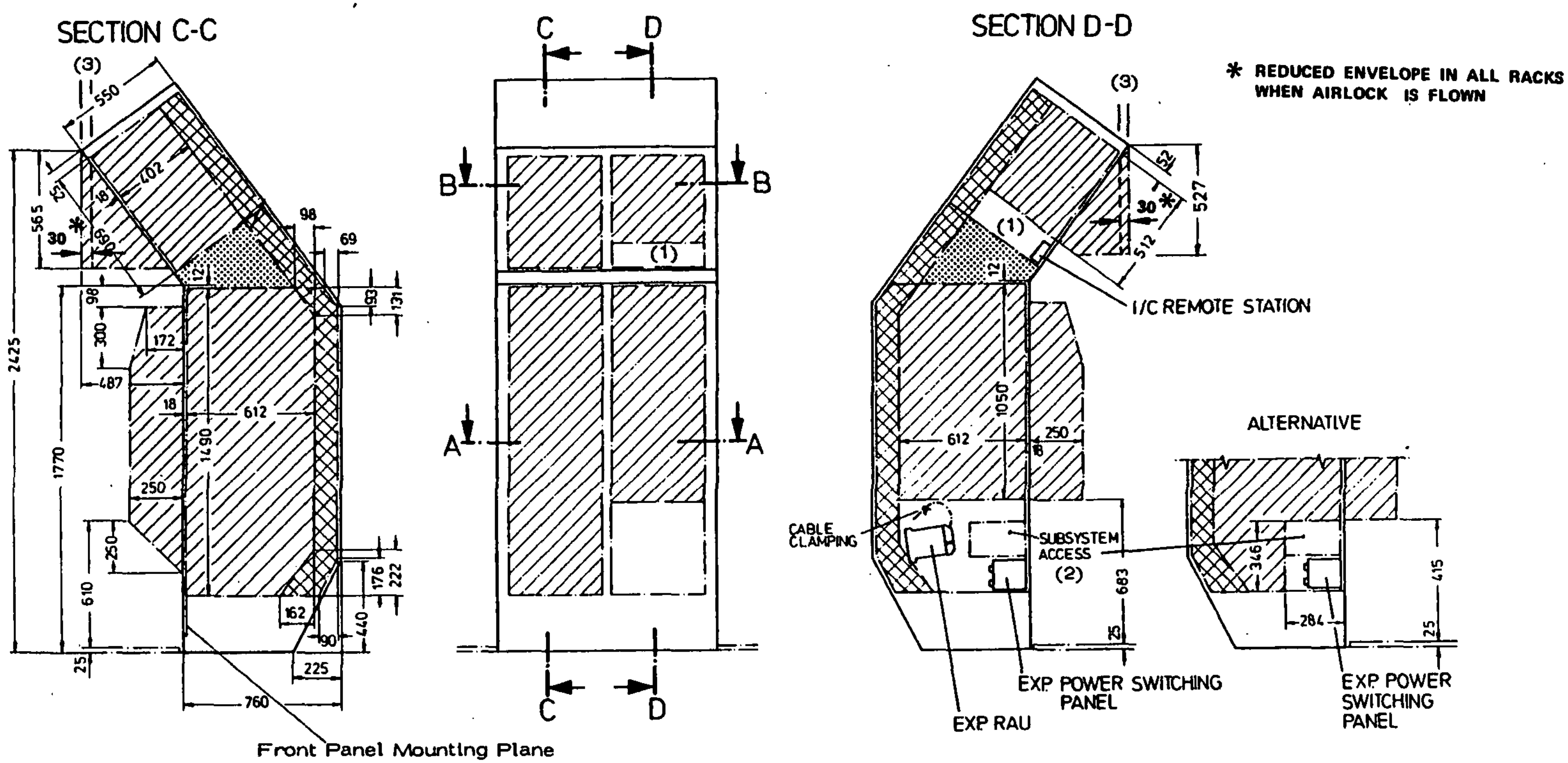
The nominal envelope for experiment and the Subsystem equipment listed in para 4.1.1.3.1 mounted in the standard experiment racks is shown in Figure 4.1 - 10. Minor protrusions of experiment equipment beyond the nominal allowable depth may be possible, if compatible with the experiment rear cabling, the spacing of the cabling support struts, and the ECS ducts.

Experiments which require no standard ECS air cooling ducts, no standard ECS fire suppression system and no rear struts for cabling attachments, may utilize the entire internal depth allowed by the basic rack structure (see Figure 4.1 - 10).

The height available for experiment (and Spacelab mission dependent) equipment is also shown in Figure 4.1 - 10.

Projections of experiments in front of the front panel mounting plane is normally limited to knobs, switches and similar small protrusions. Larger protrusions in front of the racks may be allowed, subject to case-by-case restrictions due to :

- possible interference and operational constraints with the MGSE rack-floor support braces kit.
- possible interference and operational constraints with the MGSE late access kit.
- possible interference with the floor hinged panels (Figure 4.1 - 2).
- crew habitability considerations.
- excessive aisle obstruction due to simultaneous presence of center aisle equipment.



Envelopes shown are for double racks, where the right side also represents the envelope for single racks, including the volume for the Intercom Remote Station (ICRS).

(1) Only Double Racks No 4, 7 and 10 contain ICRS's. For all other racks, the entire upper part of the rack is available for payloads

(2) When the RAU is removed, the marked up volume has to be kept clear for access to the EPDB

(3) Reduced envelope in all racks when airlock is flown

Nominal Envelope

Cable and Duct Envelope partly available to experiments requiring no struts for cabling support, or Fire Suppression Assembly (see Figure 4.1 - 10b)

This space is available for payload cabling lines and for minor protrusions to be evaluated on a case-by-case basis

Figure 4.1 - 10 a: Nominal Experiment Allowable Envelope Inside the Racks

DCN 019

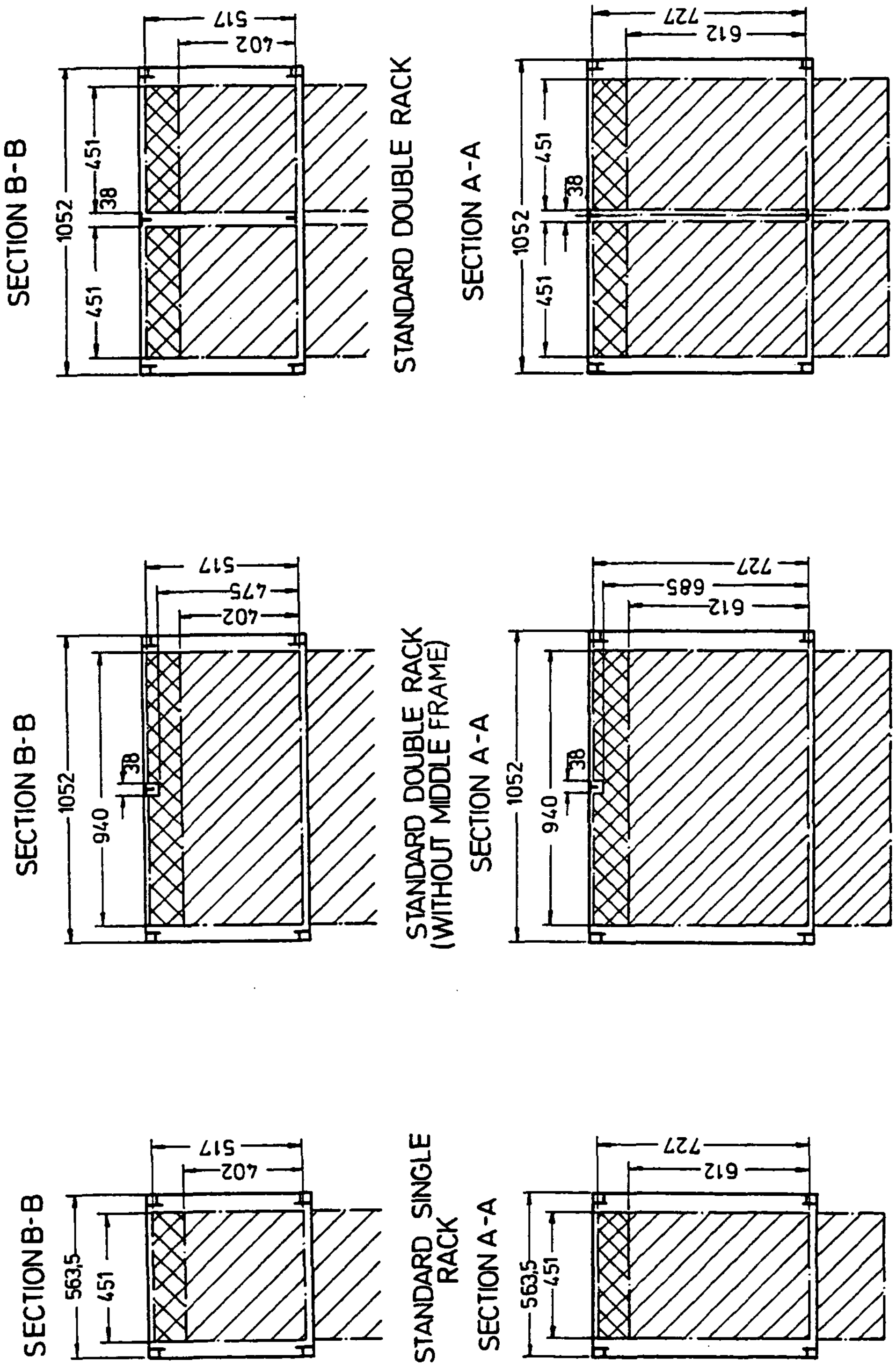


Figure 4.1 - 10 b : Nominal Experiment Allowable Envelope Inside the Racks

4.1.1.3.4 Standard Racks Carrying Capability

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

- single rack 290 kg
- double rack (overall) 580 kg
- either side of a double rack (left or right) 290 kg
- double rack (center frame removed) 480 kg

DCN 026

4.1.2 Pallet Structure

The pallet structure accommodates experiment equipment for direct exposure to space. The general structural configuration of each pallet segment is shown in Figures 4.1-11 and -13b . Pallet segments can be structurally connected to form two or three pallet segment trains as shown in Figure 4.1 - 12.

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 - 11, 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 (Figures 3 - 7 to 3 - 9, and 3 - 18), and the subsystem equipment attached to the front frame, as shown in Figure 3 - 18 and 3 - 25.

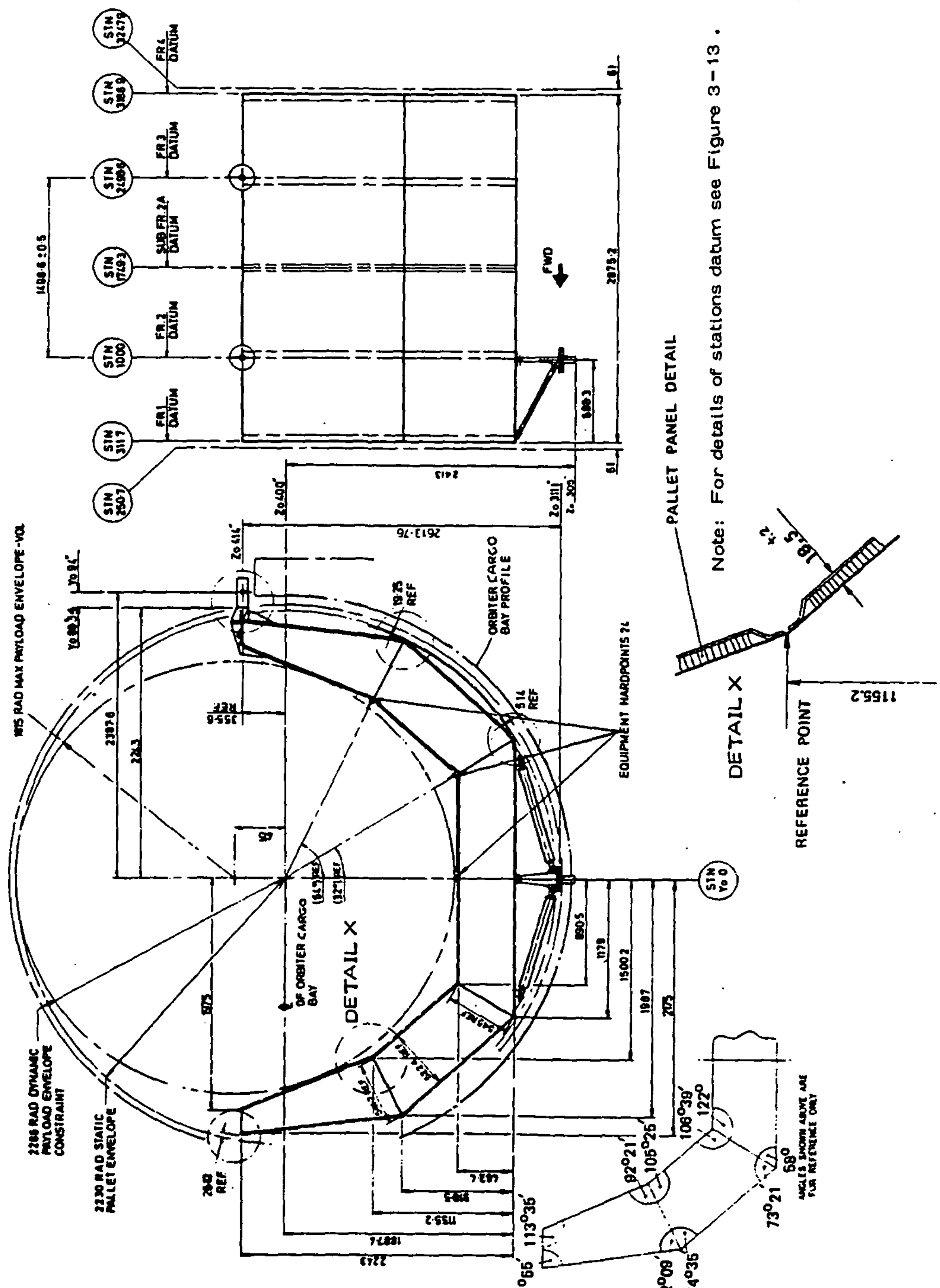


Figure 4.1 - 11: Pallet Segment Structure Geometry

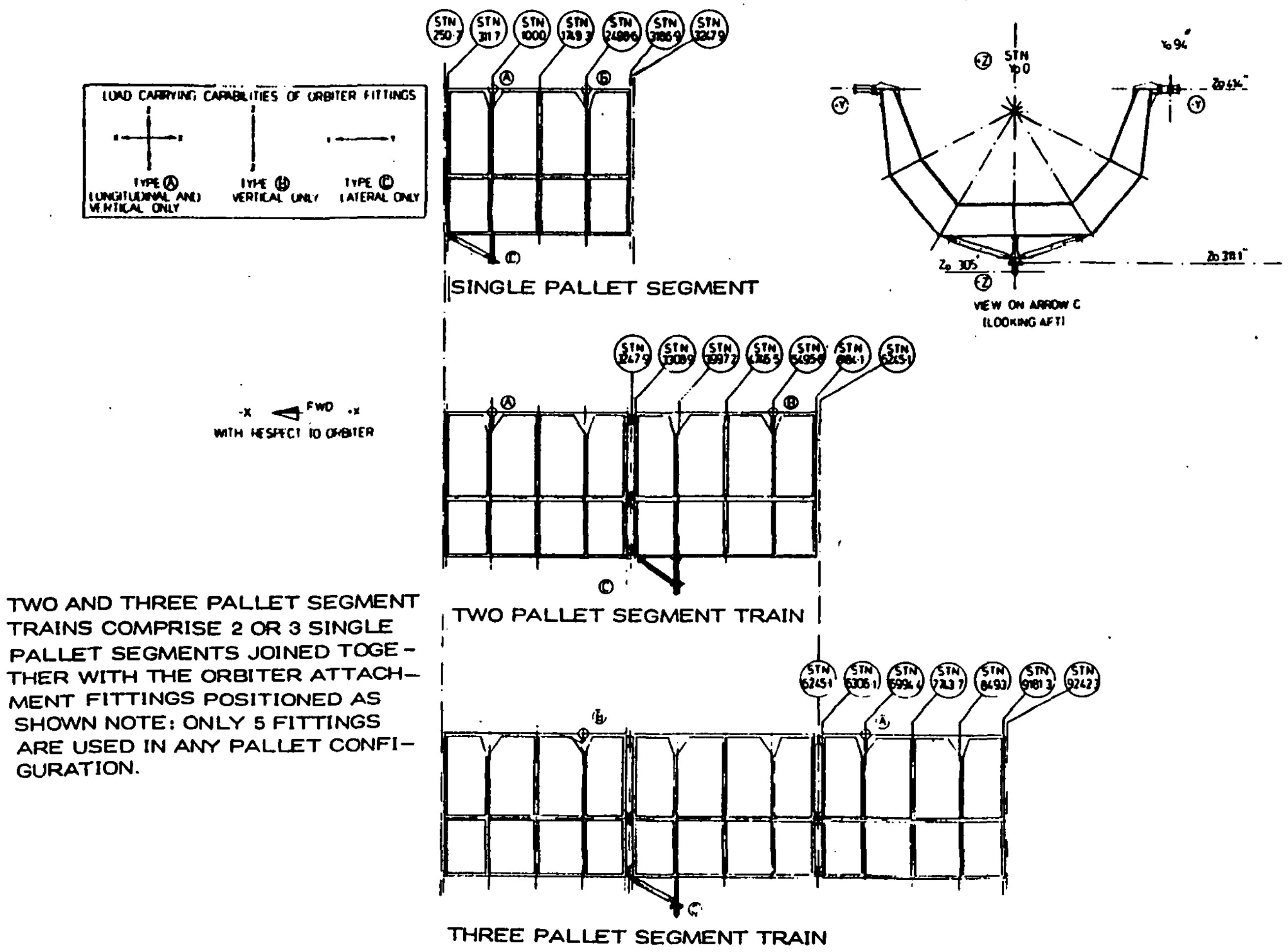


Figure 4.1 - 12 : Pallet Configurations

4.1.2.1 Pallet Panels

Experiments can be mounted on the inner skin panels. Figure 4.1 – 13 indicates the geometry of the inner panels. Actual equipment mounting is possible with inserts in the honeycomb sandwich panels. The inserts are arranged in a 140 mm x 140 mm grid. DCN 019

The inner panels can support masses of 50 kg/m^2. DCN 026
All panels are removable for pre-integration purposes. However, since all panels are structural members of the pallet structure, panels might only be removed in certain sequences. No insert pattern is provided for the outer skin panels.
No mounting provisions for experiments are foreseen at the pallet segment front and aft frames, which are reserved for subsystem mounting (e.g. igloo).

4.1.2.2 Pallet Hardpoints

For equipment masses exceeding the floor panel carrying capability, 24 standard equipment hardpoints are available on each pallet segment, located at the intersection of the frames and longitudinal members of the inner surface (see Figures 4.1 – 12 and 4.1 – 13). The actual hardpoints are inserted as shown in Figure 4.1 – 13; each hardpoint providing a spherical nut with M 20 thread, bolted to the pallet structure. DCN 019

The load carrying capability of the hardpoints is limited by the structure at the respective intersections and applies irrespective of whether the pallet is flown in a single, double or triple pallet segment train configuration.

Detailed load carrying capabilities are given in Appendix B, Structural Interface. DCN 019

It must be noted that not only the local hardpoints load capability, but also the overall mass distribution limitations must be considered, as briefly mentioned in para 4.1.2.3.

4.1.2.3 Physical Accommodation Capability

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

- The overall payload carrying capability of a single pallet is about 1000 kg/m (uniformly distributed over the pallet).

- DCN 019 The overall payload carrying capability of a 2 and 3 pallet segment train is listed in Table 3 - 7. Payloads exceeding, even considerably, the overall mass of Table 3 - 7 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-by-case basis.

  The payload mass distribution has to fullfill overall pallet segment c.g. constraints.

  Some preliminary information on the nominal c.g. range in Z_o-direction allowed for the payload design masses of Table 3 - 9 is given in Figure 4.1 - 14. Exceptions to these constraints are possible, but a special structural analysis is required for such cases.

  DCN 019 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.
  These elements, however, are not part of the present Spacelab baseline.

- A single pallet segment provides approximately 33 m^3 volume above the floor.
- The inner skin panels of a single pallet segment provide about 17 m^2 of mounting area, most of which is available for mounting payload equipment. The area for payload projected in the x-y-plane with no overhang is 11.37 m^2 .
- The pallet segment 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 into a pallet segment train.

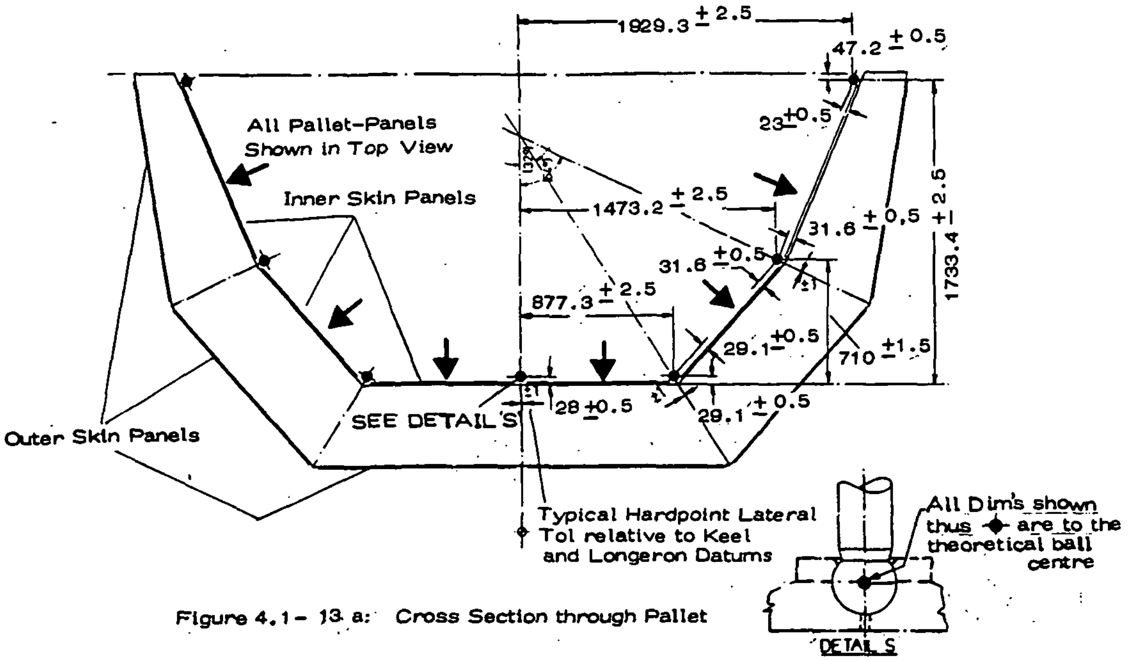


Figure 4.1 - 13 a: Cross Section through Pallet

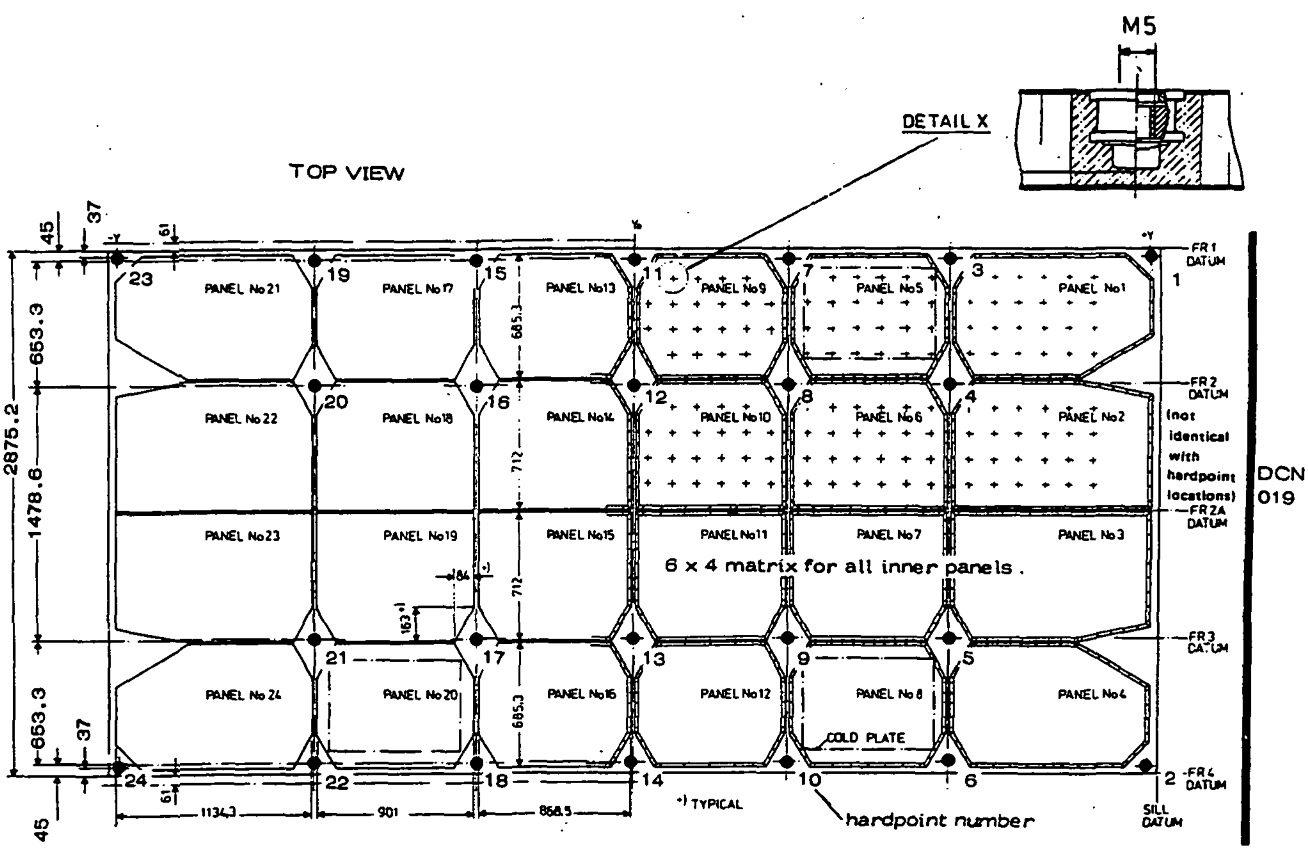


Figure 4.1 - 15 b: Inner Pallet Surface - Developed View

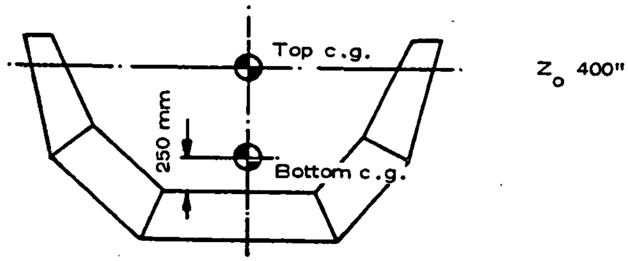


Figure 4.1 - 14: Nominal Payload C.G. Limits

4.1.2.4 Pallet Deflection

Pallet segment deflections from quasi-static inertia loads, Orbiter torsion loads and temperature effects may result in deflections in the order of 30 mm for pallet segment 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 segment or pallet segment train shall not act as a rigid bridge connection, in order to avoid unwanted secondary stresses both on the experiment and the pallet segments.

4.2 Crew Station and Habitability

The Spacelab design provides an interior lay-out for optimum 0-g crew task performance, as shown in Figure 4.2 - 1; but the mainfloor is also designed for 1-g operation. Foot restraints, handholds and mobility aids are provided throughout 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 switching off individual lights. All lights can be switched off. In the case of DC main bus power loss, one overhead lamp, which is supplied by the emergency power feeder, can be switched on again.

The acoustic, thermal and humidity environment in the module is described in Section 5.1. Tools are provided for spares replacement and contingency maintenance. Storage bags, restraining straps and writing instruments are also provided.

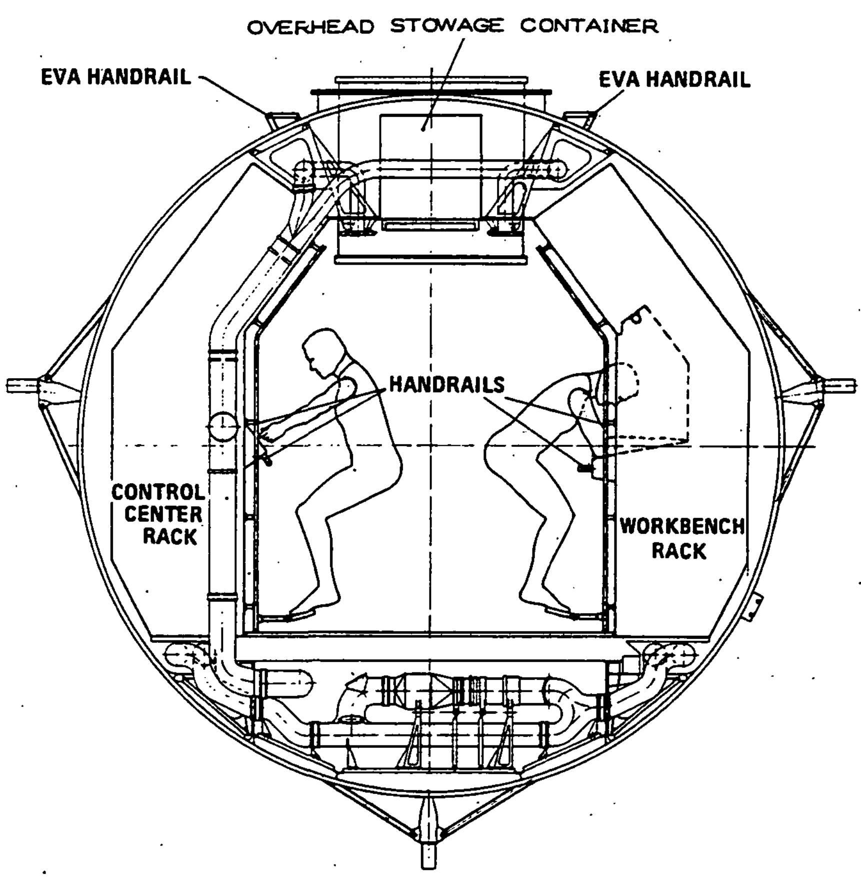
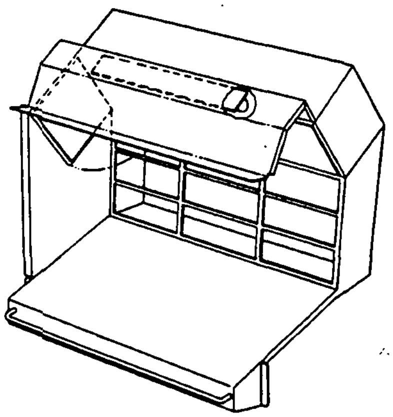
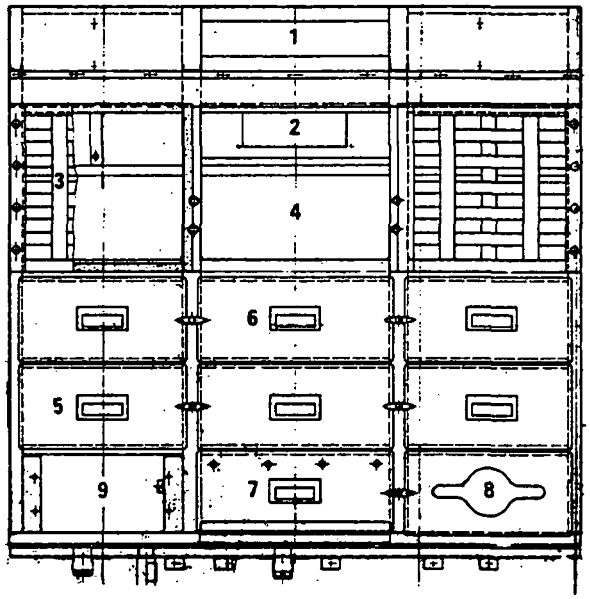


Figure 4.2 - 1: Primary Crew Working Area (Frontal View)





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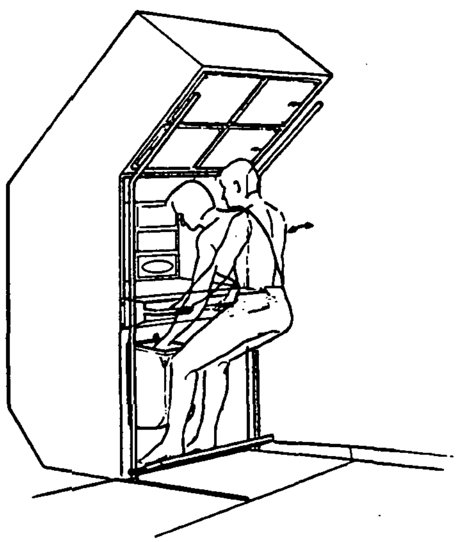

DCN 020

1 LIGHT
2 POWER OUTLET
3 CREW LOGS
4 FLIGHTDATA FILE
5 SMALL DRAWER
6 LARGE DRAWER
7 WRITING EQUIPMENT DRAWER
8 TISSUE DISPENSER
9 INTERCOM REMOTE STATION

Figure 4.2 - 2: Work Bench

4.2.1 Work Bench

A work bench is provided in the core segment. This is primarily intended to support work activities that are general in nature and not associated with a unique experiment (Figure 4.2-2), however unique experiment work at this station is not precluded. 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. DCN 020

One electrical outlet, 28 VDC - 100 W, 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.
- Intercom remote station
- File space for flight log books, etc.

DCN 020

4.2.2 Stowage Container

The stowage containers provide storage space for experiment hardware, spare parts, consumables and other loose equipment:

- 4 rack stowage containers are provided as basic equipment in the work bench and their allocation between Spacelab and experiment usage is not yet finalized, but it is expected that one container will be available for payload use on most missions (see Figure 4.2-3);
- 4 rack stowage containers are provided as mission dependent equipment for mounting in the experiment racks;
- 8 overhead stowage containers are provided as mission dependent equipment for use exclusively by experimenters. There is space for locating up to 7 of these containers in each module segment but the use of the high quality window/viewport assembly or the top airlock can reduce the number of possible ceiling containers to one in the respective segment. DCN 020 Requirements for late access to the module (on ground) through the top of the core segment will also reduce the number of storage containers possible in that segment even when the high quality window/viewport assembly is not flown.

A grid pattern of mounting holes is provided inside the containers for the attachment of internal restraints. During on-orbit access the door of each container is restrained in the open position. The characteristics are given in the following Table 4.2 - 1.

Table 4.2 - 1: Characteristics of Stowage Containers

	Size (mm) inner dimensions			Volume (m^3)	Loading capability (kg)
	height	width	depth		
Work Bench & Rack Mounted Containers	284	399	493	0.056	25
Overhead Stowage Container	521	517	302	0.081	33.5

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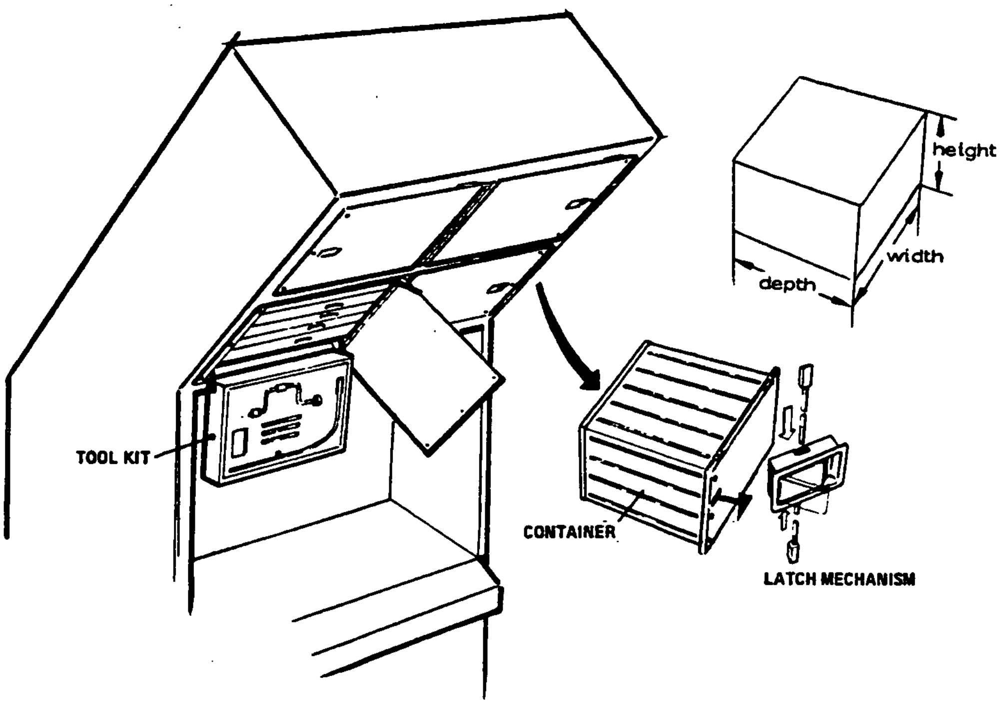

Figure 4.2 - 3: Typical Work Bench and Rack Mounted Stowage Container

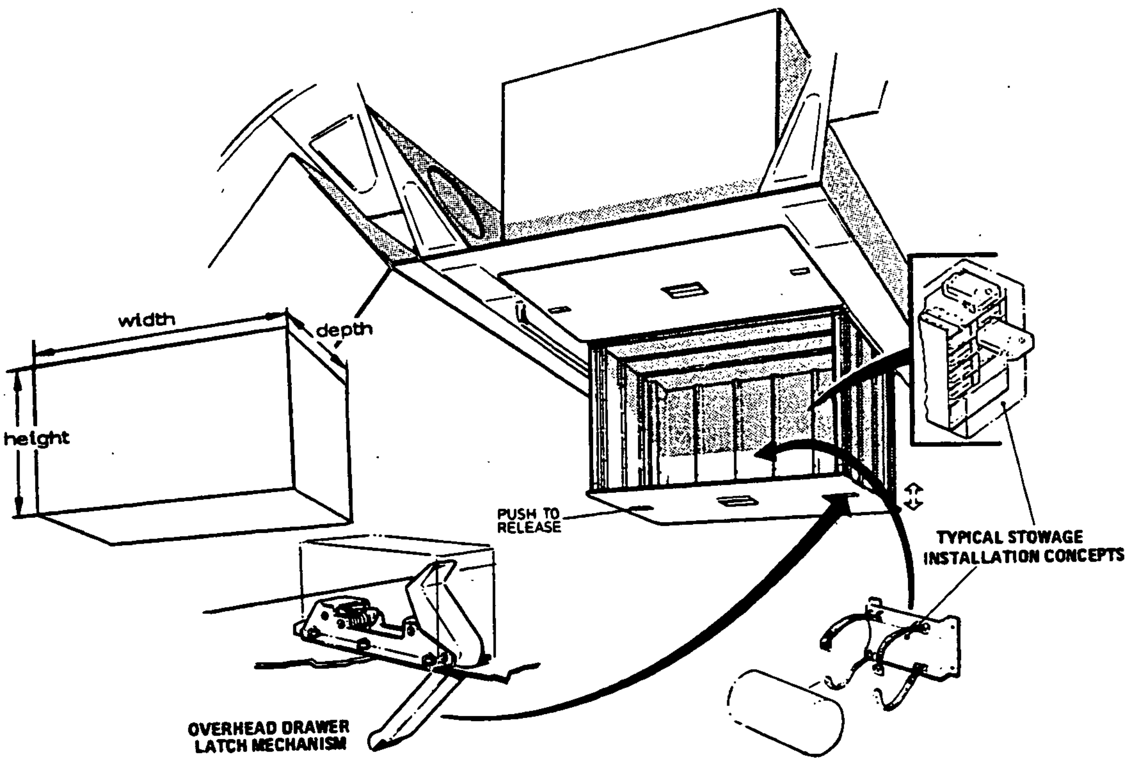

Figure 4.2 - 4: Typical Overhead Stowage Container

4.2.3 Film Storage

Films may be stored in the ceiling stowage containers and in the rack mounted stowage containers. A film storage kit is provided and consists of sheet metal separators which can be attached to the inserts inside the stowage containers as shown in Figure 4.2 - 5. The film storage kits contain sufficient separators to equip 4 overhead stowage containers and 4 experiment rack mounted stowage containers. Film may be placed in foam inserts (user provided) which can be tailored to fit each film package. User provided resealable plastic bags containing a humidity control (such as potassium thiocyanate) can be applied. DCN 020

Radiation protection for film inside the stowage containers will depend on the location of the container within the module and on the surrounding equipment (e.g. a film stowage container located near the bottom of an experiment rack will receive less radiation than a ceiling stowage container). Precise radiation levels inside the module for any mission can be determined only when the orbital parameters and the hardware composition of the mission are established. The experimenter can provide extra radiation protection for sensitive films by placing additional shielding around the individual film packages.

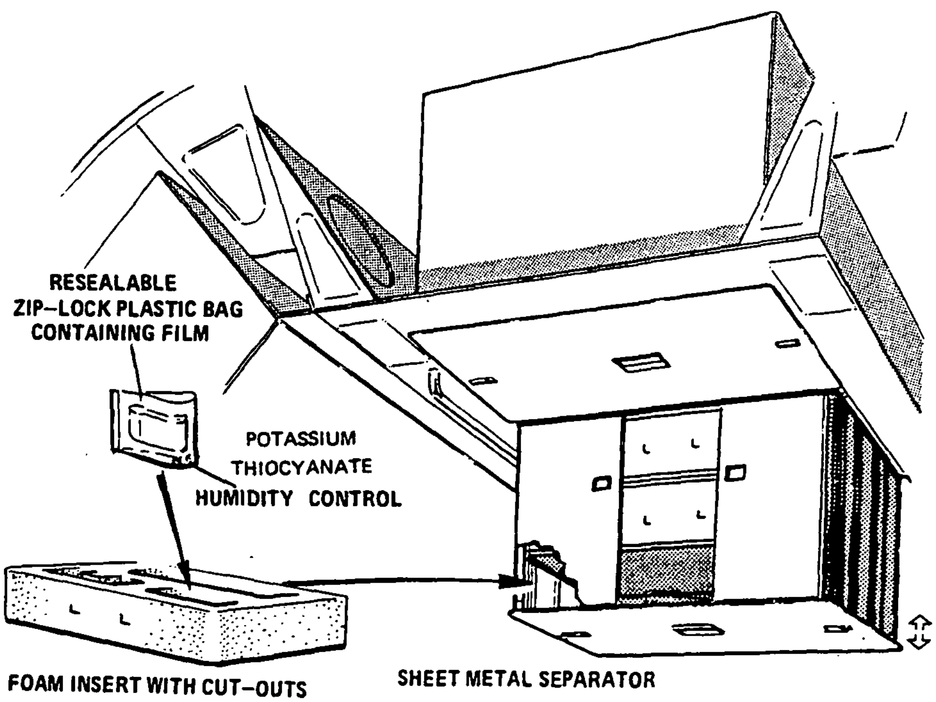

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Figure 4.2 - 5: Typical Film Storage Kit

4.2.4 Safety Equipment

4.2.4.1 Portable Fire Extinguishers

Portable fire extinguishers are provided to extinguish fire within the habitable area; they also can be used to fight fires under the main floors thru the hinged main floor access panels. Extinguishers and bracketry are of the same type as those in the Orbiter. They are mounted at both the forward and the aft end cone, as can be seen in Figure 4.2 - 6.

DCN 020

4.2.4.2 Portable Oxygen System

Portable oxygen systems plus ancillary assemblies are provided at the locations shown in Figure 4.2 - 6. Details are TBD.

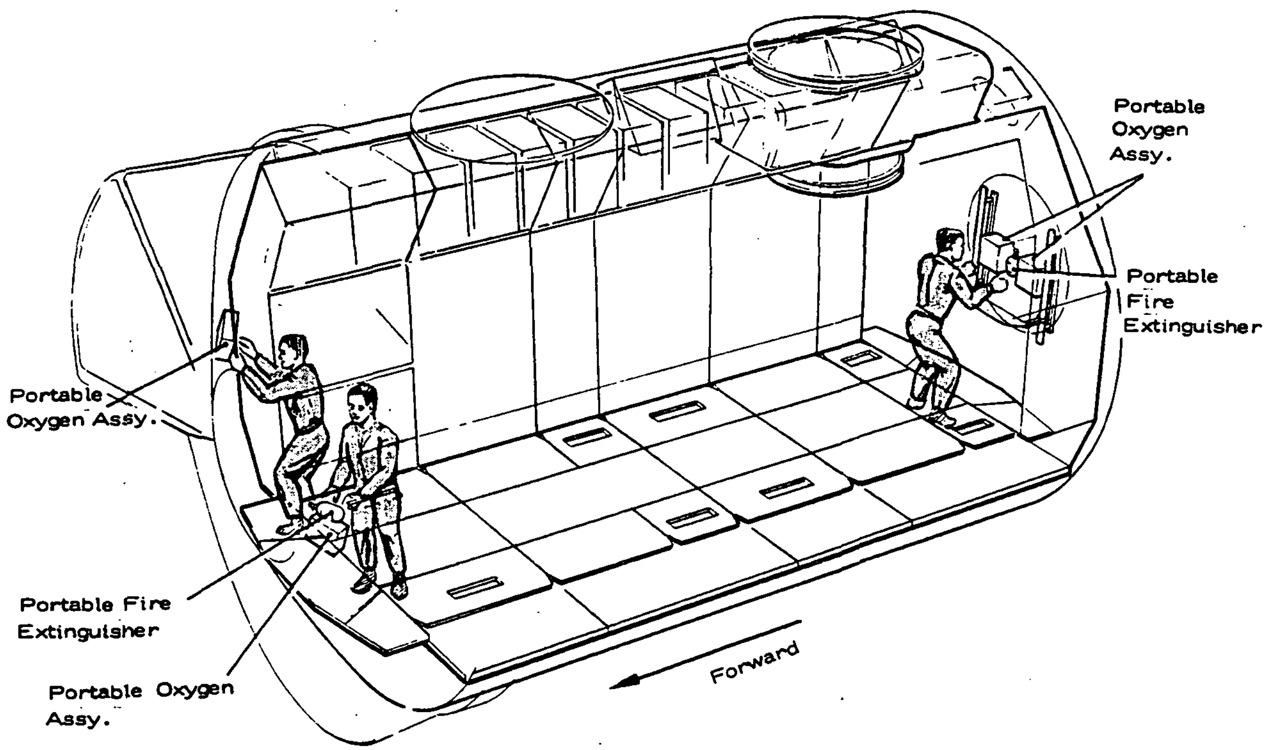

Figure 4.2 -6 : Fire Extinguishers and Oxygen Masks Located in the Module

4.2.5 Standard Equipment

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

4.2.5.1 Tool and Maintenance Assembly

A tool and maintenance assembly is provided, containing standard tools as listed in Table 4.2 - 1a. DCN 020

DCN 022

4.2.5.2 Trash Disposal Bag

The trash disposal bag shown in Figure 4.2-7 provides for collection and stowage of trash. It is also used for temporary stowage and/or transportation of loose equipment.

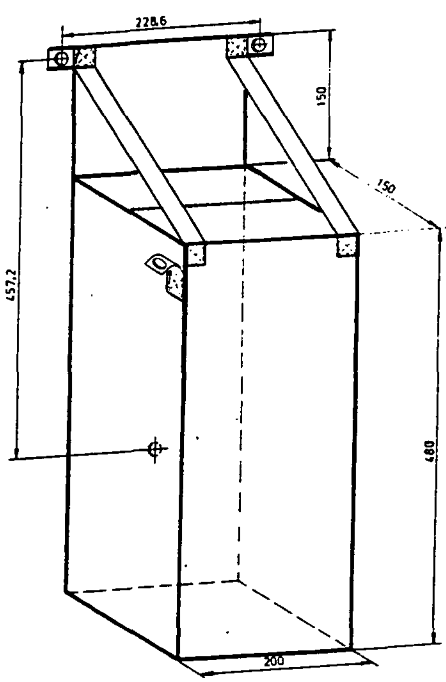

Figure 4.2 - 7: Trash Disposal Bag

Seven trash disposal bags are provided folded and packaged together at launch in a work bench stowage container. Snaps are provided in several locations in Spacelab to provide in-flight hangers for the bags while in use. Full bags are stowed in the work bench stowage containers for return (see Figure 4.2 - 8). The use of one bag per day for trash collection with two spares is assumed. The trash disposal bag will hold approximately 0.0144 m^3 of trash.

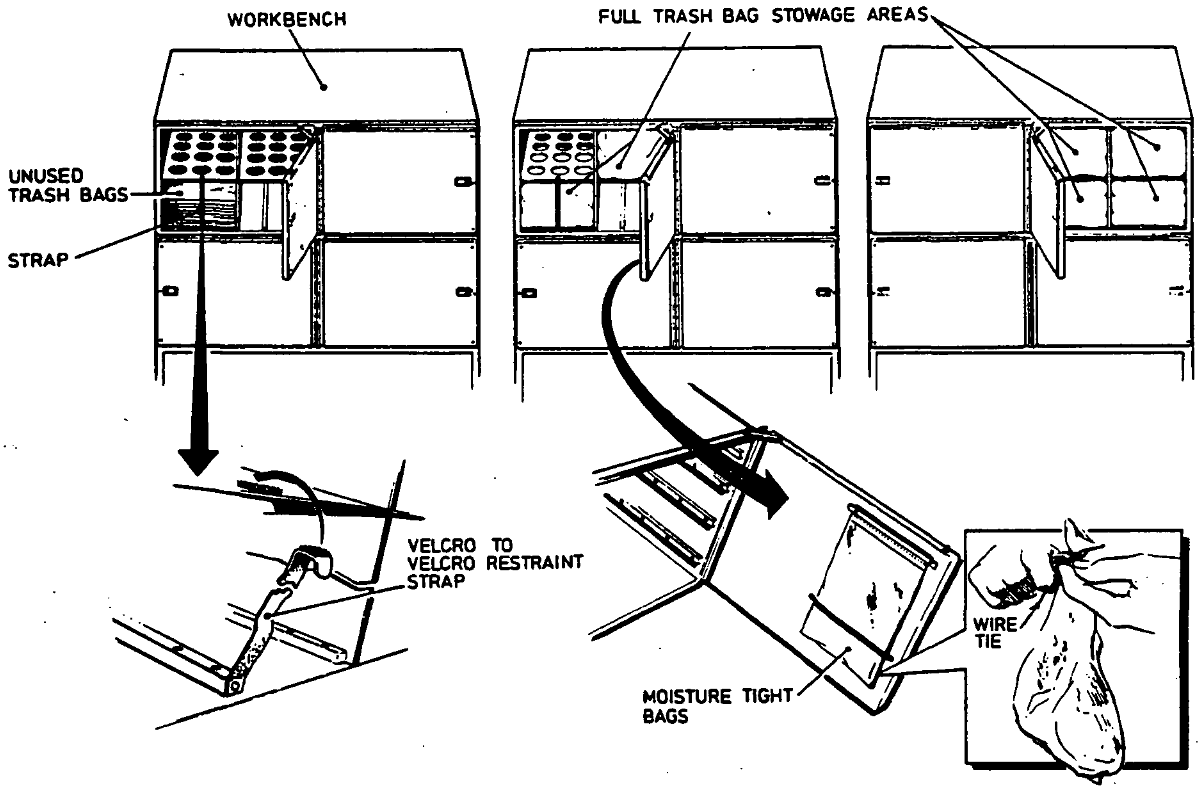

Figure 4.2 - 8: Work Bench Stowage Area

The bags are available for containing dry trash as well as controlled liquid trash (such as moist towels or sponges) but no hazardous material may be placed in the bags. For potentially hazardous waste material the experimenter must provide the hardware and a special system will be established for disposal of such waste. For reference on material control requirements see Section 7.10 of this volume.

DCN 020

For stowage of wet trash 20 waterproofed liners with simple wire tie closure will be provided.

Table 4.2 - 1a : Spacelab Flight Tool List and Maintenance Equipment

Tool	Sizes
Wrench, box and open end	8 mm 9 mm 10 mm 11 mm 12 mm 13 mm 14 mm 15 mm 16 mm 17 mm 1/4 mm 11/16 mm
Allen Wrench	1.5 mm Hex, 75 mm 2.0 mm Hex, 81 mm 2.5 mm Hex, 87 mm 3.0 mm Hex, 92 mm 3.5 mm Hex, 98 mm
Allen Wrench Set	0.05 to 0.25 1/64" Steps
Screw Driver	8 mm Blade, 150 mm 4.5 mm Blade, 100 mm 4.5 mm Blade, 25 mm 2.3 mm Blade, 75 mm
Screw Driver Phillips	No. 1, 80 mm No. 2, 100 mm No. 01, 25 mm
Phillips Offset Driver	No. 1 x 2, 100 mm
Screw Driver Bit	6 mm Hex, 33.3 mm 5 mm Hex, 4 mm Hex, 3 mm Hex, 3/16" Hex, 1/4" Hex,
Handle, Spin Type	1/4" Hex Drive
Socket to fit 3/8" Drive	1/4" Hex 3/8" Dbl. Hex 1/2" Dbl. Hex 11/16" Dbl. Hex 8 mm Hex 10 mm Dbl. Hex 11 mm Dbl. Hex
Extension	3/8" Drive, 75 mm
"	3/8" Drive, 200 mm
Handle, Ratchet *	1/2" Drive (250 mm)
Handle, Ratchet *	3/8" Drive (150 mm)
Handle, Ratchet	3/8" Drive (150 mm)
Handle, Torque	350 cm/kp, 3/8" Drive
Speeder Handle	3/8" Drive
Universal Joint	3/8" Drive
Wrench adjustable	250 mm
" "	160 mm

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* EVA-Tools, to be stored in Orbiter

DCN 022

Table 4.2 - 1a (cont'd)

Tool	Sizes
Hammer, Ball Peen	0.34 kg
Knife, Swiss Army (3)	
Mechanical Fingers	
Pin Punch	Ø 2 mm
Clamp, 'C'-Type (4)	4 "
Pinch Bar	330 mm long (20 mm Wide Blade)
Pliers, Vise Grip	175 mm
Pliers, Pin Straightener	165 mm
Pliers, Needle Nose	140 mm
Pliers, Diagonal Cutter	180 mm
Pliers, Slip Joint	8 "
Pliers, Connector	240 mm
Pliers, Channel Lock	175 mm
Scissors	160 mm
Tweezers	145 mm
Hack Saw and Blades	
Files-Large and Small Kit	
Small Bone Saw	
Dental Mirror with long handle	
Hand Mirror	
Screw and End Cutter	
Mechanics Wrench (Small pipe)	
Snap Ring Remover Inside/Outside	
Multi Meter small type	
Tape, Pressure Sen.	
Retrieval Hook	
Retrieval Mirror	
Lubricant Gen. Purpose	
Lubricant O-Ring	
Spool Assy, Twine	
Spool Assy, Wire	
EVA Tape (Wide Temp. Range)	
Glue	
Male/Female Velcro-Roll Tape Form	
Nut and Bolt Kit (Metric and American Stand.)	
Pipe Cleaner Brush Kit	
Snaps (Female) with Stick on Back (SL-Stand. Snap)	

4.2.5.3 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. Figure 4.2 - 8a shows an example of the restraints in use on the workbench desk top. DCN 020

4.2.6 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 module and on the pallet for planned and unplanned tasks.

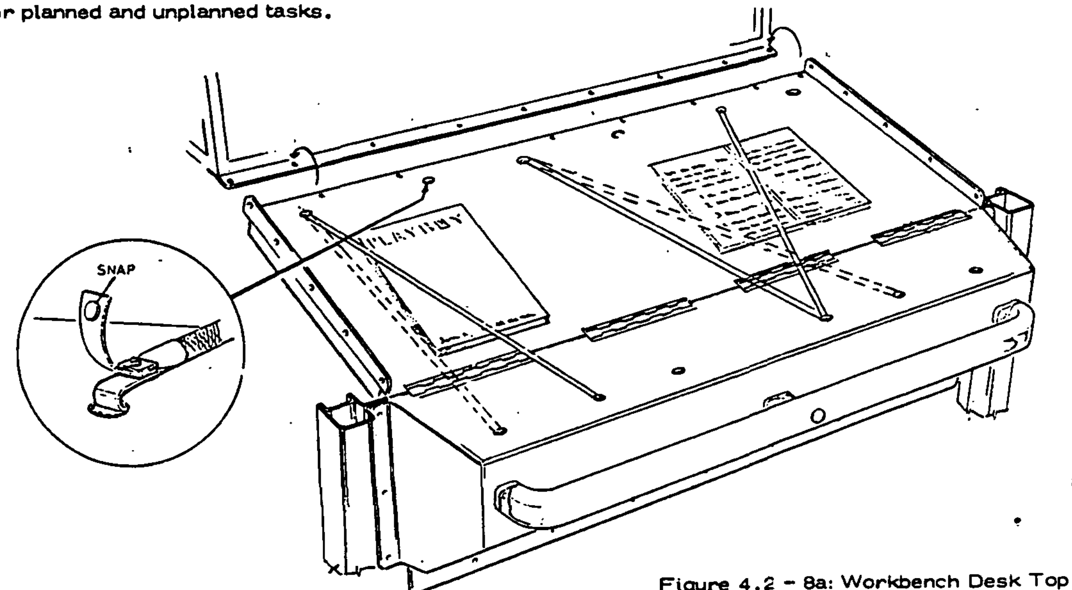

Figure 4.2 - 8a: Workbench Desk Top

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4.2.6.1 Foot Restraints

The foot restraint systems in Spacelab and Orbiter will be compatible; the basic foot restraint system used for Spacelab is the suction cup shoe. The attachment plane for the suction cup shoe is either the floor or the rack foot restraint which is a structure that can be mounted to the double rack handrails (see Figure 4.2 - 9) or two the aft end cone handrails (for operation of the airlock).

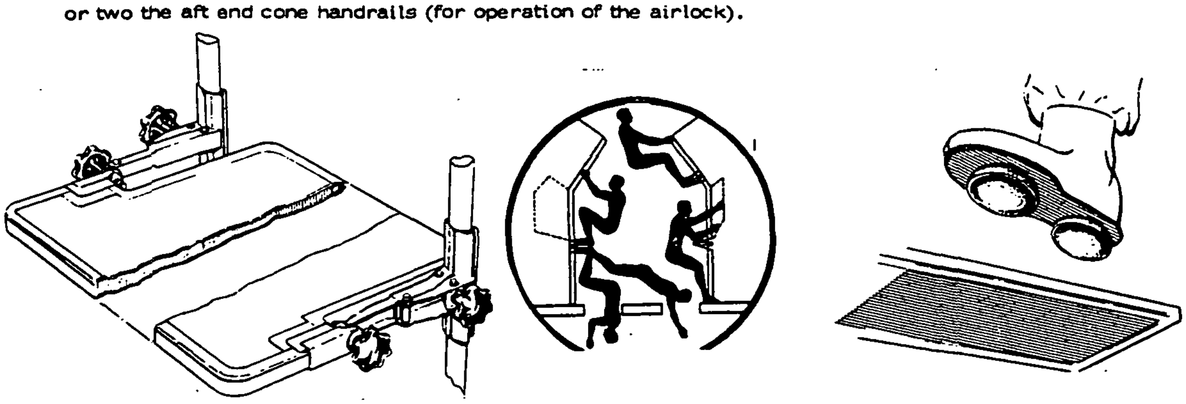

Figure 4.2 - 9: Suction Cup Shoe and Foot Restraints Positioning

4.2.6.2 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 movement through the module and also provide a means of body stabilization while performing tasks in the immediate vicinity of the handrails or handholds.

The primary standard locations of locomotion aids and handholds are:

- Vertical handrails attached to the standard racks
- Horizontal handrails attached along the overhead utilities/storage support structure

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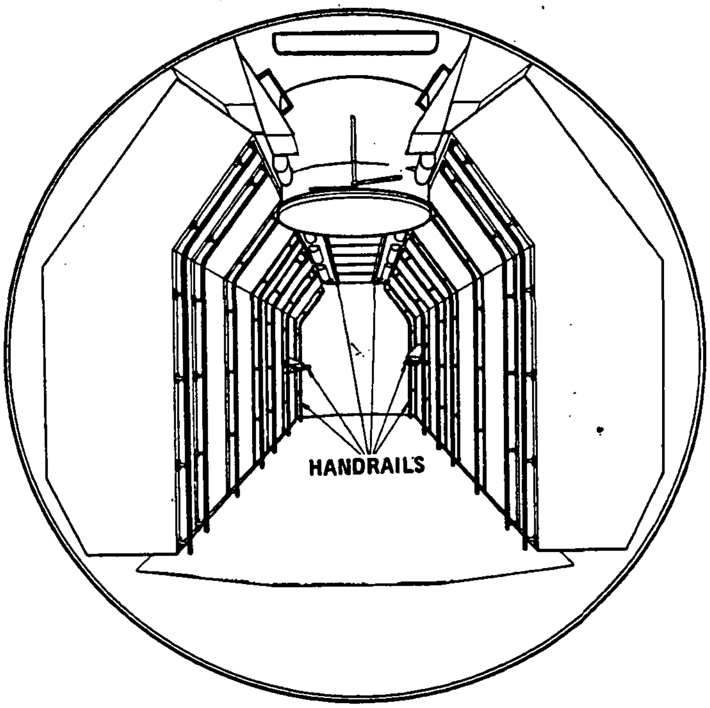

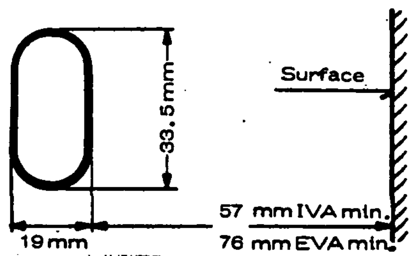

Inside Module and EVA Handhold Cross Section

Note the increased spacing in the case of EVA

Figure 4.2 – 10 Handrails Inside the Module

Other handrail locations are:

- At the inboard edge of console/work bench shelves
- Near installed airlock
- On interior structure at viewports, tunnel entrance, etc.

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4.2.6.3 EVA Restraint/Mobility Aids

EVA egress will be through the EVA airlock of the Orbiter, as described in para. 2.5.2, 2.6.1.4 and Table 2 - 17. The size of the EVA 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.

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 end cone and along the pallet as shown in Figure 4.2 - 11.

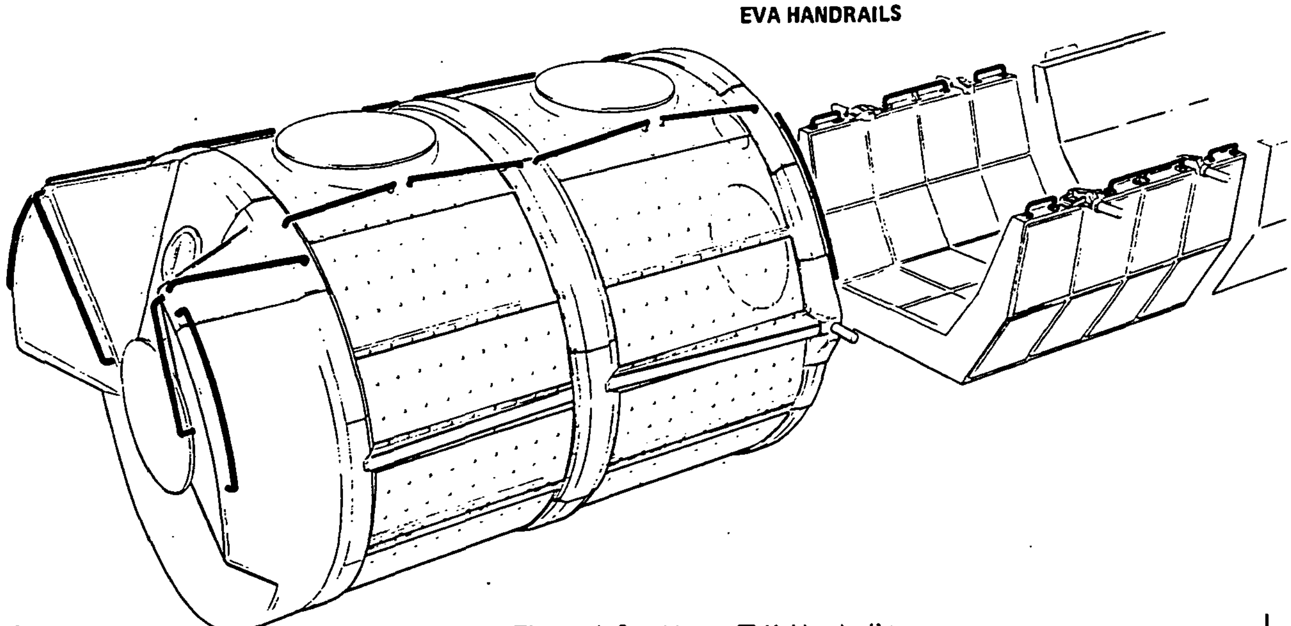

Figure 4.2 - 11: EVA Handrails

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 side of the module. The handrails on the pallet are located on the upper sill of the pallet. The standard man dimensions for EVA are shown in
DCN 020 Figure 4.2 - 12 and Table 4.2 - 2.

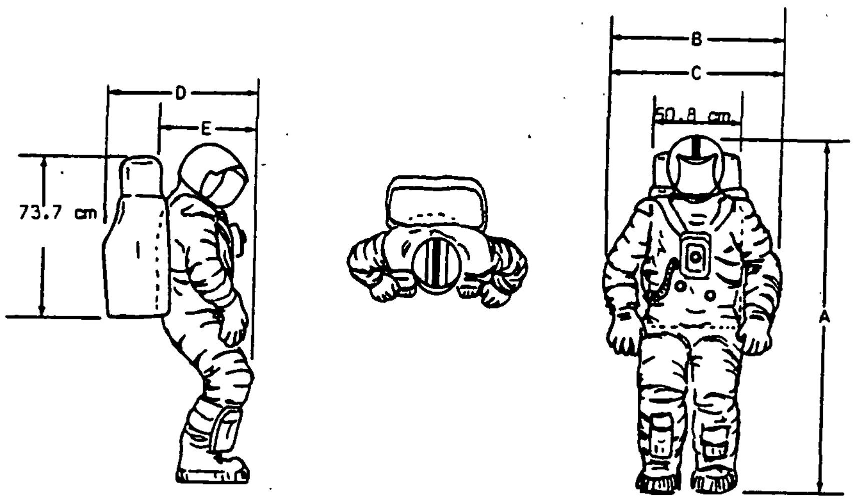

Figure 4.2 - 12: Standard Man Dimensions for EVA

Table 4.2 - 2: 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	143.3	174.8
Mass (kg), without PLSS/OPS	86.3	117.8

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4.3 Electrical Power Distribution Subsystem

4.3.1 General Description

The Electrical Power and Distribution Subsystem (EPDS) provides Main-, Essential- and Emergency DC Power and 400 Hz AC Power to Spacelab S/S and experiment equipment. The EPDS receives its primary power (28 V DC nominal) from Orbiter hydrogen/oxygen fuel cells through the Orbiter main bus system which is connected to the Spacelab Power Control Box (PCB) via a dual redundant Main DC Power Feeder. The EPDS is capable of distributing 7 kW maximum continuous (12 kW peak) to Spacelab subsystems and payload in the Orbiter cargo bay. In addition, EPDS can distribute 750 W maximum continuous (1000 W peak) in the Orbiter aft flight deck.

The total power delivered to Spacelab during different operational phases (pre-launch, ascent, on-orbit, descent) and the power available to payloads is described in detail in Section 3.3.

The EPDS equipment is summarized in Table 4.3 - 1, the experiment 400 Hz inverter is non-removable since it serves as a back-up unit for the subsystem 400 Hz inverter.

Table 4.3 - 1: EPDS Equipment

Basic Spacelab	Mission Dependent
1. Power Control Box 2. Subsystem 400 Hz inverter 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	1. Experiment power switching panels 2. Harness for Power from EPDB to EPSP Power for CPSE 3. Experiment 400 Hz inverter

The principal arrangement of the EPDS with respect to experiments is shown in block diagrams, Figures 4.3 - 1 and 4.3 - 2. The support provided to experiments is similar 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 distribution of electrical power is generally separated between subsystem and experiments.

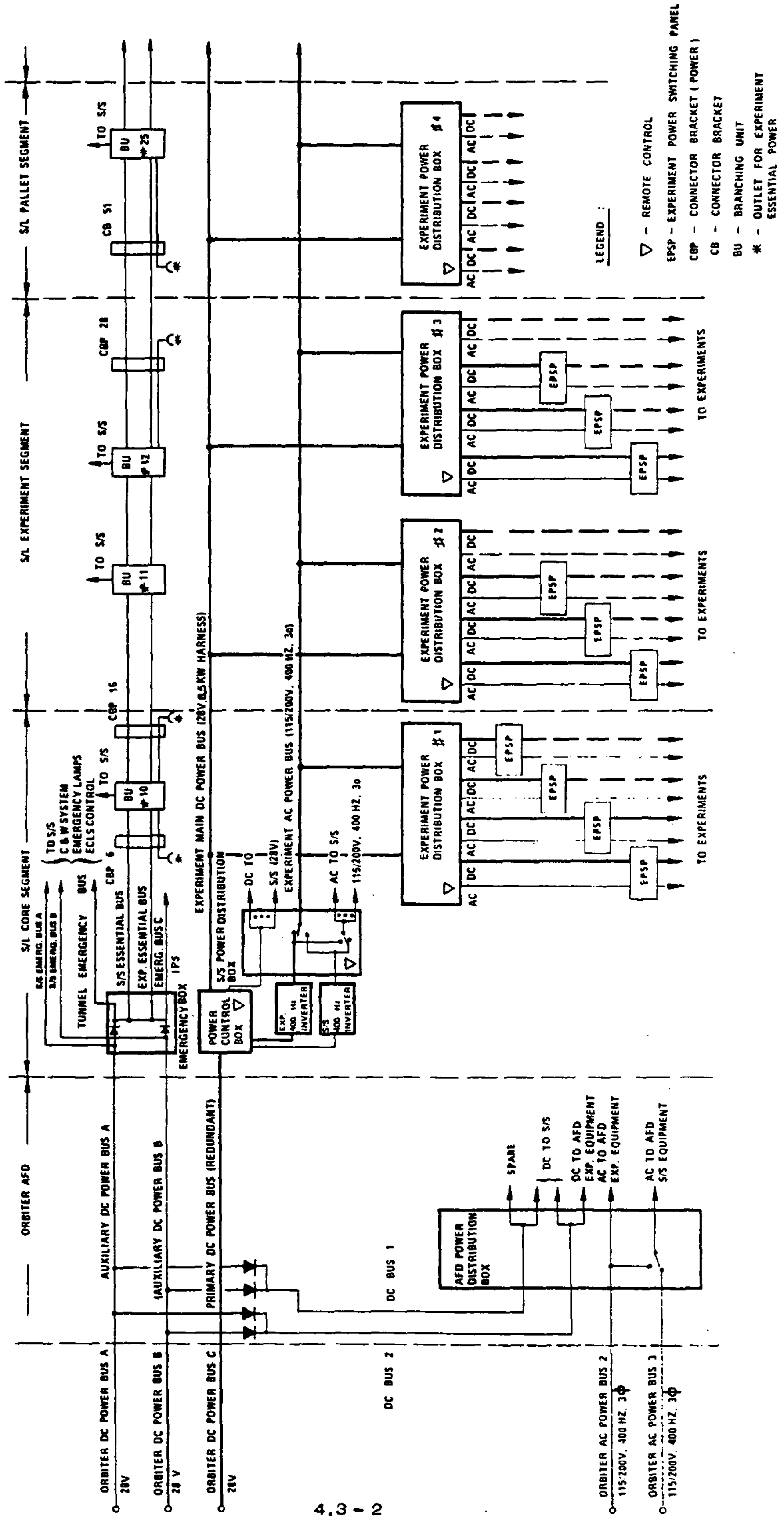

Figure 4.3 – 1: Electrical Power Distribution Subsystem for Module and Module/Pallet Mode

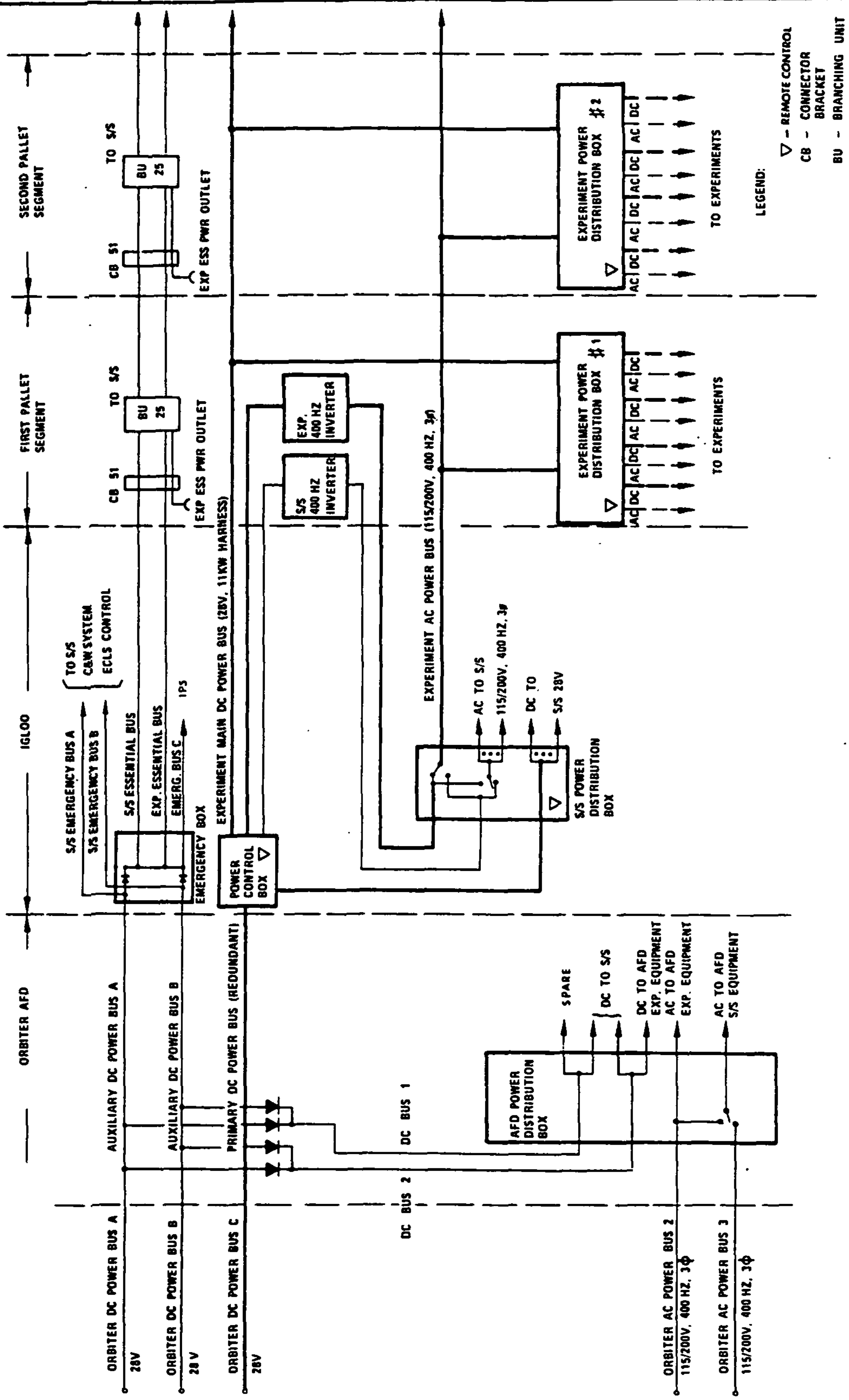

Figure 4.3 - 2: Electrical Power Distribution Subsystem for Pallet-Only Mode

The power bus system for experiment power runs through all module and pallet segments and provides the wiring for:

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

The emergency power bus will be used by experiments only to power experiment provided caution & warning equipment.

The connection of the Orbiter fuel cells to the Orbiter main bus system is managed and remotely controlled by crew actions from the Orbiter cabin. However, Spacelab normally uses one fuel cell in a dedicated mode.

In case of failure of the fuel cell dedicated to Spacelab, the Orbiter can be manually reconfigured within less than 5 minutes to provide degraded power (5 kW max. continuous) to Spacelab. (For power budgeting and power allocation during different mission phases and modes, see para. 3.6.3).

Dedicated Spacelab S/S and experiment DC/AC inverters, which receive their primary power from the main DC power bus via the Power Control Box, supply three phase AC power (400 Hz, 115/200 V) to Spacelab S/S and experiment equipment.

Electrical power for S/S and experiment equipment located in the Orbiter AFD is provided by the Spacelab AFD Power Distribution Box (AFDPDB).

4.3.2 Electrical Power Resources Available to Experiments

Electrical AC and DC power for experiments is available in the Spacelab module/pallet and in the Orbiter AFD. Concerning grounding, the module/pallet power and the AFD power are independent.

4.3.2.1 Power in the Spacelab Module/Pallet

- DC Primary Power

  This power is directly received from the Orbiter Power Bus C and is not stabilized but is protected against overvoltage by a Spacelab provided shunt regulator. The characteristics are:

Voltage:	28 ± 4 V*
Harness Layout:	8.5 kW in Module and Module/Pallet configurations 11 kW in Pallet only configurations

*23 V worst case condition, if 11 kW is drawn on last two pallets.

- AC Power

Spacelab provides a dedicated S/S and a dedicated experiment DC/AC inverter which are located in the Control Center Rack for Module and Module/Pallet configurations and on the pallet front frame for Pallet-only configurations. These inverters receive their input DC power from the DC Primary Power Bus via the Spacelab Power Control Box.

However, as shown in Figures 4.3 - 1 and 4.3 - 2, it is possible to connect the AC experiment bus to the S/S inverter and, conversely, the S/S AC bus to the experiment inverter. This feature provides flexibility in case of inverter failure.

The characteristics of the AC power are:

Outputs:	3 phases plus neutral line
Voltage line-to-neutral:	115 V rms $\pm$ 5 %
Frequency:	400 $\pm$ 1 Hz
Max. continuous power:	2.7 kVA per inverter
Peak power:	3.5 kVA for 120 s , (14 A peak absolute max. for each phase)

AC power drawn and inverter losses have to be deducted from the total primary power of 7 kW max. continuous, 12 kW peak delivered to Spacelab.

The nominal inverter losses are 175 W plus 10% of the power drawn. The actual losses depend on DC input voltage and load characteristics, and are described in Appendix A, Section 3.

- Experiment Essential Power

Experiment essential power is received from the Spacelab Emergency Box. This Emergency Box is powered by Orbiter auxiliary power through redundant auxiliary feeders A and B. The Emergency Box provides for Emergency Power, S/S Essential Power, and Experiment Essential Power. The characteristics of the Experiment Essential DC Power are

Voltage:	22.0 - 32 V in the Module
	21.5 - 32 V on Pallets

The power that can be drawn from the Emergency Box is 400 W in total. This power has to be deducted from the 7 kW primary power. Contrary to the primary DC power, the essential power is available during all operational phases.

- Experiment Emergency Power

The Emergency Box supplies a separate emergency power bus (S/S Emergency Bus C) which will be used to power IPS safing command actuators (ECR 00 157 A). It is under discussion to draw experiment emergency power (dedicated for experiment caution and warning sensors and safing command actuators) either from Bus C or from Subsystem Essential Bus.

The power characteristics are the same as for Experiment Essential Power.

4.3.2.2 Power in the Orbiter AFD

In the Orbiter AFD DC and AC power for Spacelab subsystem equipment and experiments is available through the AFD Power Distribution Box (AFD PDB). The total power available from the AFD PDB (DC and AC) is 750 W max. cont. power and 1000 W peak power (15 min/3 hours). This power is in addition to the 7 kW primary power.

- DC Power in the AFD

  DC power for experiments in the AFD is provided by a dedicated output of the AFD PDB. This output is fed directly by the DC Power Bus 2 which is connected to the Orbiter auxiliary power buses A and B.

  The characteristics of this experiment DC power are:

Voltage:	25.7 - 32 V

- AC Power in the AFD

  AC power for S/S and experiment equipment in the AFD is provided by Orbiter DC/AC inverters via the Spacelab AFD Power Distribution Box. This power is available on Orbit after main engine cut off and has the following characteristics:

Output:	3 phases plus neutral line
Voltage line-to-neutral:	115 $\pm$ 5 V rms
Frequency:	400 $\pm$ 1 Hz synchronized to the Orbiter Master Timing Unit
	400 $\pm$ 7 Hz free running

4.3.3 Experiment Power Distribution

The main electrical DC and AC power for experiment equipment in the module and on the pallet is distributed from 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 4.3 - 3). The EPDB's are attached to the main floor structure in the module and to cold plates on the 48° panels of the pallets. Within the module, DC and AC power is routed from the EPDB's to Experiment Power Switching Panels (EPSP's) located in each experiment rack (single or double rack, as shown in Figure 4.3 - 3). Inside the module the experiments normally interface with the Experiment Power Switching Panels.

As depicted in Figure 4.3 - 3, EPDB No. 1 supplies via dedicated outputs (11 through 14; for output characteristics see Section 4.3.1.1, Figure 4.3 - 5) the Experiment Power Switching Panels located in Rack No. 3, 4, 5, 6.

Power to experiments in the center aisle of the core segment can be drawn via the center aisle connector bracket CBP/S 31. The experiment dedicated power outlets are connected to the AC 13 and DC 13 outlets of EPDB No. 1. In this case Rack No. 5 cannot be supplied directly from EPDB No. 1.

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In the long module mode EPDB No. 2 supplies Rack No. 8, 10 and 12 resp. EPDB No. 3 supplies Rack No. 7, 9, 11 and CBP/S 32. EPDB No. 2 has the outlets AC 24 and DC 24 available, which may be used directly by experiments. In this case the experimenter has to provide the harness between these EPDB outlets and the experiment.

The EPSP in Rack No. 4 supplies via CBP 26 experiment equipment in the Airlock or experiment equipment attached to the High Quality Window/Viewport Assembly in the module experiment segment.

Experiment equipment attached to the aft end cone viewport will be supplied with power from an EPSP in Rack No. 10 via CBP 28. In both cases dedicated DC and AC outputs of an EPSP (see Figure 4.3 - 6) will be used.

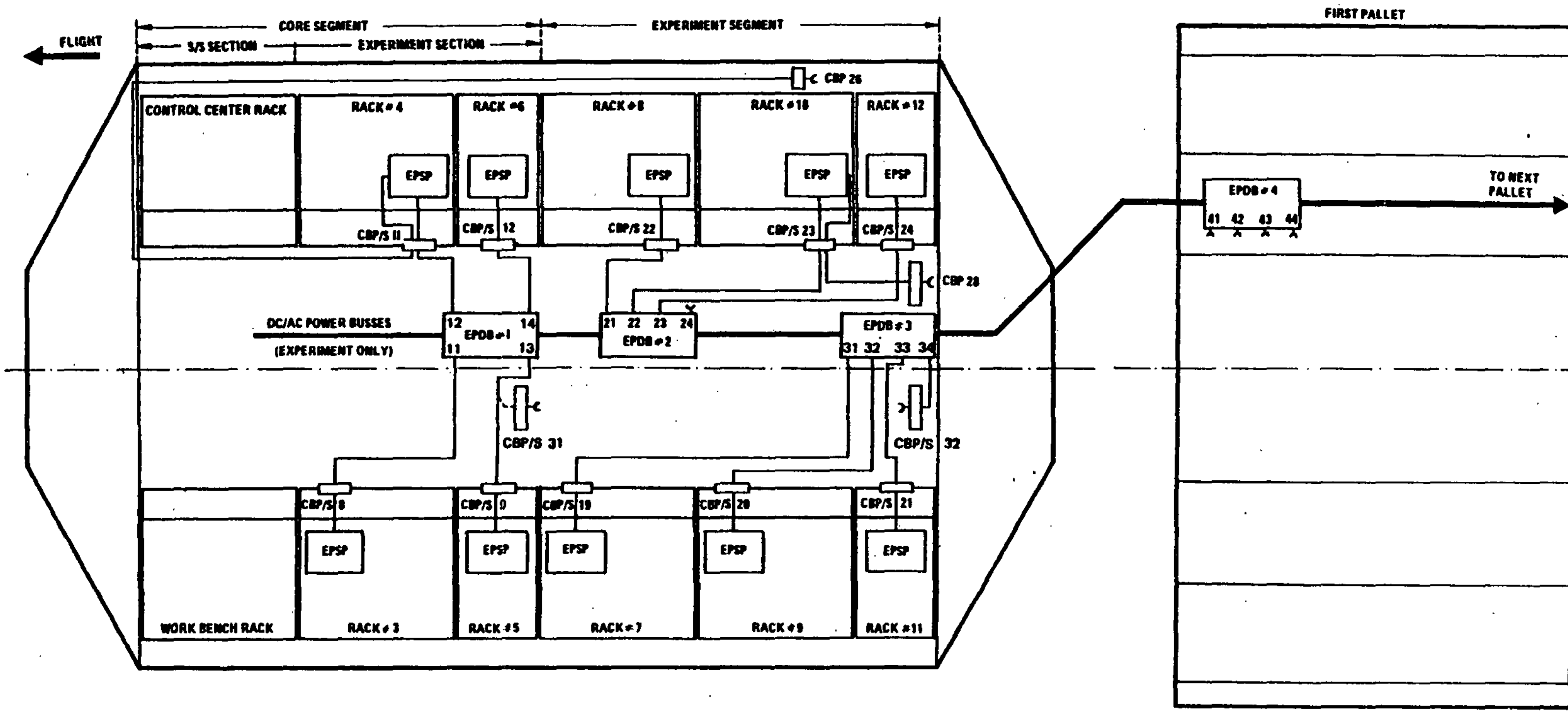
EPSP's may be attached directly to experiments which do not use the Spacelab experiment racks. In this case mounting provisions for the EPSP and any extra necessary cabling between the EPSP and the EPDB are experimenter provided. If the EPSP's are not used the experimenter has to provide any necessary cabling connectors, etc. to interface with the EPDB outlets. Experiments mounted in the Spacelab racks but which do not use the EPSP can interface with the EPDB outlets via the power connectors located at the CBP/S-XX rack connector brackets (see Figure 4.3-3).

On the pallet, experiments interface directly with the EPDB's, one on each pallet segment.

For critical experiment and S/S equipment which needs power during all mission phases and also when only degraded power is available for Spacelab a dedicated essential power bus is routed through Spacelab. This power bus receives its essential power from two outputs of the Spacelab Emergency Box, which is powered by the Orbiter auxiliary power. The interfaces to experiments are two connectors in the core segment and one connector in the experiment segment of the module and one on each pallet segment (see Figure 4.3-4). In the module these connectors are located on connector brackets below, and accessible through, the main floor. On the pallet the connector is located on a pallet connector bracket.

To supply experiment caution and warning sensors and safing command actuators with power, the experiments may draw power from Emergency Bus C or S/S Essential Bus. Details are TBD.

DC and AC power for AFD equipment will be supplied by the AFD Power Distribution Box. However, there is no auxiliary power available for experiment equipment in the AFD.



Legend

EPDB: Experiment Power Distribution Box
EPSP: Experiment Power Switching Panel
CACB: Center Aisle Connector Bracket
CBP: Connector Bracket (Power)
CBP/S: Connector Bracket (Power & Signal)

Figure 4.3 – 3: Interface Locations for Experiment Main Power in the Module and on Pallets

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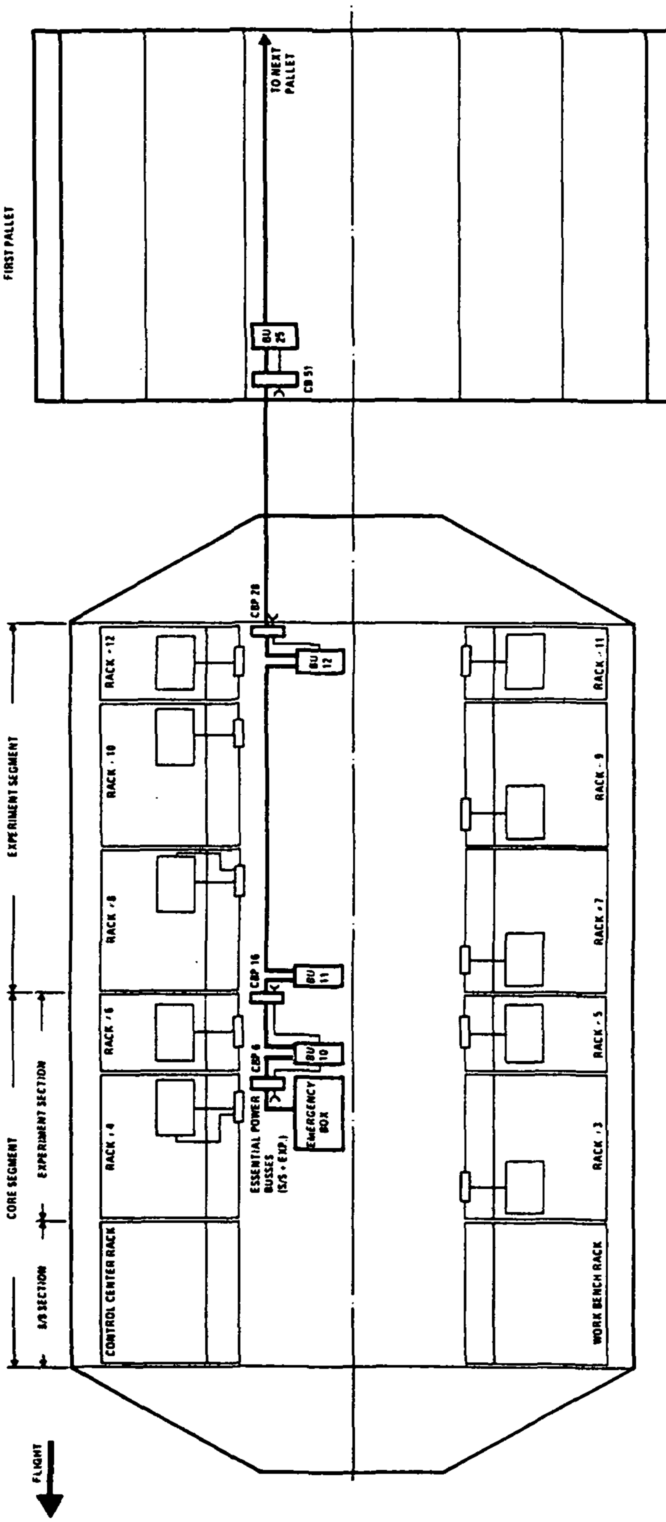

Legend

BU: Branching Unit
CB: Connector Bracket
CBP: Connector Bracket (Power)

Figure 4.3 - 4: Interface Locations for Experiment Essential Power in the Module and on Pallets

Detailed information about signal characteristics for DC and AC Main Power, Essential Power, and Emergency Power, as well as drawings of experiment dedicated connector brackets, connector notation and pin allocation of those connectors, which may interface with experiments, are given in Appendix A, Avionics Interface Definition, Section 3.

The exact location of experiment dedicated power connector brackets is given in Appendix B, Structure Interface Definition.

4.3.3.1 Experiment Power Distribution Box

The Experiment Power Distribution Boxes supply the following outputs (see Figure 4.3 - 5):

DCN 035

- 2 DC outputs, 28 ± 4 V (60 A protection per output)
- 2 DC outputs, 28 ± 4 V (both outputs together protected for 60 A)
- 4 AC outputs, 115/200 V, 400 Hz, 3 Ø
  not protected, a current limiter in the 400 Hz inverter limits the total AC peak power to 3.5 KVA.

The outputs of the Experiment Power Distribution Box may be remotely controlled via S/S RAU and Orbiter MDM. There are three circuit breakers for DC and three latching relays for AC which may be switched selectively by RAU on/off commands for each EPDB (Fig. 4.3 - 5). In addition all circuit breakers may be switched simultaneously by MDM on/off commands (simultaneously for all EPDB's).

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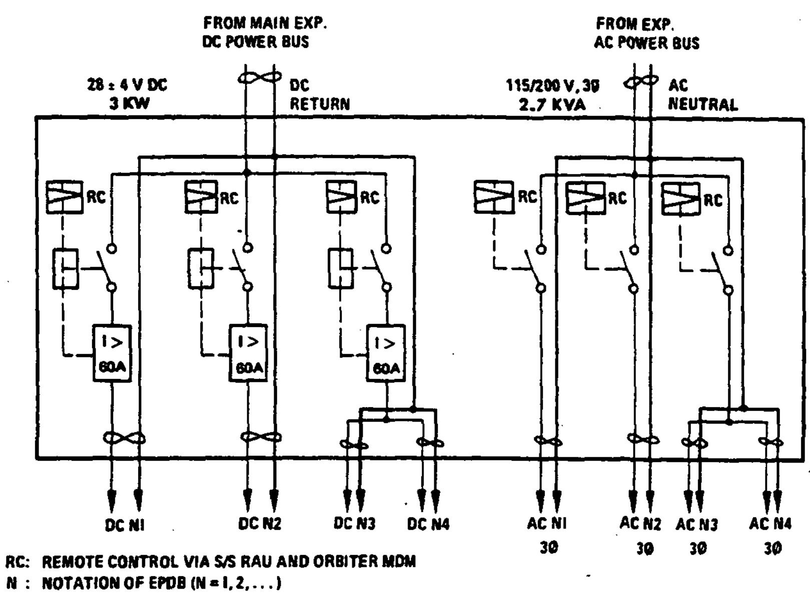

Figure 4.3 - 5: Experiment Power Distribution Box

4.3.3.2 Experiment Power Switching Panel

The Experiment Power Switching Panels (EPSP's) receive their input from the Experiment Power Distribution Boxes; therefore, the maximum total DC current is limited to 60 A. All power outputs are manually controlled via switches located on the front panel, except two DC outputs which are routed through the EPSP without any switching: DCN 035

The following outputs are available for experiments (see Figure 4.3 - 6):

- Rack internal connection
  - 4 DC outputs 28 ± 4 V (35 A protection per output by manually controlled circuit breaker).
  - 1 DC output, 28 ± 4 V, not fused in the EPSP (60 A protection in the EPDB). DCN 035
  - 1 DC output, 28 ± 4 V (7 A protection by manually controlled circuit breaker). If the EPSP is located in rack No. 4 or 10, this DC output is connected to CBP 26 or CBP 28 respectively (see para. 4.3.3 and Figure 4.3-3).
  - 1 DC output, 28 ± 4 V (redundantly wired and protected by melting fuses.) This output is normally used to supply an experiment RAU with power.
  - 2 AC outputs, 3 phases, 115/200 V, 400 Hz (protected selectively 3 A per phase). If the EPSP is located in rack No. 4 or 10, AC output No. 1 is connected to CBP 26 or CBP 28 respectively (see para. 4.3.3 and Figure 3.3-3).
- Front Outlet
  - 1 DC output, 28 ± 4 V (protected and switched together with one of the rear DC outputs, see Figure 3.4 - 6).

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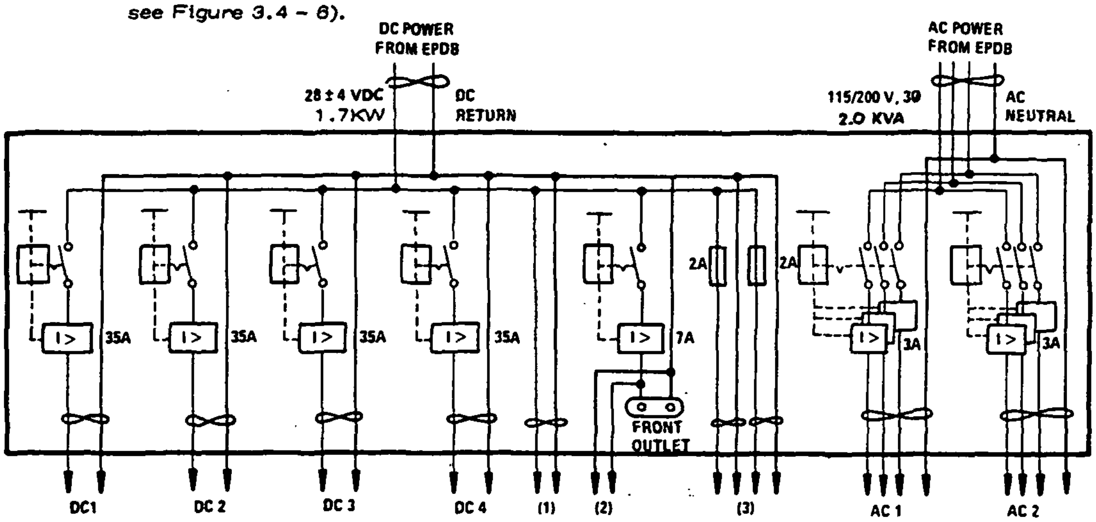

(1) Direct routing from EPDB
(2) Power outlet for airlock / one aft viewport experiment equipment
(3) Power outlet for experiment RAU (rack location), ECP-AE-50641-xx
(4) In Rack 4 / Rack 10 used for airlock/aft viewport

Figure 4.3 - 6: Experiment Power Switching Panel

4.3.3.3 Power for Experiment Equipment in the Center Aisle

For experiment equipment located in the center aisle, Spacelab provides two similar power interfaces, one in the core segment and one in the experiment segment (Figure 4.1 - 5). These power interfaces consist of connector brackets containing DC and AC power connectors located under circular cut-outs in the main floor. The cabling provided by Spacelab is routed (Figure 4.3-3) from EPDB No.1 to the connector bracket CBP/S 31. In the experiment segment the spare DC and AC outlets of EPDB No. 3 are principally used to provide power for the center aisle connector bracket CBP/S 32. As there is no free EPDB output in the core segment, one of the EPSP's in an experiment rack cannot be connected when power is provided to the center aisle of the core segment from the EPDB.

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However, if the user needs all EPSP's in the core segment or even the spare EPDB outlets in the experiment segment, he may also route his own power cabling from EPSP outputs to the center aisle connector brackets. Mounting provisions for these cables are available, but the user has to provide all necessary cables, connectors, clamps, fasteners, etc. .

Each connector bracket contains

DCN 035

1 DC output, 28 ± 4 V (60 A protected in the EPDB or 35 A protected in the EPSP)
1 AC output, 115/200 V, 400 Hz, 3 Ø (total nominal power limited to 2.7 kVA, remote control capability in the EPDB)

Details are described in Appendix A, Section 3.

4.3.3.4 Power for Experiment Equipment in the Airlock

Spacelab will provide a DC and 400 Hz AC power interface for experiments located in the Airlock. This power will be routed from an EPSP located in Rack No. 4. The following power will be available on the Airlock power connector on connector bracket CBP 26 (see also Figure 4.3 - 3):

1 DC output, 28 ± 4 V (7 A protected in the EPSP)
1 AC output, 115/200 V, 400 Hz, 3 Ø (3A protected in the EPSP).

Further details are described in Appendix A, Section 3.

4.3.3.5 Power for Experiment Equipment Attached to the High Quality Window/Viewport Assembly

Spacelab provides no experiment dedicated power connector for experiment equipment attached to the high quality window/viewport assembly in the core segment. However, power can be drawn from an adjacent rack by using experimenter provided cabling.
In the experiment segment, power can be drawn from an experiment dedicated power connector on CBP 26 (same connector as used for airlock experiment equipment).

Experiment equipment attached to the viewport in the aft end cone may use a dedicated connector on CBP 26.

The signal characteristics are the same as described in Section 4.3.3.4.

4.3.3.6 Emergency Box

The Spacelab Emergency Box (see Figure 4.3 - 7) is fed by two independent auxiliary DC power lines from the Orbiter. This power is available during all mission phases and also when only degraded power is delivered to Spacelab. The box supplies DC power to Spacelab emergency equipment via redundant lines. In addition four essential DC buses, two for subsystems and two for experiments, are routed through Spacelab. The outputs are protected by melting fuses and manually controlled circuit breakers, respectively.

The characteristics of the essential power available to experiments are:

Voltage: 22.0 to 32 V DC (Module)
21.5 to 32 V DC (Pallet)

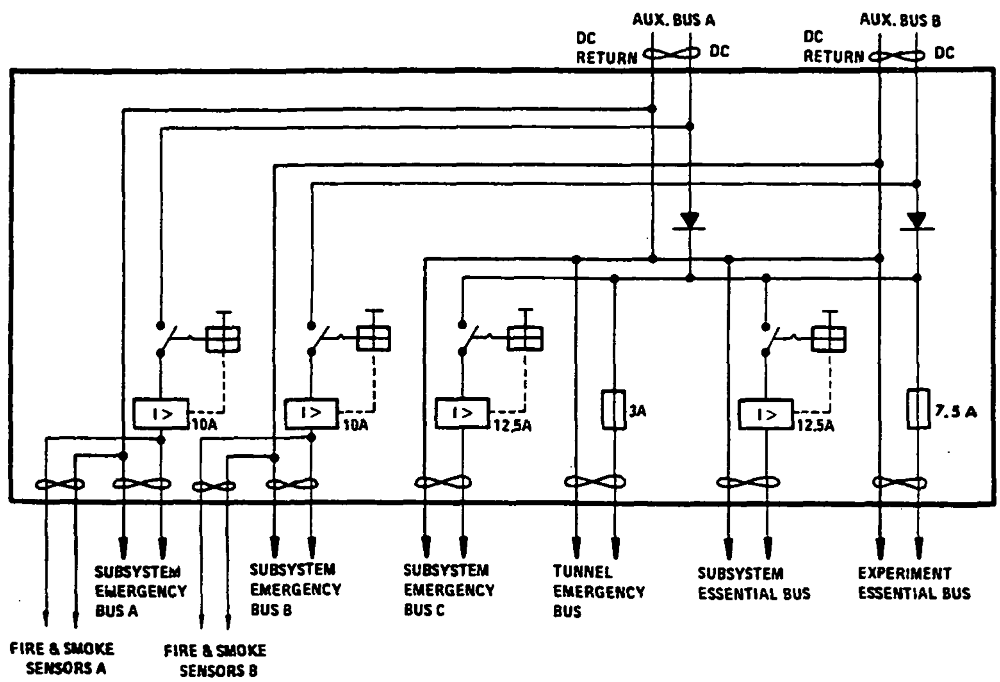


Figure 4.3 - 7: Emergency Box

Note: The Subsystem Emergency Bus C has no redundant lines; however, the power drawn from the output will be handled and controlled like Emergency Power. Bus C is dedicated for IPS Caution and Warning electronics.

Figure 4.3-7 does not reflect the number of hardwired outlets. For example subsystem and experiment essential power busses are wired to one common outlet of the Emergency Box.

4.3.3.7 AFD Power Distribution Box

The AFD Power Distribution Box (AFD PDB) is shown in Figure 4.3- 8. In addition to the power outputs for subsystems, the following outputs for experiments are provided:

- 1 DC output, 25.7 - 32 VDC (protected for 15 A in the Orbiter)
- 1 AC output, 3 phases,115/200 V, 400 Hz (protected for nominally 3 A per phase in the Orbiter)

No switching or current protection for experiment outputs is included in the AFD PDB itself; therefore, the power output may only be switched upstream of the box itself via the Orbiter switching system or downstream by switches included in the experiment hardware.

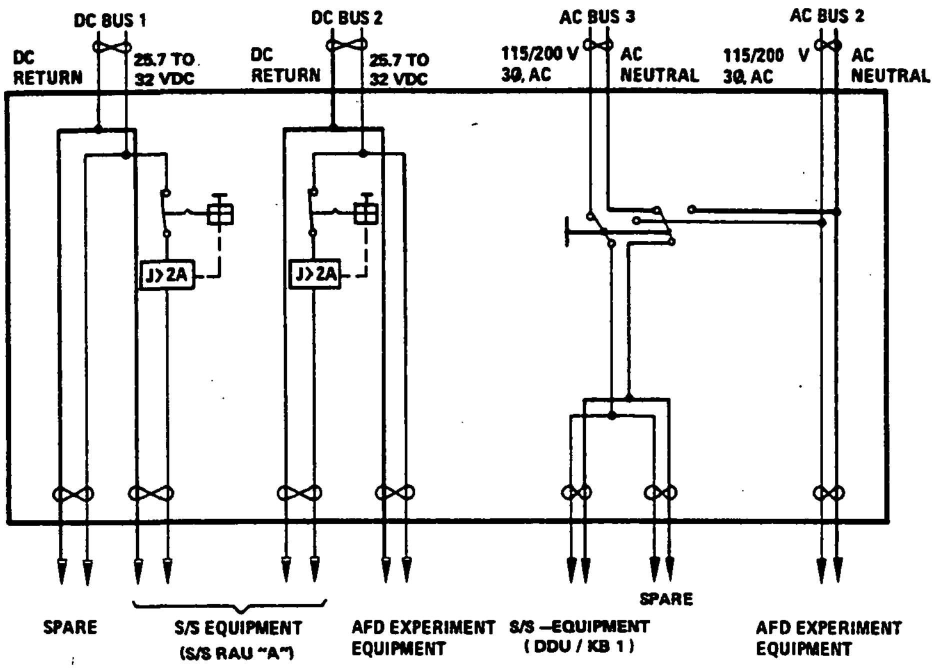


Figure 4.3 - 8 does not reflect the number of outlets.

Figure 4.3 - 8: AFD Power Distribution Box

4.3.4 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 of experiment EPDB's via Spacelab S/S RAU's.

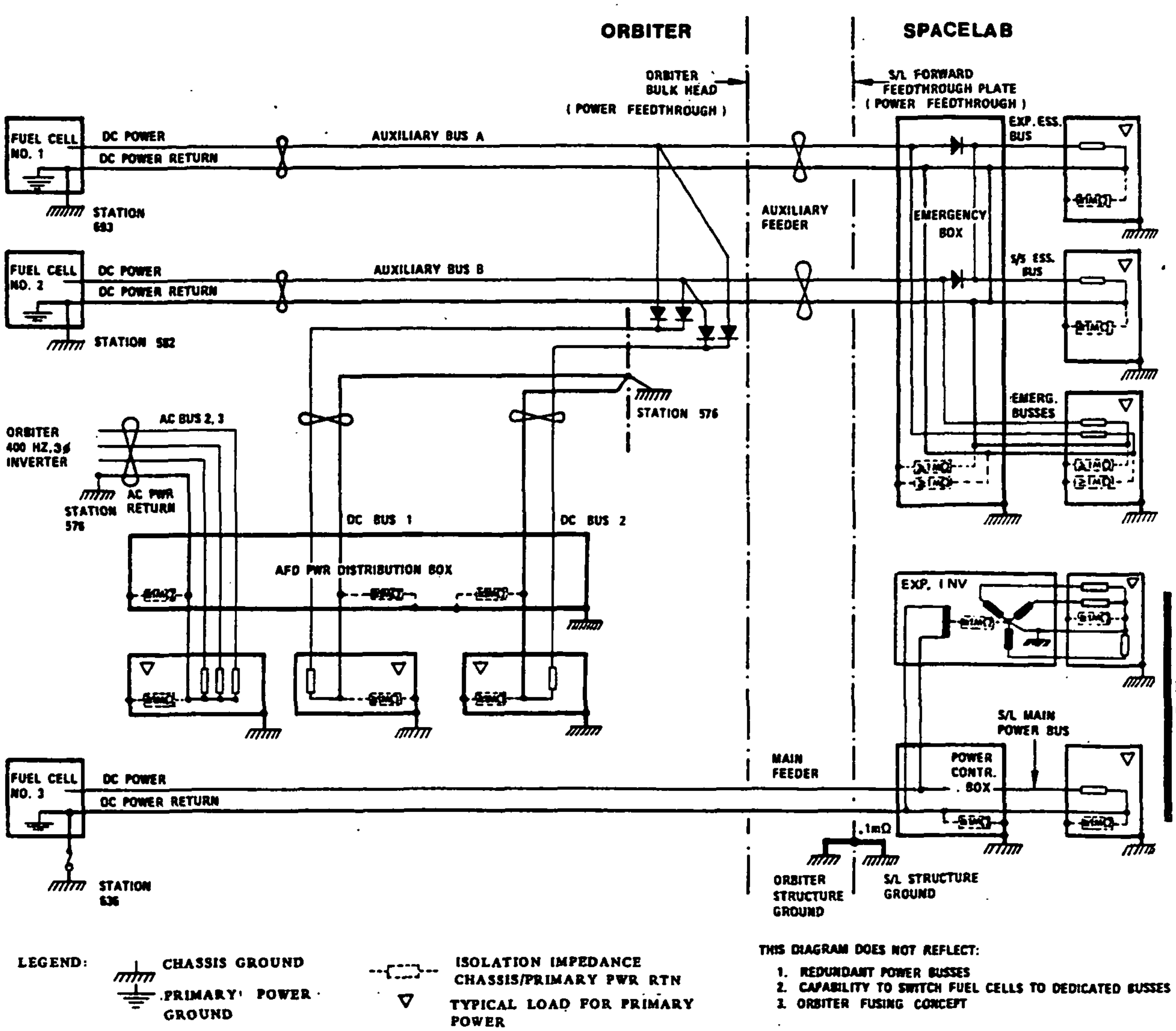
Further details on Spacelab subsystem control are given in para. 4.4.5.

4.3.5 Grounding

This section describes in detail the grounding and isolation concept used for Spacelab subsystem equipment. The purpose is to give a clear understanding of this concept to the user whose experiment design has to fit into this concept. However, the mandatory grounding and isolation requirement imposed on experiments are given in para. 7.7.2.

4.3.5.1 Primary Power Grounding Concept

Figure 4.3 - 9 depicts the primary power grounding concept of Spacelab Subsystems.



DCN 046

Figure 4.3 - 9: Primary Power Grounding Diagram

Assuming a normal operating mode, fuel cell No. 3 (FC 3) is connected via a two (2) conductor DC power bus to the S/L Power Control Box.
FC 1 and FC 2 are connected to the S/L Emergency Box.

All primary power returns are referenced to structure ground in the Orbiter only. Nevertheless, as depicted in Figure 4.3 – 9, there is no common star point for all primary power return lines.

The S/L main power return line is connected to Orbiter structure at Station No. 636; the S/L auxiliary power return lines are connected to Orbiter structure at Station No. 693 (for FC 1 power return line) and Station No. 582 (for FC 2 power return line).

As a consequence, when disconnected from the Orbiter fuel cells, all DC primary power and power return lines in the Spacelab have to be isolated from S/L structure by at least 1 MΩ impedance (DC to 25 Hz).

The Spacelab dedicated fuel cell (for example FC 3 in Figure 4.3 – 9) is grounded to Orbiter structure by a removable jumper. With the jumper removed the fuel cell is isolated from Orbiter structure by at least 1 MΩ impedance (DC to 25 Hz).

For experiment equipment in the Aft Flight Deck which is supplied with power via the AFD Power Distribution Box, DC and AC primary power return lines have to be isolated from structure inside the experiment equipment.

The 400 Hz power return lines are grounded to structure at the 400 Hz inverter boxes.

If experiment power is switched off inside EPDS units, only the respective DC power line (+ 28 V) or AC power lines (phases A, B, C simultaneously) will be disconnected. Experiment DC power return lines and AC neutral lines remain connected to the power buses.

4.3.5.2 Secondary Power/Signal Grounding Concept

Figure 4.3 – 10 depicts the secondary power/signal grounding concept of Spacelab units having an electrical interface with an Orbiter MDM. However, the interface details may differ from unit to unit. The design of Spacelab/Orbiter interface circuits is described by the detailed circuit schematics in the Orbiter/Spacelab ICD-2-05301. In general the signal circuitry is designed to avoid signal ground loops and the effects of differences in potential between structure reference points.

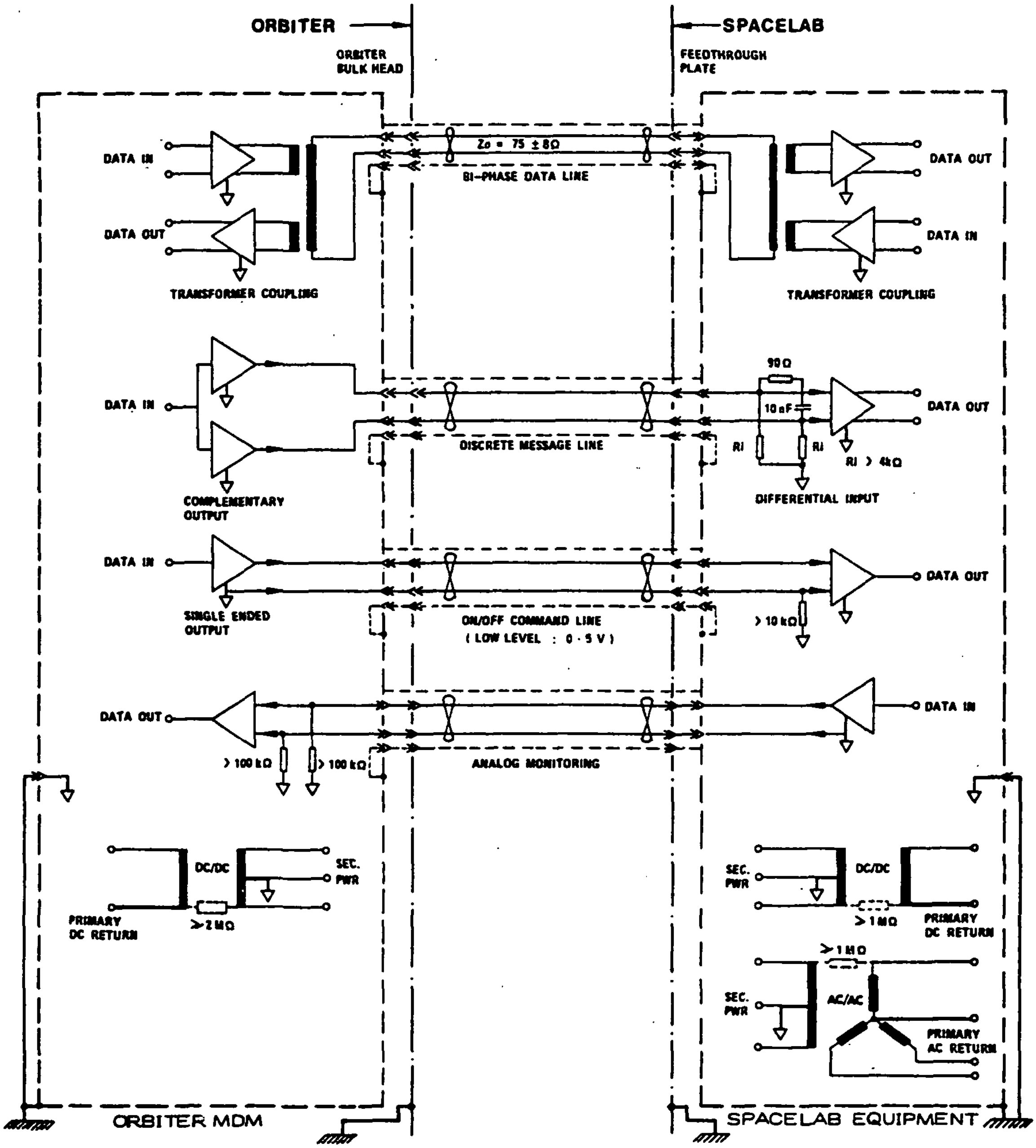
Subsystem and experiment power converters require, if disconnected from the primary power supply, an isolation between primary power ground and secondary power ground of at least 1 MΩ impedance (DC to 25 Hz) if operated in the S/L, or ≥ 2 MΩ if operated in the Orbiter.
Secondary power ground or signal ground of each unit must be connected to chassis ground at one point. The chassis itself is electrically bonded to structure.

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

- transformer coupling between input and output

or

- floating differential signal inputs is necessary.



Legend:

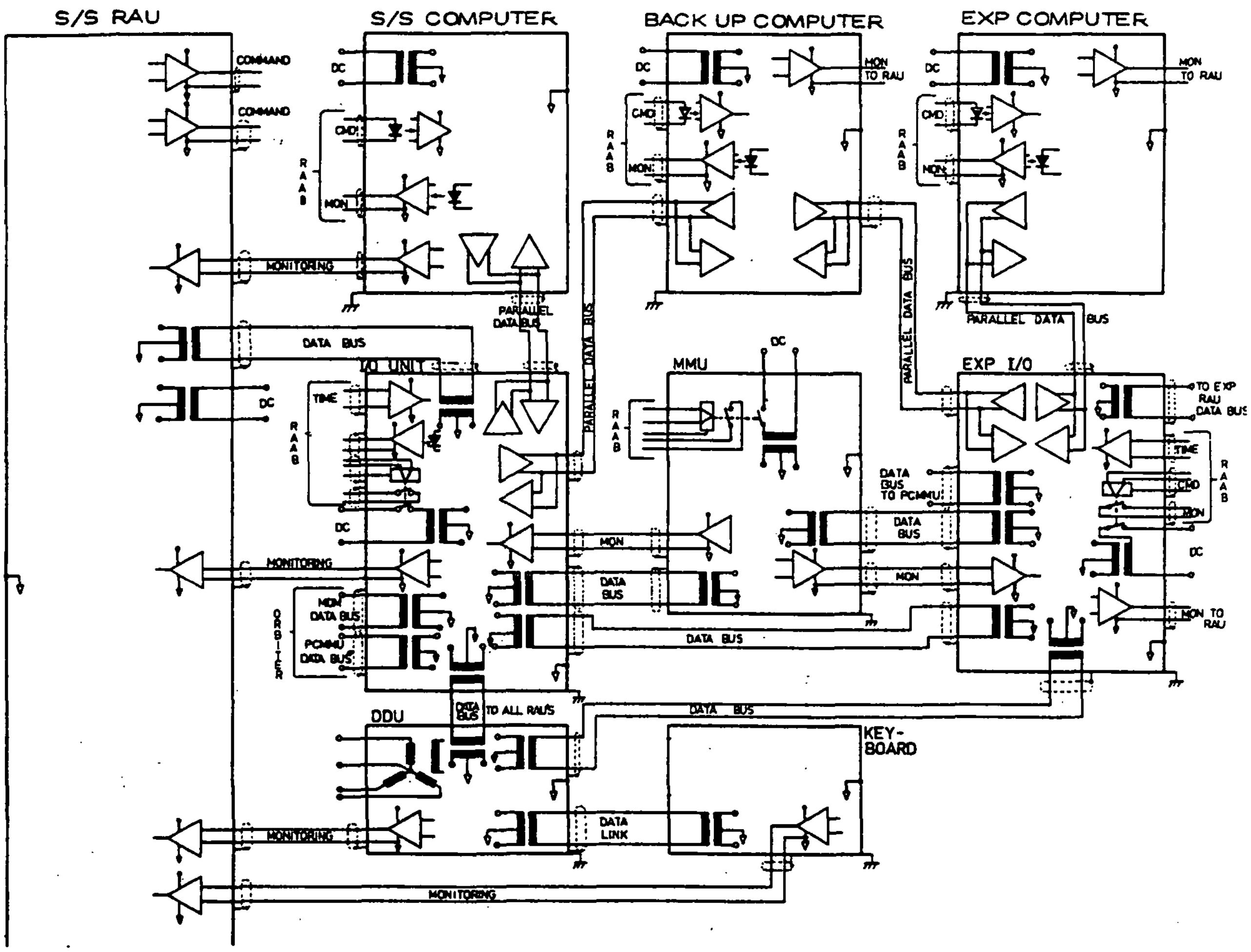
Chassis Ground

Signal/Secondary Power Ground

Isolation Impedance (DC to 25 Hz)

Figure 4.3 - 10: Secondary Power/Signal Grounding Diagram (Orbiter to Spacelab Interface)

DCN 021 The signal grounding concept of Spacelab Subsystem/Subsystem Interface circuits is depicted in Figure 4.3 - 11.



Legend:

Chassis Ground

Signal or Secondary Power Ground

Transformer Coupling

Single Ended Output

Floating Differential Input

Differential Line Driver

I/O - Input/Output Unit
RAAB - Remote Amplifier and Advisory Box
RAU - Remote Acquisition Unit
MMU - Mass Memory Unit
DDU - Data Display Unit
CMD - Command
MON - Monitoring

Figure 4.3 - 11: Secondary Power/Signal Grounding Diagram (Spacelab S/S to S/S Interface)

The Spacelab subsystem power / signal interface design shows the following characteristics:

- Primary power ground is isolated from secondary power ground by at least 1 MΩ (DC to 25 Hz).
- Secondary power ground/signal ground is connected to chassis ground at one point. The chassis is bonded to structure with a total DC resistance $\leq 10^{-2}\,\Omega$.
- Single ended or balanced output and floating differential signal input for low frequency analog signals (0 – 50 Hz) and ON/OFF command transfer.
- Balanced line drivers and differential line receivers for digital data transfer in parallel.
- Transformer coupling between input and output for bi-level serial data transfer.
- Use of opto couplers for command transfer.

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In Spacelab all shields (except coax) related to an interface between subsystem units are grounded at both ends. The shield is terminated via the connector shell. If possible, the shields are also grounded at intermediate connector brackets.

4.3.5.3 Experiment Grounding Concept

Groundrule:

Spacelab structure must be kept clean from all kind of return currents because it serves as low impedance reference for all signal sources.
Structure currents would cause common mode voltages and noise coupling.

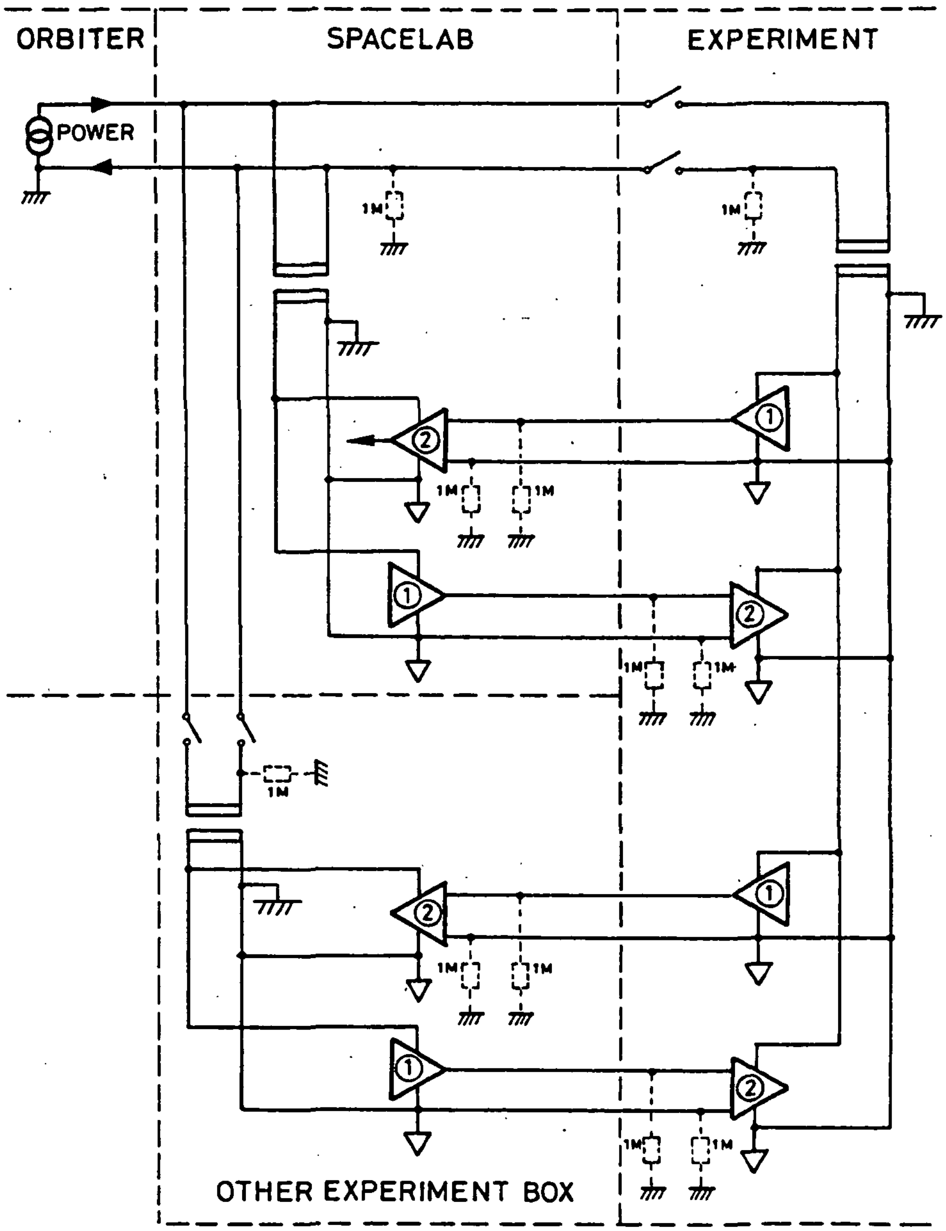
Implementation Rules:

This outlined groundrule results in the following implementation rules (see also Figure 4.3 – 12) DCN 046

- Primary power (main DC, essential power) and AC power, hot lines and return lines must be isolated from Spacelab structure under all circumstances.
- Signal receivers which form the input into a box must have floating inputs which are isolated from structure.
- Signal transmitters for signals to other boxes shall be referenced to signal ground which, for each box, is connected to structure. Signal transmitters should not be referenced to the power return line because of the noise on this line.
- Signal power shall be secondary power, derived from primary power via isolating converters, and shall be referenced to structure at the converter only (single point ground).

Detailed and quantitative EMC requirements for experiments are defined in Section 7.7.2.2.

DCN 046



(1) SYMBOL OF SIGNAL TRANSMITTER ONLY, COULD ALSO BE COMPLEMENTARY OUTPUT, BALANCED OUTPUT, SENSOR ITSELF, ETC.

(2) SYMBOL OF SIGNAL RECEIVER ONLY, COULD ALSO BE OPTO COUPLER, RELAY COIL, ETC.

Figure 4.3 – 12: Experiment Grounding Concept

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

DCN 048

The equipment provided by the CDMS to Spacelab experiments is listed in Table 4.4-1. Equipment marked by an * is non-removable and will be permanently installed.

Table 4.4-1: CDMS Equipment for Experiments

Basic Spacelab	Mission Dependent
1. Exp. Data Bus	* 1. Experiment Computer
2. Mass Memory	* 2. Experiment I/O Unit
3. Keyboard /Data Display Unit (2)	3. Experiment RAU's (8 total) (21 may be accommodated)
4. Intercom Master Station	4. Keyboard /Data Display Unit (1)
5. Intercom Remote Station (1)	* 5. High Rate Multiplexer
	* 6. High Data Rate Recorder (In Module Conf. Only)
	7. Intercom Remote Station (3)
	8. Television Monitor (NFE)

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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. The physical location of the main CDMS and EPDS equipment in the racks is shown in Figure 4.4 - 2.

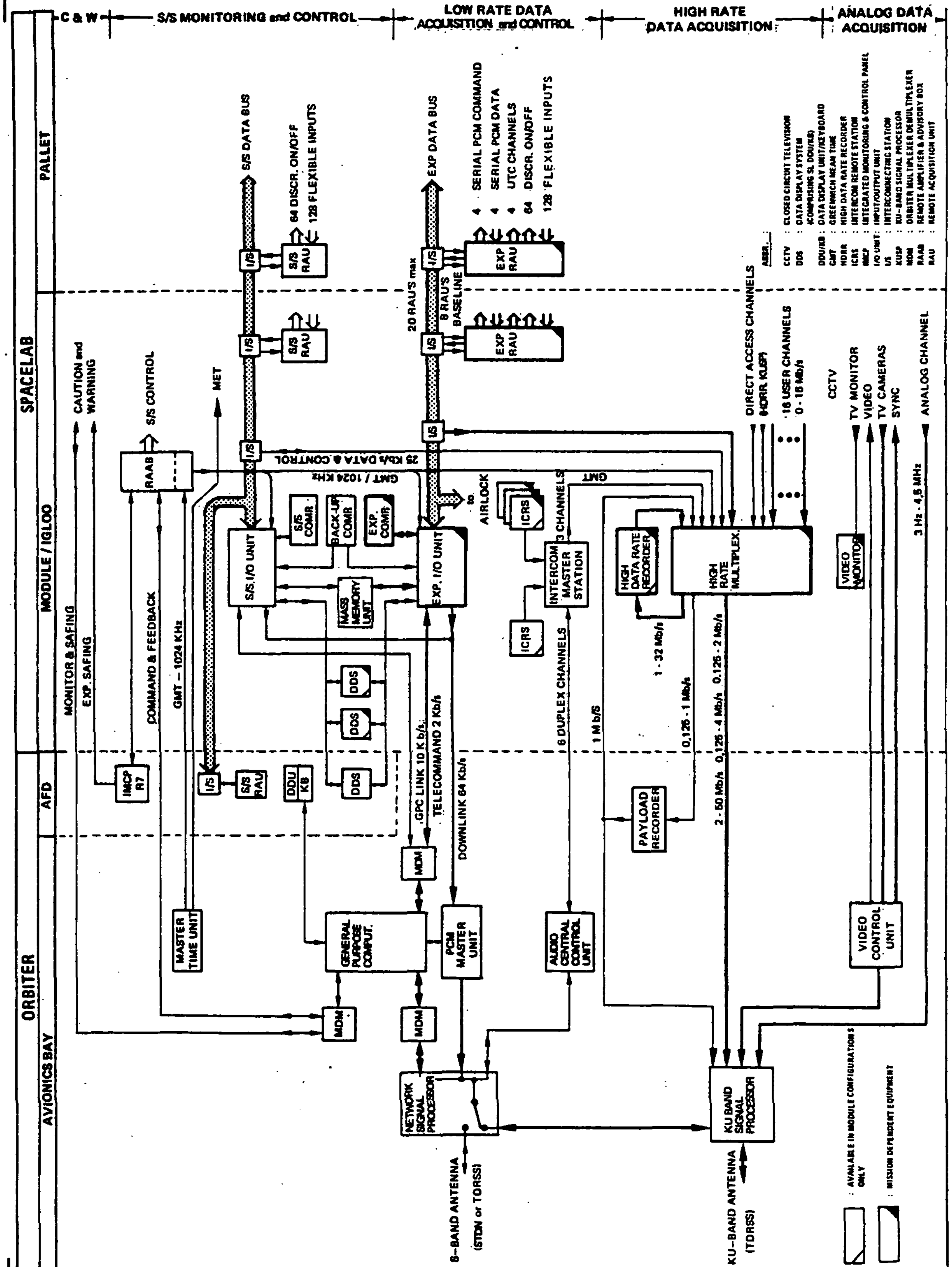

Figure 4.4 - 1: CDMS Block Diagram

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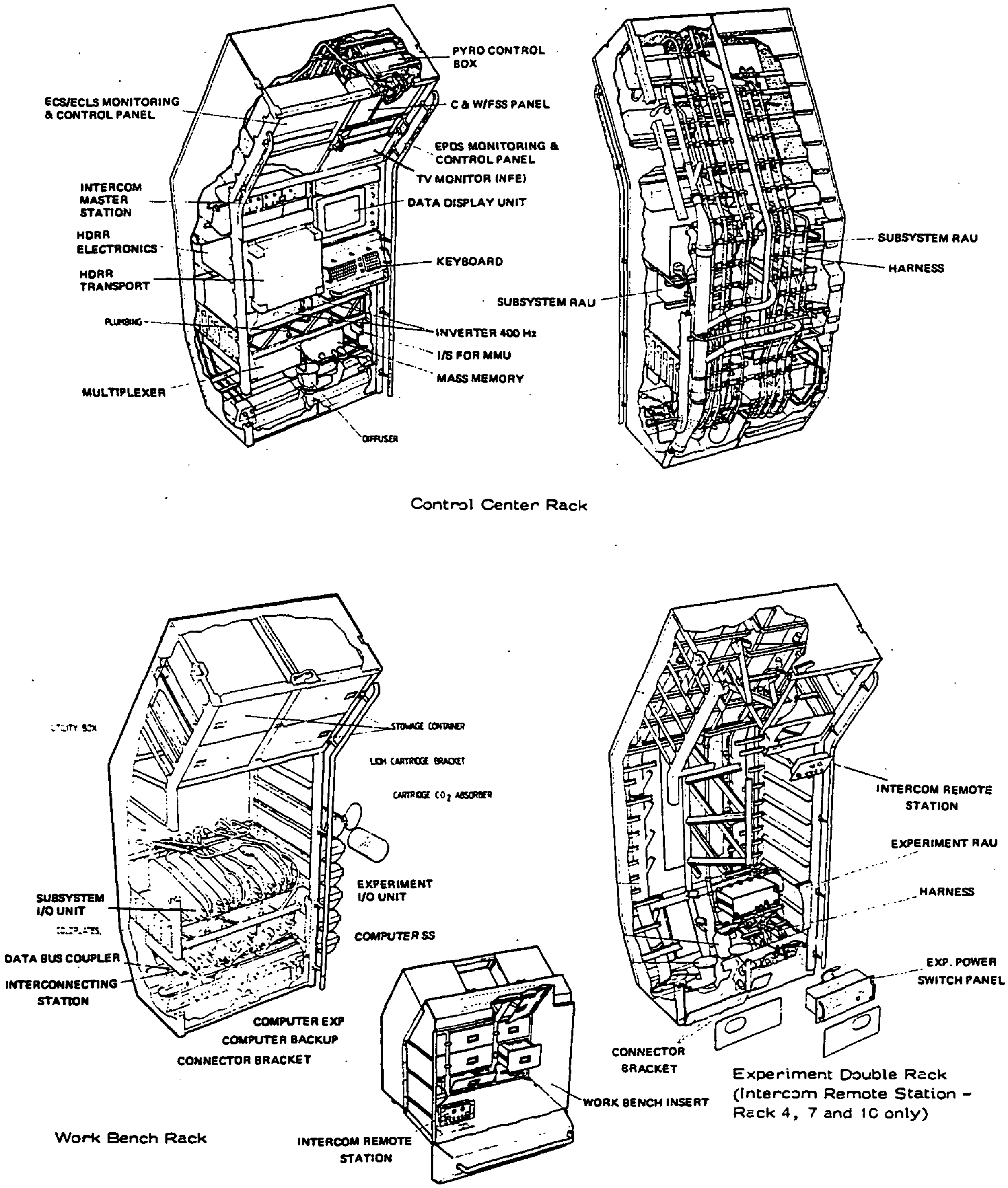

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Figure 4.4 - 2: Location of Main CDMS and EPDS Equipment in the Spacelab Module

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 experiment 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, etc. Programs for control and processing of experiments and subsystems exceeding the storage 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 are composed of 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.

Experiment and subsystem monitoring and control is in principle automatically performed by CDMS equipment. These functions are initiated automatically through pre-programmed computer sequences stored in the MMU, or semi-automatically by interaction of the keyboard/DDU with the computer, or by telecommands through the Orbiter uplink (2kb/s).

Data processed by the experiment or subsystem computer can be displayed on Data Display Units (DDU's), which have vector display capability.

Low bit rate scientific and housekeeping data processed by the experiment computer can be transmitted by the Orbiter downlink via the Tracking and Data Relay Satellite System (TDRSS).

Medium and high rate scientific data are acquired by the High Rate Data Acquisition part of the CDMS. This part consists of the High Rate Multiplexer (HRM) which includes a Voice Digitizer, the High Data Rate Recorder (HDRR), and the Orbiter Payload Recorder. This system is able to multiplex up to 16 experiment input channels and data from the S/S and exp. computer for direct downlink via the Tracking and Data Relay Satellite System or for recording (HDRR or the Orbiter Payload Recorder) during non-transmission times of the Orbiter KU-Band System. The recorded data may be interleaved with real time experiment data for transmission to ground.

The Voice Digitizer converts three analog Spacelab audio signals into a HRM compatible digital form to allow voice tagging of data multiplexed by the HRM.

Spacelab provides the necessary interfaces for CCTV equipment to form an extension of the Orbiter CCTV There are provisions to support an Orbiter common TV monitor in the control center rack. There are also electrical interfaces for experiment provide video cameras with EIA standard signal output characteristics. DCN 048

Spacelab also provides a 4.5 MHz analog channel for use by experiments, e.g. to accommodate non-EIA-standard TV signals.

One direct interface to the Orbiter Master Time Unit (MTU) is provided to supply the experimenter with Mission Elapsed Time (MET) information.

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 an analog recorder.

Duplex voice links for onboard or Orbiter ground communication are provided by the Intercom Assembly. DCN 048

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

4.4.2 Low Rate Data Acquisition and Control

For a single serial channel, the RAU can handle a data rate of up to 51.2 kbs assuming one 32 word block transfer each 10 msec Periodic Input/Output Loop (PIOL) segment. Higher rates are feasible if more than one block is transferred per 10 msec. Multiple block transfer, however, implies large user buffers to accommodate asymmetrical acquisition. However, for the entire payload, the overall data bus load together with the overall computer load imposes additional constraints, which are to be taken into account. DCN 048

Low rate data acquisition from experiments and experimental control is performed by the experiment computer through the experiment I/O unit, the experiment data bus and experiment RAU's. Although the experiments interface only with the RAU's, the following paragraphs describe the complete data and command transfer in more details. The purpose of the detailed description is to provide visibility into a system that has to be shared with many users constraining each other.

Data processed by the computer may be transmitted to the ground via the Orbiter PCM Master Unit. The low rate data link is designed to achieve a word error rate (WER) $\leq 1.7 \times 10^{-7}$ for the data flow between any RAU digital input and the Orbiter PCM Master Unit. In addition to this link, the computer output data can be transmitted via the High Rate Multiplexer (HRM).

4.4.2.1 Remote Acquisition Unit

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 distribution of commands. The data exchange between RAU's and the I/O unit is performed via simplex (dedicated bus for each direction) serial busses with 1 Mb/s clock rate. The data are encoded in a self-clocking bi-phase code (Manchester II). DCN 048

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

Output: 64 ON/OFF command channels

4 serial PCM command channels with associated clocks, code NRZ-L

4 User Time Clock channels (1024 kHz)

4 User Time Clock Update channels, 4 pulse cycles/s

A block diagram of the RAU is given in Figure 4.4 - 3. It should be noted that the measuring points shown are for bench testing only.

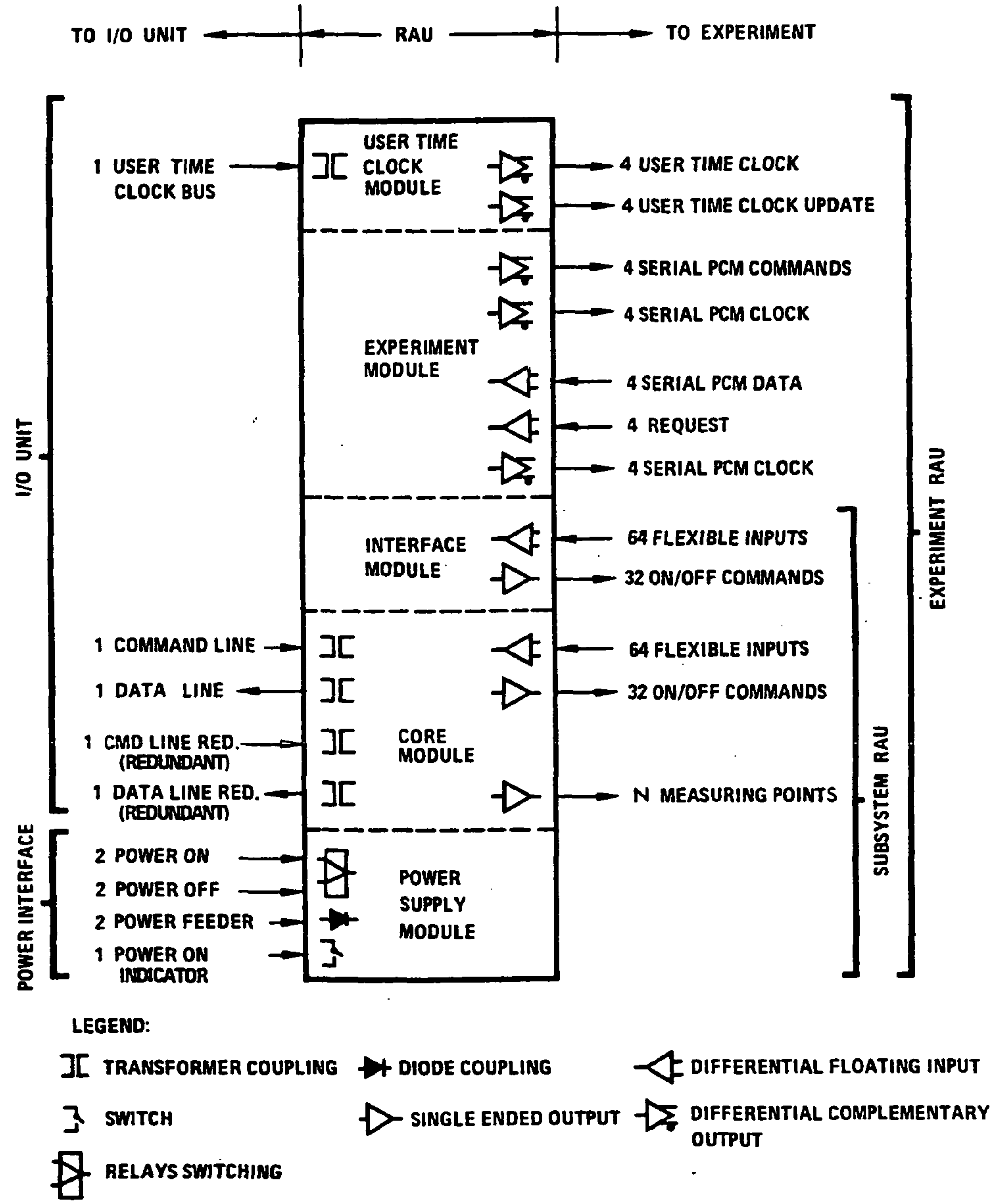

Figure 4.4 - 3: Remote Acquisition Unit Block Diagram

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 basic periods of 10 ms, 100 ms, or 1 s. Each scan cycle will be initiated and controlled by the Periodic Input/Output 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. DCN 048

4.4.2.1.2 Physical Concept

32 different addresses are foreseen for the RAU's on a bus. The address for a particular RAU is determined by a patch connector. For electrical reasons the busses (S/S and exp. bus) are split into two branches also causing a split of the 32 RAU addresses on each bus.

The split is

- for the S/S bus: AFD branch, address 0 to 7 (including IPS S/S RAU's)
  main branch, address 8 to 31
- for the exp. bus: Airlock branch, address 0 to 7 (including IPS exp. RAU's)
  main branch, address 8 to 31

The electrical characteristics of the busses allow the accommodation of up to 21 RAU's per branch. DCN 048

8 exp. RAU's is the total number of units delivered by the Spacelab program.

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 provides standard locations for RAU's in the lower part of the experiment racks. However, the concept allows the user to integrate RAU's together with his experiment equipment, if he uses his own racks and/or experiment equipment mounted directly to the center aisle or to the pallet.

In every case the user has to ensure that

- the cable length, between RAU and Interconnecting Station is below 5 m
- the applicable interface specifications of the RAU are met in accordance with EQ-MA-0003.

There are two different types of RAU. The smaller type is the subsystem RAU consisting of the power supply module, core module, and the interface module (see Fig. 4.4 - 3). The larger type is the experiment RAU consisting of the subsystem RAU modules plus the experiment module (which provides serial PCM input and outputs) and the User Time Clock (UTC) module. The functions of the RAU are described in the subsequent paragraphs.

Figure 4.4 - 4 depicts the physical characteristics and connector locations of the experiment RAU

DCN 048

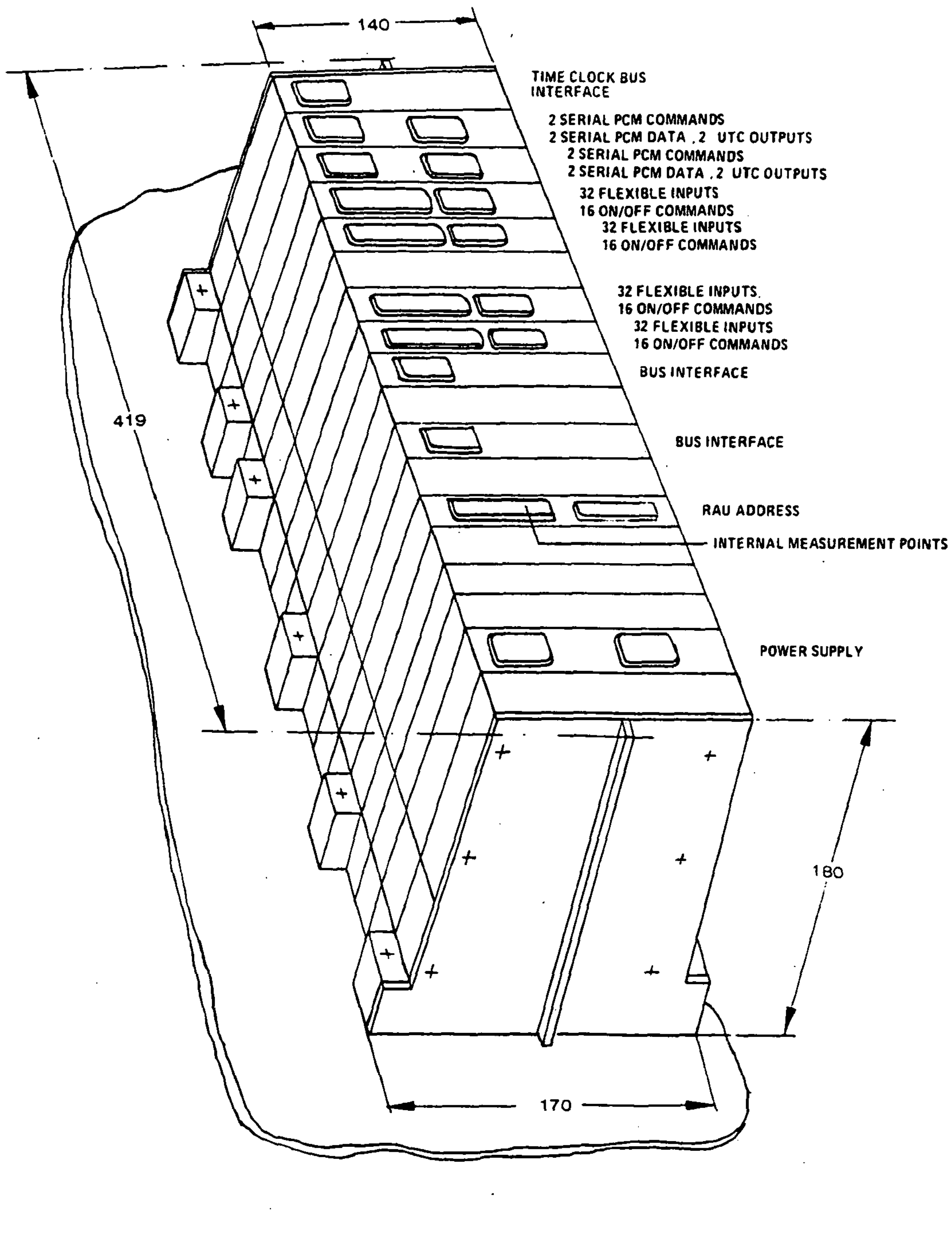

Figure 4.4 - 4: Experiment RAU Physical Characteristics

4.4.2.1.3 RAU-Experiment Links

The RAU-experiment links are depicted in Figure 4.4 - 3. The detailed electrical interface is presented in Appendix A, Avionics Interface Definition.

4.4.2.1.3.1 Data from RAU to Experiments

Serial PCM Command Channel

Four (4) RAU channels can deliver serial PCM commands to the experiments, in connection with four RAU provided 1 MHz clock pulses. The code is NRZ-L. The maximum command exchange 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, MET, ground data, Orbiter State Vector, etc. For one 32 word block transfer per PIOL 10 msec segment, a data rate of 51.2 kb/s is achieved. Higher rates are feasible by means of multiple block transfer. DCN 048

ON/OFF Commands

Each RAU will provide 64 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. The load capability of these RAU outputs is designed to drive opto-couplers directly. A software facility is provided to produce either an on/off command of 100 ms pulse length (pulse command) or an on/off command status which stays until it changed by a subsequent command (level command). DCN 048

User Time Clock Outputs

The experimenter can receive timing information from the RAU User Time Clock (UTC) module. A 1024 kHz clock (duty cycle 0.5) and an update pulse group (every 250 ms) are available. These signals are derived from the master oscillator in the Orbiter Master Time Unit (MTU) and are therefore synchronized with GMT within the accuracies of the count-down electronics chains of the Orbiter MTU.

4.4.2.1.3.2 Data from Experiment to RAU

Serial PCM Data Channel

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 data 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 experiments.

Note: The parity bit is assigned to a value that makes the number of 'ones' in each 17 - bit word an odd number.

The status ('one' or 'zero') of the experiment provided serial PCM data request lines may be scanned by the RAU on a special software request and transferred to the experiment computer. In this special case the four request lines will be handled like discretes.

Flexible Inputs

The experiment RAU provides 128 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

The use of flexible inputs as analog or discrete channels is determined by the actual software request, i.e. in principle each flexible input may be changed from analog to discrete or vice versa between two subsequent software acquisition commands.

However, in the case of discrete data acquisition only, blocks 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 blocks 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 (analog single mode) or blocks of 16 input channels (analog scanning mode) are addressable. The selection of 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 analog scanning mode up to 64 words per software request can be transmitted to the I/O unit.

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

DCN 048

Full scale voltage range:	−5.12 V to +5.08 V
Resolution (full scale voltage range):	8 bits (7 bits + sign)
Common mode input voltage range:	± 6 V
Accuracy:	0.6 % of full scale (assuming ± 6 V common mode rejection)

DCN 048

4.4.2.1.4 I/O Unit - RAU Link

The experiment RAU's are linked to the I/O unit by the experiment bus. The data bus consists of a uni-directional "command line" which carries instructions and data from the I/O unit to the RAU's and a uni-directional "data line" which carries responses and data from RAU's back to the I/O unit. The data bus and the interfaces at the I/O unit and the RAU's are dual redundant. 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. DCN 048

An additional "clock bus" is also provided which distributes the Master Time Unit (MTU) derived 1024 kHz clock and update pulses from the I/O unit to experiment 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.

4.4.2.1.4.1 Command Transfer

Serial PCM Commands

The serial PCM command transfer from the I/O unit to the RAU is depicted in Figure 4.4 - 5.

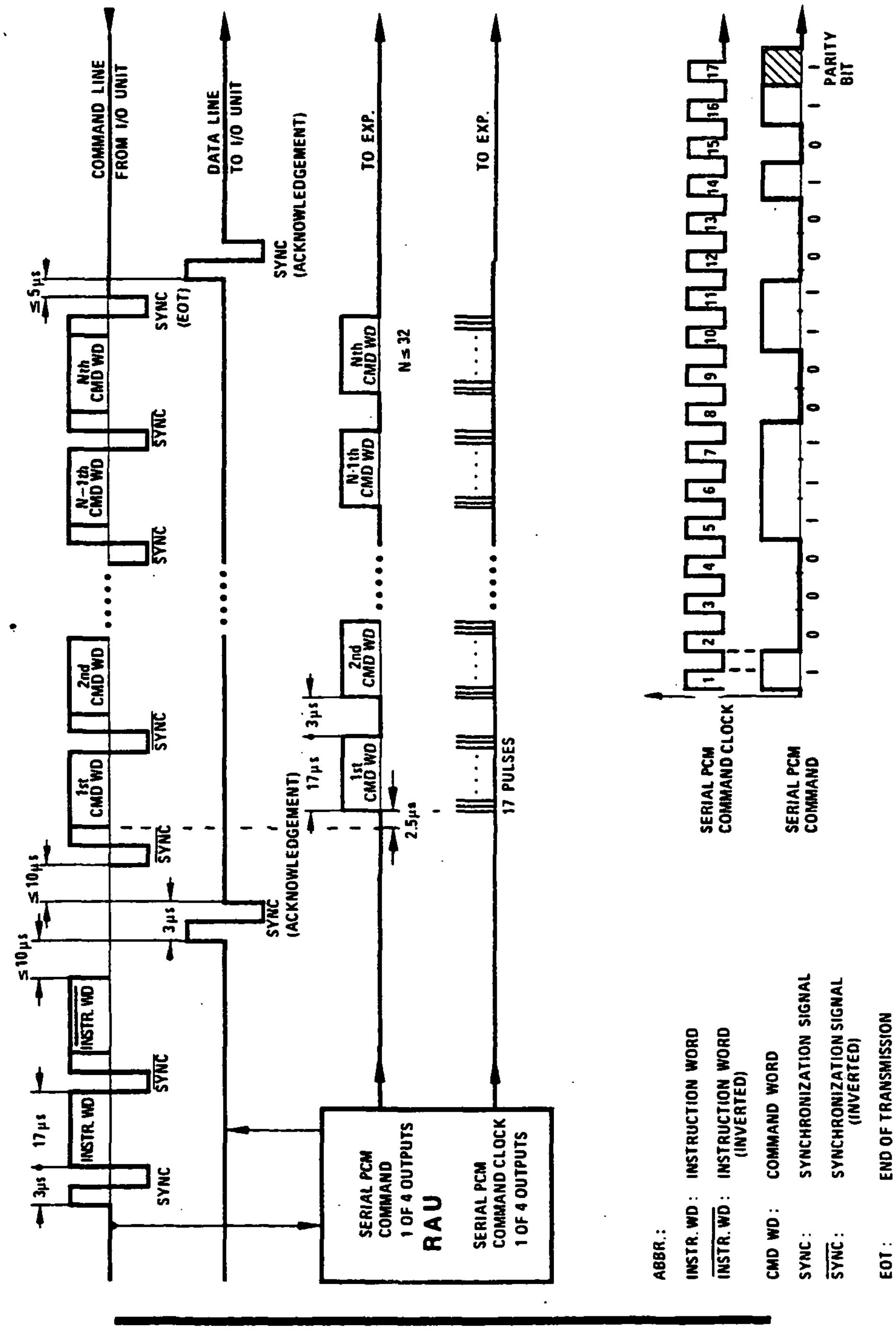

Figure 4.4 - 5: Timing Diagram for Serial PCM Command Transfer

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The transfer will start with a sync pattern and an instruction word followed by the inverted sync and the inverted instruction word. The instruction word includes RAU address, operation code, and channel address as sketched in Figure 4.4 - 6. After the acceptance of this message the RAU will send back an acknowledgement sync to the I/O unit (time delay $< 10\ \mu s$). The I/O unit now starts the transmission of command information as 16 bits + parity data words as a block with a maximum of 32 words per transaction. Each word is preceded by an inverted sync pattern and the end of the block is indicated by a non-inverted sync (EOT).

The RAU will check the received data words by checking the sync, the Manchester code pattern, and the parity while transmitting them to the experiments. In the case of an error, the RAU will shut down its output immediately. Otherwise it will send back an acknowledgement sync to the I/O unit after receiving the EOT sync (time delay $< 5\ \mu s$).

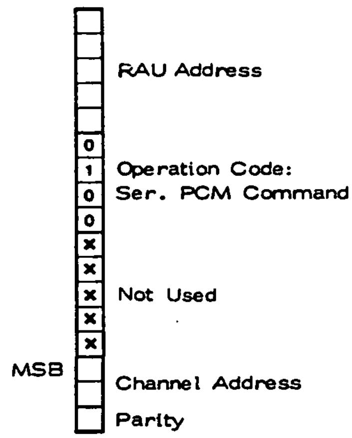

Figure 4.4 - 6: Instruction Word Serial PCM Command

ON/OFF Command

The ON/OFF command transfer from the I/O unit via the RAU to the experiments is depicted in Figure 4.4 - 7.

The transfer will start with a sync signal and an instruction word followed by the inverted sync and the inverted instruction word. The instruction word is depicted in Figure 4.4 - 7 and contains RAU address, operation code, one bit indicating the level to which the ON/OFF output of the RAU has to be set, and the binary coded channel address.

Figure 4.4 - 7 shows both an "ON" and an "OFF" command, i.e. the ninth bit in the first instruction word contains a "1" whilst this bit in the second instruction word is set to "0".

The validity of the instruction word is checked as described for serial PCM commands. If no errors are detected the RAU will send back an acknowledgement sync to the I/O unit otherwise the status of the command channel will remain unchanged.

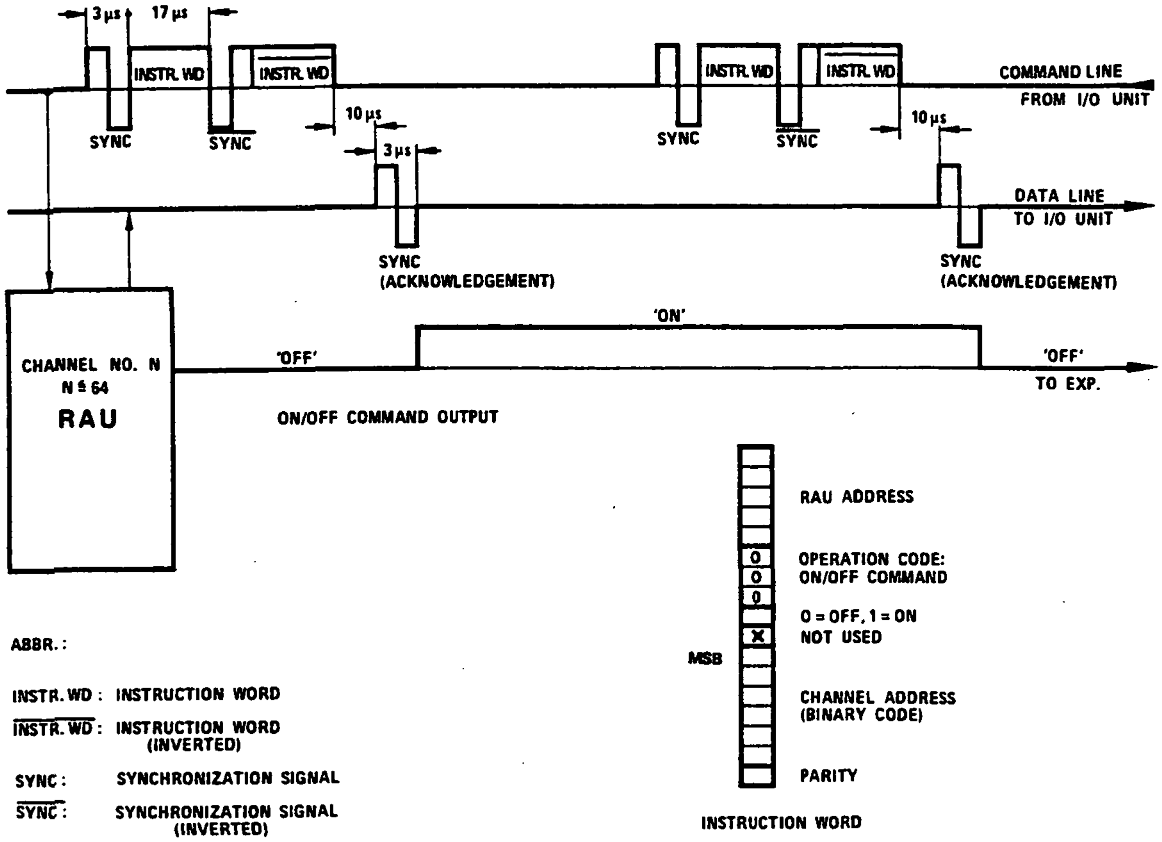

Figure 4.4 - 7: ON/OFF Command Transfer

4.4.2.1.4.2 Data Transfer

Serial PCM Data Inputs

The transfer of serial PCM data from an experiment via RAU to the I/O unit will be initiated by a software generated instruction word on the command line (Figure 4.4-8).

The structure of the instruction word is also given in Figure 4.4 - 8. After receiving this word 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", the RAU will send back a sync signal to the I/O unit within a maximum time delay of 10 μs. In this case the computer system will stop the dialogue for this channel or repeat the transfer of instruction words as determined by software.

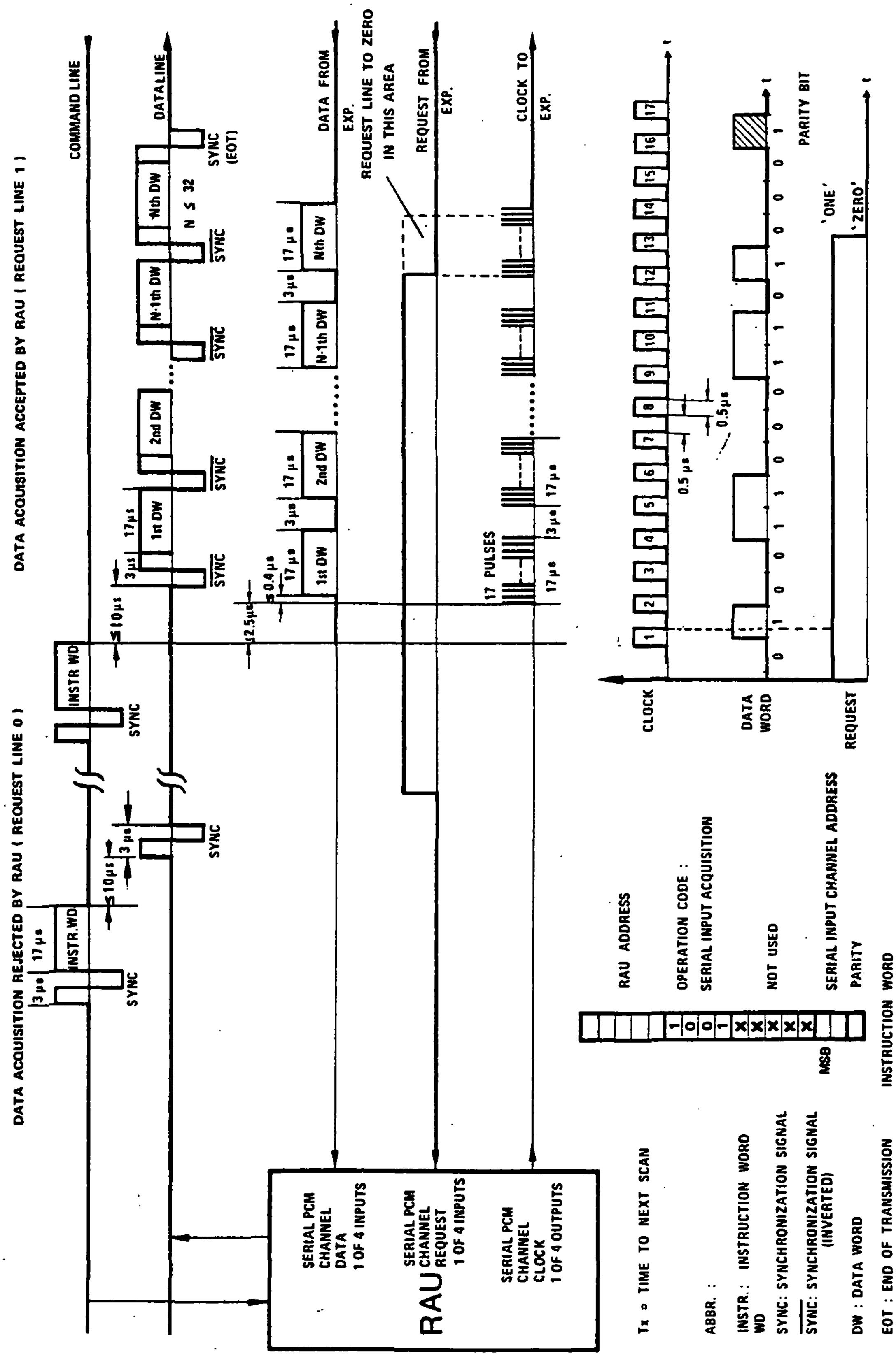

Figure 4.4 – 8: Timing Diagram for Serial PCM Data Transfer

If the status of the request line is detected as 'one' the RAU will start to deliver clock pulses to the experiment within a time < 2.5 µs. These clock pulses are grouped in blocks of 17 pulses and separated by a 3 µs time gap. Each block will take 17 µs and will be used to read out one 17 bit word from the experiment buffer. (The 17th bit always has to be the experiment generated parity bit.)
The RAU will continue to deliver these clock pulses as long as the status of the request line is 'one', the number of words transmitted from the experiment to the RAU is not greater than 32, and no parity error is detected in the user's data words.

The data words received by the RAU will be processed and transmitted to the I/O unit in real time. Processing includes check of each parity bit, conversion from NRZ-L to Manchester II (Bi-phase-level) code and the generation of sync signals at the beginning of each data word and at the end of the data transfer.

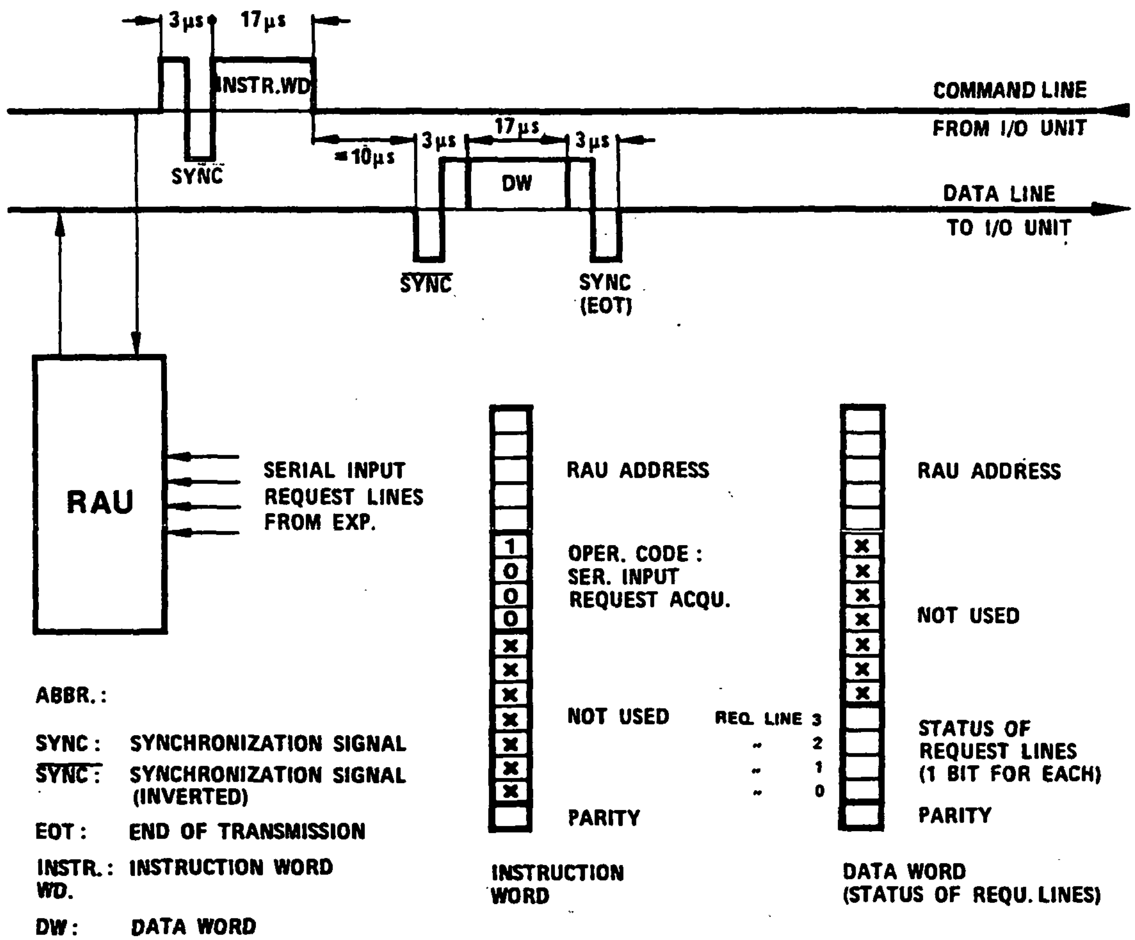

Figure 4.4 – 9: Transfer of Serial PCM Data Request Line Status

As depicted in Figure 4.4 - 9 there exists another possiblity to scan the status of the RAU request lines for serial PCM data inputs. This request line scan may be of advantage for experiments using several RAU input channels for serial PCM data with randomly distributed acquisition times. The dialogue between I/O unit and RAU for scanning the request line status of four channels of one RAU will take a maximum of 53 µs, while the data acquisition mode described above will need at least 132 µs to check four channels with request lines having "zero" status.

The dialogue will start with a sync signal and an instruction word transmitted by the I/O unit on the command line. The structure of this instruction word is shown in Figure 4.4 - 9. After a maximum time delay of 10 µs the RAU will answer with a data word as shown in Figure 4.4 - 9 preceded by an inverted sync signal and followed by a sync signal. Four bits of this data word (one for each channel) will contain the status of the request lines.

Flexible Inputs

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

- Analog Data

The acquisition of analog data is depicted in Figure 4.4 - 10.

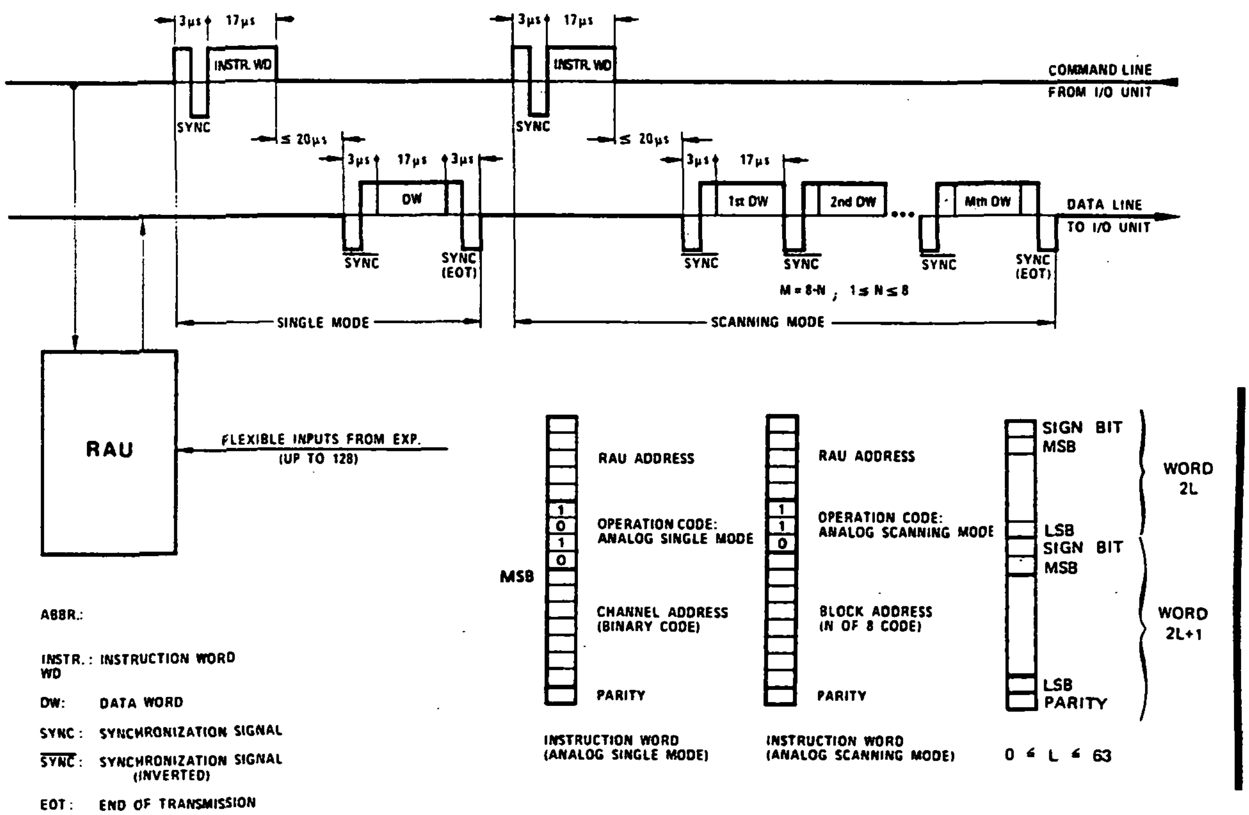

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Figure 4.4 - 10: Acquisition of Analog Signals via Flexible Inputs

The instruction word includes RAU address, RAU channel or block address and operation code. Included in the operation code is the information to acquire analog data and the sampling mode as shown in Figure 4.4-10. Two modes are possible:

- Analog Single Mode

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  This mode allows the sampling of two adjacent channels, an even one and the next odd one. In the binary channel address word the least significant bit is ignored thus selecting an even channel first, even if the following odd channel is addressed.

- Analog Scanning Mode

  This mode samples blocks of 16 input channels. The number of blocks acquired per software request is determined by the user and may vary from 1 to 8. This information is contained in the instruction word in an N of 8 code.

Each block address is directly correlated to 16 flexible hardware inputs. The correlation between hardware inputs and software channel and block address cannot be changed by the user and is described in Appendix A, Avionics Interface Definition, Section 4.1.

After receiving the instruction word the RAU initializes the 8 bit analog/digital conversion. The digitized signals of two input 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 20 µs after receiving the instruction word. As the analog/digital conversion circuitry consists of two sample and hold units and a fast A DC, there will be no time delay in addition to the transmission time determined by the RAU - I/O unit dialogue.

- Discrete Data

The acquisition of discretes is depicted in Figure 4.4 - 11.

The dialogue starts with a software generated instruction word which is sent on the command line to the RAU. The instruction word contains RAU address, operation code and the block address in an N of 8 code. The smallest unit which may be sampled is a block of 16 discrete inputs, transferred as 16 bit plus parity bit/word via the data line to the I/O unit. The number of blocks transmitted per software request may be in the range from 1 to 8. The correlation between software block address and pin allocation of the RAU is defined in Appendix A, Avionics Interface Definition (Section 4.1) and cannot be changed by the user. The transfer between RAU and I/O unit is completed by an RAU generated sync on the data line.

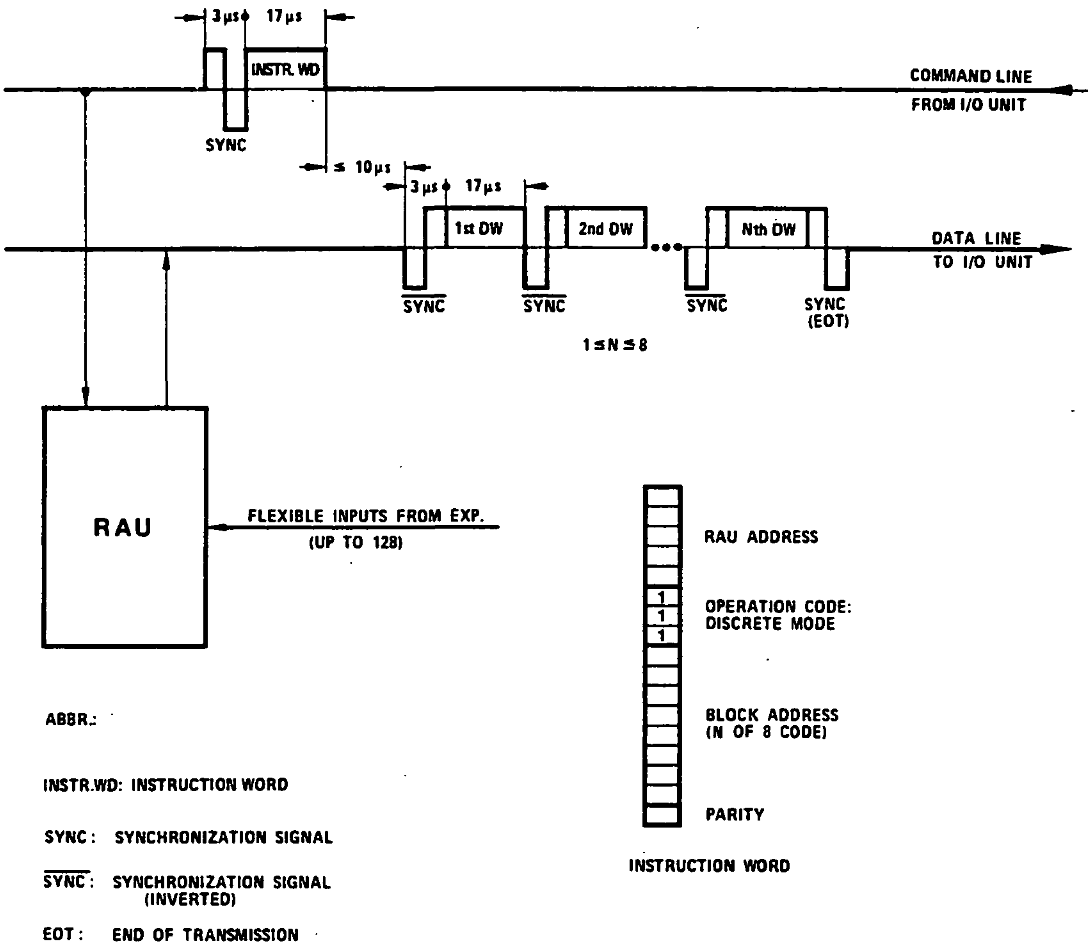

Figure 4.4 - 11: Acquisition of Discretes via Flexible Inputs

4.4.2.1.4.3 RAU Test Modes

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The RAU is designed to verify the quality of the received data. This applies to the instruction and data words from the I/O Unit and the serial data words from the experiment. In addition the I/O Unit will check the data words from the RAU.

Data and instruction words from the I/O Unit and RAU data words sent to the I/O Unit have to fulfill the following criteria:

- Word transmission at 1 Mb/s bit rate
- Manchester II coding properties
- Each word must consist of a sync (or $\overline{\text{sync}}$) + 16 data bits + parity bit
- Valid operation code (applicable to instruction words only)
- Odd parity of each word

If a word does not fulfil one of these criteria the word is considered invalid and will not be accepted by the receiving unit. In particular, the RAU will stop its work and the error is indicated in the RAU Internal Status Word (see Figure 4.4 – 12) by bit No. 12, which is set to logical level "one". In addition the RAU will not send back to the I/O Unit an acknowledge sync signal (see Figures 4.4-5 and 4.4-7) or an end of transmission signal (see Figures 4.4 - 8,9,10).

After the error has been detected, the I/O Unit (or more precisely the RAU-coupler of the I/O Unit; see also para 4.4.2.2) repeats either

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- the whole sequence if the failure is related to ON/OFF commands or acquisition of data via flexible inputs, signified by no RAU response or the reception of invalid data

or

DCN 048

- the instruction words only, if the failure is related to serial PCM commands or serial PCM data acquisition, signified by no RAU response

The results will be analysed to decide whether to switch over to the redundant part of the Experiment CDMS Data Bus (including the redundant Bus Interface Units) or to switch off the RAU.

Experiment data words at the RAU input will only be checked with respect to the experiment generated odd parity bit. If a parity error is detected the RAU will stop sending bit shift pulses to the experiment. In addition no end of transmission (EOT) sync (see Figure 4.4-8) will be sent back to the I/O Unit and bit No. 11 of the RAU Internal Status Word (see Figure 4.4 - 12) is set to level "one". In contrast to the traffic on the data bus, there is no automatic Self Test related to parity errors in the experimenters data. Therefore the user is recommended to acquire this status word periodically by his own request.

A more detailed test, including a check of the RAU analog-to-digital converter, is depicted in Figure 4.4 -13 and will be performed only on user's request.

DCN 048

If there is no RAU response or still invalid data after repetition as appropriate, the I/O unit will alert the computer with a BITE interrupt. The latter will then initiate tests to ascertain the cause of the interrupt.

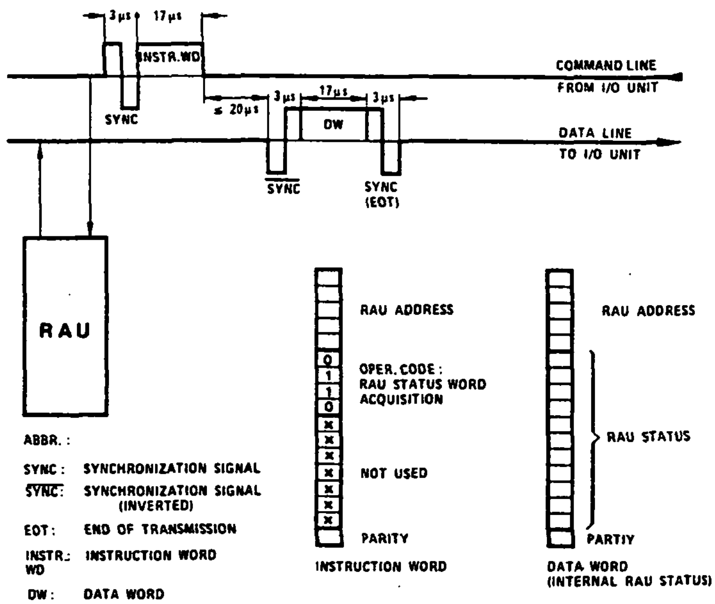


Figure 4.4 - 12: RAU Internal Status Word Acquisition

- RAU internal status word acquisition

The dialogue (Figure 4.4 - 12) starts with an instruction word on the command line. Within less than 20 μs, the RAU will return a data word containing:

Bit 0...4 RAU address

Bit 5 Primary Power Breakdown

The primary power voltage will be checked for under-voltage below 21.9 V during a period longer than 80 μs. The bit 5 will be set to "one" when the primary power recovers the normal range. This bit will be reset to "zero" by the acquisition of the Internal Status Word, and will keep this state if no voltage drop occurs. DCN 048

If no failure occurs, bit 5 will show "one" level for the first internal status acquisition, and always "zero" for the following acquisitions.

Bit 6 UTC status

This bit will be set to "one" in the case of complete or momentary absence of UTC signal coming from the I/O Unit.
This bit will be reset to "zero" after Internal Status Word acquisition.

Bit 7 Experiment Module ON/OFF Status

This bit is set to "one" if the RAU Exp. Module is powered.

Bit 8 INTERFACE Module Connection Status

This bit is set to "one" if the RAU INTERFACE module is physically connected and powered.

Bit 9 ON/OFF Command Status of the CORE Module

This bit, set to "one", will indicate that the ON/OFF command boards in the CORE module are energized.

Bit 10 ON/OFF Command Status of the INTERFACE Module

This bit, set to "one", will indicate that the ON/OFF command boards in the INTERFACE module are energized.

Bit 11 Serial PCM Input Channel Status

This bit is set to "one" if on any of the four serial input channels of the RAU

- the serial PCM clock is not working properly
- the user words show wrong parity

DCN 048

This bit is reset to "zero" after each acquisition of the status word.

Bit 12 I/O Unit - RAU Data Link Status

This bit is set to "one", if the RAU detects an error in the serial data coming from the I/O Unit.
After each acquisition of the status word, the bit is reset to "zero".

Bit 13 Spare

Bit 14 Spare

Bit 15 Spare

o RAU Test Command

The test command initiates the RAU BITE (Built-In-Test Equipment) cycle. This mode provides more detailed information about the status of the RAU including the previously mentioned Internal Status Word. The dialogue is depicted in Figure 4.4 - 13.

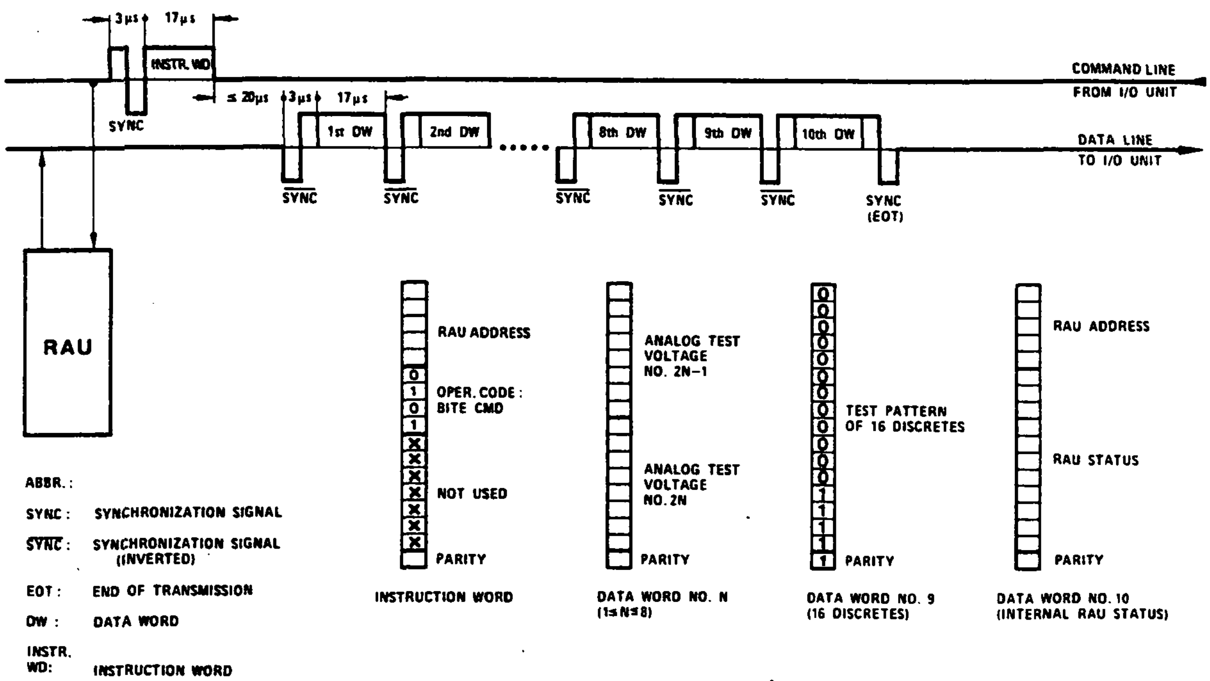


Figure 4.4. - 13: RAU Test Cycle

After an instruction word on the command line, the RAU will transmit 10 data words. The first 8 words contain 16 digitized analog reference voltages internally applied to hardware determined flexible input channels. In the same way 16 flexible inputs will be tested by applying the same voltage levels, but now handled as discretes. The result is transmitted in the 9th data word to the I/O unit. The 10th data word contains the RAU Internal Status Word.

Table 4.4 - 3 depicts the analog input voltages to be used in the RAU BITE mode.

Table 4.4 - 3 : Cross Reference between Analog Test Voltages and RAU Data Words in the BITE Mode

Test Voltage Number	Voltage Level (V)	Analog Acquisition Data Words No. 1 through 8		Discrete Acquisition Data Word No. 9	
		Affected RAU Data Word	Binary + Coded Value	Bit No.	Bit Level
1	-5.12	1	128	1	0
2	-4.48	1	144	2	0
3	-3.84	2	160	3	0
4	-3.20	2	176	4	0
5	-2.56	3	192	5	0
6	-1.92	3	208	6	0
7	-1.28	4	224	7	0
8	-0.64	4	240	8	0
9	+0.04	5	000	9	0
10	+0.68	5	016	10	0
11	+1.32	6	032	11	0
12	+1.96	6	048	12	0
13	+2.60	7	064	13	1
14	+3.24	7	080	14	1
15	+3.88	8	096	15	1
16	+4.52	8	112	16	1

+Specified Value, actual value may differ by an LSB. The actual level is specified for a certain RAU only.

4.4.2.1.5 Time Distribution

The Master Time Unit (MTU) in the Orbiter generates and distributes a central "on board time". The long term drift of the MTU will be 1×10^{-9} /day giving an accuracy better than 3 ms during a 7 days mission. The deviation of the onboard time from ground time will be controlled and 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.
From the Orbiter MTU two different time signals are derived in Spacelab and are available for experiment time tagging:

- The GMT serving as "macroscopic" time information.
  This GMT has a time resolution of 10 ms. It can be distributed to experiments via the RAU serial PCM command channels. The GMT is also inserted into the HRM data frames thus providing automatically a macroscopic time tagging of experiment data acquired by the HRM.
- The 1024 kHz User Time Clock (UTC) serving as "microscopic" time information. This UTC has a time resolution of 1 µs. It is distributed hardwired to the experiments via the RAU UTC channels.

The Spacelab time distribution system is designed to provide a relative accuracy of better than 10 µs.

Figure 4.4-14 shows a functional diagram of the time distribution system.

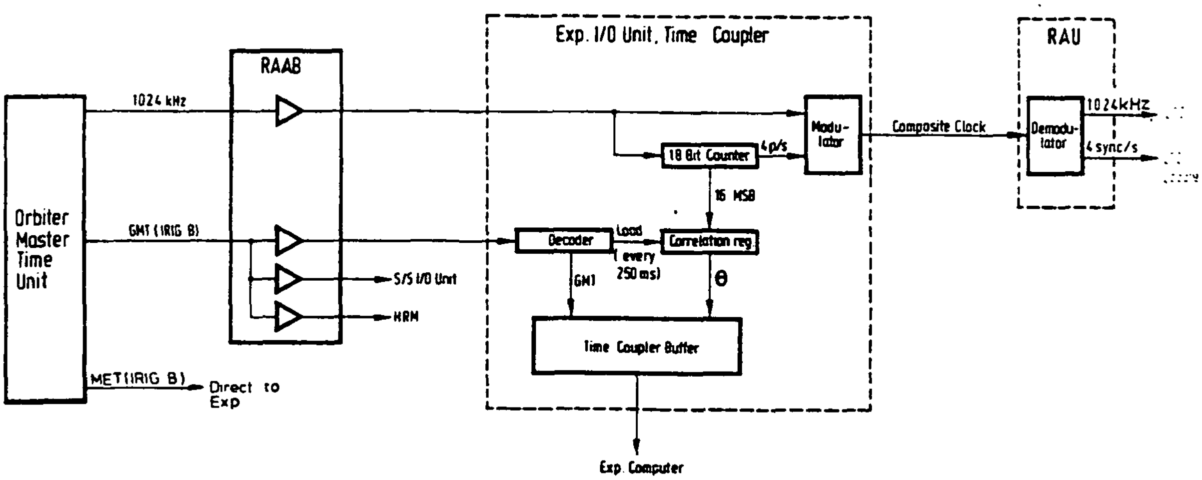


Figure 4.4 - 14: Functional Diagram of Time Distribution System

The MTU 1024 kHz signals are routed through the RAAB to the time coupler in the experiment I/O Unit.

The time coupler generates the "UTC update" which is a 250 ms signal derived from the 1024 kHz clock.

This is done by:

- o incrementing a 18 bit counter which is reset every 250 x 1024 pulses.
- o forming a composite clock by modulating the 1024 kHz clock every 250 ms, 8 pulses before the counter reset.

At the RAU level, the composite clock signal is demodulated in order to provide the two signals UTC (1024 kHz) and UTC update (4 pulse long, sync. every 250 ms). The end of the UTC update sync pattern is correlated to the 18 bit counter reset. The detailed phase relationship can be seen from Figure 4.4 - 15.

The time coupler also performs the correlation between the UTC update and the GMT. Every 250 ms, synchronously with the GMT, the 16 most significant bits (MSB) are loaded into the correlation register. This 16 bit word Θ represents the correlation between GMT and UTC update with an uncertainty of 4 µs because the last two bits of the 18 bit counter have been dropped.

Both the decoded GMT and Θ are transferred periodically via the time coupler buffer into the experiment computer.

Θ is used to time tag experiment data in the experiment computer with an accuracy of 10 µs. For this time tagging method it is assumed that the experiment contains a time counter(counting the 1024 kHz UTC pulses)which is reset by the UTC update signals every 250 ms. For each experiment event the event data have to be acquired together with the related contents of the experiment time counter. The experiment computer then, by means of Θ , calculates back the experiment time counter contents to the on board time. However, in order to relate the event unambiguously to the on board time, the data acquisition and computation has to be performed less than 250 ms after the event.

Θ also allows, together with UTC, UTC update and GMT, time tagging to be performed autonomously in the experiment. In this case Θ, or better an averaged Θ compensating GMT signal jitter, can be sent to the experiment via a serial PCM command channel for the correlation between UTC and GMT.

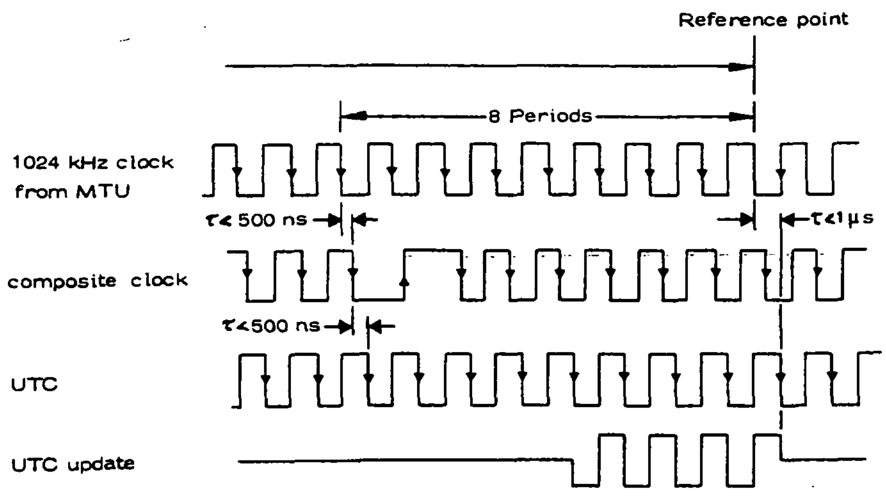
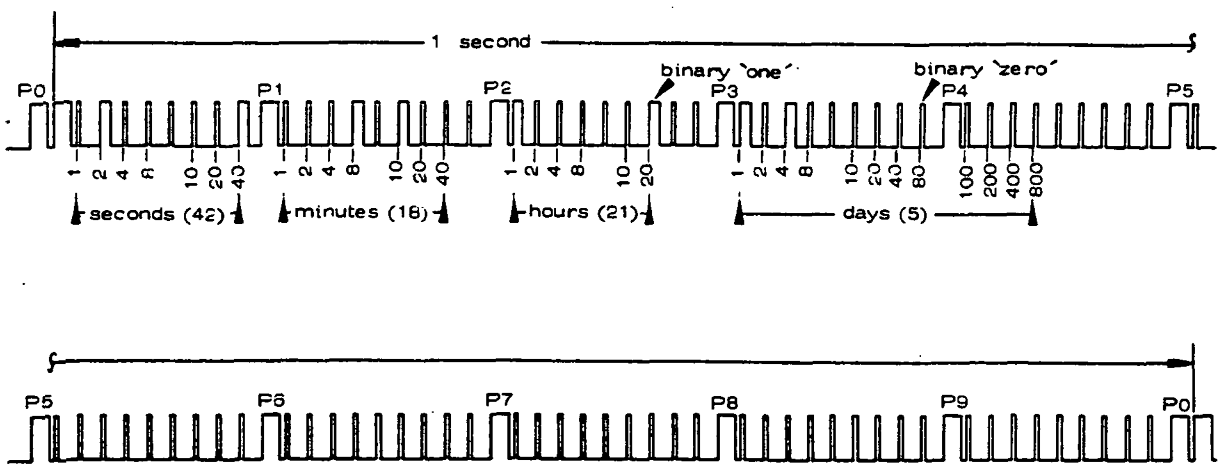


Figure 4.4 - 15: Phase Relation Between MTU Clock Signal and UTC Clock Signals

In addition to the time information distributed to the experiments through the CDMS a direct interface to the Orbiter MTU is available for experiments. The time delivered is the Mission Elapsed Time (MET). The interface is at CB 5 in module configurations and at CB 57 in pallet only configurations.



P0 thru P9 : Position Identifier

Figure 4.4 - 16: MET Output Format

As shown in Figure 4.4-16 the time information is delivered in a modified IRIG-B format, i.e. a pulse width modulated pulse train (100 pulses per second) contains the BCD coded information in seconds, minutes, hours, and days.

The main characteristics of this format are

- Frame: 1 second length
- Subframe: separated by position identifier P 0 through P 9. The first 5 frames contain seconds, minutes, hours and days BCD coded. The remaining frames are empty.
- Position identifier: pulse of 8 ms duration
- Binary 1 pulse: pulse of 5 ms duration
- Binary 0 pulse: pulse of 2 ms duration

The IRIG B Format is modified such that the "Straight Binary Seconds" which begin at Index Count 80, will not be generated. The IRIG Format will be modulated with a 100 PPS output rate and a resolution of 10 milliseconds. The IRIG B Format code will be transmitted with the least significant bit transmitted first.

4.4.2.2 Input/Output Unit

All communications between the computers and the rest of the 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 4.4-17.

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

Remote Acquisition Units and High Rate Multiplexer Data Display Units and Keyboards Mass Memory	CDMS
MDM PCM Master Unit 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 with the rest of the CDMS or Orbiter as appropriate via serial data busses. Only one coupler of a redundant pair is powered at any time. The switch-over from one coupler to a redundant one will be performed by the following procedure :

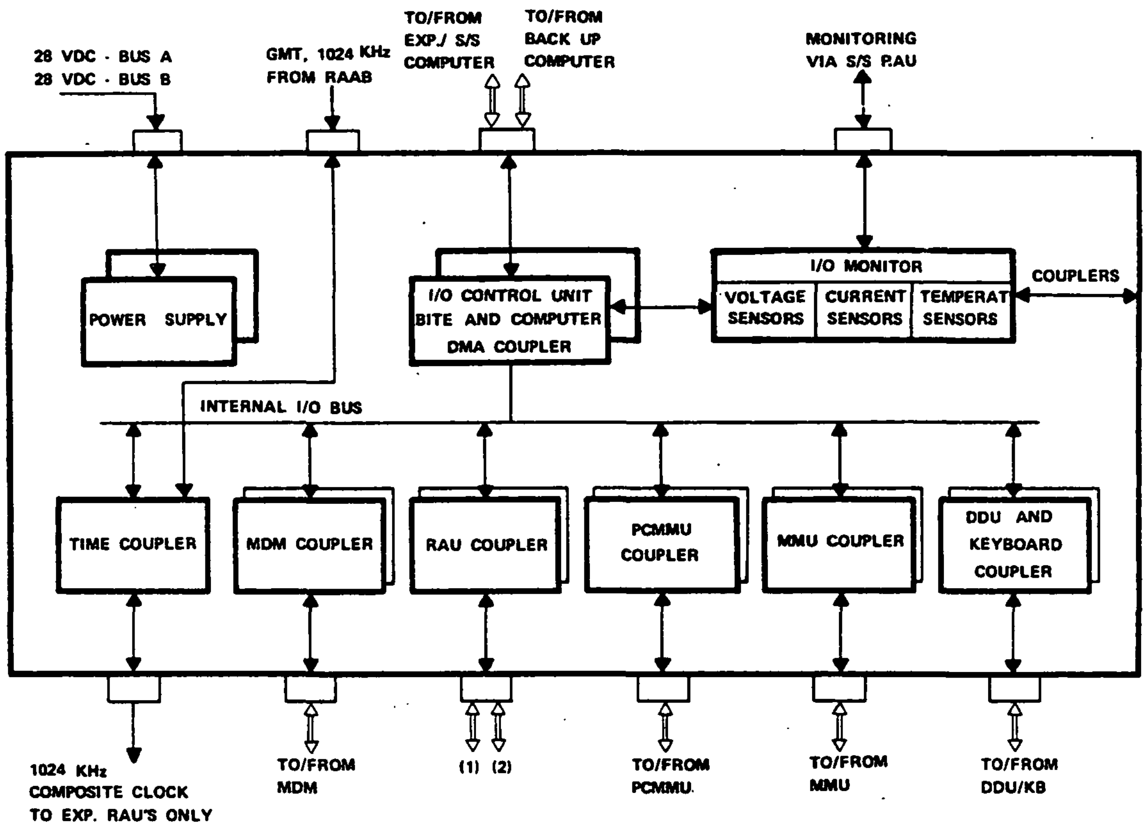
- switch off computer power
- switch over to the redundant coupler
- switch on computer power
- initiate restart procedure *)

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 which is 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.

*) Note: Present baseline requires sequence of commands via keyboard. An 'Auto-Restart' sequence will be introduced thru ECP MA - 50 556 - xx. For more details see RP - MA - 0019 .



(1) TO/FROM EXP./ S/S RAU'S CONNECTED TO MAIN BRANCH OF EXP. S/S DATA BUS

(2) TO/FROM EXP. RAU'S CONNECTED TO AIRLOCK BRANCH OF EXP. DATA BUS OR TO/FROM S/S RAU'S CONNECTED TO AFD BRANCH OF S/S DATA BUS

NOTE: THIS DIAGRAM DOES NOT REFLECT MONITORING AND CONTROL OF THE I/O UNIT VIA MDM'S

Figure 4.4 - 17: Simplified Block Diagram of the I/O Unit

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 hardwired priority basis by the I/O Control Unit. The priority levels are

1	PCMMU coupler	4	MMU coupler
2	RAU coupler	5	DDU / keyboard coupler
3	MDM coupler	6	Time coupler

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The time coupler also provides a hardwired Real Time Clock Interrupt to the computer derived from the IRIG-B coded GMT of the Orbiter MTU and programmed for a period in multiples of 10 milliseconds. These real time interrupts are used in connection with the Periodic Input/Output Loop (PIOL).

4.4.3 High Rate Data Acquisition

The High Rate Data Acquisition part of the CDMS is capable of time multiplexing digital experiment data from up to 16 different sources with a data rate of up to 16 Mb/s, together with data from the CDMS computers, voice and on-board time. To bridge mission periods with no downlink capability, the High Rate Data Acquisition Assembly includes digital recorders and provisions to interleave the playback data into the real time data stream. A single source can be accepted with a data rate of up to 50 Mb/s for direct transmission and of up to 32 Mb/s for recording.

The High Rate Data Acquisition assembly comprises the following equipment:

On board:

- The High Rate Multiplexer (HRM) to time-multiplex the input data and to perform the routing of the composite output data stream to one of the two recorders and/or one of the three KU-band processor (KUSP) inputs. In addition, the HRM includes a voice digitizer to collect and multiplex also the data from the Spacelab voice channels.
- The High Data Rate Recorder (HDRR) to store data at rates up to 32 Mb/s during mission periods with none or degraded downlink capability.
- The Payload Recorder (Orbiter equipment) serving as back-up for the HDRR for data rates up to 1024 kb/s.

On ground

- The High Rate Demultiplexer (HRDM) to demultiplex the composite data stream to recover on ground the same channels as presented at the HRM inputs on board.

The data flow within the High Rate Data Acquisition Assembly is shown in Figure 4.4 - 18.

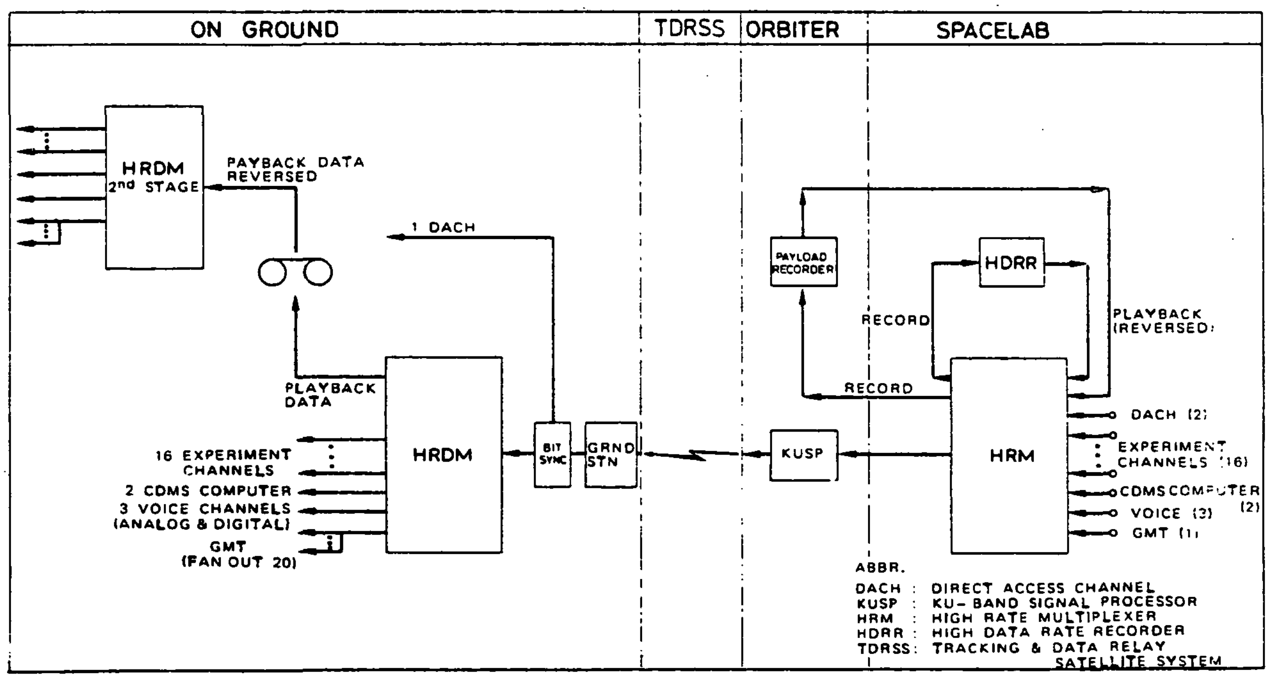


Figure 4.4 - 18: High Rate Data Acquisition Data Flow

4.4.3.1 High Rate Multiplexer (HRM)

Since the HRM represents the core of the High Data Rate Assembly, the tasks of the HRM are not constrained to the actual data multiplexing. The HRM also controls the data routing within the on-board part of the High Data Rate Assembly; it performs the voice digitizing and GMT encoding, and it provides the electrical interface circuits to the on-board interlinking equipment. DCN 048

The main characteristics of the HRM are listed in Table 4.4 - 4.

Table 4.4 - 4 High Rate Multiplexer Characteristics

Outputs to KUSP	
bit rate	48 Mb/s, 32 Mb/s to 125 kb/s in binary steps
code for 50 Mb/s KUSP input	NRZ - L + clock
code for 2 Mb/s and 4 Mb/s inputs	NRZ - L without clock
Output to HDRR	
bit rate	32-16-8-4-2-1 Mb/s
code	NRZ - L + clock
Output to Payload Recorder	
bit rate	1 Mb/s to 125 kb/s (binary steps)
code	Manchester biphase L
Input from HDRR	
bit rate	32-24-16-12-8-4-2 Mb/s
code	NRZ - L + clock
Input from Payload Recorder	
bit rate	1 Mb/s
code	Manchester biphase L
Experiment Input Channels	
number	16
nominal bit rate	
at 48 Mb/s HRM output rate	16 Mb/s to 62.5 kb/s (1)
at 32 Mb/s HRM output rate	16 Mb/s to 41.7 kb/s
at HRM output rates lower than 32 Mb/s	above rates divided by ratio 32 Mb/s to actual output rate
Direct Access Channels	
number	2
maximum bit rate	50 Mb/s
CDMS Computer Channels	
number	2 (1 for S/S, 1 for exp. computer)
maximum data rate	25.6 kb/s (2)
GMT Channel	
resolution for HRM output rates ≧ 1 Mb/s	10 ms
resolution for HRM output rates ≦ 1 Mb/s	4 frame lengths (3)
Voice Channel	
number of analog inputs	3
total bit rate	128 kb/s
algorithm	Abate variable slope delta modulation

DCN 048

DCN 048

(1) for details see 4.4.3.1.4

(2) for details see 4.4.3.1.6

(3) for details see 4.4.3.3

4.4.3.1.1 Data Routing

The overall data routing capability can be seen in Figure 4.4 - 19. In particular, the HRM is capable of performing the following routing configurations:

1. Multiplexed experiment data routed to one of the 3 KUSP-inputs for real time transmission.
2. Multiplexed experiment data recorded on one of the 2 recorders (simultaneously with real time transmission, if required).
3. HDRR output routed directly to one KUSP-input. Multiplexed data stream switched-off or routed to another KUSP input (for possible KUSP modes see Table 4.4 - 5) or recorded on Payload Recorder.
4. Same as 3, but functions of HDRR and Payload Recorder interchanged.
5. Direct access channel routed to the 2 ...50 Mb/s KUSP-input. Multiplexed data stream switched-off or routed to another KUSP input or recorded on one of the 2 recorders.
6. Direct access channel routed to the HDRR and recorded. Multiplexed data transmitted in real time or recorded on Payload Recorder.

The HRM routing modes will be commanded by the subsystem computer via the subsystem data bus and the BIU interface dependent on downlink availability, Ku-band signal processor operation mode and multiplexed data rate.

DCN 048

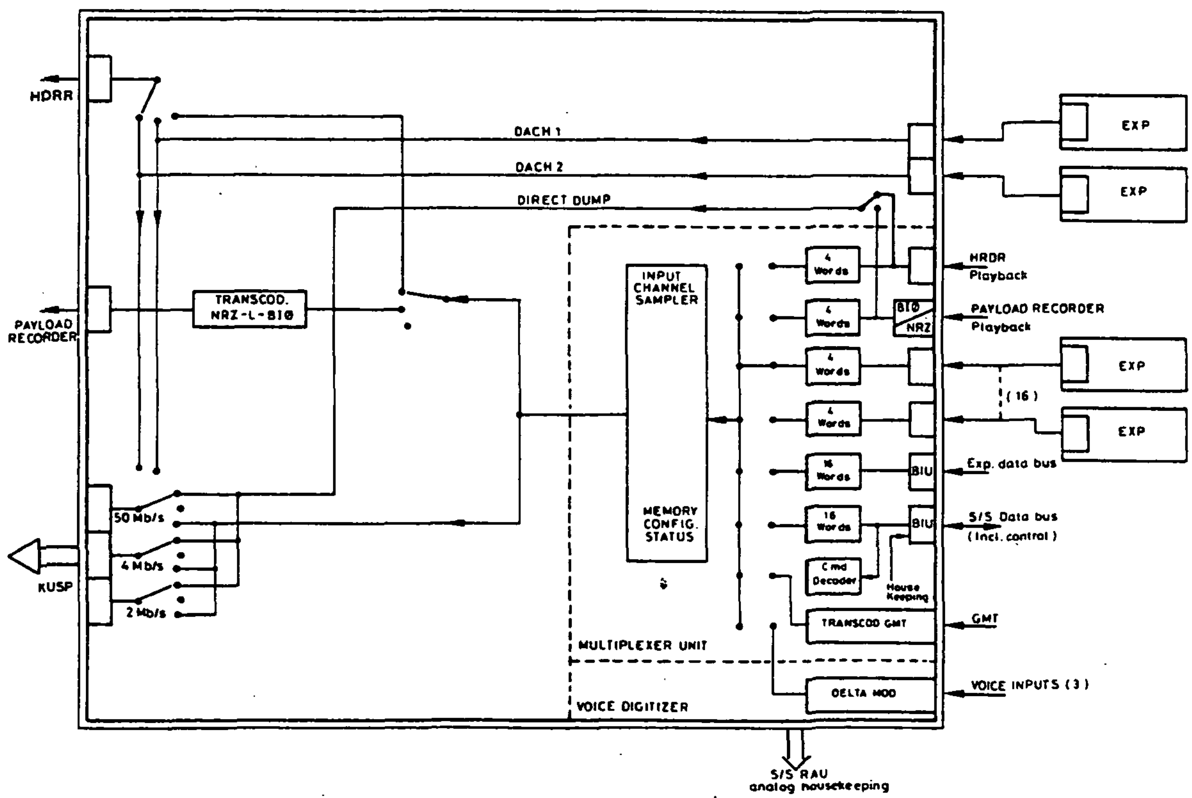


Figure 4.4 - 19: HRM Block Diagram

4.4.3.1.2 Voice Digitizer

The HRM accommodates three voice channels, coming from the SL Intercom Master Station in module configurations and, if required, from the Orbiter ACCU in pallet-only configurations.

A built-in voice digitizer performs the necessary analog/digital conversion for the three voice channels and the time division multiplexing of the digitized voice signals.

The voice digitizer transforms each analog voice signal to a 32 kb/s digital signal; delta modulation is used for A/D conversion.

Each of the three voice digitizer circuits transmits four bits in parallel at a sampling frequency of 8 kHz into an input buffer. 3 bits for the ON/OFF status of each channel plus one spare bit complete the parallel 16 bits. Thus, if a configuration with voice is programmed, the bit rate to be allocated to this input is fixed at 128 kb/s. The data flow in the HRM voice channel is depicted in Figure 4.4 - 20.

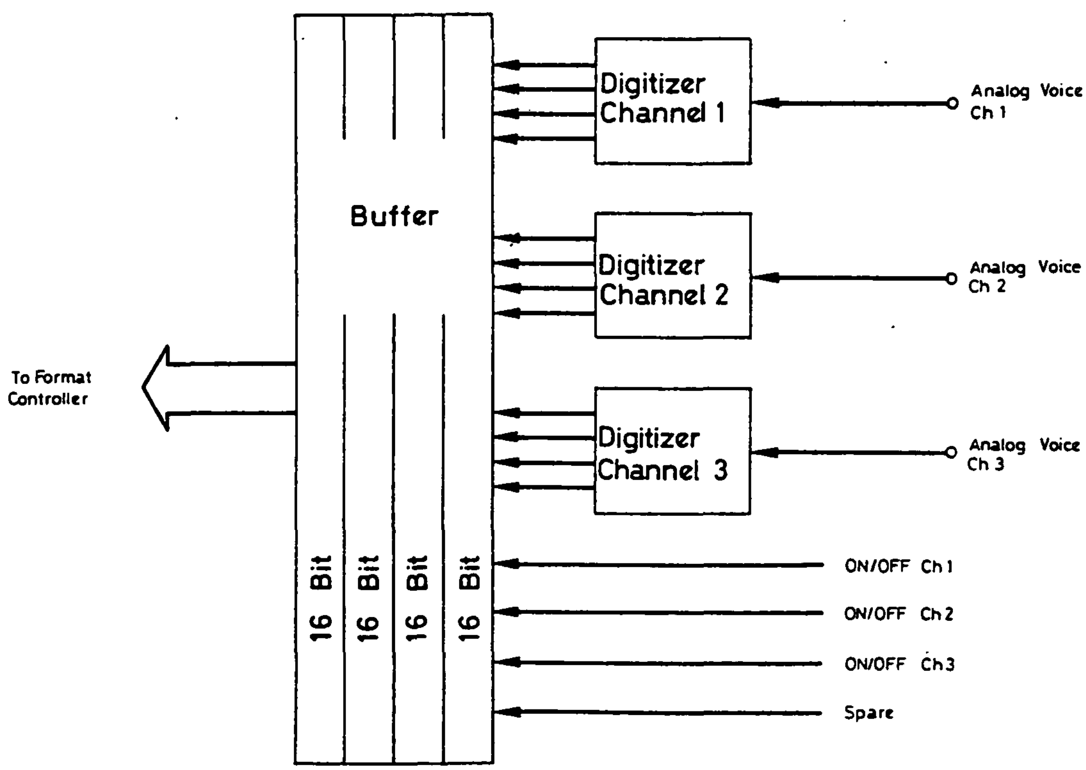


Figure 4.4 - 20: HRM Voice Channel Diagram

4.4.3.1.3 Multiplexing Concept

The HRM collects serial data from different sources, performs a time division multiplexing based on 16 bit time intervals and, finally, delivers an output of one serial data stream containing all the input data.

The main characteristic of the concept employed is the capability to accept serial data that are completely asynchronous with respect to the HRM internal clock. As shown in Figure 4.4 - 21, the decoupling of the input clock from the HRM internal clock is performed by means of 4 x 16 bit input buffers.

The user clocks in his data into a 16 bit shift register; then - after 16 bits - the content will be loaded into the 4 x 16 bit buffer.

In a sequence determined by the format loaded, the format controller fetches one 16 bit word out of the input buffer and transfers it to the output register, where it will be serialized. In the case of an empty input buffer, a fill word is introduced, which can be identified as such by means of the fill identification as a part of the frame overhead. During demultiplexing on ground, the fill words are automatically suppressed.

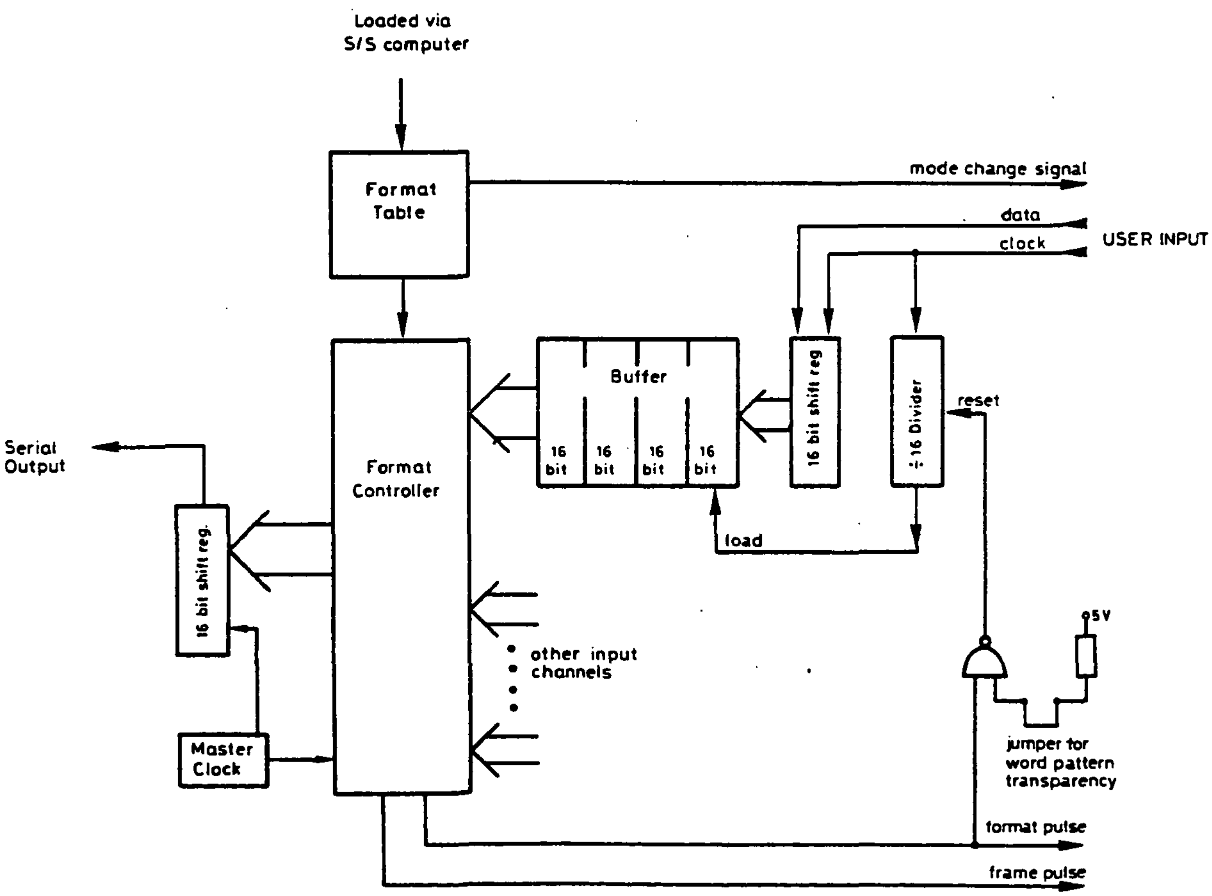


Figure 4.4 - 21: Multiplexing Concept

By this method, only two constraints are imposed on input data rates:

- The input bit rate averaged over any sequence of 64 bits shall not be higher than the nominal data rate allocated.

If the input bit rate is higher, the input buffer will overflow. Overflows will be announced to the subsystem computer.

- The peak bit rate shall not be higher than 16 Mb/s.

This constraint is due to the hardware limitation of the HRM input circuits.

The user delivering serial data to the HRM will, on ground, recover his data from the HRDM completely unchanged. This means that the user himself has to take care of the formatting and structuring of his serial data. To facilitate this task, each HRM experiment channel can operate in two different modes:

Normal Mode:

In this mode, the word structure in the HRM output frames are not at all correlated with any structure of the input data.
The serial input data are arbitrarily chopped into 16 bit words for parallel processing inside the HRM. Consequently, the user has to insert some kind of sync pattern into his serial input bit stream, in order to be able to extract on ground his scientific data out of the serial bit stream of his output channel.

- Word Pattern Transparency Mode:

In this mode, the input data can be structured in words that, after multiplexing, can be identified as words in the HRM output frames in those positions determined by the chosen format. Synchronously with the frame or format pulse, which indicates the beginning of a new frame or format respectively, experiment data can be delivered to the HRM in bursts of 16 bit words. Because the clock counter is reset at the beginning of each format, these words are identical to the internal words the HRM handles in parallel.

The HRDM in this mode delivers the data words without bit rate smoothing at the nominal bit rate allocated to the particular experiment channel.

The mode - normal or word pattern transparency - of each input channel is determined by an external HRM connector. This connector is programmed by hardwired jumpers on a mission-to-mission basis.

It should be noted that in the word pattern transparency mode the words are delivered as 16 bit bursts and not as continuous bit stream. This has to be taken into account for further on ground data handling because it might rule out the use of standard ground decommutation and recording equipment. DCN 048

4.4.3.1.4 Multiplexing Format

The format defines the arrangement of data words in the multiplexed stream output from the HRM (and the input to the HRDM on the ground).

The basic format parameters are:

- o format length : 8 frames
- o frame length : 96 words
- o word length. : 16 bits
- o sync pattern : 32 bits
- o status pattern : 32 bits

As shown in Figure 4.4 - 22, the format is structured in two different ways:

Format Structure 1 is used for HRM output bit rates of 32 Mb/s and all lower output rates being binary ratios of 32 Mb/s.

Format Structure 2 is used for the 48 Mb/s output rate only, since it is not binary ratio of 32 Mb/s.

The frames are organized in 6 lines and 16 columns in Format Structure 1 and in 8 lines and 12 columns in Format Structure 2. The last word in each line is occupied by the fill word identification.
Even frames start with a sync pattern, odd frames with a status pattern.

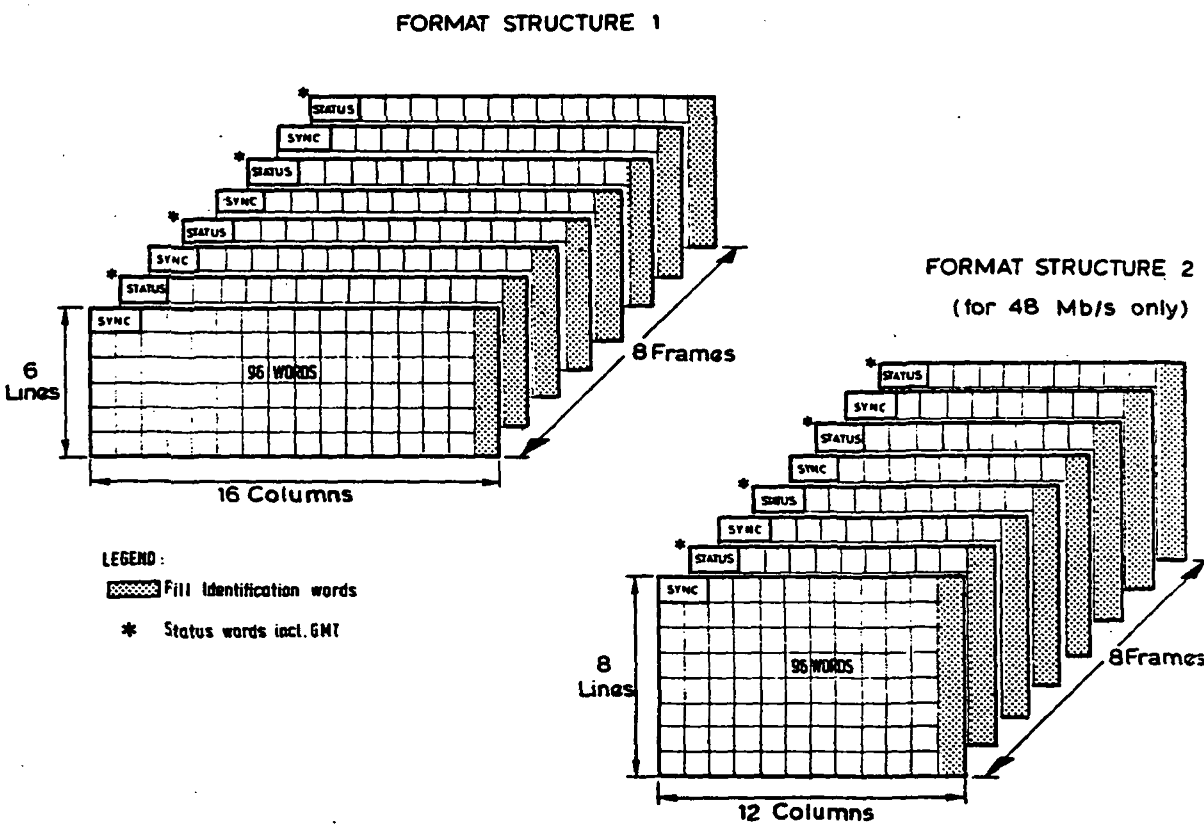


Figure 4.4 - 22: HRM Format Structure

Sync Pattern:

The sync pattern has been selected to fulfill certain requirements about sync loss. The sync word is 32 bit long, hardwired on a connector. 28 bits are actually used for the frame synchronization. The 4 remaining bits count the frames over a period of 32 frames by labeling the even frames from 0 thru 15.

Status Words:

For the transmission of HRM status information, two words are used in every odd frame. These two status words have the following contents:

- 23 bit configuration status
  - o 4 bit HRM output bit rate
  - o 3 bit HDRR reproduction bit rate
  - o 10 bit routing configuration
  - o 6 bit format identification
- 1 bit flag identifier of format change
- 8 bit GMT/flight number subcommutated over 32 frames.

DCN 048

Format Programming:

The programmer has the flexibility to assign the user's share of the HRM output data rate with 3 types of instructions:

- Mode 1, Subcommutation

  The number of data words 'n' to be sampled are specified as words per format. The resulting user's share 'f' of the HRM output data rate 'F', being the nominal data rate allocated, is given by:

  $$f = (n/768)\ F$$

- Mode 2, Commutation

  The number of data words 'n' to be sampled are specified as words per frame. The resulting unser's share 'f' of the HRM output data rate 'F', being the nominal data rate allocated, is given by:

  $$f = (n/96)\ F$$

- Mode 3, Supercommutation

  The number of data words 'n' to be sampled are specified as words per line, i.e., as columns per frame. The resulting user's share 'f' of the HRM output data rate 'F,' being the nominal data rate allocated, is given by:

  $f = (n/16)\ F$ (Format Structure 1)

  $f = (n/12)\ F$ (Format Structure 2)

DCN 048

The formatting of the HRM is accomplished with a formatting table containing 16 instruction words and 2 configuration status words.

Each instruction word contains two instructions of the same mode for two different input channels.

In Figure 4.4 – 23, an example of a particular HRM format is given. Table 4.4 – 5 shows the set of instructions defining this format. The nominal data rates allocated to each input channel as a result of this format are given in Table 4.4 – 6.

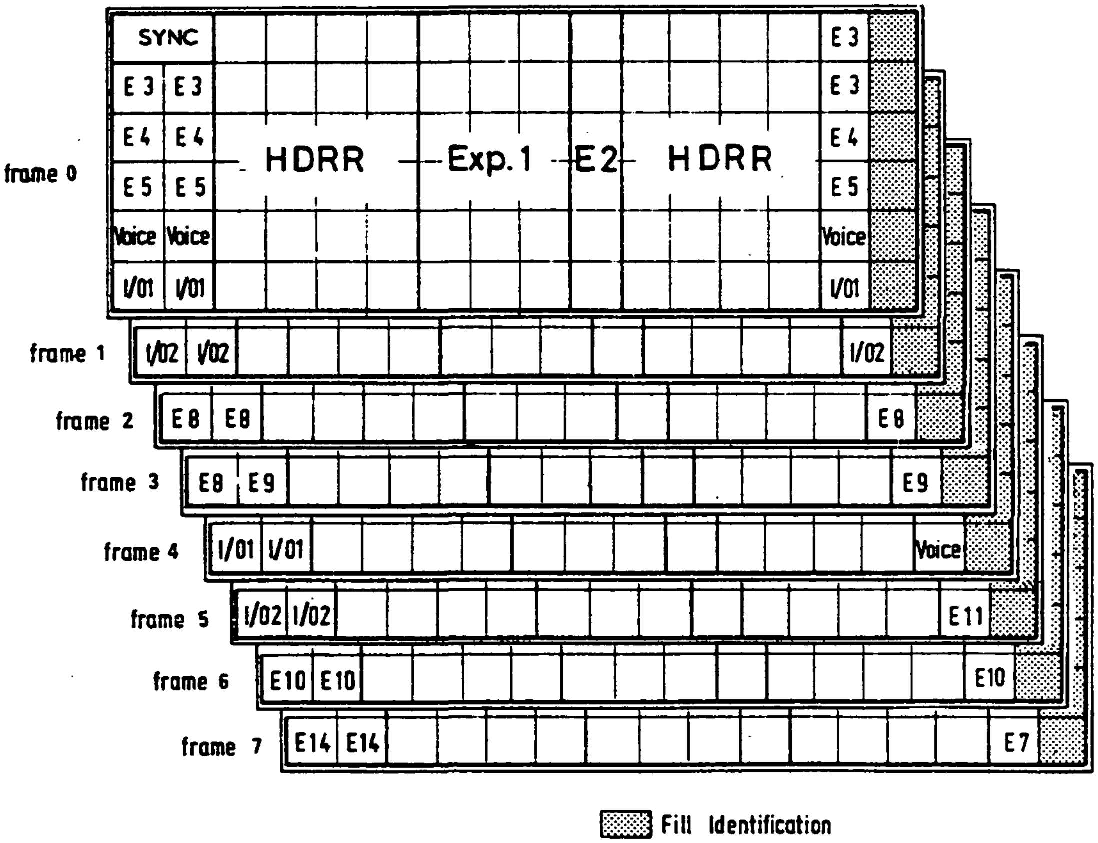


Figure 4.4 – 23: Example of HRM Format

Table 4.4 - 5: Example of Instruction Table

Instruction Word	Mode	Channel	Words	Channel	Words
1	3	HDRR	4	E 1	3
2	3	E 2	1	HDRR	4
3	2	E 3	4	E 4	3
4	2	E 5	3	Voice	3
5	1	I/01	3	I/02	3
6	1	E 8	4	E 9	2
7	1	I/01	2	Voice	1
8	1	I/02	2	E 11	1
9	1	E 10	3	E 14	2
10	1	E 7	1	Skip	0
11	0	Skip	0	Skip	0
12	0	Skip	0	Skip	0
13	0	Skip	0	Skip	0
14	0	Skip	0	Skip	0
15	0	Skip	0	Skip	0
16	0	Skip	0	Skip	0
17	Configuration Status				
18					

Table 4.4 - 6: Example of Data Rate Sharing

HRM Output Data Rate = 4 Mb/s	
Data Rate Shares	Data Rate Shares
E 1 ≤ 750 kb/s	E 11 ≤ 5.2 kb/s
E 2 ≤ 250 kb/s	E 12 = -
E 3 ≤ 166.6 kb/s	E 13 = -
E 4 ≤ 125 kb/s	E 14 ≤ 10.4 kb/s
E 5 ≤ 125 kb/s	E 15 = -
E 6 = -	E 16 = -
E 7 ≤ 5.2 kb/s	I/O 1 ≤ 26 kb/s
E 8 ≤ 20.8 kb/s	I/O 2 ≤ 26 kb/s
E 9 ≤ 10.4 kb/s	HDRR ≤ 2000 kb/s
E 10 ≤ 15.6 kb/s	PLR = -
	Voice ≤ 130.2 kb/s

(HRM Voice fixed at 128 kb/s)

4.4.3.1.5 Ku-Band Link

The constraints of the Ku-band link on the HRM outputs affect the experiment data only with respect to bit transition density. For resynchronization of data on ground, the requirements on bit transitions are as follows:

The transition density shall average 64 transitions in 512 bits with a maximum separation of 64 bits between successive transitions.

DCN 048 The HRM format structure cannot guarantee the required transitions with the sync status, and fill identification words alone.

There are two ways to assure a sufficient bit transition density:

- All experiments must accept the constraint that their data sent to the HRM shall contain not more than 15 consecutive "1"s or "0"s.
- DCN 048 The HRM provides for programming of a dummy channel with "01"s. This will increase the HRM overhead from 8.3 % for Format Structure 1 to a maximum of 33.3 % in the worst case, and from 10.4 % for Format Structure 2 to 35.4 % in the worst case.

The "Dummy Channel" can be programmed into the format like any other channel. A maximum of 4 (Format Structure 1) and 3 (Format Structure 2) columns have to be introduced into the format, evenly spaced.

The introduction of dummy channels into a HRM format will have to be based on experiment data expected and the probability of cases which could violate the signal transition density requirement. This assessment must be performed by the Payload Integration and the Mission Planning organisations.

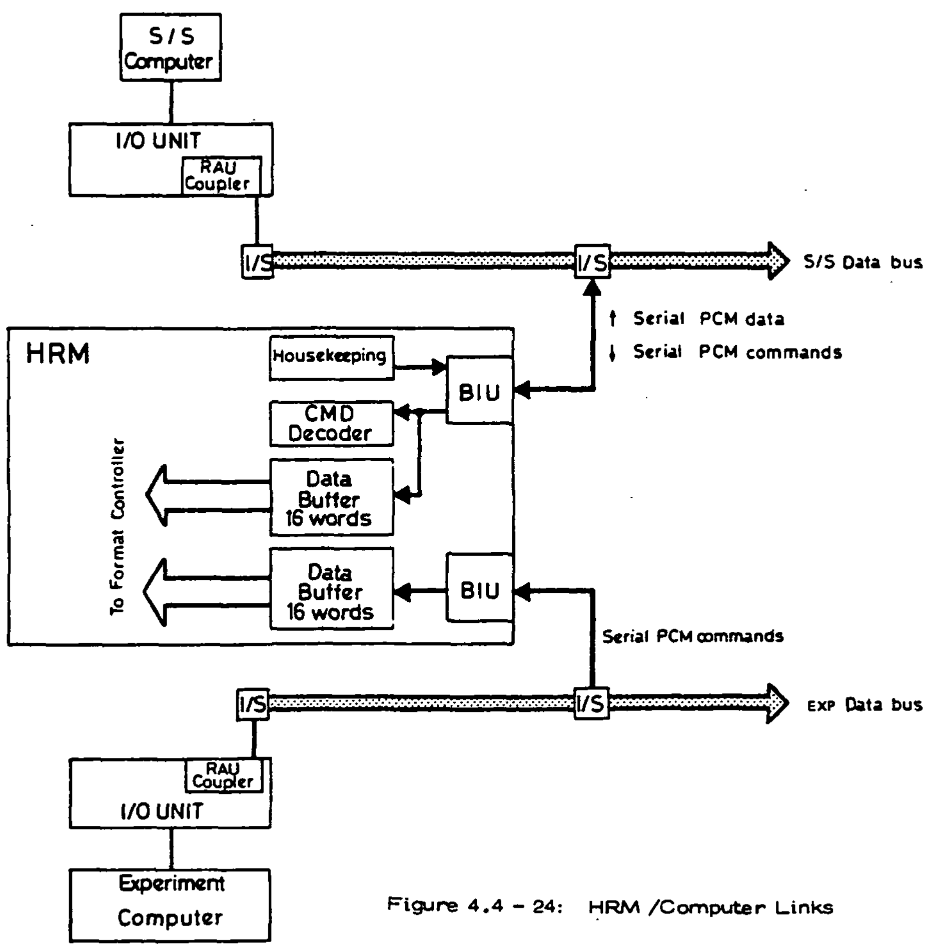
4.4.3.1.6 CDMS Computer Links

Via internal Bus Interface Units (BIU´s) the HRM is connected to the subsystem and experiment RAV busses. Thus the HRM is linked to the I/O units and the CDMS computers. The I/O units treat the HRM BIU´s like normal RAU . Data from the computers to the HRM are transfered as "serial PCM commands". The principles of the HRM / computer links are shown in Figure 4.4 - 24. DCN 048

Control and monitoring and configuring functions of the HRM are performed by the subsystem computer only. The subsystem BIU part of the HRM is capable of detecting commands in the incoming data stream and is capable of sending housekeeping data from the HRM to the subsystem computer. The experiment BIU part of the HRM serves for data transfer from the experiment computer to the HRM only.

The HRM contains 16 word input buffers at the CDMS computer input channels. Thus, blocks of up to 16 words can be transferredper GML cycle. Assuming a 10 ms GML cycle, this results in an effective data rate of up to 25.6 kb/s.

Using an HRM format that allocates a nominal data rate of 1 Mb/s to the HRM data bus input, the HRM can accept any possible multiple of 32 word blocks within a GML cycle. In this case, the size of the input buffer is no longer the limiting factor because with 1Mb/s allocated, the input buffer is emptied faster than filled. More efficient channel usage may be obtained by distributing the word blocks of data across the full period of a 10 ms GML cycle, thus reducing the allocated nominal data rate required.



DCN 048

Figure 4.4 - 24: HRM /Computer Links

4.4.3.1.7 Experiment Interfaces

Experiments interface with the HRM via:

- inputs
  - 16 experiment input channels
  - 2 direct access channels (DACH) to KU-band signal processor (KUSP) and/or High Data Rate Recorder (HDRR)
- outputs
  - 1 frame pulse output
  - 1 format pulse output
  - 1 mode change signal output

DCN 048

DCN 048 The physical location of these interfaces is at CB 50 for module configurations and at the Igloo feedthrough (CB 42) for pallet only configurations. The electrical characteristics are given in detail in Appendix A, Avionics Interface Definition.

Experiment Input Channels

These channels are the standard HRM inputs for experiment data up to 16 Mb/s. The code for the experiment input channels is NRZ - L + clock. Both data and clock are generated by the user. They will be accepted by the HRM at any bit rate or as bursts within the constraints stated in 4.4.3.1.3, since the HRM is designed for asynchronous operation.

DCN 048 Although the interface is clearly at CB 50/CB 42, the HRM design comprises the design of a high rate experiment HRM / data link (see Figure 4.4 - 25).

This may serve as support to the experiment developer, especially in case of high data rates requiring a low phase shift margin between data and clock signals.

The differential input line receivers in the HRM, which are identical for the 16 experiment input channels and the 2 direct access channels, have the following main characteristics:

type of input:	differential, DC coupled
input impedance	
line to line:	125 Ohm
each line to ground:	1 MOhm
differential input voltage range:	0.5 V to 1.5 V
common mode input range:	- 3 V to + 3 V

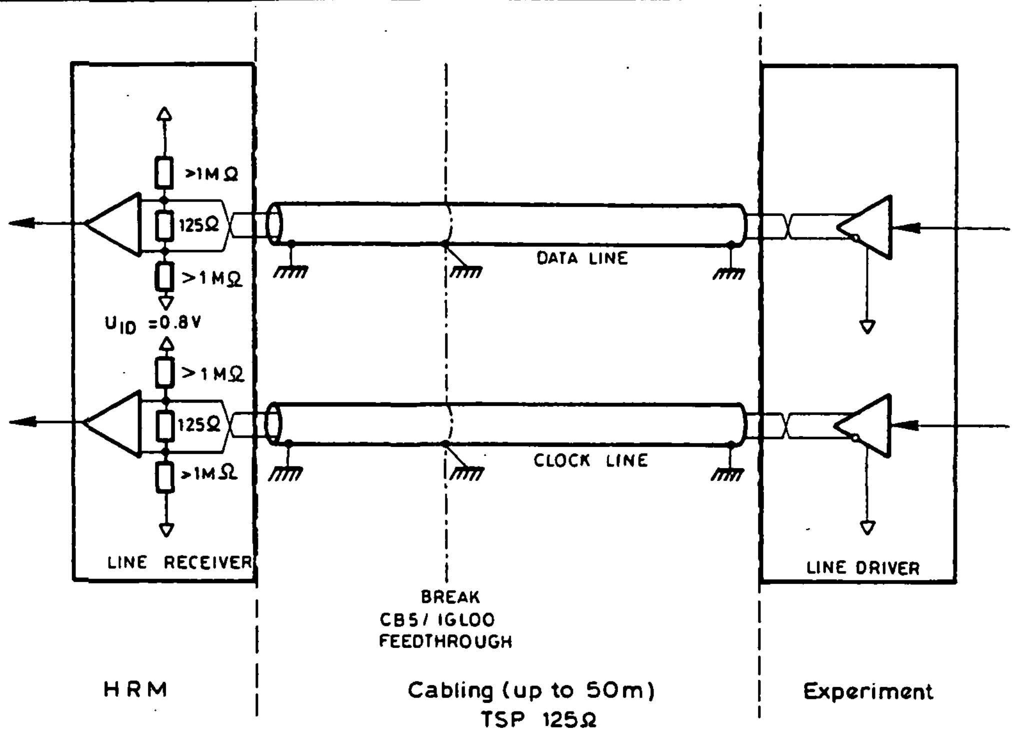


Figure 4.4 - 25: Experiment /HRM Data Link

Direct Access Channels:

For these channels the HRM acts more or less as a switching panel, i.e. after signal reconditioning the data are routed directly to the KUSP and/or the HRDR. The code of the data will be NRZ-L + clock. The frequency limits are determined by the end items themselves. For the KUSP the range is 2 to 50 Mb/s and for the HRDR the range is 1 to 32 Mb/s. As the HRM does not provide any buffering, the frequency has to be constant. The line receiver of the direct access channel is identical to the experiment input channel line receiver.

Frame Pulse Output

At the beginning of each format the format pulse is generated. This signal may serve as a sync for users employing the word pattern transparency mode.

Format Pulse Output

At the beginning of each format consisting of 8 frames, the format pulse is generated. This signal may serve as a sync for users employing the word pattern transparency mode at low data rates.

Mode change signal output

This signal indicates when a new format program loaded into the HRM is executed and thus the operational mode of the HRM is changed. DCN 048

4.4.3.2 Digital Recorders

4.4.3.2.1 High Data Rate Recorder

The principal function of the High Data Rate Recorder (HDRR) 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 HDRR and the HRM will form an integrated system. Both are controlled by the CDMS subsystem computer in a coordinated .nanner. The experiment interfaces with the HDRR via the HRM only. During recording of formatted data and reproducing all data, the HDRR will be externally synchronized by the HRM clock. When connected to a DACH channel, the HDRR will be synchronized to the experimenter clock received via the HRM.

DCN 048 The HDRR 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 reproduce the recorded data can be interleaved into the real time data stream through a recorder dedicated input channel of the HRM or directly dumped to the KUSP via the HRM, but independent of the formatter.

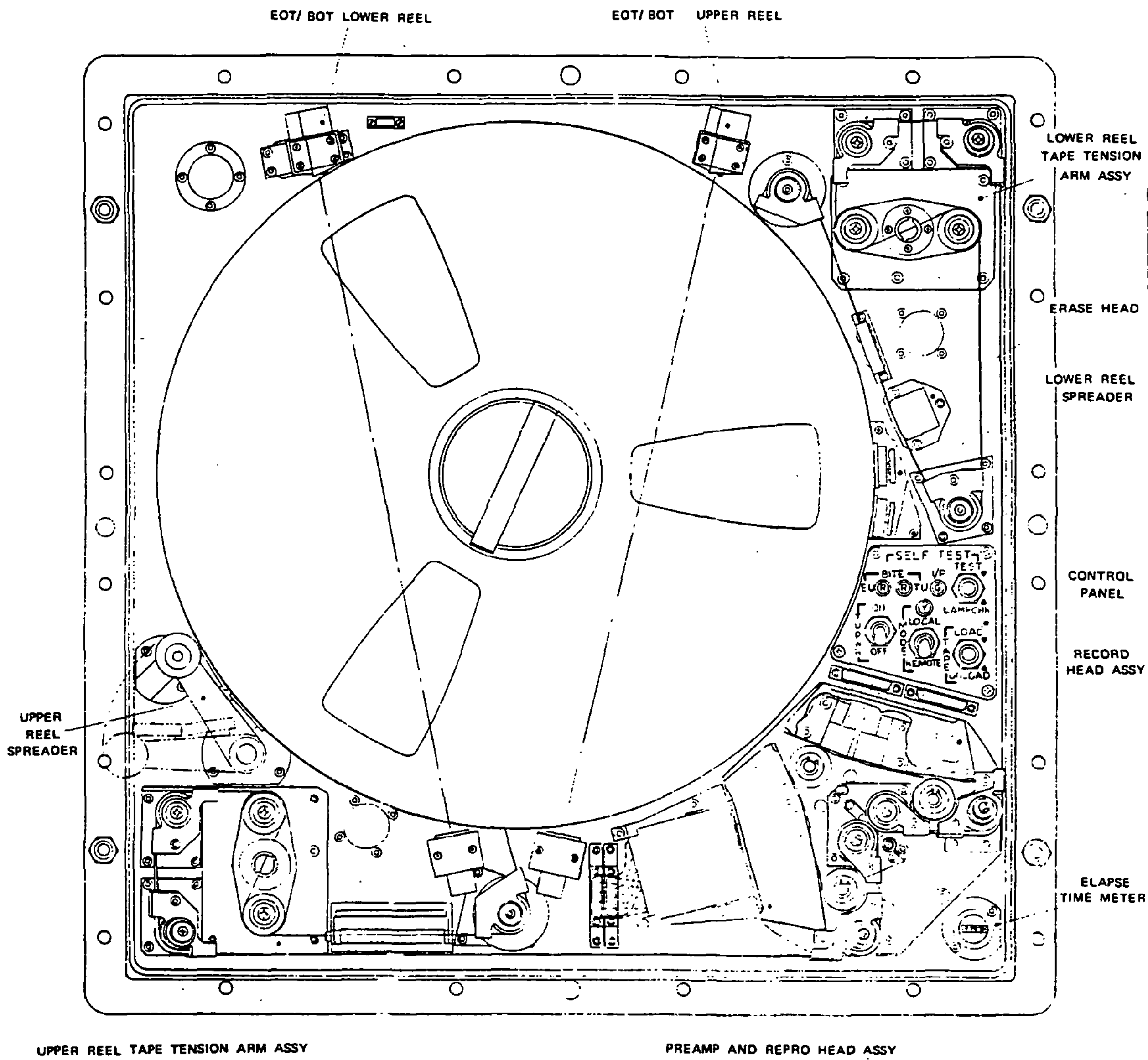
Data recording for on-board storage without transmission to the ground is only possible during periods when non-buffer capacity for transmission gap times is required. In this case the tape change capability of the HDRR may be useful for the experimenter.

DCN 048 The main HDRR characteristics are given in Table 4.4 – 7 and the front view of the HDRR transport unit is depicted in Figure 4.4 – 25a.

Table 4.4 – 7: High Data Rate Recorder Characteristics

DCN 048

Record Technique	longitudinal, 28 tracks
Data Tracks	24
Data Storage	3.8×10^{10} bits
Bit Density/Track	20 kb/inch
Data Rate Record	1,2,4,8,16,32 Mb/s or 1 thru 32 Mb/s via direct access
Data Rate Reproduce	2, 4, 8, 12, 16, 24, 32 Mb/s
Total Record Time	from 20 min at 32 Mb/s to 640 min at 1 Mb/s
Data Type	Serial in, Serial out, NRZ-L+clock, group coding on tape
Bit Error Rate	less than 1 in 10^6 bits with screened tape
Reproduce Direction	reverse to record direction
Start/Stop Time	5 s
Tape Handling	Tape Change Capability with automatic threading
Tape Handling Time	0.5 min. for attachments plus time for wind/rewind
Wind/Rewind Time	7.5 min each max.
Tape Width/Reel Diameter	1" / 14"
Tape Reel with Tape	3.8 kg



DCN 048

Figure 4.4 - 25a : HDRR Transport Unit - Front View

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In module configurations the HDRR is located in the control center rack. In pallet-only configuration no HDRR is provided in the baseline. Operational control of the HDRR will be effected via discrete commands from a subsystem RAU. However, sufficient local controls are provided on the HDRR transport unit to allow tape change and to inhibit normal control which may endanger the operator. In addition to monitoring command status and recorder housekeeping signals, the subsystem RAU also receives a parallel 8 bit word representing tape used. This information will be interpreted by software to represent tape used for display on the DDU.

The power consumption as indicated in Table 4.4 - 8 depends strongly on the actual operating mode.

Table 4.4 - 8: Power Consumption of the HDRR

HDRR Modes	DC Power
Record 32 Mb/s	79 W
1 Mb/s	59 W
Reproduce 32 Mb/s	167 W
2 Mb/s	147 W
Fast Wind/Rewind	79W
Standby	22 W

DCN 048

4.4.3.2.2 Payload Recorder

As a low rate data rate complement to the HDRR, the Orbiter provided Payload Recorder can be used (see Figure 4.4 - 15). This recorder will have a storage capacity of 3.44×10^9 bits and an input rate selectable from 64 to 1024 Kb/s. The method of recording is serial track sequencing of 14 available tracks with turn-around interrupts between 3.5 s and 6.5 s. The record time per track varies from 32 min to 4 min.
The HRM output rate to the payload recorder is 1 Mb/s to 125 Kb/s. The payload recorder reproduce rate to the HRM is 1 Mb/s.

DCN 048

4.4.3.3 High Rate Demultiplexer (HRDM)

The decommutation of the data stream received on ground via the TDRSS link is performed by the High Rate Demultiplexer (HRDM). The HRDM block diagram is given in Figure 4.4 – 26.

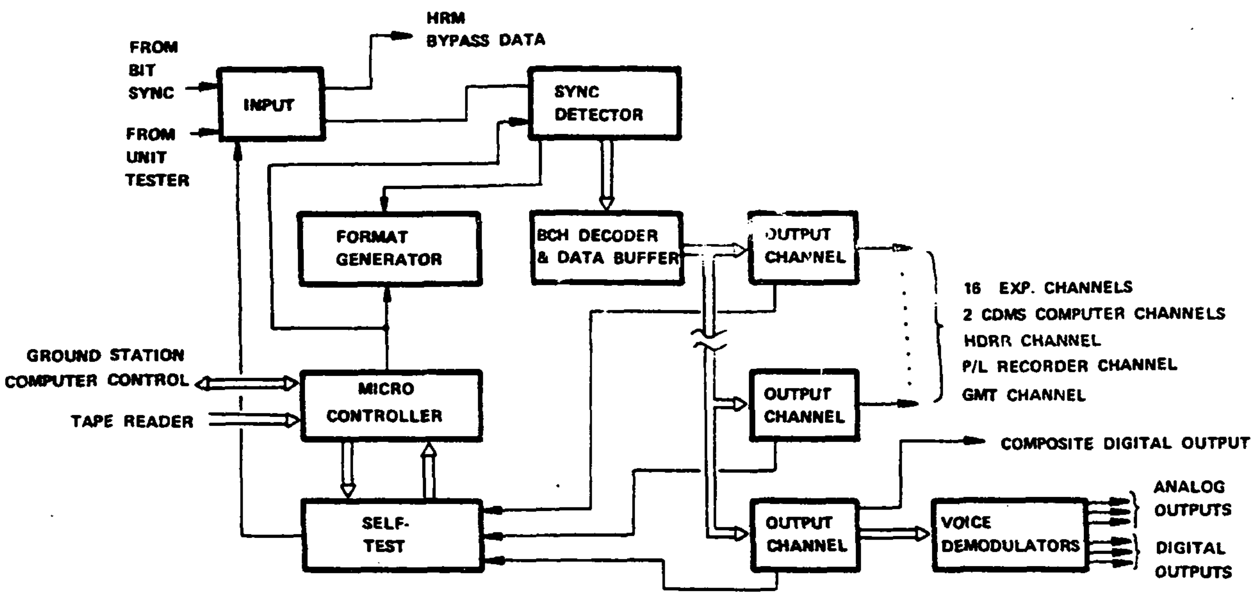

FIGURE 4.4-26 HRDM BLOCK DIAGRAM

The HRDM input circuit receives the serial data input from the bit synchronizer of the ground station. The link is composed of three lines,data, clock, and bit synchronizer lock status. As long as the lock status is not true, the input data are not considered valid. The HRDM is synchronized on the input clock and can operate at any rate up to 50 Mb/s.

For sync detection incoming data are clocked into 32 bit serial register. The 28 bit optimum code in the sync word enters the register followed by the 4 bit frame count. Initially the sync circuitry in a search mode looks for the 28 bit code. When the correct code is recognized with no more than 1 bit in error, the frame count contained in the data stream will be loaded in the frame counter. If sync is recognized in the next sync word slot again and if the received frame count is the increment of the stored frame count the HRDM will be considered in lock. The HRDM synchronization and lock procedure assures a high probability of obtaining the correct synchronization and maintaining it without loss for at least 10^{15} frames by automatic recovery in the event of many probable error contributions before loss of lock.

The format generator stores up to 16 formats in programmable read-only memories (PROMs), plus two formats in random access memories (RAM). Each format consists of 768 5-bit words. Each word represents the channel address of the corresponding word in the format. One frame of the HRDM input data consists of 96 words, so one format repeats every eight frames. The two RAM's can be loaded from a ground computer.

In the BCH decoder and data buffer, the data is buffered line by line and the fill identification word is decoded. Each line is then decommutated, fill words are removed and the detected data is sent to the appropriate output channel buffer.

4.4.3.3.1 HRDM Output Channel

As shown in Figure 4.4-27 the output buffers consist of FIFO memories capable of storing 64 16-bit words. Data are loaded in parallel into the FIFO from the BCH decoder and data buffer as they are available from the input data stream. The data words automatically bubble through the buffer from top to bottom. The data are removed from the FIFO at the appropriate rate to achieve the programmed output bit rate for every channel. Two output modes can be selected :

- The output data stream can be selected to within 1 % of the desired HRM input bit rate and will be regulated in steps of ca. $\pm$ 1.6 % of the selected frequency in order to ensure no data loss occurs through underflowing or overflowing the output fifo.
- bursts with a bit rate predetermined with $\pm$ 0.5 % granularity

DCN 048

In the continuous mode the HRDM provides a smoothing of the output data stream that otherwise might have gaps caused by commutation of other channels. A logic checks whether the contents of the buffer are more than 37, between 37 and 27 or less than 27 words. This information is input to the clock regulator that switches the clock - out frequency in three discrete steps (see Fig. 4.4 - 27).

In the burst mode the clock regulation is disabled. The data are clocked out a fixed predetermined frequencies as they are decommutated. Time delays are caused only by the intermediate line by line buffering and the output buffer bubble-through line which is about 2 μs.

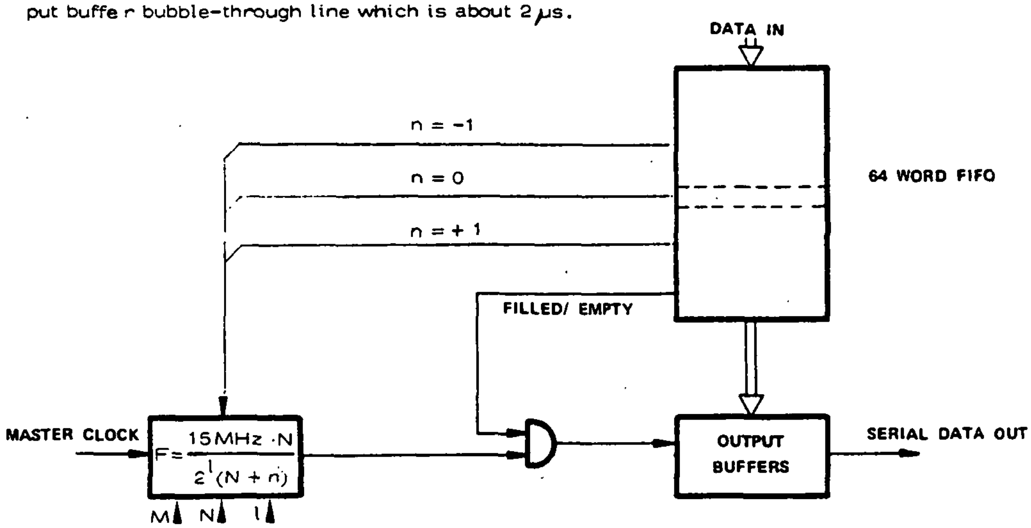

Figure 4.4 - 27 HRDM Output Channel Frequency Control

DCN 048

The selection of the output bit rate and of the mode for each channel is part of the format. For the experiment output channels the output bit rate is determined by the parameters M = 1 - 65, N = 40 - 64, and l = 0 - 16 as shown in Fig. 4.4 - 27. In summary, the HRDM provides the following outputs:

NAME	NUMBER	BIT RATE
Experiment Channels	16	200 bps up to 16 Mbps
HDRR	1	2/4/8/12/16/24/32 Mbps
P/L Recorder	1	1 Mbps
I/O Units	2	200 bps up to 0.5 Mbps

4.4.3.3.2 GMT Output

The GMT is retrieved from the down link status words and is output by a dedicated channel with a fan-out of 20. The BCD coded GMT is output every 4th format starting with the frame sync and lasting about 60 % of the length of 4 Formats. The time correlation GMT/experiment output data is given below. DCN 048

HRM OUTPUT BIT RATE	GMT
48 Mb/s	10 ms
32 Mb/s	10 ms
16 Mb/s	10 ms
8 Mb/s	10 ms
4 Mb/s	20 ms
2 Mb/s	30 ms
1 Mb/s	50 ms
.5 Mb/s	100 ms
.25 Mb/s	200 ms
.125 Mb/s	400 ms

DCN 048

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4.4.4 Analog Data Acquisition

4.4.4.1 Closed Circuit Television

As an extension of the Orbiter CCTV system, Spacelab provides a mission dependent Orbiter common TV Monitor (TVM) in the control center rack and the electrical interface to operate three experiment provided TV cameras (TVC´s) in the module (see Figure 4.4 - 27a). In pallet-only configurations the electrical interfaces are provided to operate up to three TV cameras.

The TV cameras and monitor must be synchronized with the Orbiter CCTV signals. The Orbiter provided sync signal contains also serial commands for the remote control of Orbiter common TV cameras with associated lens assembly pan/tilt unit and viewfinder monitor. The composite video signals from the TV cameras to the Orbiter Video Control Unit (VCU) and from the VCU to the Spacelab TV monitor have to be compatible with US commercial black and white 525 lines video format defined in EIA standard RS 170 and RS 330.

CN 48

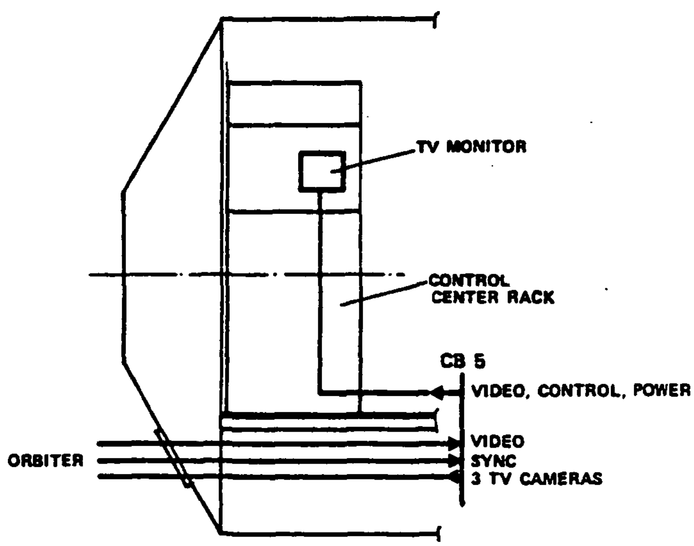


Figure 4.4 - 27a: Closed Circuit Television

The electrical provisions for experiment provided TV cameras are:

- 3 inputs for experiment generated video signals
- 1 output for synchronization of TV cameras, monitor and video recorder.
- 1 output for monitoring or recording of Orbiter/Spacelab generated composite video signals.

All signals between Spacelab and the Orbiter VCU are grouped in one connector at the experiment interface plane CB 5 in module configurations and at CB 42 (Igloo signal feedthrough) in the pallet-only configuration. DC 046

The mission dependent TV monitor can be accommodated in the control center rack. From there cables are routed to CB 5 for

Video A and B and sync/cmd	3 TSP's, 75Ω
Video identification	1 T4C
28 V DC and return	1 TP

The further routing has to be done by the experimenter/payload integrator according to the particular mission needs.

More details are given in Section 4, Avionics Interface Definition, Appendix A.

4.4.4.2 4.5 MHz Analog Channel

Spacelab provides a DC - 4.5 MHz analog channel for experiments. This channel can be used also for special non-EIA standard TV signals.

The voltage range of this analog input will be 0 - 1 V ± 10 %

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 the 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 75Ω coaxial cable routed directly to the Orbiter Ku-Band Signal Processor. The connector is located at CB 5 in module configurations or on CB 57 at the first pallet front frame in pallet-only configurations. For more details see Section 4 of Appendix A, Avionics Interface Definition.

4.4.5 Data Transmission

4.4.5.1 Network System

Figure 4.4 - 29 shows the possible transmission links to the ground. Two downlink 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.

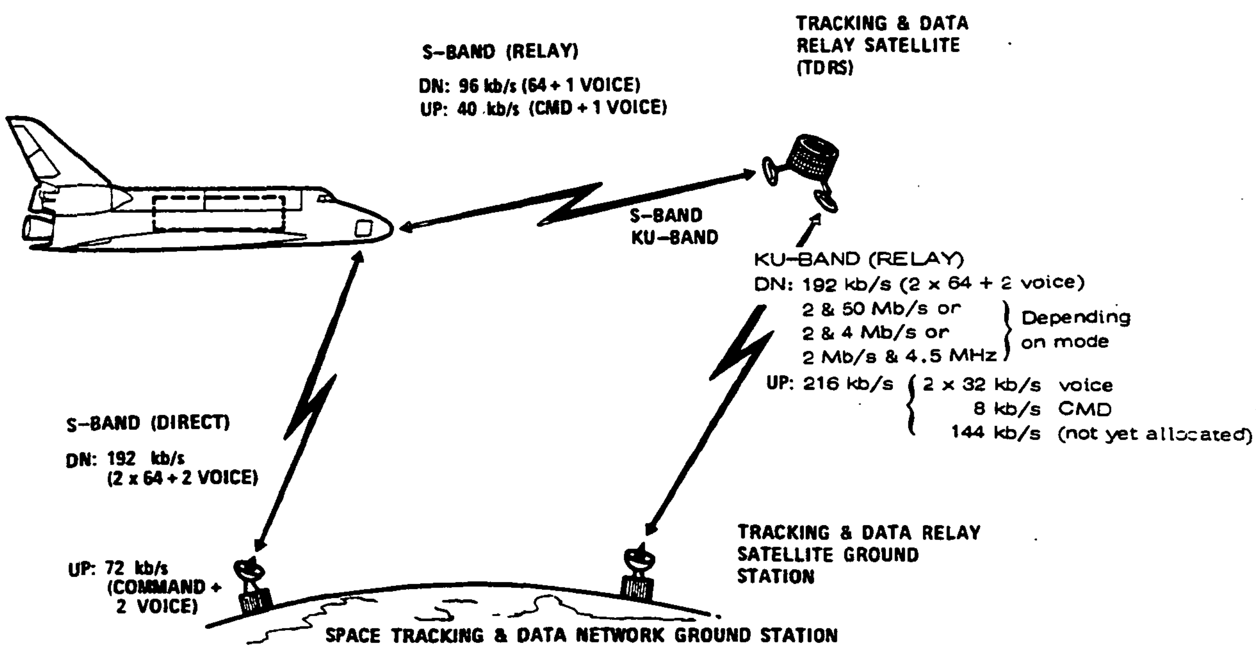


Figure 4.4 -29: Orbital Communication Links

The present NASA STDN ground stations available for direct down links are shown in Figure 4.4-18. 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 7 stations underlined in Figure 4.4-30, with Bermuda and Merritt primarily used to support the Shuttle launch phase. 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 subsystem and experiment computer. DCN 048

The TDRSS consists of two geo-stationary satellites at 41°W and 171°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 4.4-30 shows Orbiter positions where no transmission via the TDRSS is possible due to the earth geometry. The hatched circles present limit orbits without beam disturbance by atmosphere limb for two different TDRSS antennas.

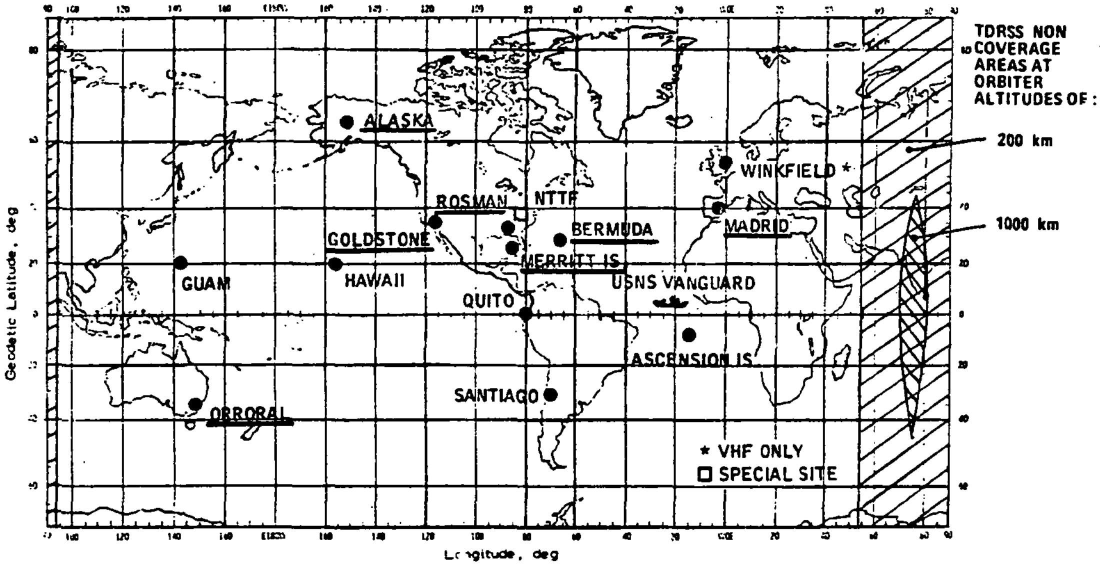
For Orbiter altitudes of 200 km and 1000 km the hatched and cross hatched areas respectively in the upper part of Figure 4.4 - 30 indicate regions on the earth surface which cannot be observed with direct TDRSS link.

Another major limitation for the TDRSS link is the Orbiter attitude dependent beam blockage by Orbiter structure (see Figure 4.4 - 31) and Spacelab payload structure.

The nominal TDRSS coverage using the Orbiter antenna is approximately 80 % over 24 hours. However, for some orbits the coverage per orbit may be only 50 %. DCN 048

The exact coverage will depend on the actual mission profile and computer programs are being developed to perform the necessary analysis of the various factors affecting coverage, such as

- TDRSS gaps over the Indian Ocean (see Figure 4.4 - 30)
- Ku-Band antenna masking by Orbiter, Spacelab and payload (beam blockage) (for Orbiter blockage, see Figure 4.4 - 31)
- Flux density limitations (transmission constraint)
- TDRSS antenna adjustment through S-Band link
- RF transmission factors - bit error probability and data rate vs power gain/loss factors (quality)
- Solar interference - TDRSS data reception interruption



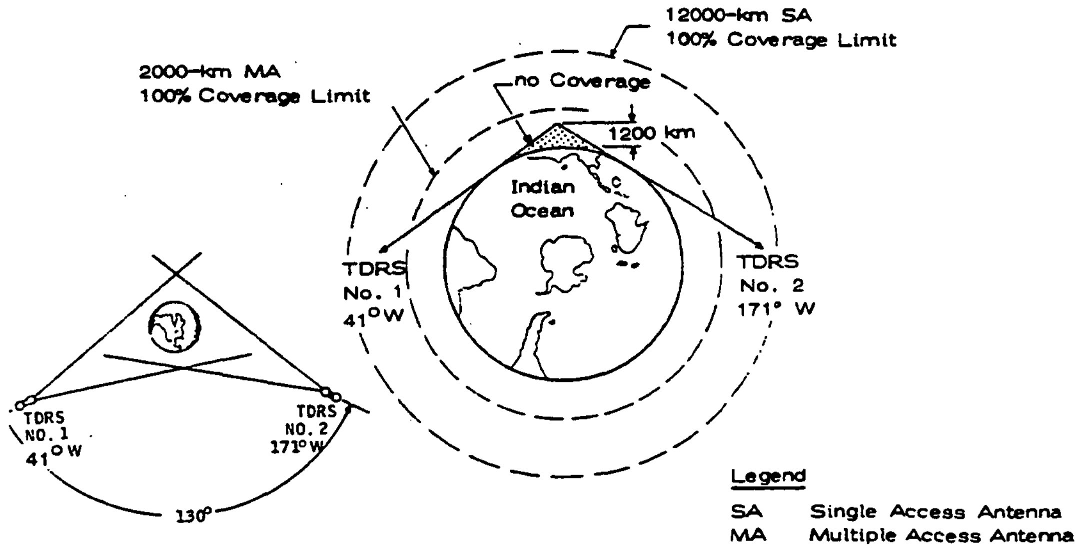


Figure 4.4 - 30: STDN Stations and TDRSS Coverage

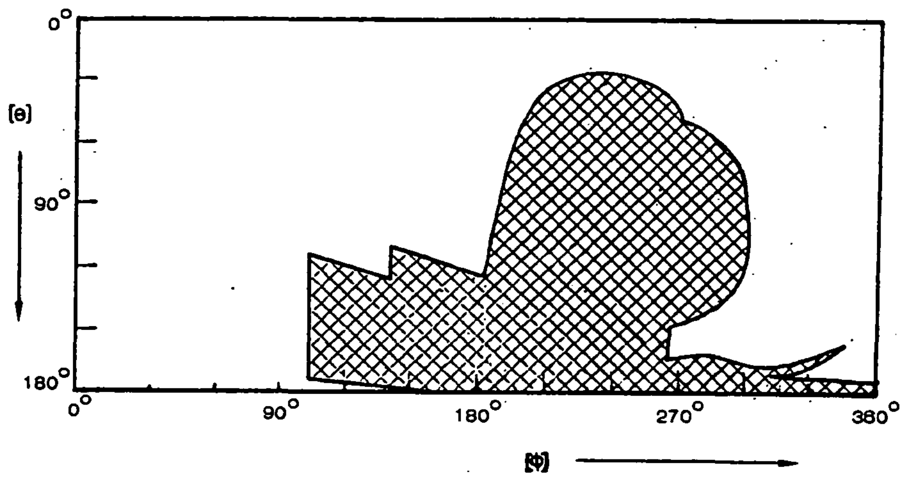
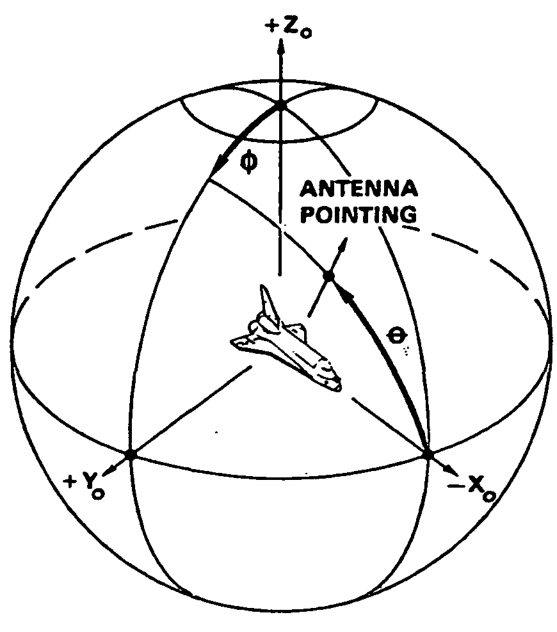


Figure 4.4 - 31: TDRSS Antenna Blockage Due to Orbiter Structure



Ø, θ Coordinates

4.4.5.2 Down Link

The transmission of data generated by Spacelab or Spacelab payload is performed by the Orbiter Avionics (see Figure 4.4- 32).

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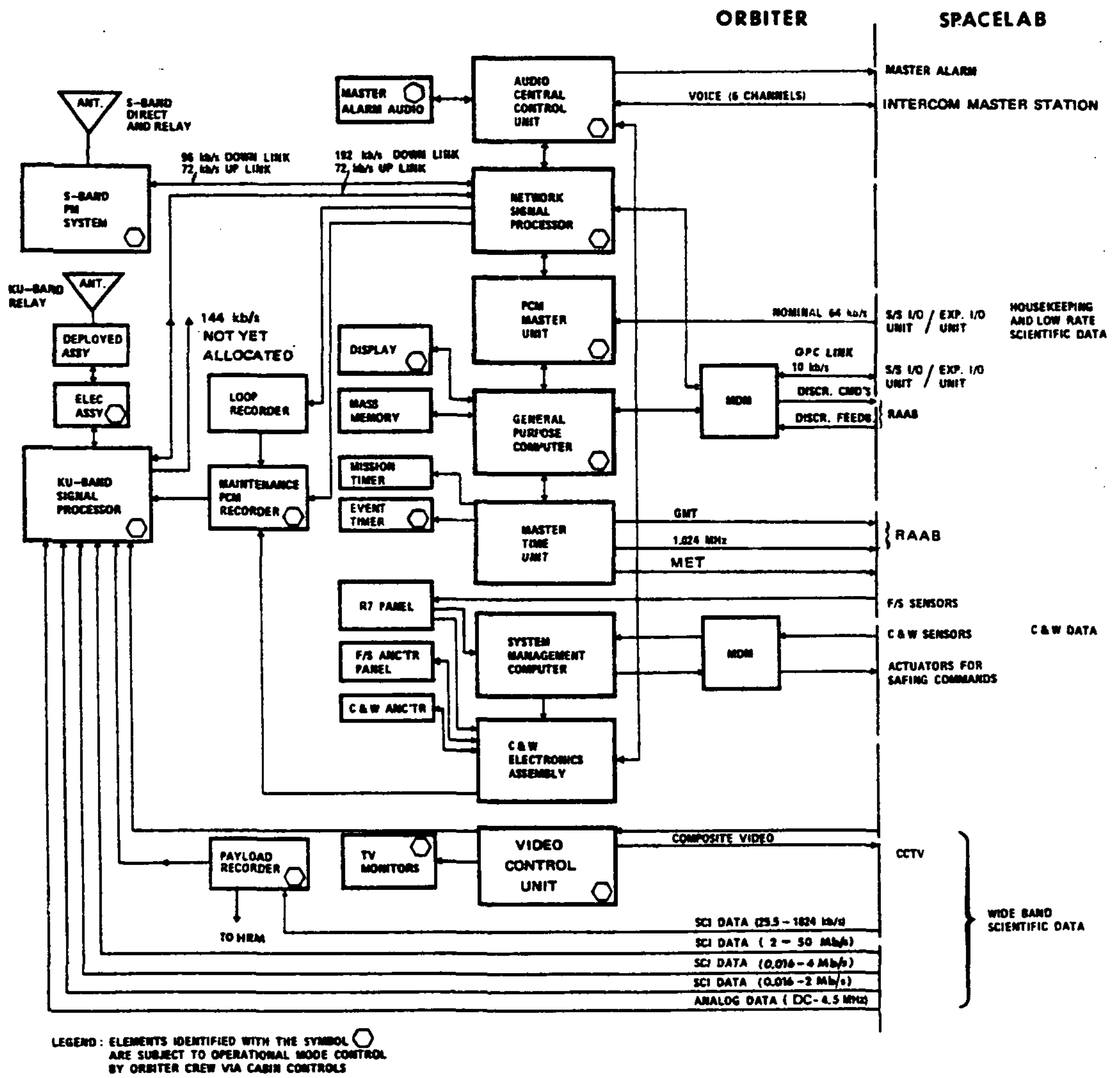

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Figure 4.4 - 32: Orbiter Avionics Functional Diagram for Payloads

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.

It should be noticed that low rate scientific data using this link are subject to stringent formatting:

- o The PCM Master Unit can request data from the CDMS computers up to 2000 times per second.
- o Upon one request, up to 10 data words can be transferred.
- o The requests can address data in a 2 K subsection of each CDMS computer core memory. The specific area in core memory that can be addressed by the PCM Master Unit is assigned at system initialization. DCN 048

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 or via TDRSS KU-Band 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 Mission Control Center 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.

The TDRSS S-Band link provides only a 96 kb/s downlink capability (64 kb/s data + 32 kb/s voice) which has to be shared between Orbiter and Spacelab on a case by case base.

- 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 4.5 MHz channel. These Wide Band Scientific Data are transmitted to ground only via the Ku-Band of the TDRSS. For the digital data, TDRSS non-coverage periods are bridged by the Spacelab HDRR and the Orbiter Payload Recorder (see Section 4.4.3). Means to bridge the transmission of CCTV and analog signals are not provided. DCN 048

The Orbiter controlled mode selection and channel allocation of the Ku-Band down link is performed by the Ku-Band signal processor.

The functional flow chart in Figure 4.4 – 33 indicates the switching capabilities to combine the various inputs to the KU-Band signal processor.

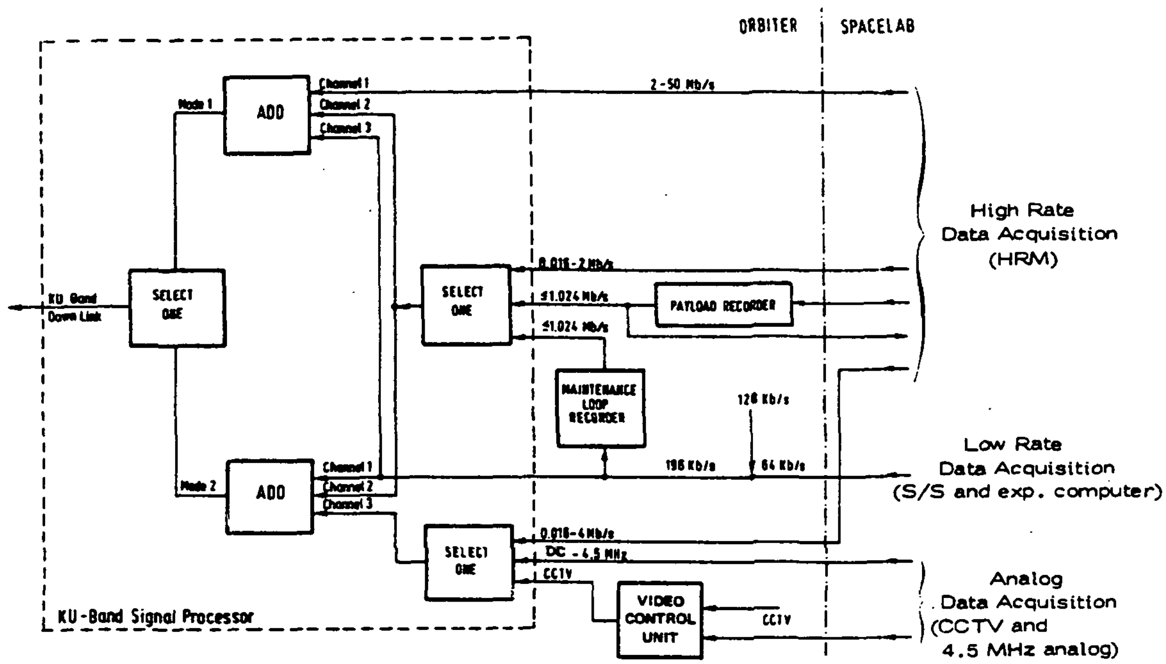

Figure 4.4 – 33: Functional KU-Band Data Processing

The channels available in the two KU-Band modes are summarized in Table 4.4 – 9.

Mode 1 is a phase modulated transmission line providing one 192 kb/s channel for telemetry data with 64 kb/s out of it dedicated to Spacelab subsystem and experiment computer output data, one 0.016 – 2 Mb/s channel interfacing with the HRM or the Payload Recorder output and one 2 – 50 Mb/s channel interfacing with the HRM. All these channels can be operated in parallel.

Mode 2 is a frequency modulated transmission line providing one 192 kb/s channel (same as in mode 1) one 0.016 – 2 Mb/s channel (same as in mode 1) and one channel accepting either digital or analog signals. The digital data (0.016 – 4 Mb/s) are delivered from the HRM output. The analog signals are delivered from the CCTV or from the DC – 4.5 MHz analog channel directly.

The KU-band link requires a minimum density of bit transitions (see para 4.4.3.1.5). To fulfill these bit density requirements, it may be necessary to have some additional overhead in the data stream transmitted via the KU-band link.

Table 4.4-9: Ku-Band Mode Description (Orbiter to TDRSS)

MODE	CHANNEL		
	1	2	3
1 (PM)	Digital: 192 kb/s (64 kb/s and voice from Spacelab)	Digital : 0.016 - 2 Mb/s	Digital : 2 - 50 Mb/s
2 (FM)	Digital: 192 kb/s (64 kb/s and voice from Spacelab)	Digital : 0.016 - 2 Mb/s	Digital : 0.016 - 4 Mb/s or Analog : CCTV or 4.5 MHz Channel

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The further transmission and processing of Wide Band Scientific Data after having been received at the TDRSS ground station at White Sands, New Mexico, is still under discussion.

4.4.5.3 Up Link

The Orbiter Avionics System provides an up link channel of 216 kb/s for command and voice via the STDN S-Band and via TDRSS KU-Band. There is no TV uplink to the Orbiter or Spacelab. The 216 kb/s uplink line is split into two voice channels with 32 kb/s data rate each and one command channel with 8 kb/s. The remaining 144 kb/s are not yet allocated. The command channel transmits 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.

The TDRSS S-Band uplink provides only a degraded uplink capability of 40 kb/s (command + 1 voice).

All Spacelab payload commands generated on ground have to be routed to the Payload Operation Control Center (Houston) prior to actual need. This means that the user has to deliver his experiment commands and uplink data at the Payload Operation Control Center where the complete telemetry frames will be generated and transmitted to the Orbiter via the Mission Control Center .

4.4.5.4 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. In addition the air to air link of the Orbiter intercom system provides the capability of a duplex voice channel to detached payloads.

4.4.6 Data Processing

This section applies only to experiment data acquired by RAU's.

4.4.6.1 Computer

The CDMS has three identical CIMSA 125 MS general purpose computers.

The three computers are used as Subsystem Computer (SSC), Experiment Computer (EXC) and Back-up Computer (BUC). S/S and Subsystem Experiment Computers are connected to the CDMS equipment each via its own I/O unit, data Bus and RAU's. There is no direct link between each computer.

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The third computer is available as a back-up either for the subsystem or the Experiment Computer and can be switched over by MDM command initiated either on board or on ground.

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 associated peripherals.

Normally the back-up computer is loaded with subsystem software (operating system and application software) since a subsystem computer failure is more critical with respect to the overall performance of Spacelab. However, in case of Experiment Computer failure, the experiment software must be loaded from the Mass Memory Unit (MMU) by MDM command (see Section 4.4.6.2).

In the Module or Module/Pallet configurations, the computers are located in the Work Bench Rack. The location in the Pallet-Only configurations is the Igloo.

The computer facilities allow general purpose processing by user provided software written in assembler or a high order language (see Section 4.5) for such purposes as: DCN 048

- Checkout of Experiments
- Sequencing of Experiment Operations
- Monitoring and Control of Experiments
- Generation of displays on DDU DCN 048
- Processing of data acquired by Experiment RAU's

Examples of Data Processing are:

- Filtering
- Data Reduction
- 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 via keyboard entries or directly via uplink commands, a running sequence of operation steps may be stopped or changed or a new program may be initialized to be executed in the Experiment Computer.

4.4.6.1.1 Computer Architecture

The 125 MS is part of the CIMSA militarized 15 M/computer range designed for real-time data processing under severe environmental conditions.

Functionally, it corresponds to the ground version MITRA 125 S which is also equipped with a Fast Arithmetic Operator (FAO).

The 125 MS is fully compatible with the MITRA 125 S computer (same architecture) and has "commonality" (same real-time behavior of individual instructions and programs in both computers). The architecture of the computer is shown in Figure 4.4 - 33a.

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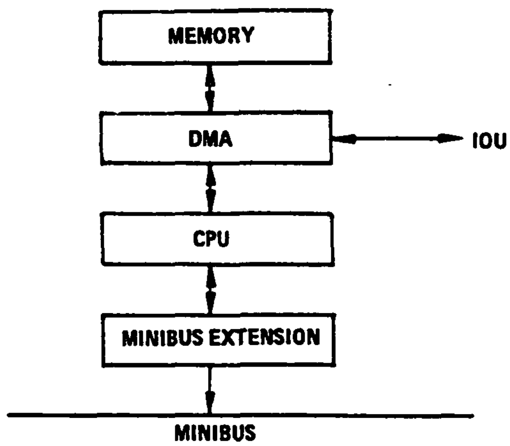

Figure 4.4 - 33a: 125 MS Architecture

The memory organized into 18-bit words (16 bit center, 1 bit parity, 1 bit protection) has a bus on which a Central Processing Unit (CPU), and a Direct Memory Access channel (DMA), is connected. The Central Processing Unit is built from a structure identical to the MITRA 125 S and includes mainly a microprogrammed control logic that permits executing a very complete and extendible instruction set; a Fast Arithmetic Operator (FAO) is integrated in the Central Processing Unit and enables executing 200,000 floating-point operations/sec.

The CPU communicates with the outside via a peripheral bus (the Minibus) on which can be connected, depending on the case, either the 15M/ type militarized peripheral range or the MITRA 125 peripheral range, both for ground checkout only. An interrupt system and a suspension system are associated with this bus, the latter system acting like the former but at the micro-machine level.

Peripheral coupling functions can thus be optimally distributed between a physical "coupler", a coupling microprogram and a programmed "handler", providing operational flexibility. For very fast peripherals like the IOU connection to the Direct Memory Access (DMA) channel results in a 700 K word/sec. maximum exchange rate.

The 125 MS is housed in a single .19 x .28 x .50 m enclosure and includes:

- Central Processing Unit (CPU)
- 4 K words of control memory
- Fast Arithmetic Operator (FAO)
- 64 K words memory
- Direct Memory Access (DMA) channel for IOU
- Real Time Clock
- 8 Hardware Interrupts (4 in use by IOU) (can also be used by software)
- 16 Software Interrupts
- DC Power Supply
- Driver for Minibus control
- Remote control and monitoring interface

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The configuration used for the computers in Spacelab is shown in Figure 4.4 - 33b

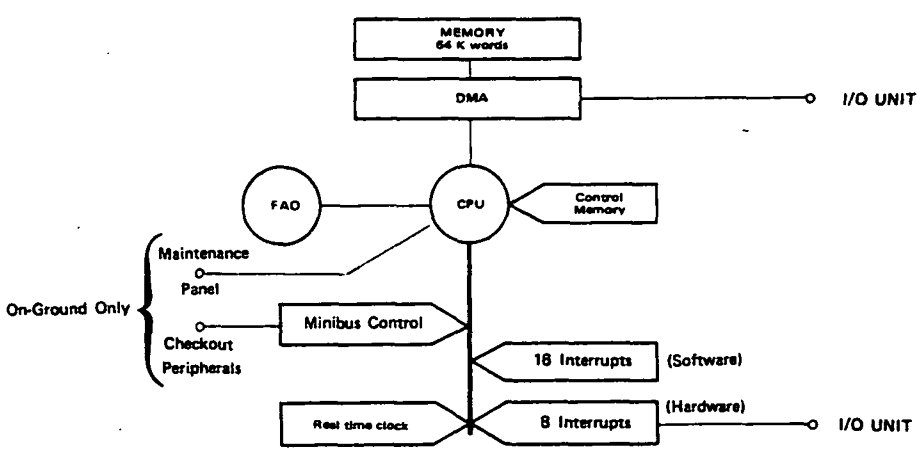

Figure 4.4 - 33b: Spacelab Computer Configuration

The software for the Spacelab CDMS computers is described in Section 4.5.

The 125 MS computer is fully compatible with the MITRA 125 S computers and can, therefore, use the same basic software, i.e. the Spacelab provided SCOS or ECOS together with the user provided application software.

4.4.6.1.2 Computer Memory

The 125 MS memory is a ferrite core (18 mil) memory (2.5 D) extremely compact and made up of 18-bit words (16 data bits, 1 parity bit, and 1 protection bit).

The 125 MS main memory capacity is 64 K words: however, its addressing system can address up to 512 K words. The memory permits a Central Processing Unit and a Direct Memory Access channel to simultaneously access data and programs.

Memory protection is provided by a specialized bit associated with each memory word and acting as a "lock".

Program status associated with each task includes a Memory Protection (PM) indicator with the same function as a "key"; if this indicator is at 1, the task can access only the unprotected memory areas; any memory protection violation triggers a trap.

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Extra -protection is provided in the CPU by a system of base and length associated with the context reciprocally protecting the various elements of a system (segments) and in which each task associated with a base has a preset length.

A special bit is associated with each memory word so that the memory word can be tested for odd parity.

Any memory parity error either during program running or input/output data exchange triggers a trap.

4.4.6.1.3 Central Processing Unit

Central Processing Unit functions are distributed between a certain number of elements, in particular, an arithmetic operator associated with a working register, a set of general registers acting as a private memory, and a microprogrammed control logic for both executing 125 MS coded instructions and coupling peripherals.

The Central Processing Unit also includes a peripheral bus - called the "Minibus" - to which interrupt and suspension systems are associated as well as a certain number of indicators that facilitate computer utilization and a trap system that insures constant system operating surveillance.

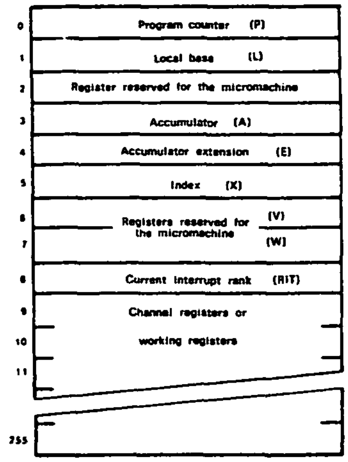

Figure 4.4 - 33c: Central Processing Unit Registers

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General Registers

The Central Processing Unit has 256 general registers divided up into 32 eight-register blocks. Each register contains 16 bits and is accessible in approximately 60 nanoseconds.

Generally, the first registers are assigned to the program context. The other registers are used as channel registers for the integrated peripheral couplers (microprogrammed data exchanges).

The 125 MS has, in addition, 32 base registers and 32 length registers used to relocate addresses (20 bit addressing) and for memory protection, and to make complex system execution easier (see Table 4.4-11).

Table 4.4 - 11: Additional 125 MS Registers

Registers	Function
S	Supervisor base
G,GL	General base and associated length
Q,QL	Shared program base and associated length
C	Program context base
Z,ZL	Shared data base and associated length
H1, TH1	Bases and length reserved for the micro-machine
S1, P1 to S25, P25	Bases and protection associated with suspensions

Program Indicators

The Processing Unit includes program accessible indicators, as shown in Table 4.4-11a.

Table 4.4 - 11a: Processing Unit Accessible Indicators

Indicator	Function
I1	Carry or operation test
I2	Overflow or operation test
MA	Interrupt masking
PM	Memory protection key
SV	Supervisor/user mode
SP	Shared/unshared program mode
PV	Privileged/unprivileged mode

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Microprogramming Memories

The microprogrammed control logic is associated with three types of microprogramming memories:

- The control memory consisting of 4,096 words, each word comprising 16 and/or 20 bits; this memory contains the microprograms which execute either 125 MS coded instructions or coupling functions.
- Executive memory has 2 sets of 88 executive words each; this memory is addressed by the various control memory microinstructions and allows executing microcommands.
- Memory Associated with Suspensions (MAS) consisting of 32 twenty-four bit words; this memory defines suspension rank and level as well as the entry point of corresponding microprograms in the control memory and the block of general registers used.

Interrupt System

The Central Processing Unit has a system of interrupts organized into 32 independent and hierarchical levels (priorities increase with the level).

A context is associated with each level; the context address is defined by a context pointer located in a table stored in main memory.

Context is composed of the following values: indicators, program registers, base and length registers, parameters of the stack associated with the task.

When a priority interrupt is triggered, it causes an automatic exchange of the interrupting and interrupted task context in less than 40 μs. Instruction DIT deactivates the current interrupt and changes the context according to a similar mechanism. Instruction XCTX permits, in addition, exchanging the context without deactivating the current interrupt.

Suspension System

Suspensions have a function similar to interrupts, but at the micromachine and microinstruction level. The 125 MS Central Processing Unit has 25 external suspensions and 7 internal Suspensions divided into 4 "levels" - level 0 being reserved for instructions execution - and each level is hierarchically divided into "ranks".

Micromachine context stacking devices insure very fast suspension acceptance (approximately 300 ns).

Trap System

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A trap is produced whenever an incident is detected during either a processing or input/output micro-program . The effect is to create a trap status word and save in memory data related to the current program before calling a supervisor module which would attempt to correct the incident (e.g. simulation of illegal instructions). (see Table 4.4-11b)

Table 4.4 - 11b: Trap Status Words

Trap	Meaning
PG	Program or input/output trap
AI	Non-existent memory address
PA	Memory parity error
PM	Memory protection violation
DT	Size overflow
VM	Mode violation
II	Illegal instruction
ES	Incorrect input/output or non-existent coupler
OP	FAO trap

Instructions

The 125 MS has an instruction set operating on very varied formatted data:
- Binary format: 1 , 8, 16 or 32 bits,
- Fixed point: 16 or 32 bits,

- Floating point: 32 bits,
- Character strings with 32 Kbyte length maximum.

Generally instructions have a fixed, 16-bit word format:

13 addressing modes permit instructions to reach either data related to the various bases, other instructions, or parameters contained in the instruction itself (immediate addressing):

- Immediate, indexed immediate addressing,
- Addressing relative to the P (program) counter: forward or backward,
- Addressing relative to the Local base L: direct, indirect, indexed indirect,
- DCN 050 Addressing relative to the General base G: direct, indirect, indexed indirect,
- Indirect addressing via a generalized pointer; the indirect address may be relative to the General base G or the Common data base Z.

Special instructions make interprogram communication easier: calling supervisor modules, common modules, processing stacks and queues, etc...

The 125 MS has a 135-instruction set (see Table 4.4-11c)

Fast Arithmetic Operator (FAO)

The fast arithmetic operator built in the Central Processing Unit executes fixed point operations in single or double precision and floating point operations in single precision. A 32-bit floating point multiplication is carried out in less than 6 μs.

Table 4.4 – 11c: Instructions

1 - memory reference instructions

Symbolic code	Function
ADD	Addition
ADM	Addition memory word
AND	AND operation
BAN	Branch if A negative
BAZ	Branch if A equal to zero
BCF	Branch if carry false
BCT	Branch if carry true
BOF	Branch if overflow false
BOT	Branch if overflow true
BRU	Branch unconditional
BRX	Branch with index
CMP	Compare
CPS	Compare byte with a string
DIV	Algebraic division
DLD	Load registers E and A
DST	Store registers E and A
EOR	Exclusive OR
IOR	Inclusive OR
LBL	Load byte left of A
LBR	Load byte right of A
LBX	Load byte right of X
LDA	Load register A
LDE	Load register E
LDR	Load register
LDX	Load register X
LEA	Load effective address
MUL	Algebraic multiplication
MVS	Move byte string
SBL	Store byte left of A
SBR	Store byte right of A
SPA	Store program address
STA	Store register A
STE	Store register E
STR	Store register
STS	Store selective of register A
STX	Store register X
SVB	Subtraction
TRS	Translate byte string

2 - register instructions

Symbolic code	Function
AAE	A and E in A
AAX	Add A in X
ACE	Add carry with E
AEA	Add E in A
AEE	A exclusive OR with E
AIE	A inclusive OR with E
AXA	Add X in A
CAA	Compare A left with A right
CAE	Compare A with E
CBA	Change bit K of A
CBE	Change bit K of E
CCA	Copy complement of A
CCE	Copy complement of E
CHX	Copy half X
CMZ	Count most significant zero of A
CNA	Copy negative A
CNE	Copy negative E
CNX	Copy negative X
DCE	Decrement E
DCL	Decrement L
DCX	Decrement X
ICE	Increment E
ICL	Increment L
ICX	Increment X
LDB	Load A with base
LDC	Load A with context
LDI	Load A with indicators
LNE	Load negative E
RBA	Reset bit K of A
RBE	Reset bit K of E
SAX	Subtract A in X
SBA	Set bit K of A
SBE	Set bit K of E
SRP	Save and reset parity
STB	Store A in base
STC	Store A in context
STI	Store A in indicators
TBA	Test bit K of A
TBE	Test bit K of E
TSX	Test X
XAA	Exchange A left with A right
XAE	Exchange A with E
XAX	Exchange A with X
XEX	Exchange E with X

3 - shift instructions

Symbolic code	Function
NLZ	Normalization
PTY	Parity
SAD	Shift arithmetic double
SAS	Shift arithmetic single
SLCD	Shift left circular double
SLCS	Shift left circular single
SLLD	Shift left logical double
SLLS	Shift left logical single
SRCD	Shift right circular double
SRCS	Shift right circular single
SRLD	Shift right logical double
SRLS	Shift right logical single

4 - system instructions

Symbolic code	Function
BRK	Break
CLM	Clear mask
CLP	Clear protection
CLO	Call common
CLS	Call section
CSV	Call supervisor
DELQ	Delete element in queue
DIT	Deactivate IT
EXEC	Execute
HLT	Halt
INQ	In queue
INQP	In queue with priority
LDG	Load G and TG with segment descriptor
LDZ	Load Z and TZ with segment descriptor
NBP	Normalize byte pointer
OUTQ	Out of queue
PULL	
PUSH	
RD	Read direct
RSV	Return supervisor
RTD	Return from trap
RTO	Return from common sub-programs
RTS	Return section
STM	Set mask
STP	Store memory protection
TES	Test and set
TESQ	Test first element of queue
WD	Write direct
XCTX	Exchange context

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5 - additional FAO instructions

Symbolic code	Function
DAD	Fixed-point double length addition
DDV	Fixed-point double length division
DMU	Fixed-point double length multiplication
DSU	Fixed-point double length subtraction
FAD	Floating-point single length addition
FDV	Floating-point single length division
FMU	Floating-point single length multiplication
FSU	Floating-point single length subtraction
NF	Normalization

6 - special DMA instructions

Symbolic Code	Function
RDIO	Read register IO
WDIO	Write register IO
STIO	Start input/output on IO

4.4.6.1.4 Input / Output

The CPU communicates with the outside via the peripheral bus or the Direct Memory Access coupler (DMA).

Direct Memory Access (DMA)

Peripherals are coupled by an Input/Output unit connected to the computer via the DMA. This coupler provides a bidirectional 18 bit link on which addresses and data are routed in time sharing mode. Maximum throughputs of 700 K words/sec. approximately are obtained in Read Direct and in Write Direct.

This link is also used in programmed mode (instruction RDIO, WDIO, STIO) to initialize transfers in the I/O Unit.

Peripheral Bus (for ground use only)

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The peripheral bus is physically a PC board support in which peripheral coupler and/or interface PC boards are plugged in at non-specialized locations. An extension chassis allows increasing the system facilities if necessary, the remote bus setup being controlled by the transmission and reception PC Boards.

A board called "system" permits specifying the various options offered by the couplers; this board is also plugged into the board support but at specialized locations.

Depending on the input/output rate and reaction delay, exchanges can be carried out in three ways with identical peripheral connection to the Minibus:

- Single programmed I/O: word by word exchanges controlled by instructions (RD or WD) and, therefore, synchronous with the program being processed.
- Programmed I/O on interrupts: exchanges by blocks initialized by program and executed by instructions (RD or WD), but triggered by interrupts, and, therefore, asynchronous with the program being processed.
- Microprogrammed I/O on suspensions: exchanges by blocks executed by microprograms and triggered by suspensions; therefore, asynchronous with the program being processed.

Usually, a peripheral is coupled by combining these three types of exchanges: transfer initialization by a program (Handler 1), exchange execution by a microprogram triggered by a suspension transmitted by the coupler, transfer termination by a program (Handler 2) triggered by an interrupt transmitted via the coupling microprogram.

Maximum data rate obtainable range from 50 Kwords/sec in programmed mode, and 200 Kwords/sec in microprogrammed mode, and 800 Kwords/sec in DMA. The coupler and interface set includes, in particular

- Fast channels AMC, NTDS,...
- Digital interfaces: inputs, outputs, external interrupts or suspensions,
- Synchronous or asynchronous telecommunications lines,
- Couplers for militarized peripheral connection: typewriter, tape punch and reader, magnetic disk, minitape, fast analog chain,...
- Couplers for data processing center type peripherals (MITRA 125 range): punched cards, printers...

4.4.6.1.5 System Protection

Safety is insured at different levels: hardware, microsoftware, and software; the various safety systems protect information in memory (instructions and data) as well as processing in the Central Processing Unit and inputs/outputs (see Table 4.4 - 11d).

Table 4.4 - 11d: System Protection Matrix

Levels	Information	Processing	Inputs/ Outputs
Hardware	—Memory parity —Memory protection bit	—Operation indicators —Traps	—Checks executed by the couplers —Watchdog
Microsoftware	Memory microprogrammed test Proper Operation Test (TBF)	—Proper operation test (TBF)	Checks performed by coupling microprograms.

On the other hand, in addition to the control functions a remote control and monitoring interface allows
- Monitoring via go/no-go of TBF power line voltage, temperatures, internal voltages and
- Measuring CPU and memory temperatures, power line current and secondary voltages.

The Proper Operation Test (TBF) is a device to detect clear breakdowns occurring in a 125 MS computer during operation. This device along with the global safety system is designed to insure high reliability thanks to an extremely high failure detection probability.

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The TBF principle is to run sequences of microprogrammed tests each one implementing a certain number of logic circuits whose failures can be thus detected. The standard test has 15 sequences-each sequence lasting less than 20 μs for a total of approximately 512 microinstructions.

Total test duration is variable and depends, in particular, on the number of passes of each sequence and the sequences selected (Total duration 5 ms approximately).

Should an error occur during the running of the Proper Operation Test, the computer would loop onto a microinstruction or would exit the microprogram; an alarm would be triggered by a monostable which could not be reinitiated in due time (time = 250 ms)

The TBF can be used in 4 different ways:

- Periodic triggering via a real-time clock: whenever a suspension is triggered by the clock, the 125 MS executes a TBF sequence and then resumes the suspended program. This is the normal operation of the TBF in the "integrated periodic self-test".
- Triggering by program (using instruction WD) of all sequences one after another (maximum duration 3 ms). This triggering mode can be used during the running of a program requiring high operating safety (the TBF is then entirely run before critical operations are executed; it can also be used when the Central Processing Unit is waiting for a resource or an event (in this case, the waiting time is used to test the computer).
- Power restore triggering as a preventive action before any 125 MS system processing begins.
- Manual triggering using the technical console during maintenance operations to facilitate failure localization.

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4.4.6.1.6 General Characteristics

Physical Characteristics:

- Enclosure
  - o width : 295 mm
  - o height : 193.5 mm
  - o depth : 498 mm
  - o volume : less than 27.5 litres
  - o weight : 30.5 kg
- Power supply
  - o Voltage : 24 - 32 V DC with overload protection
  - o Maximum continuous power consumption : 397 W
  - o Peak power consumption: 427 W

The general physical layout is shown in Figure 4.4 - 33c.

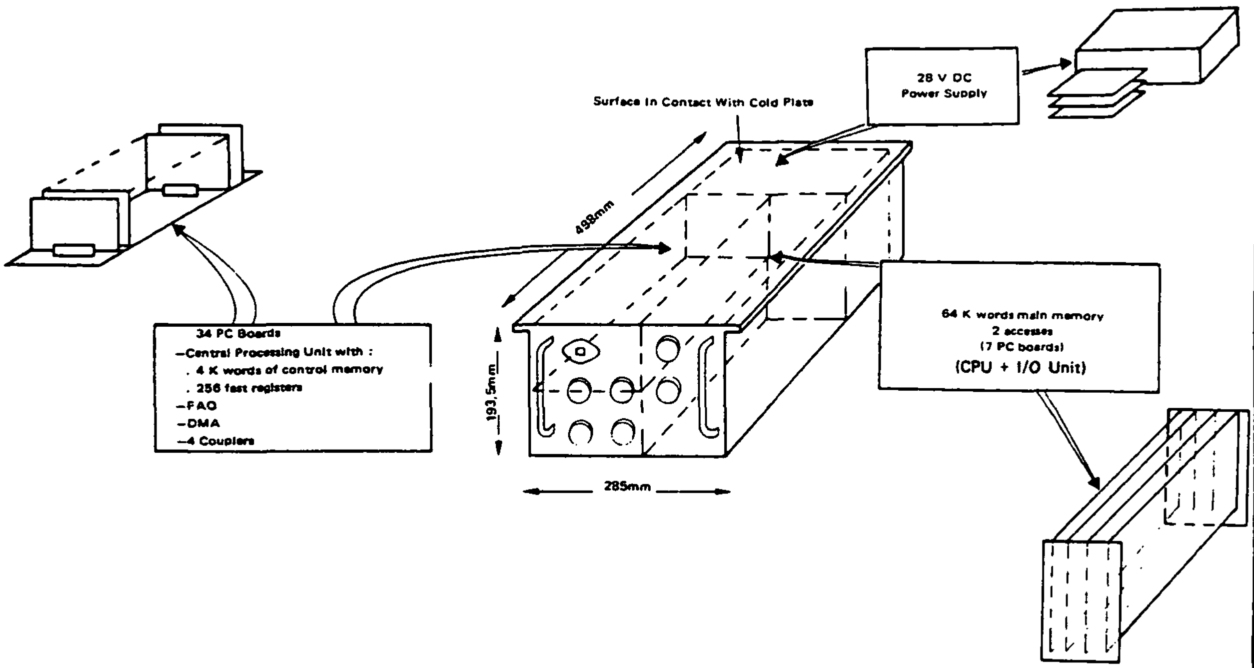

Figure 4.4 – 33d: General Physical Layout

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Memory

- Type : 2.5 D ferrite core memory.
- Organization: 18-bit words consisting of 16 data-bits, 1 protection-bit, and 1 parity-bit.

- Memory access: byte, word, or double-word.
- Read/write cycle : $\leq$ 950 ns.
- Usual capacity : 16 to 64 Kwords in 16 Kword increments.
- Number of memory channels 2 (CPU + DMA).
- Access time : $\leq$ 500 ns.

Central Processing Unit

- Type : microprogrammed with built-in Fast Arithmetic Operator (FAO) and direct access memory channel.
- Addressing capacity : 1 M Bytes.
- Arithmetic : binary, 2's complement.
- Microprogramming memories :
  control memory : 4,096 words with 20 bits,

- o executive memory : 2 sets of 88 words with 44 bits each,
- o memory associated with suspensions : 32 twenty-four bit-words,
- o duration of one microinstruction : approximately 300 ns,
- o functions performed : execution of the instruction code and peripheral coupling,
- o types of memories : ROM and PROM.

- Fast registers :
  - o 256 general registers with 16 bits each and 64 base and length registers,
  - o access time : approximately 60 ns,
  - o utilization : Central Processing Unit general registers: I/O channel registers, working registers for the micromachine.
- Interrupt system :
  - o 32 independent and hierarchical levels.
- Suspension system :
  - o 25 external suspensions and 7 internal on 4 levels,
  - o microprogram context acceptance and switching duration: approximately 300 ns,
  - o 1 internal signal for automatic traps in case of incidents.
- Data representation :
  - o binary : 1, 8, 16 or 32 bits,
  - o fixed point : 16 or 32 bits including the sign,
  - o floating point : 32 bits of which 7 bits are for the characteristic and 1 for the sign,
  - o character (byte) string length limited to 32 Kbytes.
- Instruction format : generally on one 16-bit word.
- Addressing modes (13 modes):
  - o immediate, indexed immediate,
  - o relative to the program counter : forward and backward,
  - o relative to the local base, general base, or common data base: direct, indirect, or indexed indirect.
- Number of instructions: 135 (see Table 4.4 - 11e).

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Table 4.4 - 11e: Allocation of Instructions

Type	Number
Memory reference	38
Registers	44
Shifts	12
System	29
FAO (32 bits)	9
DMA	3

- Processing speed

Fixed Point 16 Bits

Add/Sub	Direct	2 μs
	Indirect	3 μs
Mul	Direct	3 μs
	Indirect	4 μs
DIV	Direct	3.4 μs
	Indirect	4.4 μs

Fixed Point 32 Bits

Add/Sub	Direct	6.7 μs
	Indirect	7.4 μs
Mul	Direct	8.0 μs
	Indirect	9.8 μs
DIV	Direct	8.0 μs
	Indirect	8.8 μs

Floating Point 32 Bits (24 + 8)

Add/Sub	Direct	4.4 μs
	Indirect	5.4 μs
Mul	Direct	5.4 μs
	Indirect	6.4 μs
DIV	Direct	5.9 μs
	Indirect	6.9 μs

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Gibson Mix 3.2×10^5 Operations/Second

Inputs / Outputs

- Type : parallel 8/16 bit transfers on peripheral or memory bus.
- Transfer modes :
  - o single programmed, programmed on interrupts, microprogrammed on suspensions, or direct memory access (DMA).
- Transfer rates :
  - o 700 Kwords/sec for DMA channel
  - o 50 - 200 Kword/sec for minibus (on ground only)

4.4.6.2 Mass Memory Unit

4.4.6.2.1 General Description

The Mass Memory Unit (MMU) is a coaxial reel tape recorder for storage of all basic and flight application software for the subsystem and the experiment computers, thus enabling the CDMS to reload and periodically load the computer memories from the MMU through the associated IOU. The MMU will also be able to accommodate the data display skeleton formats and pre /in-flight stored experiment data for usage within experimenter provided programs.
The MMU will also provide a capability to load subsystem or experiment programs that were not contained in the initial program load.
About 50 % of the MMU Storage capability is available for experiments.

Writing into the MMU is allowed only on tracks specifically pre-selected for this purpose by an external connector with hardwired enable jumpers.Write protection can also be implemented by software means in the MMU handler of the operating system.

DCN 050 The IOU's communicate with the MMU through a half duplex serial databus with an effective data transfer rate of 500 kBPS.

4.4.6.2.2 Tape Organization

The Tape is organized in Files, Subfiles, Blocks and Words. The smallest unit for a read/write operation is a block. The layout of the tape is shown in Figure 4.4 - 33d.

Table 4.4 - 11f shows how many words, blocks, etc. are contained in a word, block, subfile, etc.

Table 4.4 - 11f: Tape Organization Matrix

	bit	Words	Block	Subfile	Track	File	Tape
bit	1	16	8 K	256 K	16 M	16 M	$1.34 \cdot 10^8$
Words		1	512	16 K	1024 K	1024 K	8192 K
Block			1	32	2048	2048	16384
Subfile				1	64	64	512
Track					1	8	8
File						1	8
Tape							1

K = 1024
M = 1024 · 1024

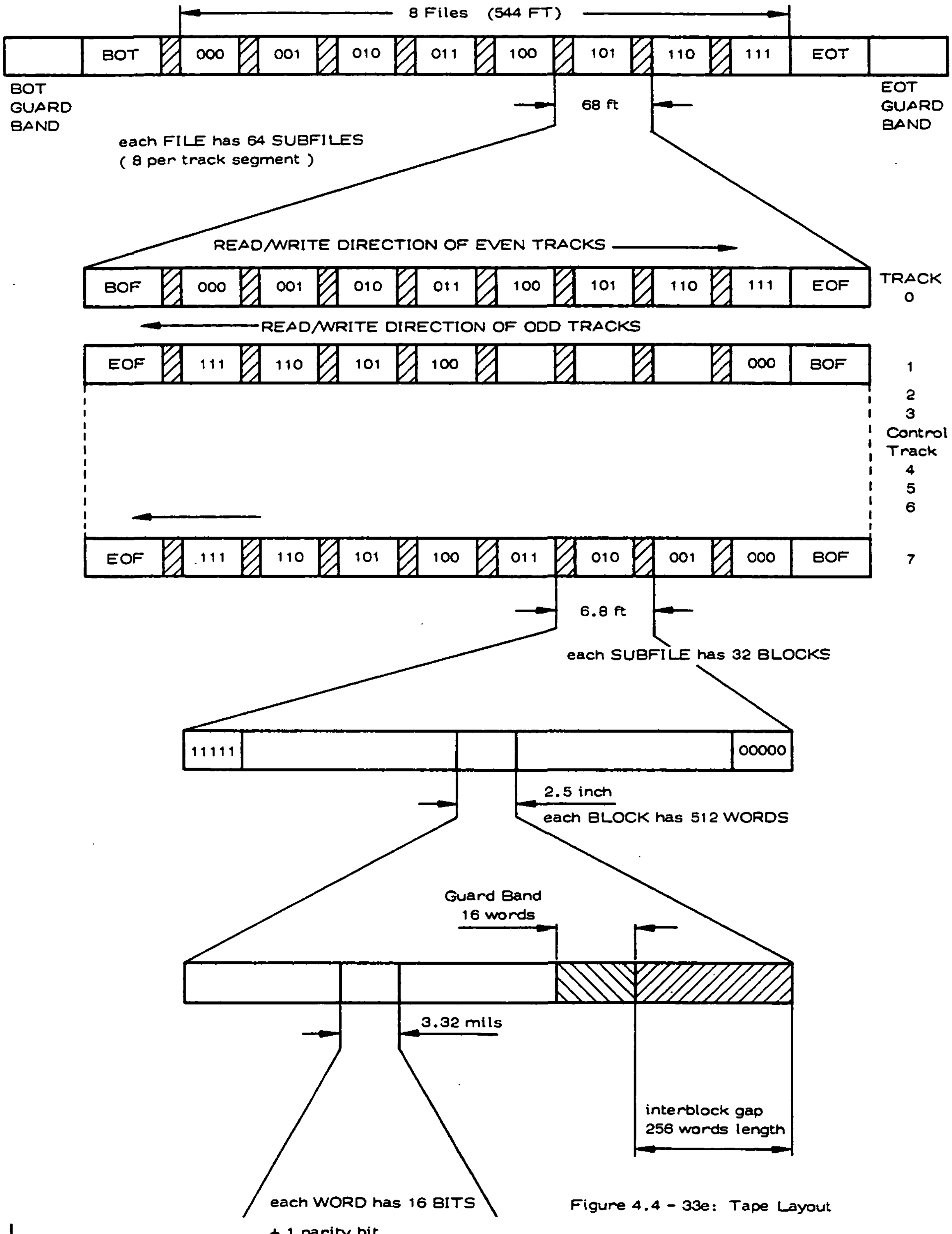

Figure 4.4 - 33e: Tape Layout

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4.4.6.2.3 MMU Operation

There are two different operational modes of the MMU

- Read / write
- File select

When in the read/write mode, the MMU executes any subsequent read or write command and stops the tape in the subfile or gap following the subfile where the last read or write operation occurred.

The file select mode is provided for positioning the tape from one file to another or to position within a file. The MMU enters this mode upon receipt of the position tape command. When the tape is correctly positioned, the MMU stops and returns to the standby condition.

Operations within a File

When data are to be read or written within the file in which the tape is positioned, it is necessary to preposition the tape only when the data block being addressed lies within the current subfile or behind it for the required direction of travel. Positioning may also be used to minimize access time.

DCN 050 As an example, assume that the tape is positioned at subfile 5 of track 2. If 16 blocks of data are to be read from track 2, beginning with subfile 6 block 30 of the file, the CDMS computer issues a single read command for track 2, subfile 6, with a starting block address of 30 and a block count of 15. (A block count of zero is equal to one block of data).

When this command is received, the MMU starts tape motion in the proper tape direction, monitors data locations from the control track until the specified location is reached, and then starts transmitting the recorded data to the CDMS computer. As soon as all the requested data are transmitted, the MMU inhibits data transfer, stops the tape within the next subfile (in this example it is the EOF). The MMU is then placed in standby awaiting the next request.

If the tape was initially positioned in subfile 6, 7, or EOF of track 2 at the start of the example, the CDMS computer read command would not be executed. For this case, prepositioning to a location before subfile 6 is necessary.

If more than 16 blocks of data are to be transferred, the CDMS computer must issue a command to extend the block count. This command allows up to 256 blocks of data to be read (or written) with one read (or write) command.

All read and write operations are restricted to a single file and a single track within the file. Data may be read or written irrespective of gaps and boundaries within any given file, but read or write operations may not cross a file boundary or switch between tracks. If a file gap is encountered while a read or write operation is being executed, the MMU will cease the operation, stop the gap, and indicate an error condition.

The sequence for writing data into MMU is similar to the sequence for reading data with the following exceptions:

1. A write command is used instead of a read command.
2. The write circuitry in the MMU must be enabled both by hardware enabling and by write enable command.
3. A search complete word (SCW) is transmitted to the CDMS computer a specified time before the designated location is reached (the timing for this sequence is shown in Figure 5). The SCW is transmitted before each new block address during a multiple block operation.

Write commands issued without the required enabling of the MMU write electronics will result in no action taken and an error indication being set. DCN 050

Operations Outside Present File

When data are to be read from a file different from the present MMU position, the CDMS computer issues commands to position the MMU tape in the proper file location before a Read or Write command is issued.

If it is assumed that the MMU is initially positioned in file 2, subfile 3, and that 129 blocks of data are to be read from track 2 of file 6, starting with the first block in subfile 3, the following command sequence would be issued by the CDMS computer:

POSITION TAPE

File 6
Subfile 2
Track 2

EXTENDED DATA

Block Count = 128

READ

Track = 2
Subfile = 3
Starting Block Address = 0
Block Count = x

The position tape command contains track, file, and subfile bits that are used to determine proper tape direction relative to the present position. In response to the position tape command, the MMU becomes not ready, tape motion is initiated, and the tape moves to the subfile specified by the position command. The tape stops in the center of the designated subfile (2), and the MMU becomes ready. The extended data command sets the block count to 128, allowing 129 blocks of 512 data words to be transferred by the following read command. The read command initiates tape motion to accelerate to normal speed and access the designated location. When block 0 of subfile 3 is reached, data transmission is started. Data transmission continues until 129 data blocks are transmitted (subfiles 3, 4, and 5, 12 blocks of subfile 6). The MM continues to move tape and then stops in the middle of the next subfile (7). Sequence and timing are similar to the previous operation (Figure 4).

The write sequence in this case follows the description given for read operations with the exceptions previously noted.

File Select

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File select is commanded to position the MMU tape in the file and subfile area in which subsequent operations are to be requested. This mode is enacted by the position tape command. As a response to this command, the MMU positions the tape in the designated location and stops. File select may be performed either separately or as part of a sequence requesting an immediate read or write operation as described in the previous paragraph, Operations Outside Present File.

Access speed

The time needed for tape positioning is for

- any subfile on the tape	1.1 to 65 sec
- any subfile in the file	0.7 to 8 sec
average	4.4 sec

The read / write time is for

- 1 subfile (16 K words)	0.9 sec
- 2 subsequent subfiles	1.8 sec
- 3 subsequent subfiles	2.7 sec
- 4 subsequent subfiles	3.6 sec

4.4.6.2.4 MMU Characteristics

The main MMU characteristics are listed below:

- Pressurized Coaxial Reel Tape Recorder (Spacelab/Orbiter Common Hardware)
- Tape Tensioning System with negator spring
- 1/2 inch certified tape 602 ft (200 M) long
- 8 data tracks and 1 pre-recorded control/address track
- total storage capability
- Tracks are read/written in alternate directions DCN 050
- Read/write speed 88 inch/sec (2.2 M/sec)
- Maximum data bus transfer rate 500 kb/s
- Bit error rate during MMU life less than 1 in 10^8 bits
- Tape/head wear over 20000 full length tape passes
- Negator spring wear over 20000 tape direction reversals
- Power
Standby	15 Watt
Read	70 Watt
Write	75 Watt
Fast	92 Watt
- Coldplate cooling $17^{o}C - 49^{o}C$
- Mass (29 LBS) 13 KG

4.4.6.3 Data Display System (DDS)

The operator/computer interface is performed via the data display system comprising the data display unit (DDU) and an associated keyboard (KB). Spacelab provides one DDU/KB in the Module (Control Center Rack) and one DDU/KB in the Orbiter AFD. The experimenter may operate a third Spacelab provided DDU/KB in an experiment rack. Spacelab provides all necessary signal interfaces routed to connector bracket CB 5 to operate a DDU/KB. The experimenter will be responsible for the power harness and the signal harness between CB 5 and the DDU/KB. The details are given in Appendix A, Avionics Interface Definition.

All DDU/KB's are connected both to the subsystem I/O unit and the experiment I/O unit by means of redundant display buses similar to the data buses.

DCN 051 Each DDU can display information from both computers simultaneously and the display format is chosen and determined by software. Each KB can communicate via the DDU and the display buses through the I/0 units with the SS and experiment computer by means of a manual switch. Each KB has also the ability to call either subsystem or experiment information for display on any of the three DDU's by software procedure.

4.4.6.3.1 Keyboard (KB)

DCN 051 The keyboard functions are shown in Figure 4.4 - 34. The alpha-numeric keys cover the full ASCII capability. In addition, 25 function keys are provided that generate an ASCII code which is recognized by the operating system software. The actual functions of ten keys are dedicated for user provided application software and can be chosen according to the user requirements.

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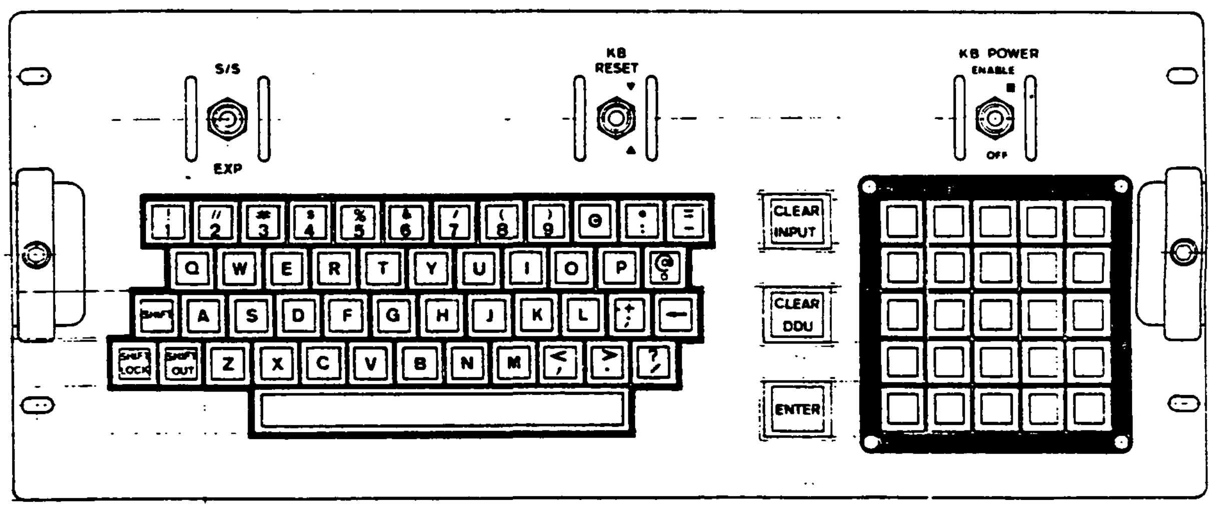

Figure 4.4 - 34: Keyboard

The keyboard hardware scans all key-switches (hall-effect type) sequentially, converts a key-position into an ASCII code, adds parity, and transmits the 8-bit information to the KB interface in the DDU. This KB interface is polled every 30 m sec by the operating software in the computers. The symbol is transmitted in a 16-bit word to the keyboard user interface routine (KBD) via the display bus under control of the operating system software (keyboard entry system, KES). The ASCII code is interpreted by KBD and sent back to the DDU for display on the operator scratch pad line under control of the operating system software (DDS handler (DDS) and display updating task (DISP)). DCN 051

A limited number of editing facilities for the operator input line are available (backspace, error message). When the statement on the input line is complete, the operator sends the statement to the operating software in the subsystem computer (SCOS) or experiment computer (ECOS) by means of the enter key. The operating software can pass statements to application software.

4.4.6.3.2 Data Display Unit (DDU)

The data display unit front panel is shown in Figure 4.4 - 35.

The DDU has a tricolor (green, yellow, red) penetration type cathode ray tube with a 12 inch diagonal screen. The DDU can display 128 different symbols with a total number of 999 symbols on 22 lines of 47 symbol positions and has a vector display capability. A power saving mode of operation is implemented in which the DDU is on standby, waiting for information from the computers. After receipt and storage of the information the DDU is in full operation within 2 - 5 seconds. A manually adjustable timer (30 sec - 5 min) switches the DDU back to the standby mode, awaiting new information. This feature can be inhibited by the operator.

The major hardware characteristics of the DDU are summarized in Table 4.4 - 12.

Table 4.4 - 12: DDU Hardware Characteristics

• Buffer memory for control address, words, symbols and vectors:	1024 words of 16 bits
• Available symbol:	128
• Display capability:	max. 999 symbols
• Available standard positions for symbols:	22 lines of 47 symbol positions
• x-y position matrix in viewing area for vectors and symbols:	820 x 620
• Size of symbols:	4.8 x 3.2 mm or 7.7 x 5.1 mm
• Space between symbols:	1.1 mm or 1.7 mm
• Space between lines:	1.6 mm or 2.6 mm
• Refreshing rate:	60 Hz nominal, 30 Hz minimum
• Colors:	red, yellow, green, overbright green
• Power consumption (DDU plus KB)	115/200 V AC, 400 Hz 3 phase
off:	5 W
standby:	70 W
operational average	220 W
maximum continuous:	296 W

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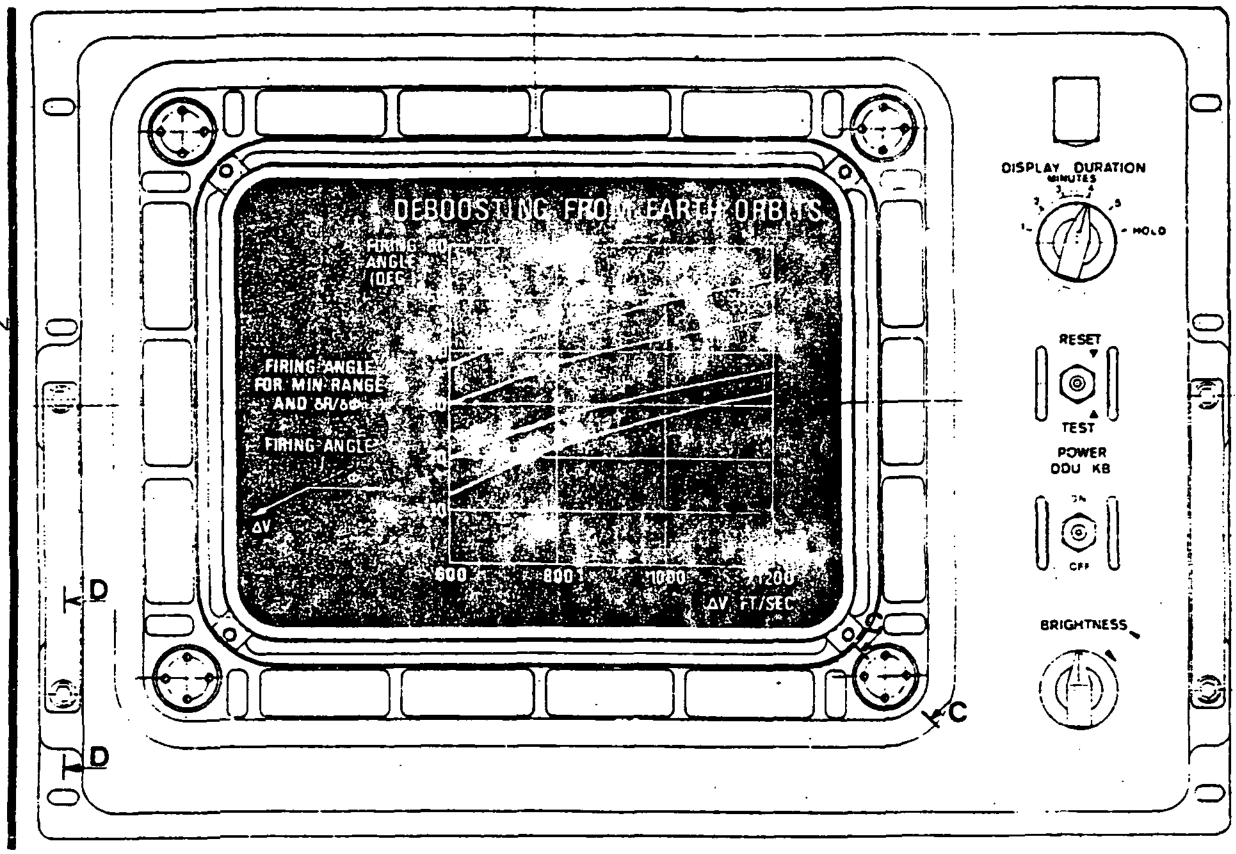

Figure 4.4 - 35: Data Display Unit Front Panel

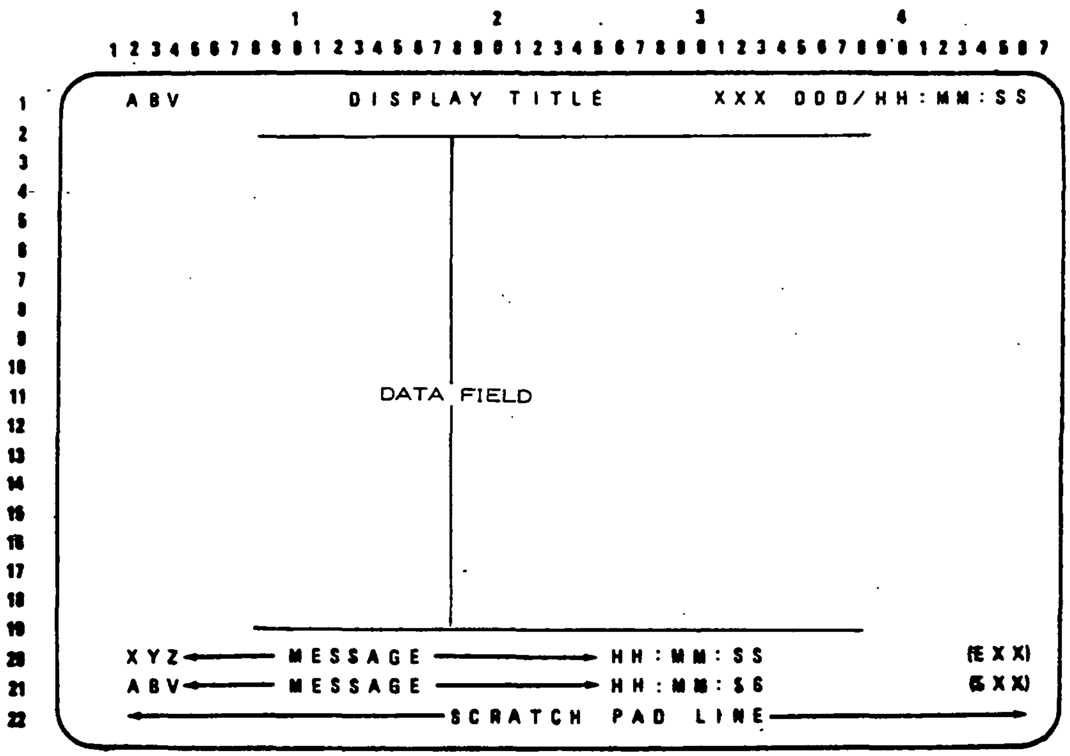

Figure 4.4 - 36: General DDU Monitoring Format

The system management information on the DDU is displayed in a display skeleton format with dynamic real time information. These skeleton formats are stored in-flight in the mass memory unit.

The display concept will be described in more detail in Section 4.5 Software.

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The DDU has a buffer memory of 1024 memory locations of 16 bit words. The hardware scans this buffer memory 60 times per second nominal (999 green symbols). In cases where the buildup of the picture is very time consuming (due to the low writing speeds in red, switching color, or writing long vectors at different angles), the refresh rate will automatically be reduced to a minimum of 30 Hz (in the case of 999 red symbols). At 30 Hz the resulting flicker of the DDU warns the operator of an overload situation. The first word in the buffer memory is the status word which contains parameters like color, size, dash, flash, blanking, and overbright.

DCN 051

The status word defines the status of all subsequent symbols and vectors up to the next status word:

The following two words are the x and y position of the first symbol. The DDU automatically increments the x-position. The y-position is incremented by a carriage return control symbol, automatically setting x to the beginning of the line

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X: there are 47 symbol positions nominal or
+ and − 410 positions referenced to Xo

Y: there are 22 line positions nominal or
+ and − 310 positions referenced to Yo

Buffer memory words 4 through 1023 can be used for:

- alpha-numeric symbols (128 different possibilities):

  A through Z/ a through z / α β γ δ ε ρ ω σ π Θ Σ Φ Ω Δ ν /
  .⊙ through 9 / ! " % & ' ( ) [\] , ∧ . / : ; ? □ → ← { }
  .↑↓ ▽ ▷ ◁ · ÷ ·· # $ * + - < = > @ ° ═ ∼ _ | ± ≦ ≧

  DCN 051

  each requiring one buffer memory word.
  Each symbol is micro-programmed in a 6 x 4 matrix, by means of a segment stroke writing method

- vectors (1024 different lengths, 4096 different angles)

  Each vector is defined by origin, length, cos and sin of angle (maximum 5 words).
  It is not necessary to repeat each of the 5 parameters for each vector, if this parameter is defined in a preceding memory location in the buffer memory. Vectors can be dashed.

DCN 051

The alpha-numeric symbols and vectors can be displayed in a flashing mode on an individual basis without changing the status word.

The last word of the display picture is the end-of-picture word which causes the electronics to go back to the first word of the buffer memory (elapsed time 16 - 32 m sec for one scan).

4.4.7 System Activation and Monitoring (SAM)

4.4.7.1 Control Concept

Most of the 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 to subsystem equipment. Similarly, the RAU is capable of receiving analog or digital monitoring signals. DCN 051

Those functions which require control prior to the activation or after de-activation of the subsystem computer are performed by SAM. The major hardware components of the SAM are:

- the IMCP - R7 panel in the Orbiter AFD providing manual switches and signal displays
- the Remote Advisory and Amplifier Box (RAAB) located on the Spacelab main floor extension in the core segment/Igloo. The RAAB provides for signal conditioners and signal amplifiers.

The functional flow of the SAM is shown in Figure 4.4 - 37.

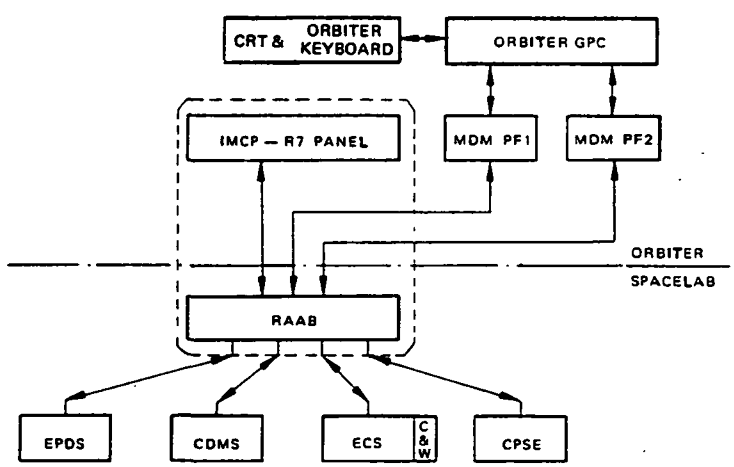

Figure 4.4 - 37: System Activation and Monitoring Block Diagram

The R 7 panel is the hardware device to control Spacelab subsystems by manual switches and to monitor the subsystem status by signal lamps and a digital display. An MDM under the control of the Orbiter computers is capable of transmitting ON/OFF commands to Spacelab subsystem equipment. This 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's, 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 operational modes are possible; however, precise control modes will be established for each Spacelab mission, supported by documented procedures.

The lay-out of this Integrated Monitoring and Control Panel R 7 is depicted in Figure 4.4 - 38.

4.4.7.2 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 the Orbiter keyboard at the Mission Station (see Figure 2 -21).

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) through MDM commands.

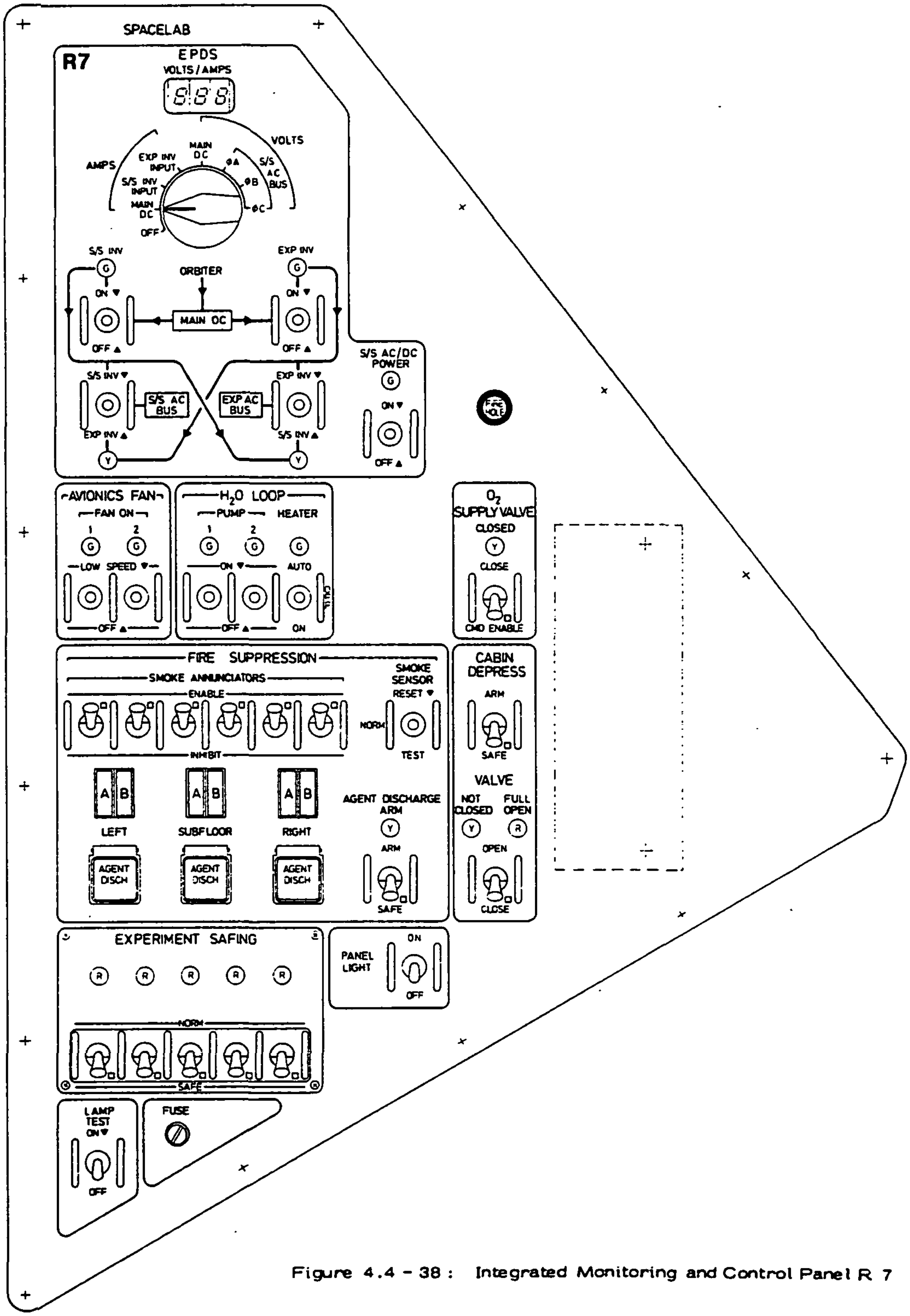

Figure 4.4 - 38 : Integrated Monitoring and Control Panel R 7

c) After computer operations have been verified and the subsystem software has been initiated, the subsystem computer with I/O unit and RAU's can be used to distribute further commands to the subsystem equipment. These types of command are normally operator-initiated from the Spacelab CDMS 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 are controlled from the keyboards via the subsystem computer and RAU's, thus:

- cabin temperature control
- cabin oxygen/nitrogen control
- humidity control, cabin fan
- experiment computer including I/O unit
- experiment RAU's and experiment power distribution boxes

4.4.8 Intercom

The Orbiter Audio Distribution System and the Spacelab Intercom Assembly are fully integrated, employ the same headset/umbilical assembly and have operational commonality.

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The Spacelab Intercom system comprises an Intercom Master Station (ICMS) in the control center rack and an Intercom Remote Station (ICRS) in the work bench rack as basic subsystem equipment which will fly in all Module modes. A further three remote stations can be fitted into dedicated experiment racks (as mission dependent equipment). In the short module configuration an ICRS is located in experiment rack number 4, but in the long module configurations ICRS's are foreseen in experiment racks 4, 7 and 10. The ICMS has provisions to connect up to six remote stations. The front panels of the ICMS and the ICRS are shown in Figure 4.4 - 39. A remote loudspeaker, for paging only, is attached to the aft end cone.

The Spacelab intercom includes main & emergency D/C to D/C converters which are separately fixed in the ICMS for connection to the Main DC and Emergency DC power buses. Additionally, main or emergency DC power is distributed within the ICMS and to the ICRS's for supplying power to the Orbiter common headset umbilical assemblies facilitating full operational capabilities from either power source.

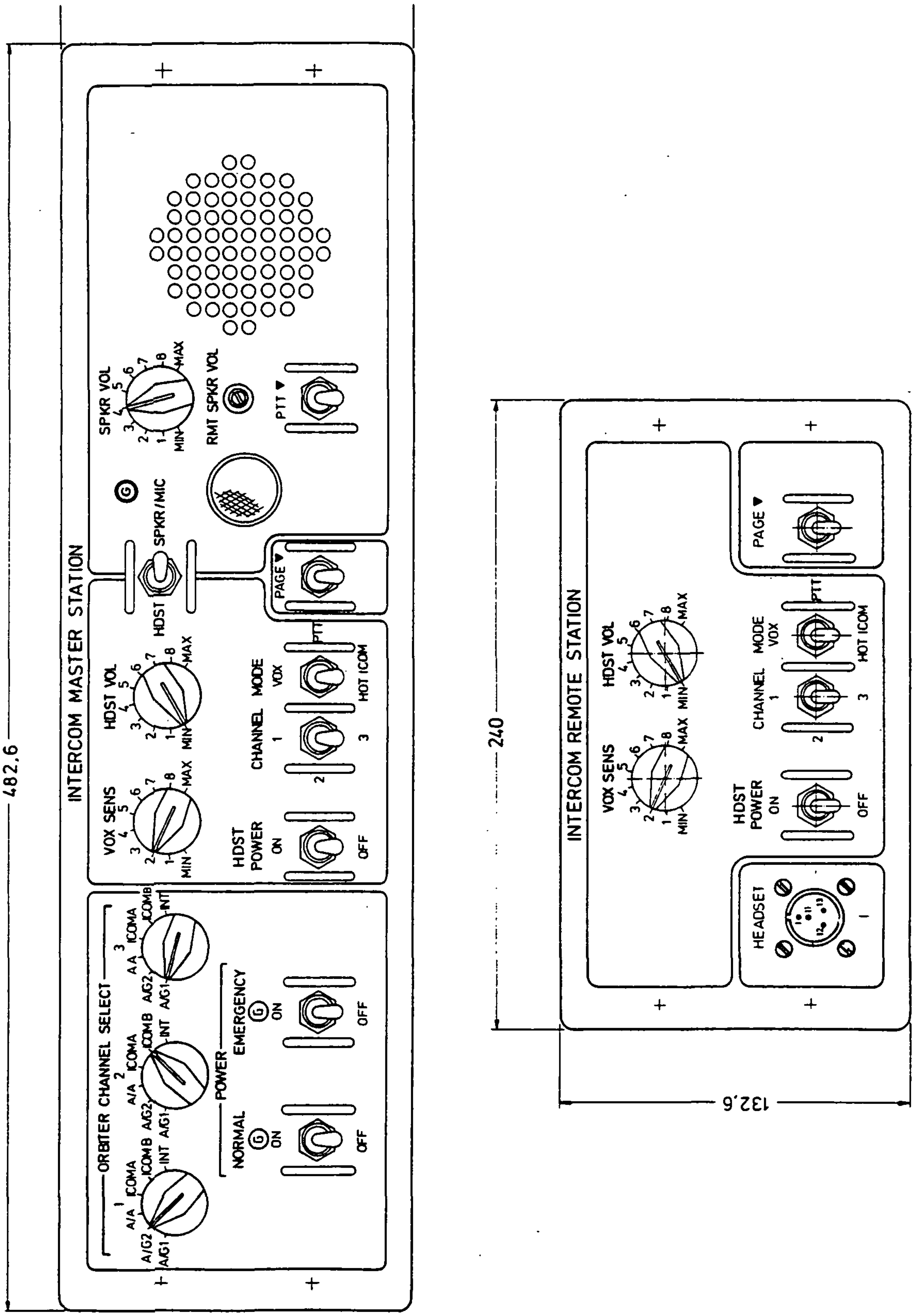

Figure 4.4 - 39: Intercom Master and Remote Station

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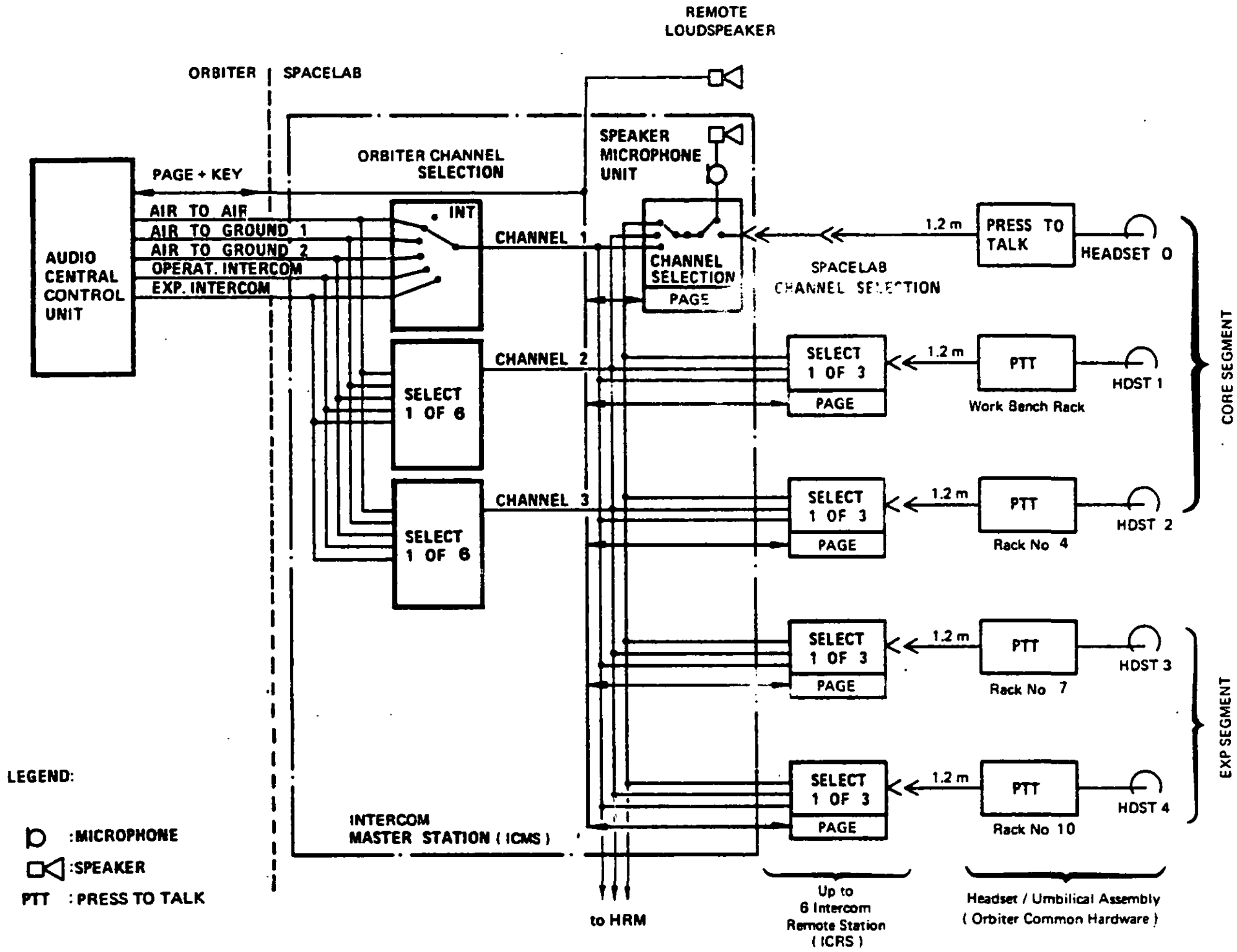

Figure 4.4 – 40: Functional Intercom Block Diagram

4.4.8.1 Channel Routing

The overall channel routing capability is depicted in Fig. 4.4 - 40. The ICMS, which is the control and audio signal processing center of the system, interfaces with the Orbiter Audio Central Control Unit (ACCU) and Orbiter EVA/ATC transceiver (for Air to Air transmitter keying) to facilitate communications on the following full duplex (simultaneous talk and listen) audio channels:

• Air to Ground 1	A/G 1
• Air to Ground 2	A/G 2
• Intercom A (Orbiter/SL internal)	ICOM A
• Intercom B (Orbiter/SL internal)	ICOM B
• Air to Air (EVA)	A/A
• Page (Orbiter/SL internal)	PAGE
• Internal SL	INT

Each of the above Orbiter channels, with the exception of Page, may be selected on each of three Spacelab full duplex channels, which are distributed via interface cards in the master station to each ICRS and the audio facility included in the ICMS.

Page signals for general address or calling purposes originated in the Orbiter or within Spacelab or in both locations are superimposed on each channel at a level of + 6dB above the nominal value. The page signals are routed to the headsets, the loudspeaker in the master station and the remote loudspeaker. DCN 051

Access to the Page Talk line is obtained by operation of a special PAGE (PTT) switch mounted on all Intercom stations.

Communication within Spacelab is provided by feeding back channel talk signals onto the channel listen lines for distribution to any Intercom Station selected to the same channel (INT channel).

Spacelab channel talk and listen lines are combined for distribution to the Voice Digitizer in the High Rate Multiplexer (HRM) for all three Spacelab channels.

The Spacelab channel allocation and the Page superimposition is performed by logic circuits in the ICMS that receives discrete commands from each ICRS or the ICMS itself.

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4.4.8.2 Microphone Modes

Each Intercom Station (ICMS, ICRS) provides for connection of an Orbiter type Headset/Umbilical assembly and includes beside the switches for a local selection of a Spacelab channel, switches for different microphone modes.

DCN 051

The logic circuits in the master station receive discrete commands from each intercom station for channel and microphone mode selection.

Microphone signals are gated onto the channel talk circuits in the following different modes

DCN 051

- PTT - only after activation of the press-to-talk switch in the headset umbilical or on the master station control panel. By selecting PTT and Pressing biased switch.
- VOX - only after exceeding the VOX threshold which is adjustable on the Intercom control panels.
- HOT ICOM - continuously, but only onto the Orbiter ICOM A and B. (If an invalid switch position is selected, the system automatically reverts to the PTT mode).

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The ICMS includes, in addition to the headset which is identical to an ICRS, a speaker/microphone unit.

The speaker/microphone unit can be selected in place of the remotely connected headset for channel access. In this case, however, system operation is limited to VOX or PTT and a special PTT switch is provided.

- In the HOST mode normal channel communications are provided at the ICMS via a remotely connected headset/umbilical and the loudspeaker carries only paging signals.
- With the speaker/microphone selected, normal channel communications are switched to the speaker/microphone unit (in addition to page signals).
  Access to the channel talk line in this case is via a panel mounted PTT switch.
  A green LED indicates when the microphone is activated by PTT vox. or page.
  Operation of the PTT switch on the ICMS disconnects the ICMS loudspeaker for listen and page. Page is always present on the remote loudspeaker.

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4.4.9 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 and discrete C & W signals is performed (software controlled) in the Orbiter GPC and in the Spacelab Subsystem Computer for redundancy. An overall functional block diagram of the C & W system is given in Figure 4.4-41 for module configurations and Figure 4.4 - 42 for pallet only configurations .

4.4.9.1 Emergency Signals and Safing Commands

Emergency signals of Spacelab apply only to fire and rapid pressure loss in the Module.

There are two types of sensors foreseen:

- $\Delta p/\Delta t$ sensors indicating rapid cabin depressurization.
- Three redundant pairs of Smoke sensors. One pair each is located in the left side and the right side of the avionics loop and in the cabin loop.

The Spacelab $\Delta p/\Delta t$ sensor output is hardwired to the Orbiter Caution & Warning Electronic Assembly (CWEA). The input of the CWEA is connected to the Orbiter sensor only during ascent/descent and to Orbiter & Spacelab Sensors on orbit.

An emergency tone (Klaxon) will be generated by the Orbiter CWEA when a $\Delta p/\Delta t$ is detected.

For redundancy reasons, the Smoke sensor outputs are routed independently

- through the subsystem RAU's B and C to the Spacelab subsystem computer.
- through the Integrated Monitor and Control Panel (IMCP - R7 panel) to the Orbiter CWEA, to the Orbiter Fire & Smoke Annunicator Panel, and via MDM - PF 2 to the Orbiter System Management GPC. In addition Fire and Smoke conditions detected in the Spacelab subsystem computer are annunicated to the GPC through the PCM Master Unit.

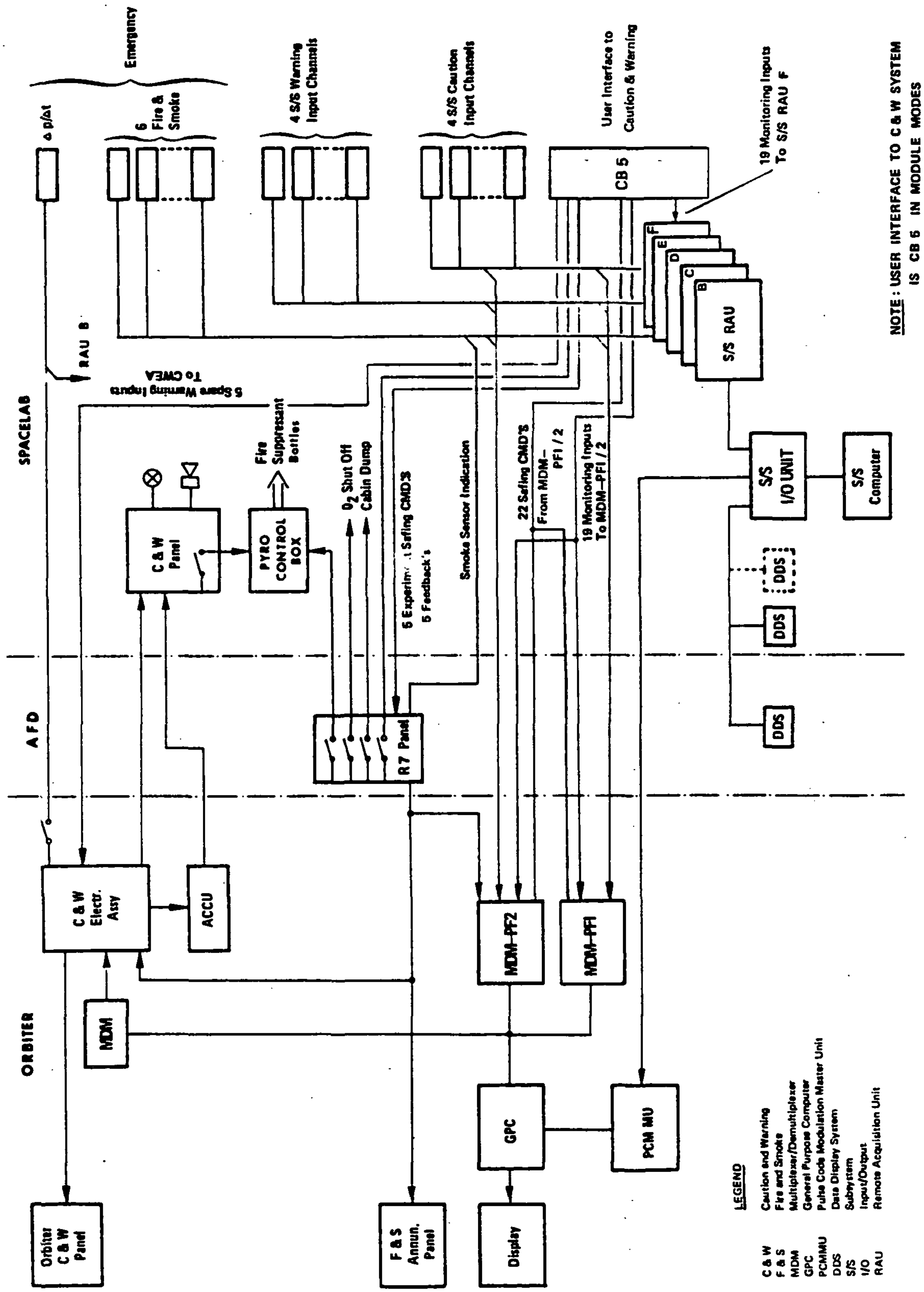

Figure 4.4 - 41: Caution and Warning Functional Block Diagram (Module Configurations)

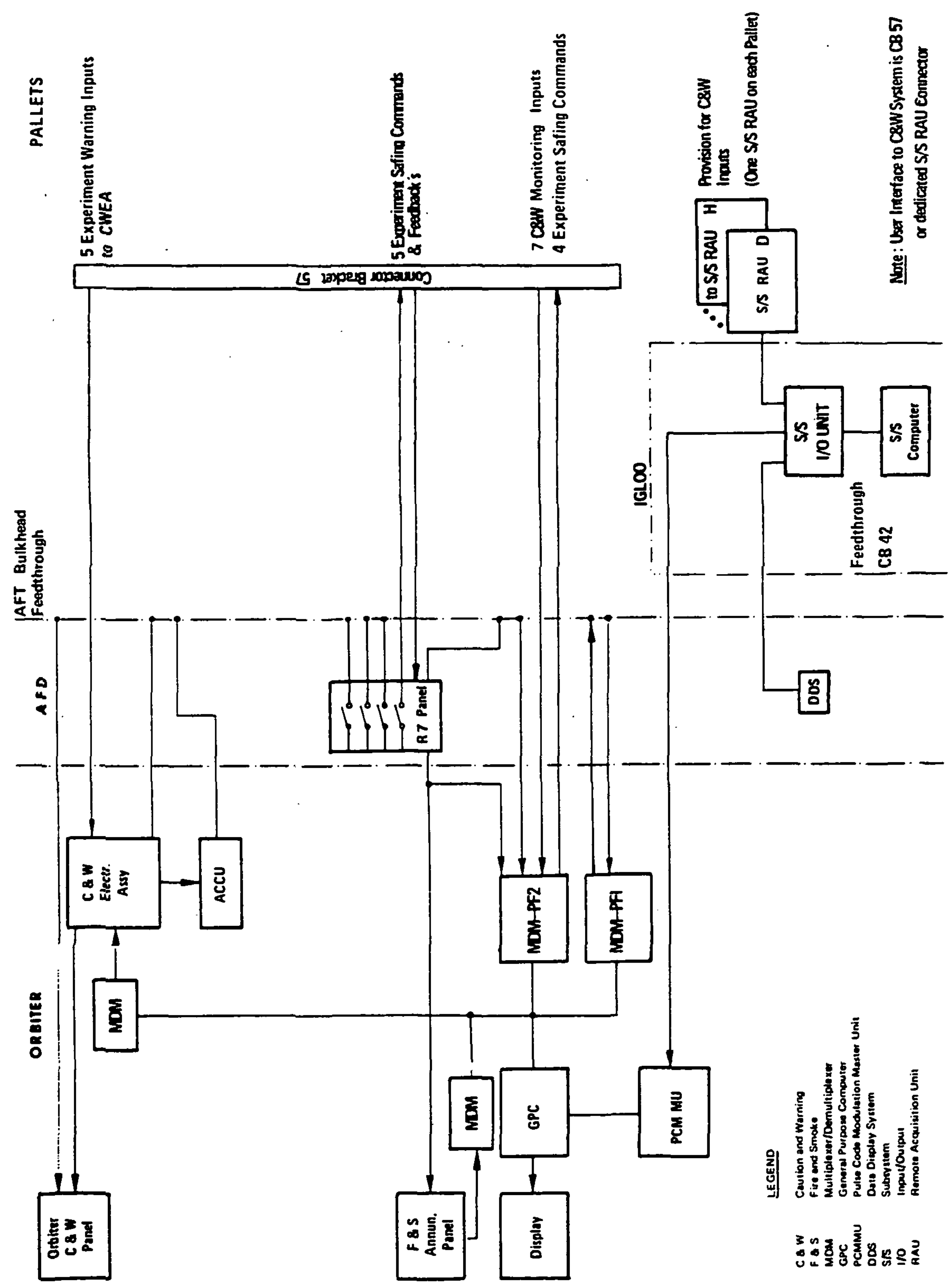

Figure 4.4 - 42: Caution and Warning Block Diagram (Pallet-Only Configurations)

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From the Orbiter CWEA, emergency conditions are signaled at the Orbiter C & W panel and at the Spacelab Caution & Warning/Fire Suppression System panel (C & W/FSS panel) in the module.

To act on the detection of Spacelab Fire & Smoke conditions, manual switches on the IMCP-R 7 panel allow for activation of the Spacelab suppression system, the O_2 shut-off valve, and the cabin dump valve. The fire suppression system in addition can be manually activated from the Spacelab C & W/FSS Panel in the Control Center Rack. or can be manually activated from the Spacelab C & W/FSS Panel in the Control Center Rack.

An emergency tone (siren) will be generated by the Orbiter CWEA when a Spacelab Fire/Smoke signal is detected.

4.4.9.2 Caution and Warning Signals

The Spacelab Caution & Warning System can accommodate caution & warning sensors (from a hardware viewpoint, caution signals and warning signals are treated identically). Currently there are 4 caution and 4 warning sensors dedicated to monitor Spacelab subsystems. 19 C & W input channels of the remaining input channels are available for experiments. In addition five direct warning inputs to the Orbiter Caution & Warning Electronic Assembly are available for experiments.

Redundancy in detecting caution & warning conditions is achieved by routing the C & W sensor outputs via independent signal conditioners through the

- MDM's PF 1 or PF 2 to the Orbiter System Management GPC. C&W conditions detected by the GPC are displayed on the Orbiter Display Unit and the Orbiter C & W Panel.
- Subsystem RAU's B, C, D, E, F to the Spacelab subsystem computer. C&W conditions detected by the subsystem computer are displayed on the Spacelab Data Display System. Experiment C& W sensors are connected to S/S RAU-F in module configurations, (see Figure 4.4 - 41) or to the dedicated inputs of pallet S/S RAU's in pallet-only configurations (see Figure 4.4 - 42).

These two branches of the C & W system, through MDM to Orbiter and through RAU to Spacelab, are interlinked.

- The link from the Orbiter GPC to Spacelab for detected C & W conditions is through the CWEA and the ACCU to the Spacelab C & W/FSS Panel.
- The link from the Spacelab subsystem computer is through the subsystem I/O unit and the PCMMU to the Orbiter GPC.

4.4.9.3 Caution and Warning Safing Commands

The Orbiter will provide a maximum of 36 safing commands to be used in response to Spacelab Caution & Warning conditions. 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 28 V) to Spacelab via MDM - PF1 and PF2. 22 MDM safing commands are reserved for experiments.

Five safing commands, manually switched from the R 7 Panel in the Orbiter AFD are available exclusively for experiments to act on Caution & Warning conditions.

4.4.9.4 Experiment/Caution & Warning Interface

To interface with the C & W system through the 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, active signal conditioning (for the S/S. RAU and for the MDM link); these signal conditioners have to be powered by the emergency bus.

Also actuators controlled by the safing commands have to be powered by the emergency bus.

The physical location of the experiment interface to the Caution & Warning System is depicted in Figures 4.4 - 41 and 4.4 - 42.

For module configurations the location of these interfaces is the connector bracket CB 5, and for pallet-only configurations connector bracket CB 57 and a dedicated connector of the pallet subsystem RAU.

Connector notation, pin allocation and signal characteristics at the Spacelab/experiment C&W interfaces are given in Appendix A, Avionics Interface Definition.

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4.5 Software

The Spacelab Computer Software comprises the software used for Spacelab during software development, 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 and Integration 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 defined functions.

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, check-out language interpretation, subroutine library, etc.

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. To a minor extent,he also might deal with the ATE computer software for experiment integration purposes. Also available is the interpreter package which comprises the monitor functions, the operator's communication function and the check-out language interpretation function. Furthermore, means are provided to support the experimenter in compiling, testing and integrating his software. Table 4.5 - 1 gives an overview of the Spacelab Software Environment.

4.5.1.1 CDMS Computer Operating System

The Subsystem Computer Operating System is at present the same as the Experiment Computer Operating System (SCOS/ECOS). However, since ECOS has to accommodate a variety of experiment applications, the requirements for ECOS are under review and a new version will be developed by NASA.

The development schedule and the time of availability of the new ECOS is not yet established and possibly the present SCOS will be modified to use as ECOS for the early missions.

The SCOS/ECOS allows the performance of both asynchronous and synchronous tasks. The executive performs initialization, scheduling and termination of tasks. It assures time scheduling, loading of tasks - including memory allocation to them - and management of the various data tables for program control and housekeeping. It controls the allocation of the computer peripherals such as mass memory, keyboards, data display unit, telemetry channels, and RAU data bus. The executive allows for initialization of the computer system and for convenient recovery after system failure. The executive includes a self check function for computer, I/O unit and RAU's 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). They transmit data to the DDU's for presentation to the crew and experimenters and receive and process external event messages based upon usage of the keyboard(s). 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 and for communication with the Orbiter general purpose computer (GPC), including loading of software from ground. They check the status of the peripherals (parity checking, data ready bits, data available bits as applicable).

SCOS/ECOS supports a limited subset of HAL/S real time language features (see 4.5.2.2).

The functions summarized as general facilities are functions common to all or most of the application programs and include services such as conversion of raw data into engineering units, library of mathematical functions, etc.

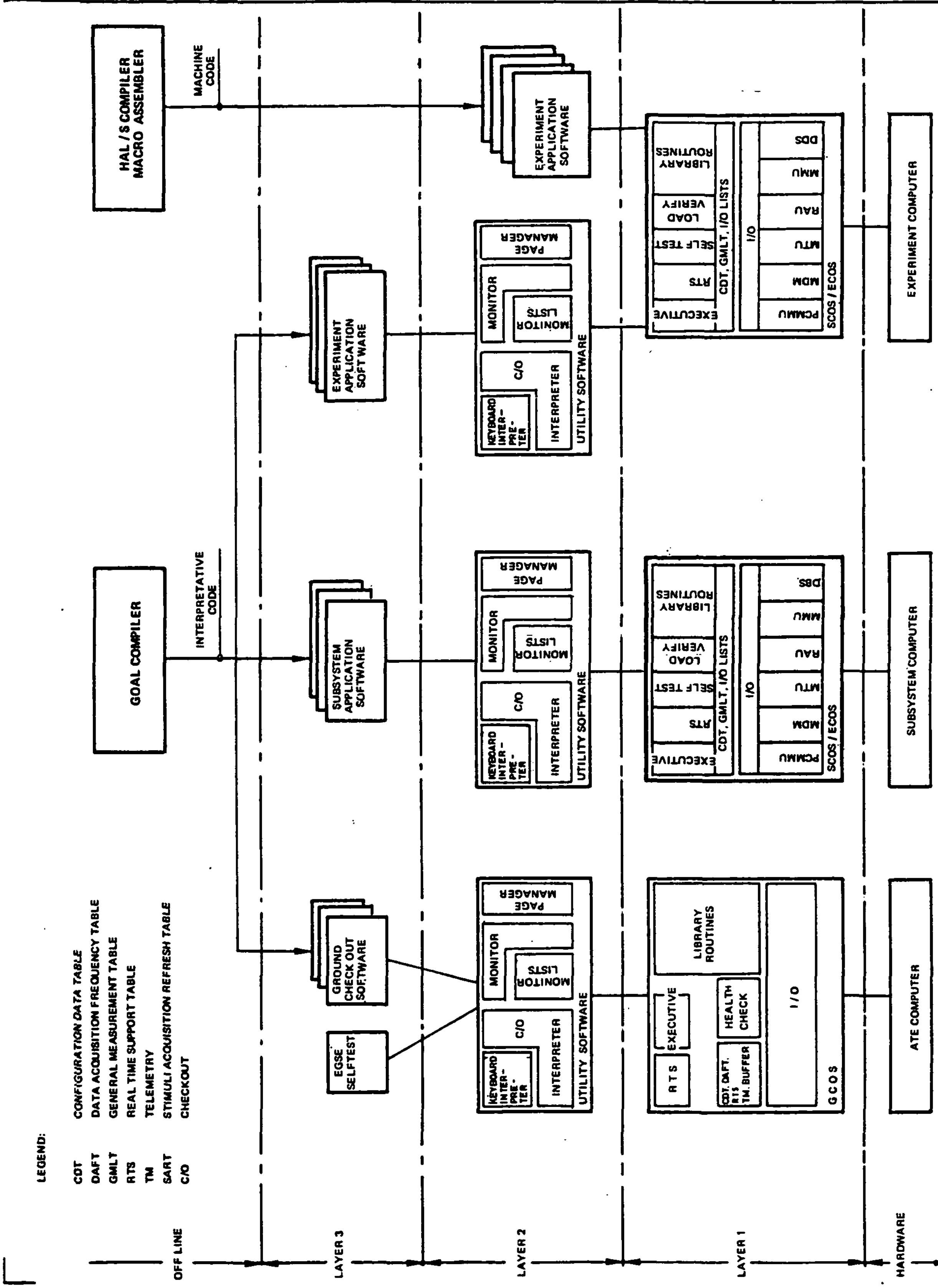

Table 4.5 – 1: Spacelab Software Schematic

Table 4.5 – 2 : Core Resident Software for the Subsystem Computer

	CDMS-S/S-COMPUTER FOR GROUND CONTROL & FLIGHT SIZE (K WORDS)		CDMS-EXP.-COMPUTER FOR GROUND CONTROL & FLIGHT SIZE (K WORDS)		ATE-COMPUTER EGSE FOR GROUND CONTROL SIZE (K WORDS)	
CORE RESIDENT S/W	Current Size	Nominal	Current Size	Nominal	Current Size	Nominal
Operating System	21.8	17.0	21.8		62.5	54.0
Configuration Data Table	4.0	4.0	8.0		12.0	8.0
Load & Verify	2.0	2.0	2.0			
Interpreter including Monitor support & Page Management	22.6	14.0	22.6		22.6	16.0
(Interpreter reduced)	N/A	N/A	(15.0)		N/A	N/A
Page Display Buffer	3.5		3.5		3.9	
Monitor Lists	4.0		4.0		4.0	
		12.0				16.0
GOAL Reload Area	8.0		2.1		8.0	
(Non-GOAL Reload Area)	N/A	N/A	(9.7)		N/A	N/A
Total Actual S/W	65.9	49.0	64.0		113.0	94.0
Computer Memory Size	65.5		65.5		131.1	

The current sizing and timing estimates for the subsystem computer and SCOS/ECOS are presented in Tables 4.5 - 2 thru 4.5 - 5 to give a first idea of the potential core requirement.

Average operating system overhead is estimated to be 20 % of CPU time. Reaction time on external events is estimated to be 100 µ sec maximum.

The S/W - S/W interface between SCOS/ECOS and experiment application packages is managed by supervisor calls and data tables.

The S/W - H/W interface between SCOS/ECOS and the peripheral hardware is handled via handlers in the SCOS/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 SCOS/ECOS provides the interface between the operator and the computer system. The language is based on a subset of GOAL. The SCOS/ECOS will be capable of displaying monitoring data on the DDU in a tree-like structure, as described in para. 4.4.6.3.2. The interpreter package will update these displays in accordance with the latest sensor readings.

The functions involved are calling for computer status display, initiation and termination of experiment tasks, transfer of messages to application programs and changes to experiment modules.

Table 4.5 - 3: System Timing

	CDMS-S/S-COMPUTER FOR GROUND CONTROL & FLIGHT (msec)			CDMS-EXP.-COMPUTER FOR GROUND CONTROL & FLIGHT (msec)			ATE-EGSE-COMPUTER FOR GROUND CONTROL (msec)		
	PRESENT	LAST	DELTA	PRESENT	LAST	DELTA	PRESENT	LAST	DELTA
Asynchronous Read									
from RAU - Assembler	1.8	1.8	-	1.8	1.8	-		N/A	
- HAL/S	2.0	2.0	-	2.0	2.0	-		N/A	
from MSU - Assembler & HAL/S		N/A			N/A		5.3	11.0	- 5.7
from/to SCCD - Assembler & HAL/S		N/A			N/A		6.8	6.5	+0.3
Synchronous Read									
from GMLT - direct access	0.01	0.01	-	0.01	0.01	-		N/A	
- access via CDT	0.3	0.3	-	0.3	0.3	-		N/A	
from CDT, DAFT, TM		N/A			N/A		2.0	2.0	-
Command Issue Time									
- Assembler		N/A			N/A		6.8	11.0	- 4.2
- HAL/S	2.0	2.0	-	2.0	2.0	-	6.8	11.0	- 4.2
Throughput Rate for GOAL-Application S/W	(Goal-statements per second)								
- worst case	114	125	- 11	114	125	- 11	11	0	+ 11
- average	130	141	- 11	130	141	- 11	77	40	+ 37

Table 4.5 – 4: CPU Load for Subsystem Computer

Consumer	CPU load (%) worst	CPU load (%) average	peculiarity cyclic	peculiarity random	Remarks
GML	22.2	22.2	X		activated every 10 msec
Clock task	3.7	3.7	X		activated every 10 msec, 1 delay to handle
MDM	5.0	0.5		X	assumption is an average of 1 MDM request per second, worst case 10 requests
Monitor					see design goals
	1.6	1.6			activated every 10 ms
	2.1	2.1	X		activated every 100 ms
	2.6	2.6			activated every 1 second
Selftest		TBD			under negotiation
PCM	0.01	0.01	X		
KBD	0.6	0.2		X	average: 1 char/sec peak: 3 char/sec
DDU A	1.1	0.2		X	average: 1 page per minute, 20 lines/page, 0.33 line/sec worst: 1 page per sec., 20 lines/sec
DDU B	0.8	0.6		X	average: time to be updated on one DDU worst: time to be updated on three DDU's
Subtotal basic tasks	39.7	33.7			
OS overhead	10	10			
CPU-time lost for Cycle stealing	4.7	4.3			
Total	54.4	48.0			
Available for GOAL tasks	45.6	52.0			
Grand total	100	100			

Table 4.5 – 5: Interpreter Timing for Subsystem Computer

FUNCTION	ESTIMATE FOR NUMBER OF EXECUTED HAL/S STATEMENTS	EXECUTION TIME in msec	REMARKS
Monitor one analog value or a group of 16 discretes	8 ± 2	0.15	To be repeated n times each 10 msec, n will be determined at system generation time
Process an exception flag by monitor auxiliary	100	2	Happens normally for updates of monitor table due to commands, happens abnormally for an out of limit condition
Update a page with one line and indicate update to OS	100	2	Normal processing for display output, happens at maximum once per second
Analyse a token from the keyboard	1000	20	Normal processing for keyboard input, happens at maximum once per second
Interpret a checkout statement	150	3	After interpretation system subroutines of OS services will be involved A checkout program comprises about 2000 statements

4.5.1.2 Interpreter Package

The Interpreter Package being part of the experiment computer basic software is a utility software package that includes the following functions:

- Checkout interpreter function to analyze and execute interpretive code generated by the GOAL compiler from GOAL check-out programs written by the experimenter.
- Keyboard interpreter function to handle operator inputs. These inputs can be GOAL statements which will be converted to interpretive code and then be transferred to the check-out interpreter function. The inputs can also be direct executable system commands.
- Monitor function which - following a list given by the user - continuously reads sensor values and compares them against limits. If out-of-limit condition occurs, a message to the operator is given on the DDU.
- The Interpreter utilizes configuration dependent data tables generated off-line.
- Page managing function to generate on DDU display pages which belong to GOAL programs, the keyboard interpreter, and to the monitor function.

4.5.1.3 Facilities Available for Application

The experiment application software for the experiment computer is the software executed by the ECOS and the Interpreter. The Spacelab provided part consists of the check-out Software for the experiment portion of the CDMS. All experiment related software packages to be loaded in the experiment computer are the responsibility of the experimenter or payload integrator.

4.5.1.4 Software Integration

For integration of experiment application software packages, the experimenter or payload integrator will be supplied with the following software (para. 4.5.2):

- Interpretive computer simulator (ICS)
  This software simulates the Mitra 125 S/MS on a host computer IBM 370.
- Input/Output Box and Peripheral Simulator (IOBPS)
  This software will simulate on a host computer IBM 370 the CDMS environment as seen by the CDMS Software.
- Experiment computer operating system (ECOS)

The IOBPS and the ICS can be integrated in order to simulate the complete CDMS on the host computer.

4.5.2 Software Development Aids

These software packages will be used to support the effective development and maintenance of all Spacelab software, i.e. operating systems, ground checkout packages and the flight application packages.

The experimenter, in developing his experiment software, should utilize the facilities provided as far as possible. The intention is to ascertain compatibility with the real Software environment for flight application and ground check-out as early as possible. For debugging, the simulator software will be employed (see para. 4.5.1.4).

4.5.2.1 Host Software System

The host software system comprises support software useful for the development of experiment software and executes on a host computer (IBM 370). The following items will be available:

- HAL/S - 360 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 interpretive code. The interpretive code can be executed by an interpreter running on a Mitra 125S/MS computer. The compiler itself will run on an IBM 370.

- Interpretive Computer Simulator (ICS)
  The ICS will simulate the Mitra 125 S/MS on machine instruction level. 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 will work together with the ICS. It will execute on an IBM 370.

4.5.2.2 Programming Languages

The availability of programming languages for experiment software is highly dependent on the operating system actually used. The SCOS design will support the following languages:

- HAL/S
  Experiment dedicated software requiring real time features should be written in HAL/S, a real time programming language which allows the scheduling and synchronization of program steps. The language also allows the manipulation of vectors and matrices and data structures in a simple manner.
  A wide range of mathematical functions is available with HAL/S (see HAL/S Language Spec EQ - ER - 0018).
- Experiment software may be written in CII MITRA 125 S/MS assembler language.

- Software for on-board and ground check-out, as well as experiment dedicated sequencing software should be written in GOAL. This language is oriented towards the convenient specification of checkout procedures by scientists and engineers.

4.5.3 Software Development Guidelines

Experiment software development guidelines and standards will be provided at a later date.

There will be two main topics: One covers the technical management aspects such as verification (reviews and acceptance) and configuration control. The other specifies the necessary guidelines 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.

Additional guidelines, e.g. safety requirements, constraints on memory size and CPU load and mass memory requirements, will be included.

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4.6 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
- Trace contaminant removal (CO_2 , certain trace contaminants)
- Air and liquid cooling for module-mounted equipment
- Liquid cooling for pallet-mounted equipment
- Passive thermal control
- Airlock repressurization
- Experiment venting
- Fire detection and suppression

Table 4.6-1 lists the basic and the mission dependent ECS equipment and provisions.

Table 4.6-1: ECS Provisions and Equipment for Experiments

Basic Spacelab	Mission dependent
Atmosphere control inside the module	Exp cold plate - Pallet (8) Exp cold plate - Module (1)
Trace contaminant removal inside module	Thermal capacitors (4)
Air cooling for experiments	
Airlock repressurization assembly	Experiment dedicated heat exchanger inside the module
Experiment vent assembly	Thermal blankets

The Spacelab ECS is designed to provide a shirt-sleeve 1 atmosphere environment for up to 4 crewmen, 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 smoke detection, fire extinguishing and protection for the module against overpressure and negative differential pressure. Figure 4.6 - 1 presents an overall ECS schematic, showing the major ECLS and TCS components and interfaces.

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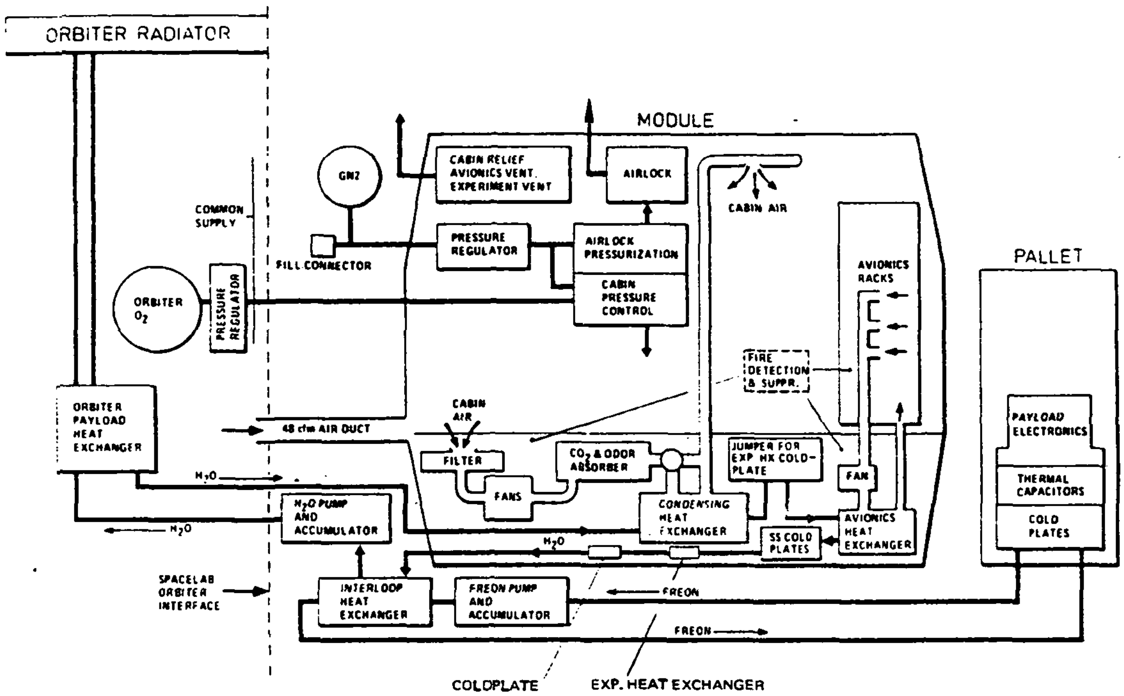


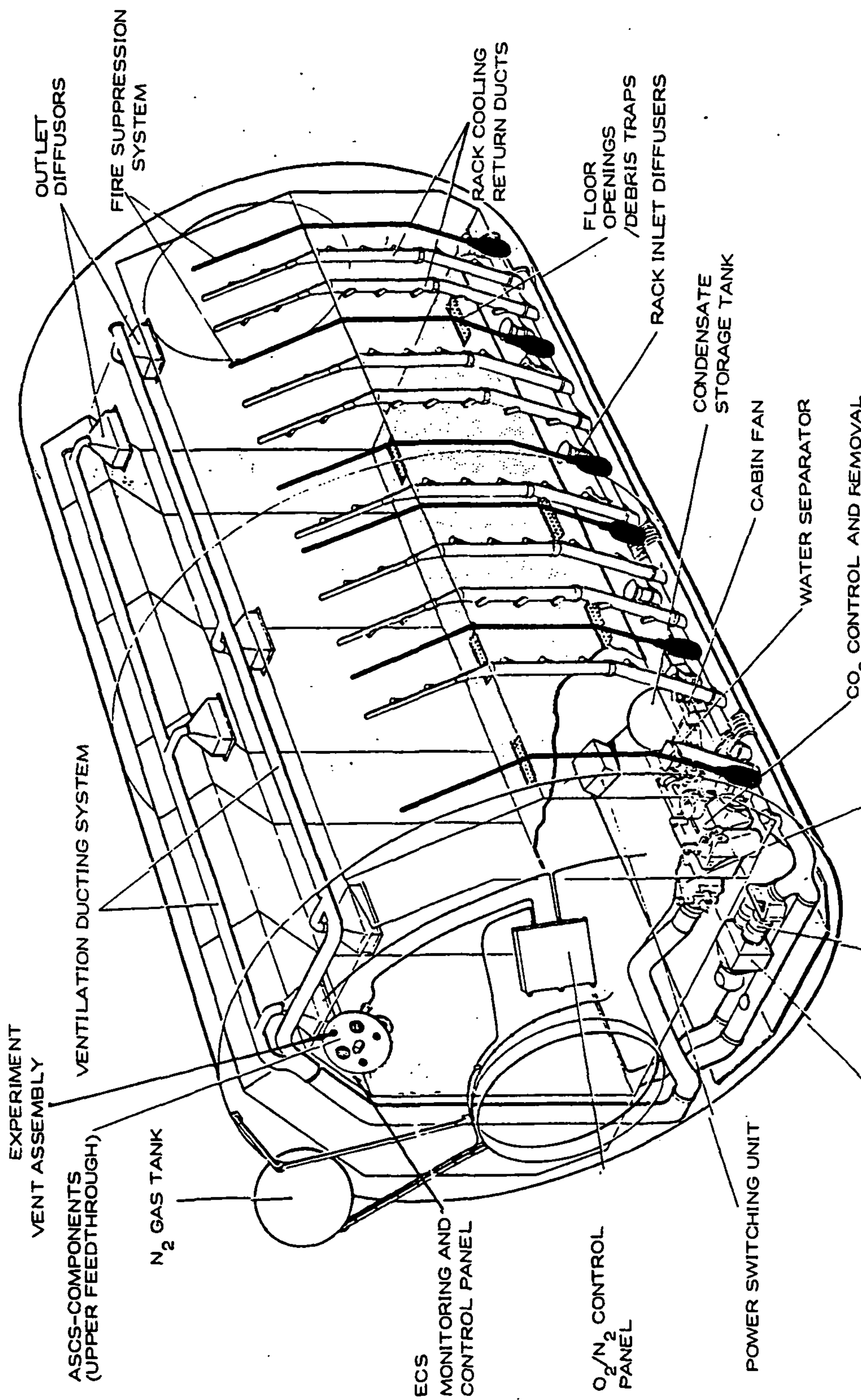
Figure 4.6 – 1 : ECS Schematic

4.6.1 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. However, the ECLS design does not preclude provisions for extended missions (see para 3.7).

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 and is described under para. 4.6.3.

A general overview of the ECLS subsystem is given in Figure 4.6 – 2, showing the arrangement of major ECLS components and the air ducting system inside the module.



Note: In order to simplify the diagram not all of the available air ducts are shown.

Figure 4.6-2: Environmental Control Life Support Subsystem

Table 4.6 - 2 lists some basic parameters characterizing the ECLS capability.

More detailed information on the environment provided by the ECLS is given in para 5.2.1.1 and 5.2.1.2. The Spacelab ECLS also provides the protection of the pressurized module for the emergency cases of overpressure and negative differential pressure.

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Table 4.6 - 2: Major ECLS Design Data

- ECLS Design Characteristics

Mean Radiant Temperature	max 30° C
Max. Touch Temperature	max 45° C
Air Atmosphere Leakage	1.35 kg/day
CO_2 Control	Nom. 0.0067 bar or less Max. 0.01 bar
Air filtration	280 micron Filter nominal, 300 micron absolute
Airlock Repressurization	1.02 m^3, 7 times total for a 7 day mission

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- Standard Metabolic Parameters for ECLS: nominal 0 - 3 men, max. 52 man hours/day

Metabolic Heat Loads	Max. 176 Watt / man Nom. 164 " Min. 118 "
CO_2 Removal Heat Load (sensible plus latent)	Nom. 35 Watt/man
Metabolic Oxygen Consumption	Max. 1000 g / man day Nom. 840 " Min. 770 "
CO_2 Generation Rates	Max. 1180 g / man day Nom. 990 " Min. 910 "

4.6.1.1 Atmosphere Storage and Control Section (ASCS)

The ASCS consists of one N_2 high pressure (228 bar) storage tank containing 22.4 kg at 146 bar of nitrogen, a high pressure regulator and an atmosphere pressure controller. O_2 is supplied via a line connected with the Orbiter O_2 supply system.

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.

4.6.1.2 Atmosphere Revitalization Section (ARS)

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. necessary for crew comfort. Experiment heat may also be rejected in the cabin loop as described in para. 4.6.3.

In the cabin loop (Figure 4.6 – 6), air is drawn through main floor openings, which are closed with debris traps, into a filter assembly (280 micron filter) upstream of redundant cabin fans. Check valves prevent recirculation through the inactive fan. Downstream of the cabin fans lithium hydroxide (LiOH) canisters provide carbon dioxide control. They also contain activated charcoal to remove certain trace contaminants in the cabin. Two canisters are provided in the air loop and operated simultaneously. The airflow through each canister is 51 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; the dried air is then returned to the cabin. The capacity of the storage water tank is sufficient for nominal 7 day missions. Overboard dumping is possible for contingency cases or after mission termination.

4.6.2 Thermal Control - General Description

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 heat rejection capabilities are given in Section 3.6.4.

DCN 036 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 and a nominal load of 7.4 kW).

The total heat transfer capability is allocated on the Spacelab side to various cooling loops and heat exchangers 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).

DCN 036 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 from excessive temperature variations and to avoid excessive heat leaks to and from Spacelab.

Schematic ATCS diagrams are shown in Figures 4.6 - 3. 4.6 - 4 and 4.6 - 5 for module-pallet, module-only and pallet-only configurations.

A brief description of the various cooling loops and heat exchangers follows:

a) Module-Only Configurations (Figure 4.6 - 3)

The heat generated by Spacelab and its payload inside the module is collected by a water cooling loop. The water loop contains several heat exchangers, connecting secondary cooling loops to the water loop, and cold plates in series as listed below:

- Cabin Air Loop/Condensing Heat Exchanger

  Condensing heat exchanger used in the cabin air loop for metabolic heat loads, some Spacelab subsystem loads and for experiment heat rejected in the cabin air (see para. 4.6.3.1).

- Avionics Air Loop/Avionics Heat Exchanger

  Avionics heat exchanger used in the avionic air loop which cools rack mounted experiments and some subsystem equipment (see para. 4.6.3.2).

- Spacelab Subsystem Cold Plates

  Spacelab subsystem cold plates for thermal control of Spacelab basic and mission dependent subsystem equipment such as computers, I/O units, mass memory, multiplexer and 400 Hz inverters.

- Experiment Heat Exchanger and Experiment Cold Plate

  Experiment dedicated heat exchanger and cold plate (both mission dependent) to provide liquid cooling and cold plate capability to experiments (see para. 4.6.3.3 and 4.6.3.4).

To accommodate peak heat loads a thermal capacitor assembly is provided and is installed on the outside of the forward end cone.

The water loop includes the water pump package (also located at the forward end cone) and the heat is finally transferred to the Orbiter heat exchanger.

b) Module Pallet Configurations (Figure 4.6 – 4)

The water loop inside the module is the same as in the module only configuration. However, an interloop heat exchanger is added to the water loop to collect heat from the freon loop (Freon 21) on the pallet. The freon pump package and the interloop heat exchanger are located at the forward end cone. The freon loop collects heat from experiments and subsystem equipment located on the pallet by means of experiment standard cold plates.

It should be noticed that the first experiment standard cold plate on each pallet segment is needed for subsystem equipment. DCN 036

Up to four thermal capacitors may also be mounted on pallet standard experiment cold plates, to accommodate peak heat loads.

The mission dependent experiment heat exchanger (shown in Figure 4.6 –4) may also, in principle, be connected in the freon loop although no standard location or mechanical provisions (such as extra freon lines and connectors) are presently provided by Spacelab.

c) Pallet-Only Configurations (Figure 4.6 – 5)

The pallet freon loop interfaces directly with the Orbiter heat exchanger and has a lower mass flow rate than in the module + pallet configuration (since freon rather than water interfaces with the Orbiter heat exchanger). Experiment standard cold plates are available for experiment and subsystem cooling on the pallet as in module + pallet configurations. In addition, other cold plates are located upstream of the experiment standard cold plates for cooling of subsystem equipment mounted in the Igloo and directly on the front frame of the first pallet segment. DCN 036

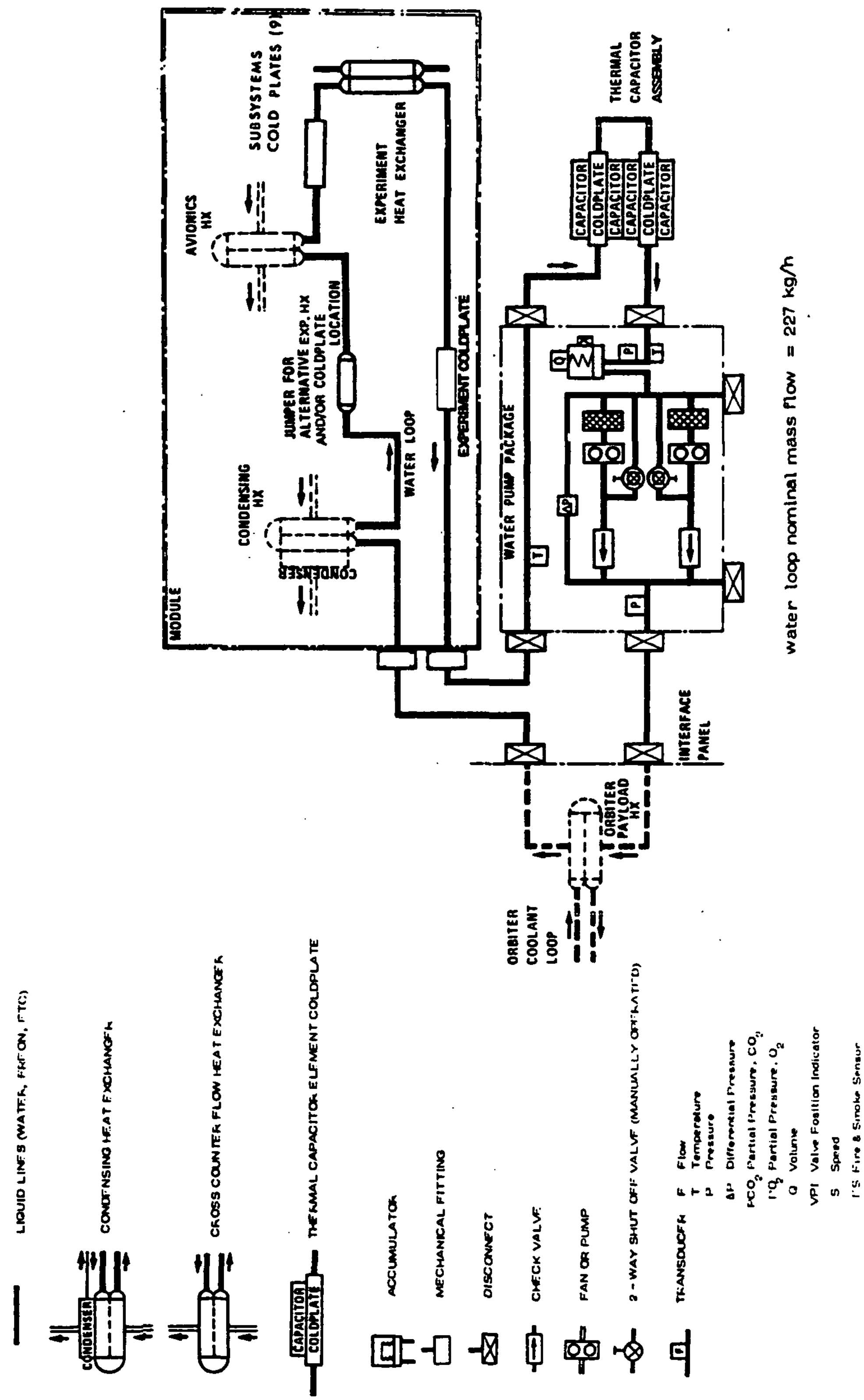
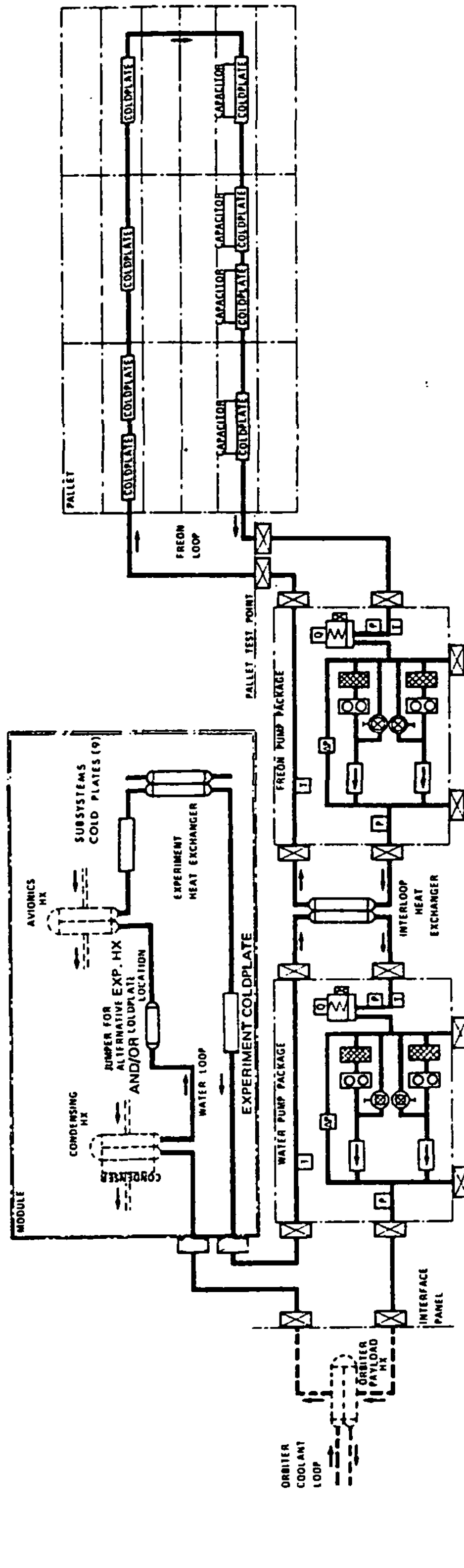


Figure 4.6 - 3 TCS Schematic Module-Only Configurations



Water loop nominal mass flow = 227 kg/h
Freon loop nominal mass flow = 1368 kg/h

Figure 4.6 - 4 : TCS Schematic Module + Pallet Configurations

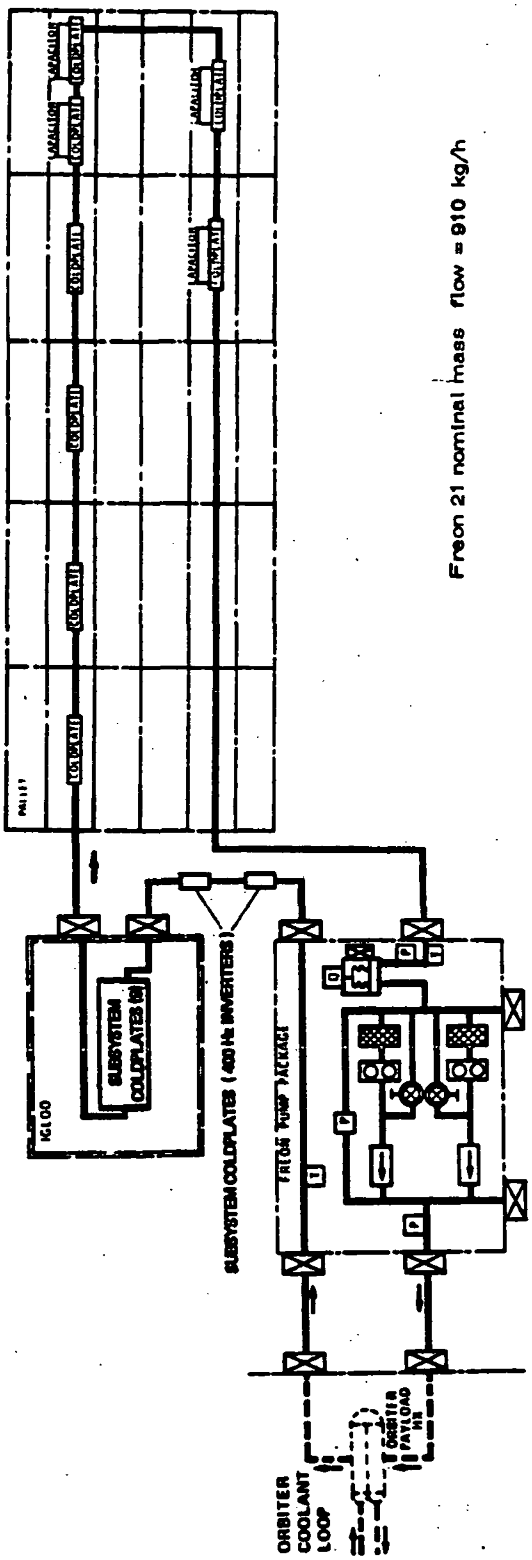


Figure 4.6 - 5 : TCS Schematic Pallet-only Configurations

4.6.3 Experiment Thermal Control

Experiment equipment may reject heat into Spacelab by the following means:

Module-Only Configurations

- Cabin air loop as described in para. 4.6.3.1.
- Avionic air loop as described in para. 4.6.3.2.
- Experiment heat exchanger as described in para. 4.6.3.3.
- Experiment dedicated cold plate in the module as described in para. 4.6.3.4.

Pallet-Only Configurations

- Cold plates in the freon loop as described in para. 4.6.3.5.
- Experiment heat exchanger mounted in the freon loop as described in para. 4.6.3.3.

Module + Pallet Configurations

The heat rejection capabilities of the module only and of the pallet only configurations are both available.

In addition Spacelab payload equipment located in the Orbiter Aft Flight Deck may also reject heat as described in para. 4.6.3.7 by air cooling. Passive thermal control capabilities available to experiments are described under para. 4.6.3.6.

4.6.3.1 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 air temperature can be adjusted within the range 18° to 27° C, and will be controlled within $\pm$ 1 $^\circ$C of the set point at full heat load. The cabin loop components and ducting are shown in Figure 4.6 - 2 and are described in Section 4.6.1. Figure 4.6-6 shows a simplified scheme of the cabin air loop. The nominal capacity of the cabin loop is 2.782 kW.

The heat loads in the cabin loop are as follows:

a) Total metabolic heat rejection including heat generated by the CO_2 removal:

199 W heat per man - nominal

b) Heat leaks to/from the space environment depend on orbits and Orbiter attitudes. These heat leaks are assumed to be zero in the nominal case of para 3.6.4.

Figure 4.6 - 7 shows the definition of reference cold and hot cases and the related estimation of module heat leakage. Such worst case estimated heat leakages may give some guidance for the assessment of the heat leakages in real cases.

c) Heat leaks from the avionic loop through the rack front panels; the amount depends largely on experiment configurations and has to be evaluated case by case. In the nominal case of para 3.6.4, the amounts from rack mounted subsystem equipment are 111 W for Short Module and 168 W for Long Module.

d) Heat rejected in the cabin loop by subsystem equipment, mostly on the subfloor.

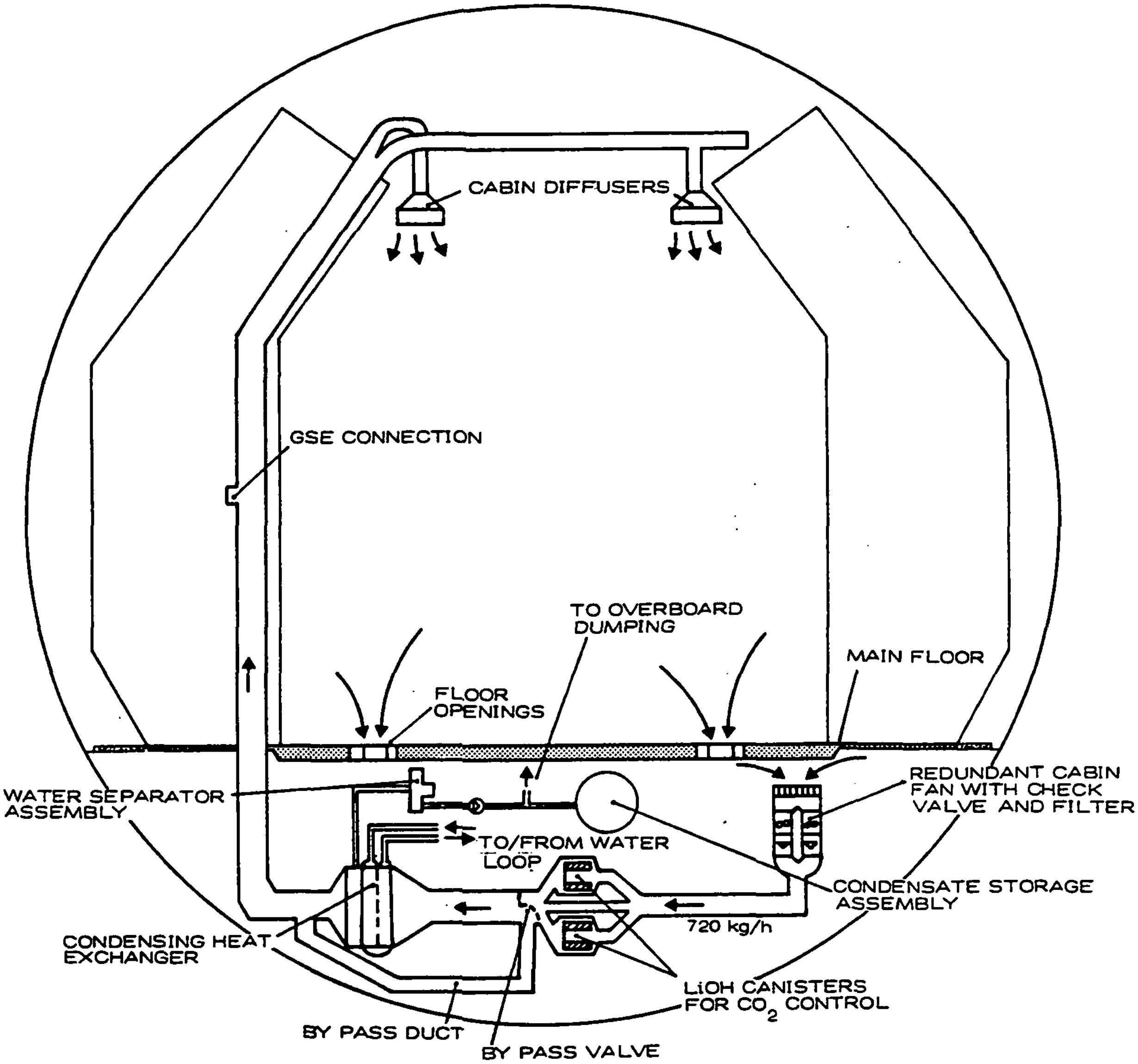
e) Heat rejection available to experiments

The cabin air loop can be used to cool experiment equipment which cannot be readily cooled using the avionics air loop - in particular, equipment mounted in the following locations:

- module center aisle
- high quality window / viewport
- airlock (during operations with the inner hatch open)

The airflow velocity in the cabin is determined by crew comfort requirements (5 - 12 m/min) and, for experiments that require increased airflow for cooling, the experimenter may have to provide local fans. For equipment mounted in the centre aisle, there are cut-outs in the floor so that hot air from the equipment can be ducted into the subfloor area to avoid undesirable airflow disturbances in the habitable area.

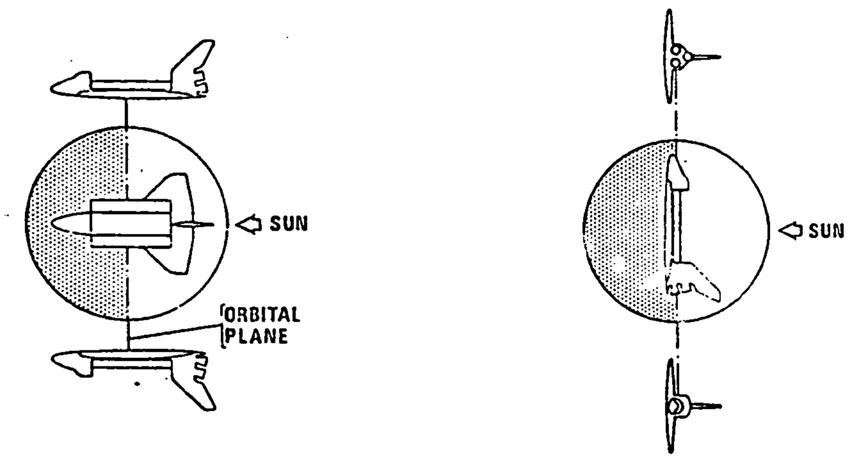
The nominal heat rejection capability available to experiments from the cabin air loop is given in Table 3 - 13. The actual heat rejection available on any particular mission is significantly dependent on the mission characteristics (orbiter attitude, instantaneous distribution of heatloads throughout Spacelab etc.) and can only be precisely established following a preliminary mission /payload definition. Approximately 300 W of the experiment allocated heat rejection capability can be used to take out latent heat loads (e.g. water vapour resulting from animal respiration and perspiration).



Note: Noise reduction provisions are not shown.

Figure 4.6 - 6: Cabin Air Loop Simplified Scheme

Reference COLD CASE | Reference HOT CASE



Heat leak to/from Orbiter tunnel is included

Heat leakage out of: Short Module ≃ 900 W; Long Module ≃ 1200 W

Heat leakage into: Short Module ≃ 840 W; Long Module ≃ 1250 W

Figure 4.6 - 7: Reference Cold and Hot Cases

4.6.3.2 Avionics Air Loop

4.6.3.2.1 Avionics Air Loop - General Description

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

The avionics loop components and ducting system are depicted in Figure 4.6 - 2. A simplified scheme of the loop is also shown in Figure 4.6 - 8. The avionics fan assembly establishes the airflow through the ducting system and the racks.

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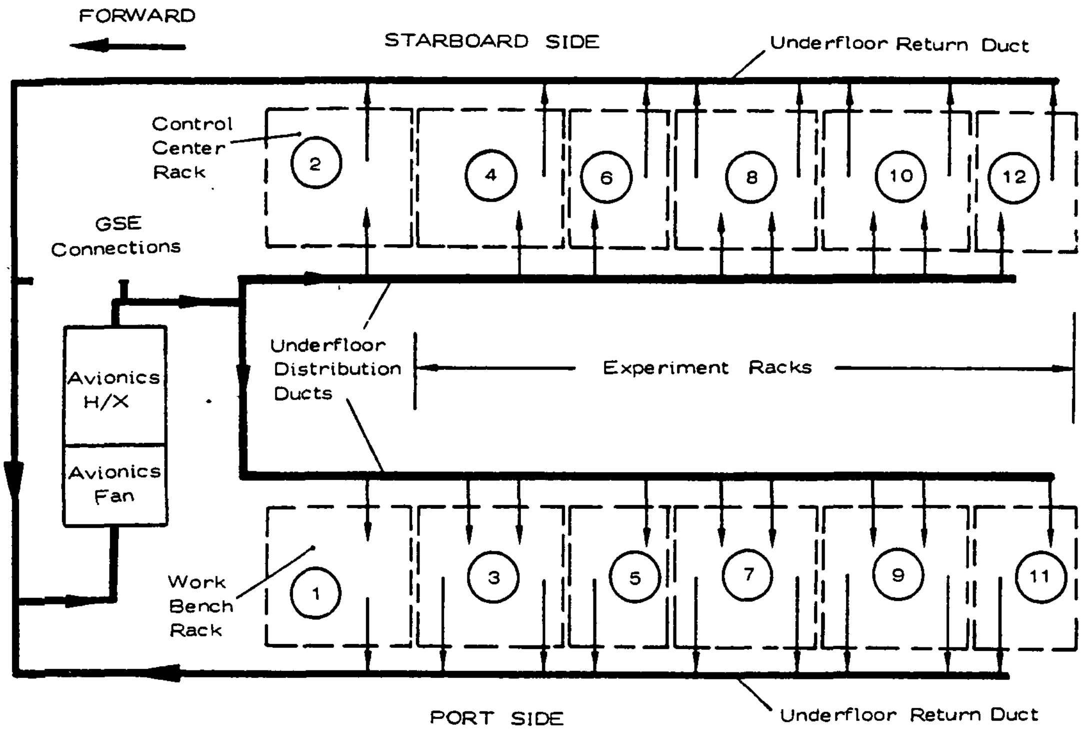
For ascent and descent as well as low-power modes in orbit, the avionics fan is designed for switching from 4 pole to 8 pole operation. The airflow will thus be reduced to approximately 50 % at a fan power reduction of approximately 20 % of the 4 pole operation.

The fan assembly consists of two redundant fans and contains filters (280 micron) for particulate removal. Downstream of the fan assembly the airflow 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 diffusers at the bottom of the rack and, after cooling rack equipment, it is sucked through the return ducts inside the racks into the return ducts under the main floor, and back to the avionics fan.



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Shut off valves at the bases of racks are not shown

Figure 4.6 - 8: Schematic of the Avionics Loop

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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 and one supply diffuser in each single rack and in each section of an experiment double rack.

Cooling of rack mounted equipment is possible in two ways as shown schematically in Figure 4.6 - 9: (1) surface cooling for open equipment, and (2) ducted cooling for enclosed equipment.

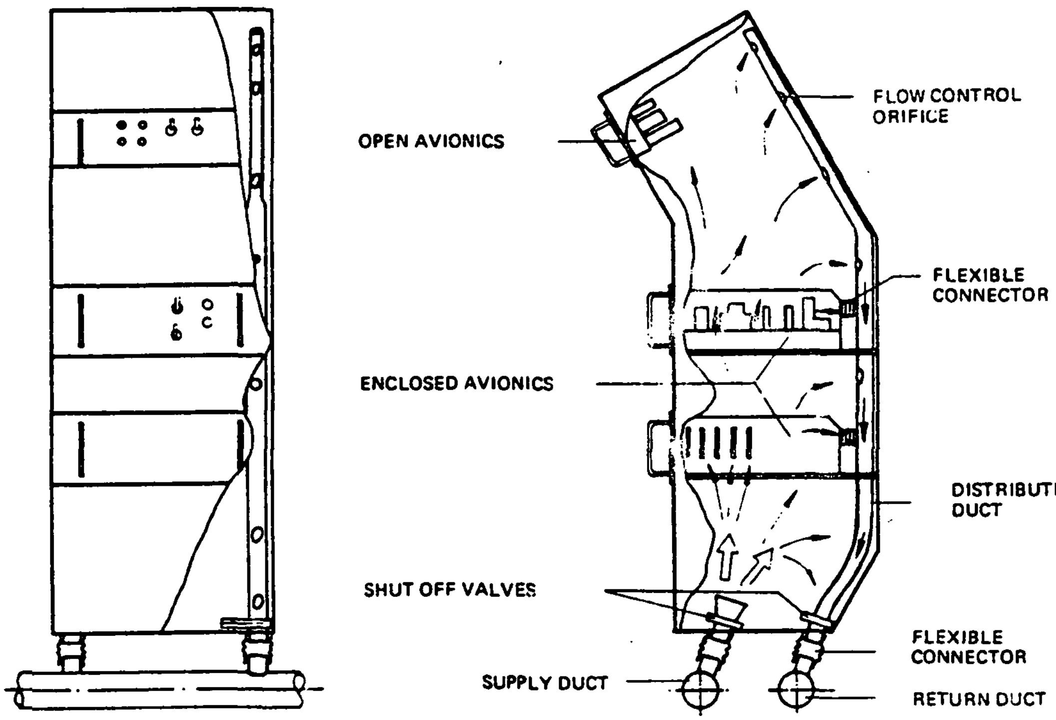


Figure 4.6 - 9 : Rack Equipment Cooling Concept

The normal mode of experiment cooling is ducted cooling. Air enters the experiment equipment through cut-outs 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.

Details of the mechanical interface with air cooling ducts inside racks are given in para. 4.1.1.3.1 and Figure 4.1 - 9

The airflow distribution may be adjusted to the specific payload needs by means of:

- Adjustable butterfly valves installed at each inlet of the rack return duct.
- Rack shut-off valves located at the bottom of racks, in the supply and return ducts as shown in Figure 4.6 - 9.

It is noted that the butterfly valves at the inlets of the return ducts may be adjusted or capped off only during integration on the ground. During ground operation the setting of the open status of each shut-off valve may also be adjusted to a preset position corresponding to the cooling requirements of the equipment in the rack.

During orbital operations the shut-off valves can be manually operated and placed in either the off position or the preset on condition, depending on the operating requirements of the experiment equipment in the racks. In this way, and according to a pre-arranged operating timeline, the avionic air can be diverted to the particular heat loads existing at any time during the mission. Certain other constraints, in addition to experiment requirements, may also affect the operation of these valves e.g. the need to maintain the overall avionic loop airflow and pressure drops within their normal operating range, and for fire detection purposes.

The return air ducts inside the experiment racks can be removed and, in this case, experiment equipment inside the racks may interface directly (at floor level) with the return air duct which runs under the main floor. Experiments not using the experiment racks may similarly interface (at floor level) directly with the supply and return air ducts under the floor.

4.6.3.2.2 Avionics Air Loop Cooling Capacity

The avionics loop nominal capacity is 4.51 kW of subsystem (including 650 W of fan power) and experiment heat transferred to the water loop. However, the actual heat that can be collected is constrained by the total maximum continuous heat transfer to the Orbiter, and by the actual heat load distribution in all Spacelab cooling loops as mentioned before.

The heat loads in the avionic loop are the following:

a) Heat rejected by subsystem equipment as shown in Section 3.6.4.

   It includes the heat from aircooled equipment in the control center rack, fan power, and some heat leakage from equipment cooled by subsystem cold plates in the work bench rack and in the control center rack; it also includes heat generated by subsystem equipment in the experiment racks (experiment RAU, intercom stations, EPSP).

b) Heat rejected by experiment equipment located inside the standard experiment racks or otherwise connected to the avionics loop. The nominal heat rejection available for experiments is shown in Section 3.6.4.

c) Heat taken in or out of the avionic loop, due to leakage through the rack front panels as discussed under 4.6.3.1.

4.6.3.2.3 Cooling Airflow and Pressure Drops

In addition to limitations due to heat rejected through the avionics heat exchanger further limitations are due to the airflow available in the loop.

The avionics fan is sized for a nominal airflow of 872 kg/h and a pressure rise of 17 m bar.

The nominal airflow available for experiments is 625 kg/h.

The nominal pressure drop available for experiments is:

- 2.5 m bar through single experiment units (including return duct orifices) located within standard experiment racks.
- 3.85 m bar between floor level inlets and outlets (normally connected to the racks) for experiments not using the standard experiment racks.

The actual airflow/pressure drop available to experiments depends on the distribution of the airflow between different racks.

Favourable airflow distributions are those resulting in lower pressure drops through the distribution ducts; this allows an overall airflow slightly higher than nominal or higher pressure drops allocated to the experiments. The opposite is valid for unfavourable air distributions. As a result the overall airflow ranges between approximately 800 kg/h and 920 kg/h and the portion available to experiments between 550 kg/h and 670 kg/h. This corresponds to an experiment equivalent heat rejection capability (based on the standard ARINC airflow of 21.8 kg/h per 100 W) ranging between approximately 2.5 and 3.1 kW.

Table 4.6 – 3 shows some cases of airflow distribution based on the nominal 2.5 m bar pressure drops through the experiment units.

The cases shown are all relatively unfavourable because the experiment airflow is concentrated only in one or two racks and on the same side of the module. Favorable airflow distributions are those resulting in an equal allocation of the total airflow between the right and left side of the module and when the racks requiring airflow are near the forward end of the module.

Table 4.6 – 3: Examples of Avionics Airflow Distribution

		Total Air-flow	AIRFLOW DISTRIBUTION kg/h											
			PORTSIDE RACKS						STARBOARDSIDE RACKS					
			1 WBR	3	5	7	9	11	2 CCR	4	6	8	10	12
CASES	Design	872	17	–	–	–	425	200	230	–	–	–	–	–
	1	815	17	–	–	–	–	–	230	–	–	–	382	186
	2	797	17	–	–	–	–	–	230	–	–	–	550	–
	3	890	17	643	–	–	–	–	230	–	–	–	–	–

Note:
- Rack numbering is in accordance with Figure 4.6 – 8
- Values shown are estimated

Low-speed Mode of Avionics Fan

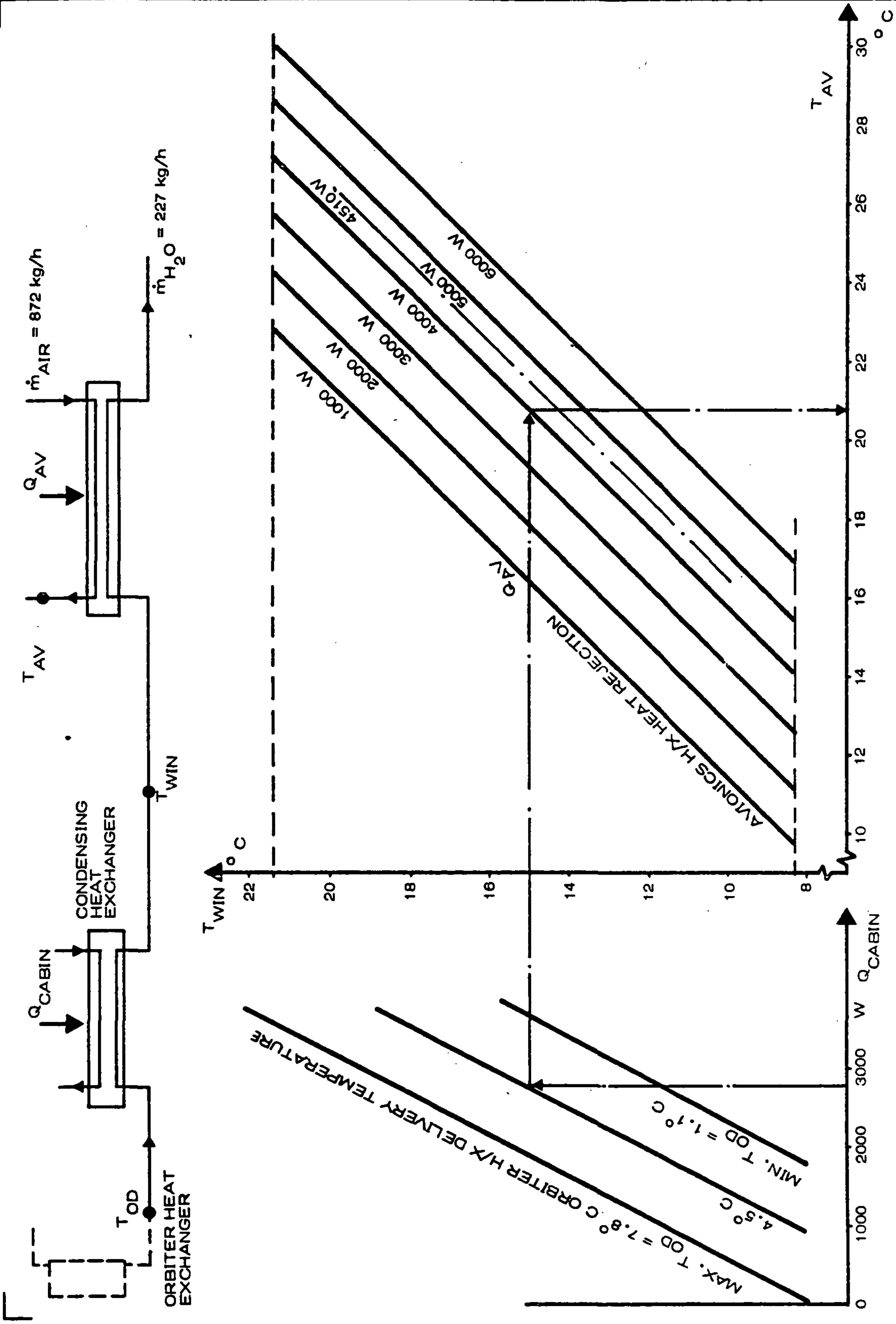
(N.B. The analyses and tests associated with this operating mode of the avionics fan are not completed and the resultant capabilities for experiments are preliminary estimates only)

When the 2-speed avionics fan is switched to the low-speed (power saving – para 3.6.3.2.1) mode the airflow to the experiment racks is substantially reduced. This mode of operation may be advantageous when:

- use is made of the experiment heat exchanger and/or cold plate in the module
- the main experiment power dissipation is on the pallet DCN 036
- experiment equipment is cooled by the cabin air loop

Since the pressure drop capabilities for experiments are reduced to typically 1.5 to 2.0 mbar the Arinc standard (based on 2.5 mbar) will not be maintained for experiment equipment with ducted cooling in the racks. With the use of dedicated low power mode experiment racks (with the other experiment racks switched off) a total airflow of approximately 175 kg/hr is then available for experiment equipment. Certain constraints, to be evaluated on a case-by-case basis, will be necessary on both the location of these dedicated racks and on the location of equipment inside them to ensure that the avionics fan operates in its correct pressure drop/flow rate regime.

For ascent/descent phases of the mission, the avionices fan can operate in the low-speed mode (to provide sufficient air flow for fire detection purposes). During these mission phases the partially activated Spacelab subsystem equipment consumes less cooling air than for the on-orbit low-speed mode and the total airflow available for experiment equipment in the racks will be approximately 200 kg/hr.



Notes: • Q_{AV} includes Subsystems and Experiment heat loads in racks and ≈ 650 W of fan power
• Experiment H/X upstream of the avionics H/X is not considered

Figure 4.6 - 10: Temperatures in the Avionics Loop

4.6.3.2.4 Avionics Air Loop Temperatures

The avionic air loop inlet temperatures are dependent on the heat load on the cabin loop, on the Orbiter heat exchanger delivery temperatures and on the heat load on the avionic loop itself.

Figure 4.6 - 10 gives T_{AV}, that is approximately the temperature at the inlet of the racks as a function of the Orbiter delivery temperature, the overall cabin loop heat load, and the overall heat load in the avionic loop itself. It is noted that Figure 4.6 - 10 is based on the nominal mass flow of the water loop; variations up to ± 5 % may occur.

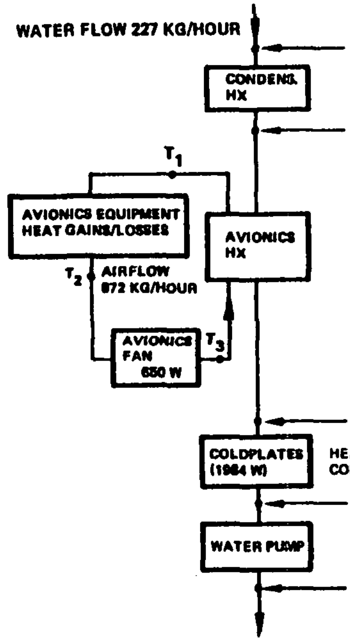
Figure 4.6 - 10 shows as an example that, with a nominal Orbiter delivery temperature of 4.5° C and with the nominal cabin heat load of 2.872 kW, a rack inlet temperature of approximately 21° C results for a heat load in the avionics loop of 4000 W.

It must be emphasized that this example relates only to one particular set of operating conditions. The actual operating conditions for any particular Spacelab mission can only be determined when the payload complement, orbital parameters, equipment operating timelines etc. have been established. Depending on all these input parameters the rack air inlet temperature will be somewhere in the range specified in Section 5, Table 5 - 16.

In addition, Table 4.6 - 4 illustrates some cases of load distribution. Shown are four different heat load distributions, all resulting in 8.5 kW total heat rejection to the Orbiter, for a module-only configuration and without using the experiment dedicated liquid cooling heat exchanger.

Four different heat loads are selected 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 at 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). The table is shown only as an example of various possible heat load conditions.

If the Orbiter water supply temperature is different from 7.3 °C (the actual temperature ranges from 1.1 °C to 7.8 °C), all the other temperatures quoted in Table 4.6 - 4 are shifted by approximately the same amount.



CASES	1.	2.	3.	4.
WATER SUPPLY TEMP.	7.3 °C			
CABIN LOOP	3262 W	2782 W	1782 W	782 W
WATER TEMPERATURE	19.7 °C	17.8 °C	14.0 °C	10.3 °C
AVIONICS EQUIPMENT HEAT GAINS/LOSSES *	2689 W 242 W 2931 W	3189 W 242 W 3431 W	4189 W 242 W 4431 W	5189 W 242 W 5431 W
AVIONICS FAN	650 W	650 W	650 W	650 W
AVIONICS HX	3581 W	408. W	5081 W	6081 W
AIRLOOP TEMPERATURE T_1	24.9 °C	23.7 °C	21.4 °C	19.0 °C
AIRLOOP TEMPERATURE T_2	36.9 °C	37.8 °C	39.8 °C	41.1 °C
AIRLOOP TEMPERATURE T_3	39.6 °C	40.6 °C	42.3 °C	44.1 °C
WATER TEMPERATURE	33.3 °C	33.3 °C	33.3 °C	33.3 °C
HEAT LOAD FROM COLDPLATES TO WATER LOOP	1574 W			
WATER TEMPERATURE	39.2 °C			
WATER PUMP	63 W			
WATER TEMPERATURE	39.4 °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

Table 4.6 – 4: Cooling Loop Heat Load Distribution Module-Only Configuration

4.6.3.3 Experiment Heat Exchanger

4.6.3.3.1 Experiment Heat Exchanger - General Description

Location in the Module

An experiment heat exchanger is provided for experiments that need liquid cooling. The heat exchanger is a mission dependent item which is connected to the module water loop as shown in Figures 4.6 - 3 and 4.6 - 4. The connection to the water loop may be made either downstream of the subsystem cold plates or just upstream of the avionic heat exchanger according to the requirements of a particular mission. These alternative connections are made only on the ground. The experiment heat exchanger is identical in design to the interloop heat exchanger which is used to transfer heat from the freon loop to the water loop in module + pallet configurations. Figure 4.6 - 11 shows the external features of the heat exchanger.

The experiment heat exchanger can be located only in the experiment double rack No. 4 which is adjacent to the control center rack. Figure 4.6-15 shows the location in the lower part of the rack. The heat exchanger may be used in addition to the mission dependent cold plate in the same rack or without it (see para 4.6.3.4 for more details).

Location on the Pallet

The design of the heat exchanger is such that it may, in principle, be connected into the pallet freon loop. Such a location, however, would require some extra freon plumbing, connectors, mounting attachments etc. which do not form part of the Spacelab baseline. The pressure drop capability of the freon loop would allow use of the heat exchanger in series with cold plates on the pallet. The performance characteristics of the heat exchanger when used with freon (or a similar fluid) on both sides are not provided as part of the Spacelab baseline.

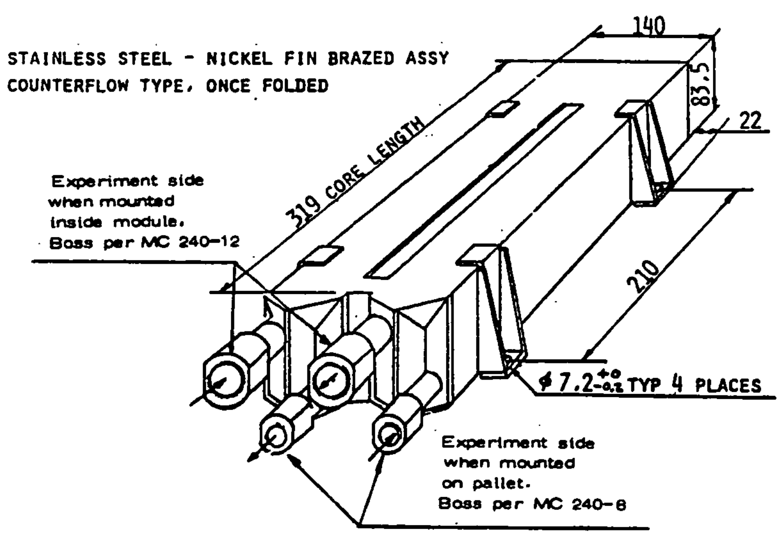


Figure 4.6 - 11 : Experiment Heat Exchanger

4.6.3.3.2 Experiment Heat Exchanger Cooling Capacity

The nominal capability of the heat exchanger is 4 kW. This value represents the performance of the heat exchanger itself and the actual heat that may be transferred from an experiment cooling loop will, of course, depend on the experiment heat load which is separately transferred to the cabin loop and the avionics loop. Further constraints on the actual heat transfer capability are:

- the total Spacelab heat transferred to the Orbiter (8.1 kW to 8.5 kW as described in 3.6.4).
- the maximum temperatures allowed in the module water loop downstream of the experiment heat exchanger. When the experiment heat exchanger is connected upstream of the avionics heat exchanger there is a considerable constraint due to the maximum temperatures allowable for subsystem equipment which is cooled by the avionics air loop and subsystem cold plates (The related limitation on the maximum water temperature at the inlet of the avionics heat exchanger is shown in Figure 4.6 - 12).

The actual operating temperatures and heat transfer capabilities of the experiment heat exchanger can be determined for any particular Spacelab mission when all the Spacelab and experiment heat loads are established for that mission. Meanwhile, an estimate of the expected heat transfer and temperature range at the inlet of the experiment heat exchanger (Spacelab side) may be obtained from Figure 4.6 - 12.

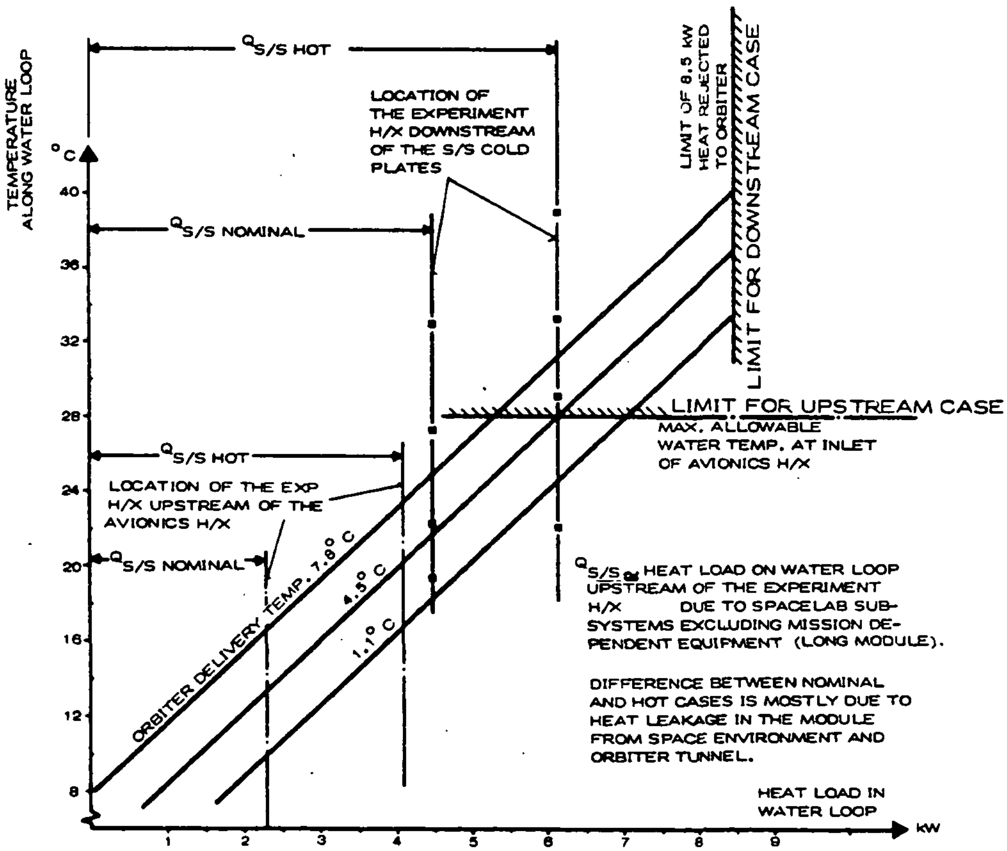


Figure 4.6 - 12 : Experiment Heat Exchanger Temperature Ranges

For the upstream position, the available experiment heat exchanger /coldplate heat load is determined by starting on the curve at a point defined by the heat loads upstream of the experiment heat exchanger / cold plate (cabin loop only) by the maximum allowable water loop temperature at the inlet of the avionics heat exchanger.

For the downstream position, the available experiment heat exchanger / cold plate heat load is the difference between the 8.5 kW limit and the heat loads upstream of the experiment heat exchanger /coldplate (cabin loop, avionics loop and subsystem cold plates). DCN 036

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4.6.3.3.3 Experiment Heat Exchanger Pressure Drop Characteristics

Pressure drop characteristics of the experiment side of the experiment heat exchanger are given in Figure 4.6 - 14

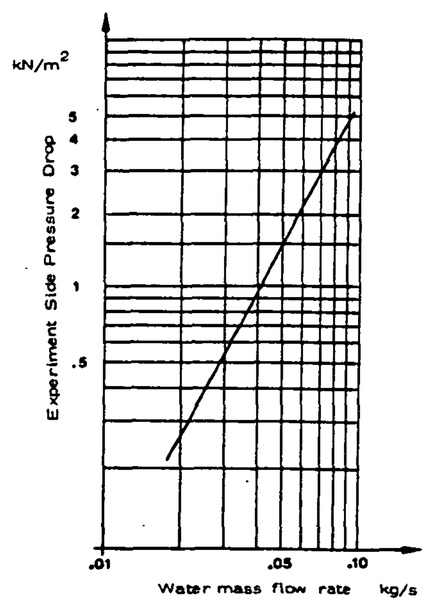

Figure 4.6 - 14 : Experiment Heat Exchanger Pressure Drop

4.6.3.3.4 Experiment Heat Exchanger Mechanical Interface with Payloads

Interfaces with the experiments are provided at the inlet and outlet connectors at the heat exchanger. Any plumbing, pumps, cooling liquid etc. which are necessary to form a cooling loop for experiments have to be provided by the user. The normal working liquid for this secondary loop is water, but any coolant liquid allowable in manned spacecraft (and which complies, in particular, with the requirements specified in Section 7) can be used.

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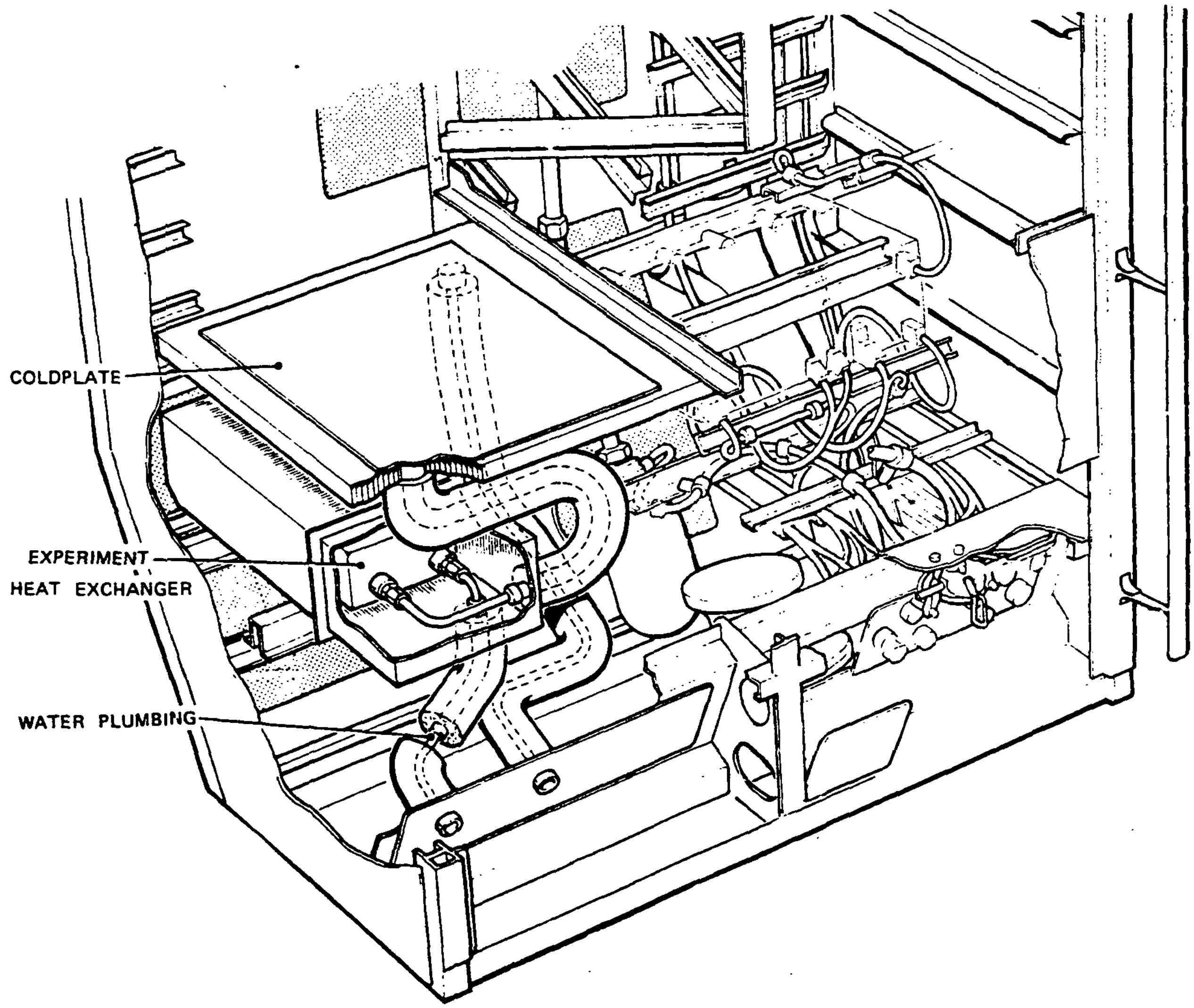

Figure 4.6 - 15: Experiment Heat Exchanger and Cold Plate Location

4.6.3.4 Experiment Cold Plate Located in the Module

One mission dependent cold plate is available in the module for experiment use and is connected to the module water loop as shown in Figures 4.6-3 and 4.6-4. The cold plate location in the lower part of the experiment rack is shown in Figure 4.6.-15 (note that location is possible only in the double experiment rack next to the control centre rack). The cold plate can be used simultaneously with the experiment heat exchanger (in which case the two items are connected in series) or either item can be flown separately (in which case there is a jumper connection to the module water loop). When both items are flown, however, the combination of cold plate plus heat exchanger will share the same connection to the module water loop, i.e. it is not possible to connect the cold plate upstream of the avionics H/X and the experiment heat exchanger downstream of the S/S cold plates or vice versa.

Some characteristics of the cold plate are given in Table 4.6 - 5. Experiments may be mounted to cold plates either with or without filler. When filler is not used, the contact area between the experiment and the cold plate can be assumed to be a maximum of 6.5 cm^2 around each attachment bolt. For increased heat transfer rates, the use of filler ensures heat transfer over the whole cold plate area under the experiment interface. DCN 035

Table 4.6 - 5: Experiment Cold Plate in the Module

Mechanical Characteristics	Cold Plate Dimensions	L x W x H 500 x 390 x 4.4
	Envelope Clearance on Basic Area	L x W x H 612 x 451 x 1100
	Hole Pattern on Cold Plate	70 x 70 mm
	Wet Weight	4.5 kg
	Number of Holes in Cold Plate	42
Thermal Characteristics	Heat conductance from the equipment heat transfer area to the cold plate water without filler	$\simeq$ 1 W/°C per bolt area
	Heat conductance from the equipment heat transfer area to the cold plate water with filler	0.07 $W cm^{-2}\ {}^{o}C^{-1}$

4.6.3.5 Experiment Heat Rejection via the Freon Loop - Experiment Standard Cold Plates and Thermal Capacitors

4.6.3.5.1 General Description

Experiment equipment located on pallet segments in module-pallet and pallet-only configurations and which need active thermal control can use Spacelab provided cold plates (experiment standard cold plates) and thermal capacitors. The freon loop is designed to accommodate nominally up to eight experiment standard cold plates (in addition to subsystem cold plates in the igloo in series).

The freon flow rates for these configurations are as follows:

DCN 036
- 1, 368 kg/h ± 10 % for the Module/Pallet configuration
- 912 kg/h ± 10 % for the Pallet-Only configuration

The cold plate (with or without a thermal capacitor) is mounted to an intermediate support structure which is then attached to the pallet structure (see Figure 4.6 - 18). This support structure consists of a honeycomb plate which is mounted to the pallet via fibre glass webs which provide thermal isolation between the plate and the pallet structure. The cold plate/thermal capacitor (and experiment equipment interfacing with it) is attached to the support plate with bolts which pass through the holes in the cold plate /thermal capacitor (and experiment) and into inserts in the support plate.

Baseline locations of the cold plates are the 48° sections of the pallet segments. It is possible either to mount all eight experiment standard cold plates on one pallet segment or to distribute them over several pallet segments.

Spacelab provides 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 two configurations are shown in Figure 4.6 - 16. Other configurations may be possible if additional (outside the present baseline) plumbing and/or cold plate attachments are provided and if the necessary analyses of pressure drops and volumes (accumulator) are performed. Alternative configurations with cold plates mounted directly onto experiments, for example, or with an experimenter provided heat exchanger mounted into the freon loop may be possible but would have to be evaluated case by case.

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One experiment standard cold plate is normally required to be located on each pallet segment to provide cooling for subsystem equipment (RAU, EPDB and interconnect stations). The other experiment standard cold plates are mission dependent.

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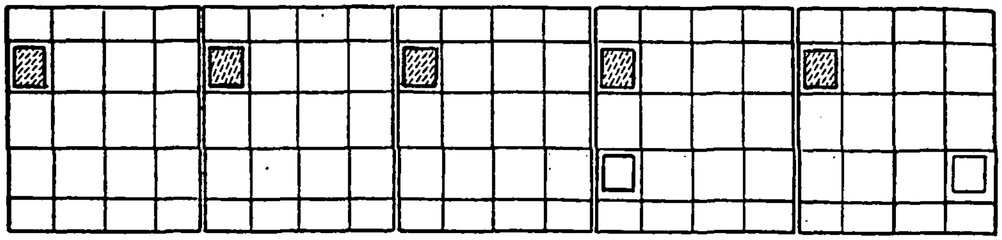
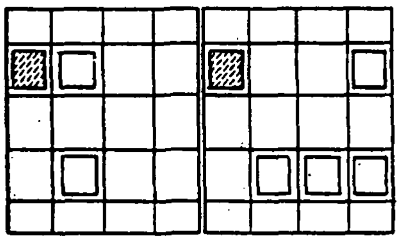
5 Pallets



Note: Shaded cold plates are required for subsystem equipment (EPDB, Subsystem RAU and Interconnecting Station). DCN 035

2 Pallets



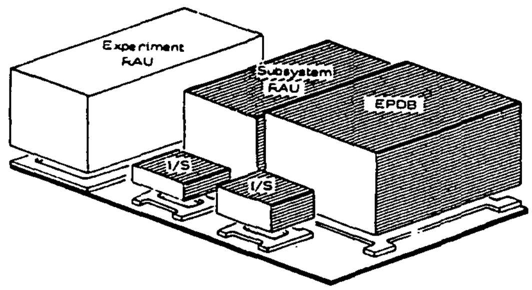


Figure 4.6 - 16: Baseline Configurations of Experiment Standard Cold Plates

4.6.3.5.2 Freon Loop Cooling Capacity

The actual heat rejection capability of the freon loop for experiments is dependent on the Orbiter heat rejection capability and on the heat loads present in the other Spacelab cooling loops. The heat loads on the freon loop itself comprise the following:

a) Heat loads due to the subsystem package (EPDB, S/S RAU, IS) on the first standard experiment cold plate of each pallet segment.

b) Heat leaks to/from the pallet structure and to/from the space environment.

Conduction heat leaks to the structure are low due to the thermally isolated mounting of the cold plates. Radiation heat exchange with the space environment may be reduced if necessary by the use of thermal blankets described under para. 4.6.3.6.

c) Heat loads due to experiments on experiment standard cold plates.

d) Heat load due to experiments using the experiment heat exchanger, if this is used on the pallet.

The above mentioned heat loads are relevant for both module plus pallet and pallet-only configurations. In the latter case, there are additional heat loads on the freon loop upstream of the experiment standard cold plates, due to subsystems on the front frame of the first pallet segment (mostly inside the Igloo) which have to be taken into account.

4.6.3.5.3 Temperature along the Freon Loop

An exact determination of the temperature along the freon loop is possible only after an analysis for each mission. Some information is presented below to give some guidance on typical temperatures to be expected for cold plate mounted experiments.

Module + Pallet Configurations

Figure 4.6 – 17 allows the determination of the steady-state temperature at the beginning of the freon loop for module + pallet configurations as a function of the Orbiter delivery temperature T_{OD}, the overall heat load Q_W on the water loop upstream of the interloop heat exchanger, and the heat load on the interloop heat exchanger itself Q_F. The example shown gives a freon temperature T_F of ~ 24° C for 4 kW of heat load on the water loop, T_{OD} = 4.5° C and Q_F = 3 kW.

The overall heat load Q_W may be determined in specific cases using the information given on heat loads in Section 3.

The fluid temperature increase along the freon loop is approximately 2.45° C/kW of rejected heat. The presence of some Spacelab subsystem equipment on the first cold plate of each pallet (≃ 30 W) also has to be considered.

Based on Figure 4.6 – 17 and on the above mentioned 2.45° C/kw, the max. temperature on the last cold plate, at the full overall Spacelab heat load of 8.5 kW may be up to ~41° C in steady-state condition.

Pallet-Only Configurations

Coolant temperatures at the standard experiment cold plates may be determined based on the Spacelab subsystem heat loads upstream of the first cold plate (see para. 4.6.4). The presence of some Spacelab subsystem equipment on the first cold plate of each pallet segment also has to be considered.

The temperature increase along the freon loop is approximately 3.64° C/kW.

It should be noted that the temperature information mentioned above is based on the nominal water and freon flow rate. A tolerance of ± 5 % is presently foreseen for the water loop flow rates; a similar tolerance is also expected for the freon loop.

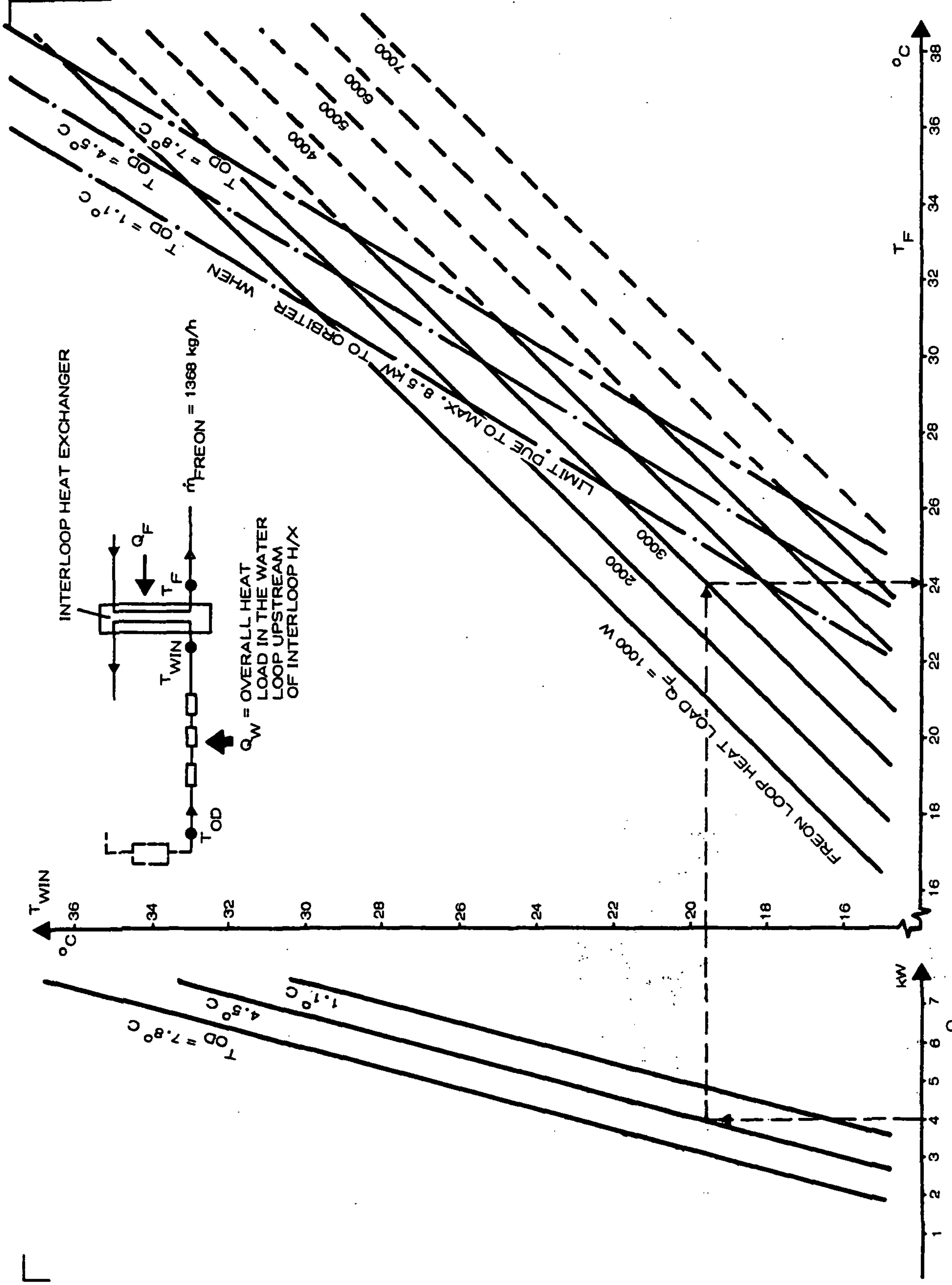


Figure 4.6 - 17: Temperatures at Beginning of the Freon Loop

4.6.3.5.4 Experiment Cold Plates on the Pallet - Thermal Capacitors

The cold plates are designed for a nominal heat rejection capability of 1 kW each. The thermal capacitors are designed for a nominal heat storage capacity of 0.25 kWh each. Cold plate and thermal capacitor physical dimensions are shown in Figures 4.6 - 19 and 4.6 - 20.

Table 4.6 - 6 gives the major thermal characteristics of cold plates and thermal capacitors. Experiments may be attached either with or without filler (see 4.6.3.4).

Table 4.6 - 6: Cold Plate and Capacitor Characteristics

	Without Filler	With Filler
• All experiment surfaces in contact with cold plate shall be in a plane having flatness within the bolt pattern of 70 x 70 mm.	0.05 mm	0.05 mm
• Overall flatness (max. deviation between hole pattern fields)	0.2 mm	0.2 mm
• The portion of the experiment surface that serves as heat transfer area shall be	6.25 cm^2/bolt	All
• Heat conductance from the experiment heat transfer area to cold plate coolant is	$0.15 \frac{W}{°C\ cm^{*2}}$	$0.08 \frac{W}{°C\ cm^2}$
• Heat conductance from the experiment heat transfer area to capacitor wax is	$0.15 \frac{W}{°C\ cm^{*2}}$	$0.08 \frac{W}{°C\ cm^2}$
• Melting range of capacitor wax	39 - 40.5 °C	

Note: Values shown are for 1369 kg/h of freon-flow. * bolting area

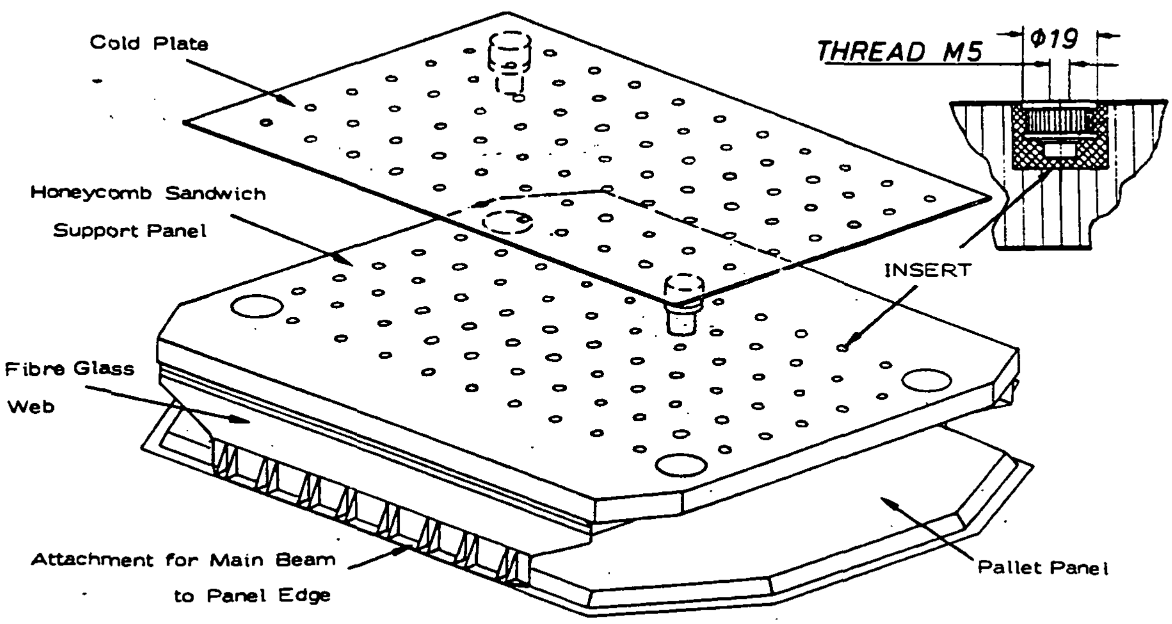


Figure 4.6-18: Experiment Cold Plate Support Structure

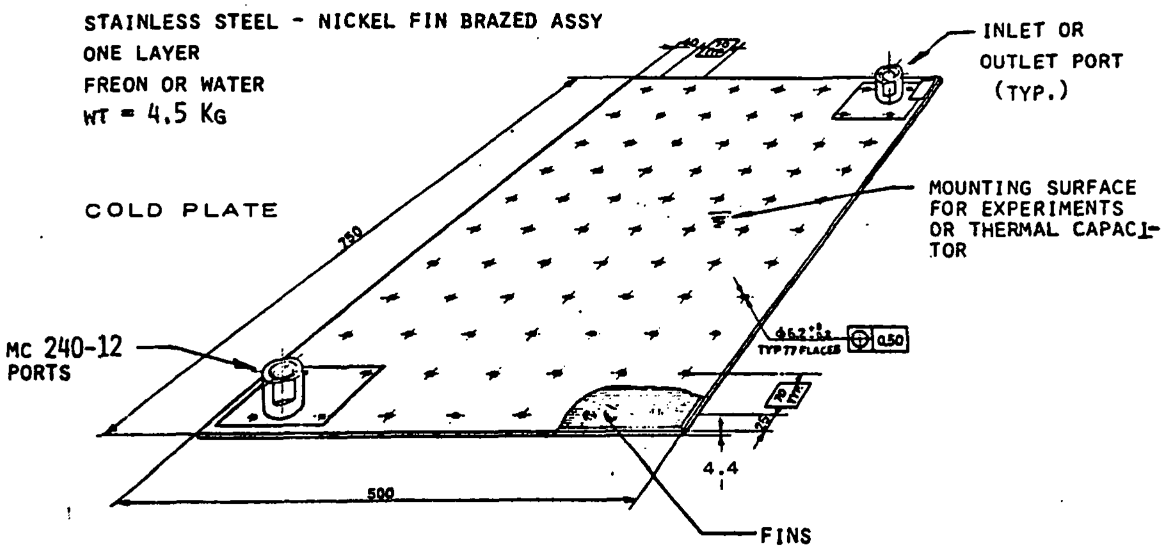


Figure 4.6 - 19: Experiment Cold Plate

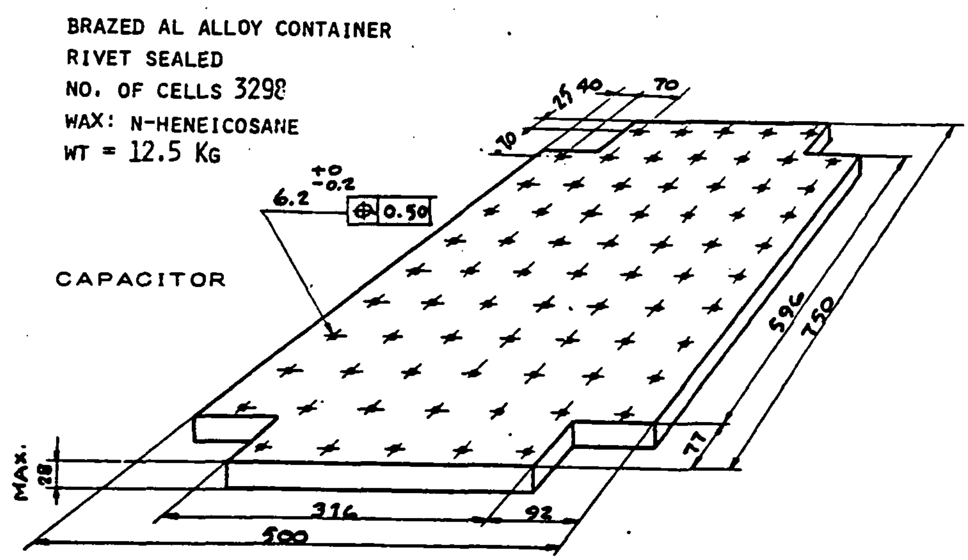


Figure 4.6 - 20: Thermal Capacitor.

4.6.3.6 Passive Thermal Control

The Spacelab passive thermal control employs insulations, surface coatings and thermal covers to protect the module, pallet segments, utility lines and other externally mounted subsystem equipment. Experiment passive thermal control (including the provision of thermal blankets if required) is primarily the user's responsibility.

DCN 036 The pallet structure might be used as heat sink for low heat rejection payloads directly attached to the pallet panels, subject to a case by case analysis.

4.6.3.7 Orbiter Aft Flight Deck Thermal Control

Experiment equipment located in the Orbiter aft flight deck is cooled by the Orbiter cabin air. This air is drawn through openings in the experiment enclosure and is ducted through experiment outlets to the Orbiter manifold via Orbiter provided flex hoses and flow restrictors. Non-ducted cooling is also possible for experiments with low dissipation. The air inlet temperature to experiments is in the range 18 to 27°C (except for peaks of 35°C) and the outlet temperature from the experiment equipment may go up to 54°C. The maximum pressure drop between experiment inlet and outlet is 2.5 mbar and the total airflow available for experiments is approximately 120 kg/hr. Section 3.6.4 gives the total heat rejection capability for various conditions.

4.6.4 Subsystem Heat Rejection in the Various Loops

The subsystem heat rejection into the various loops may be evaluated from the complementary information of para 3.6.4.3 and Tables 3-13, 3-14, 3-4.

4.6.5 ECS Capabilities During Ground Operations

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 (for details on ground operational phases see Figure 3-27). 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. A 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).

4.6.6 Experiment Vent Assembly

4.6.6.1 General Description

The experiment vent assembly allows the venting of gases from experiment vacuum, processing etc. chambers located in the module. The vent assembly is located in the upper part of the top feed through plate in the forward end cone (see Figure 3-16).

Figure 4.6-21 shows the configuration of the vent assembly . A manually operated butterfly valve provides flow restriction with a maximum orifice area of 11.4 cm^2. This valve can also be placed in two intermediate detent lock positions between closed and fully open. The experiment vent assembly is additionally used for reducing the module air total pressure under certain abnormal over-pressure conditions. For this purpose a quick disconnect cap (female) incorporating a cabin bleed valve is normally fitted to the cabin side of the assembly. When experiment venting is required this cap is replaced by a similar quick disconnect (not provided by Spacelab) which is attached to the user provided vent line from the experiment chamber.

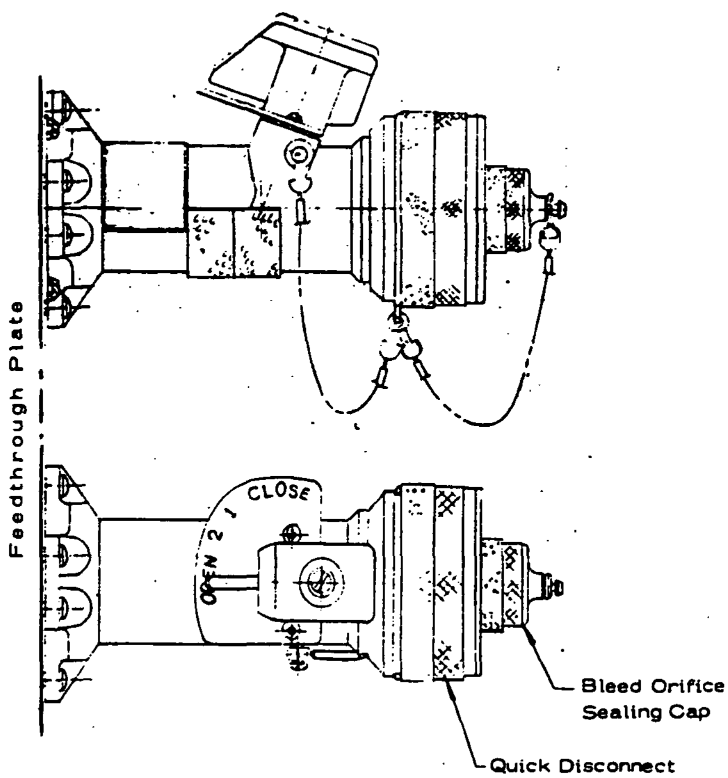


Figure 4.6 - 21: Experiment Vent Assembly

4.6.6.2 Vent Assembly Characteristics

– Pressure	– operating	: 0 to 1.1 bar
	– proof	: 1.65 bar
	– burst	: 2.20 bar
– Max flow rate at $21^{\circ}C$ and ΔP = 1.013bar	– valve fully open	: 0.18 kg/sec
	– valve position 1	: 0.12 kg/sec
	– valve position 2	: 0.06 kg/sec
– Leakage	– internal	: 3 sccm at ΔP = 1.034 bar
	– external	: 0.07 sccm at ΔP = 1.034 bar
– Cross sectional area	– maximum	: 11.4 cm^2
– Temperature	– operating limits	: –74 to + 71 $^{\circ}C$

4.6.6.3 Compatible Gases

The compatibility of various vented gases with the vent assembly is largely dependent on their compatibility with the two types of silicone rubber (Silastic 675 and Silastic LS-53) used in the assembly seals. The design is known to be compatible with oxygen, nitrogen, air and carbon dioxide and is probably compatible with methane, ethane, carbon monoxide, hydrogen and helium (with increased leakage). Freon is not considered to be compatible. It may be necessary to test each anticipated gas with the silicones at actual expected temperatures, pressures and durations to confirm compatibility.

4.6.6.4 Venting Rates

The vent flow rates defined in para 4.6.6.2 are applicable only to the vent assembly alone. The actual flow rates when the vent is connected to experiment equipment will depend on the characteristics of the total system of vent, vent line, chamber, valves, etc and an analysis will be necessary for each system configuration.

4.7 Common Payload Support Equipment

The Common Payload Support Equipment (CPSE) consists totally of mission dependent equipment (Table 3-4). It is composed of the following items:

- one (1) airlock (1 m dia., 1 m length)
- one (1) high quality window/viewport assembly
- one (1) viewport assembly

The installation of this equipment into the module is at locations indicated in Figure in Figure 4.7 - 1. Flanged holes at the top of each segment of the module allow the installation of any of the CPSE items which may be required for a particular mission. The airlock, however, can not be used in the core segment because of constraints due to access to the module interior during ground operations (see para. 3.3.1.1). For missions where CPSE is not required cover plates are provided to close-out the holes in the module segments.

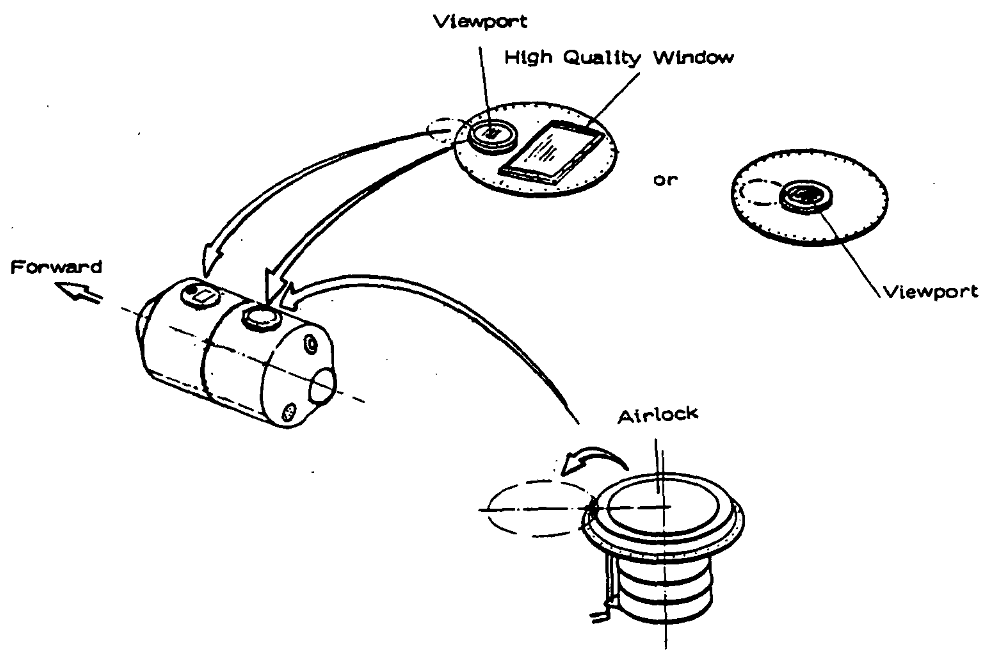


Figure 4.7 - 1 : Common Payload Support Equipment for Module

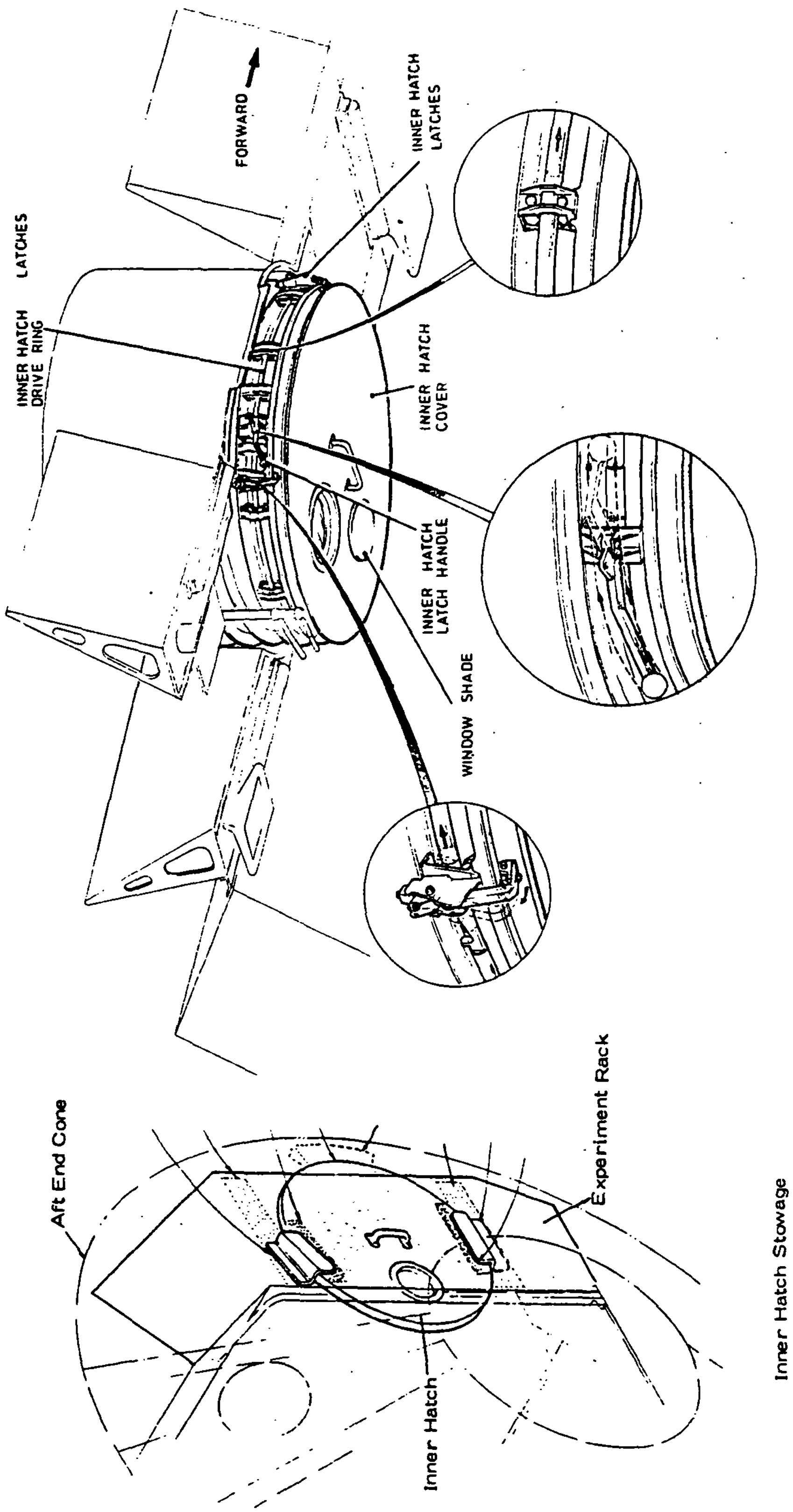


Figure 4.7 - 2 : Common Payload Support Equipment - Airlock

4.7.1 Airlock

4.7.1.1 Description

The airlock consists of a 1 m long by 1 m diameter cylindrical shell closed at both ends by circular honeycomb sandwich hatches which open away from the shell. The outer hatch is hinged at the edge of the shell while the inner hatch, which contains a viewport, is completely removable. Experiments are normally attached to a sliding table which can be extended into space or into the module. Figures 4.7 - 2 and 4.7 - 3 show the external configuration of the airlock located in the top of the long module. All airlock controls are manually operated and a control panel and a pressure gauge provide for monitoring the airlock operations. DCN 016

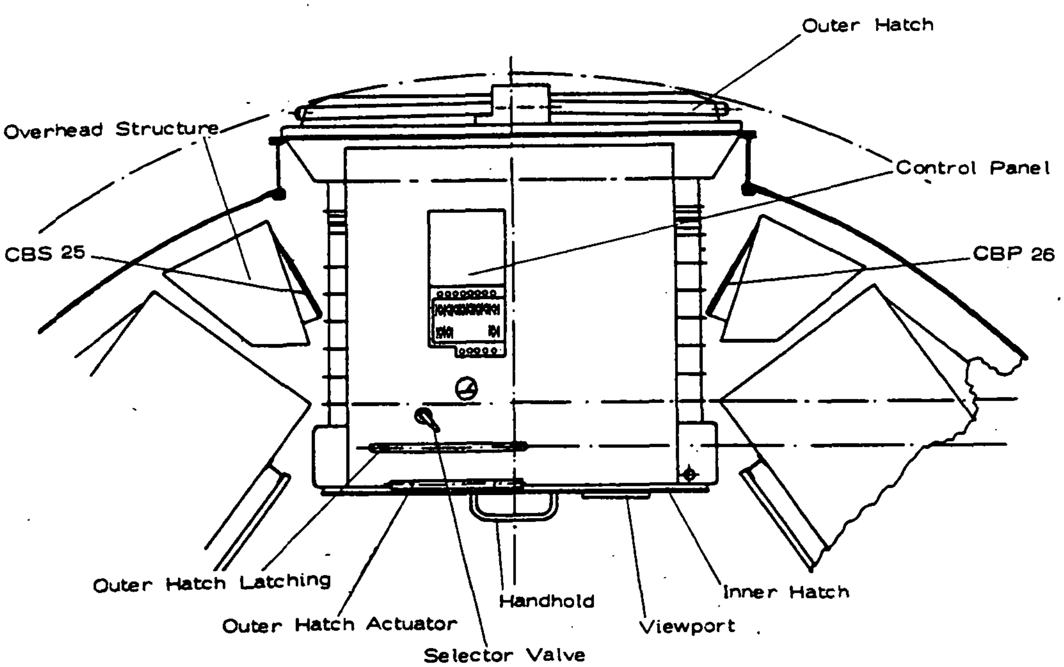


Figure 4.7 -3 : Airlock Inboard Profile - Looking from Aft

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.

DCN 016 The experiment table, when extended, and the outer hatch penetrate the Orbiter cargo bay dynamic 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 experiment table and the hatch by means of EVA.

4.7.1.2 Experiment Dynamic Envelope

DCN 016 The dynamic envelope inside the airlock available for experiments is shown in Figures 4.7 - 4 and 4.7 - 5. The airlock lamp and the experiment table protrude into this envelope but both items can be removed when a voluminous experiment is flown.

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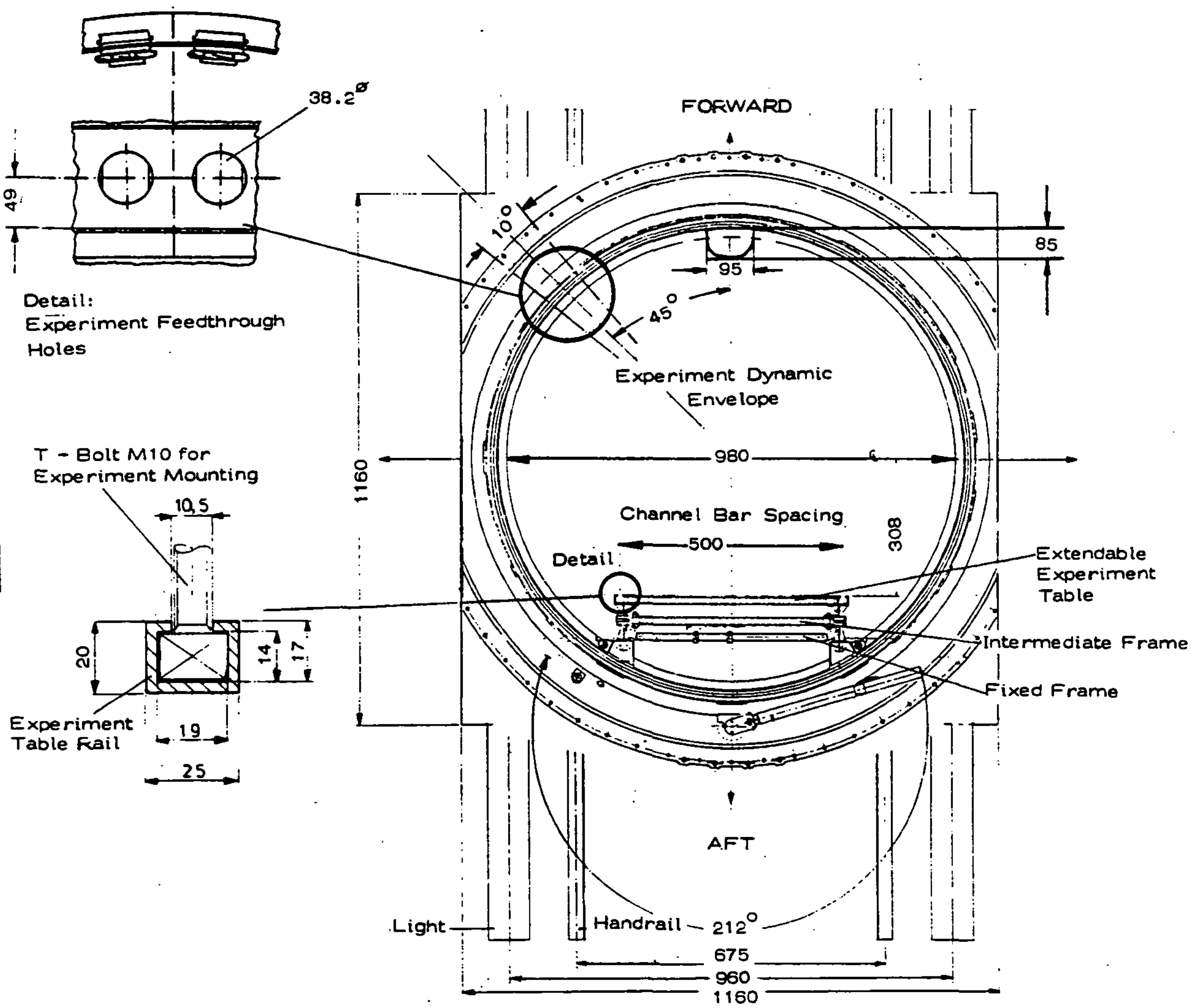

Figure 4.7 - 4: Airlock-View Up

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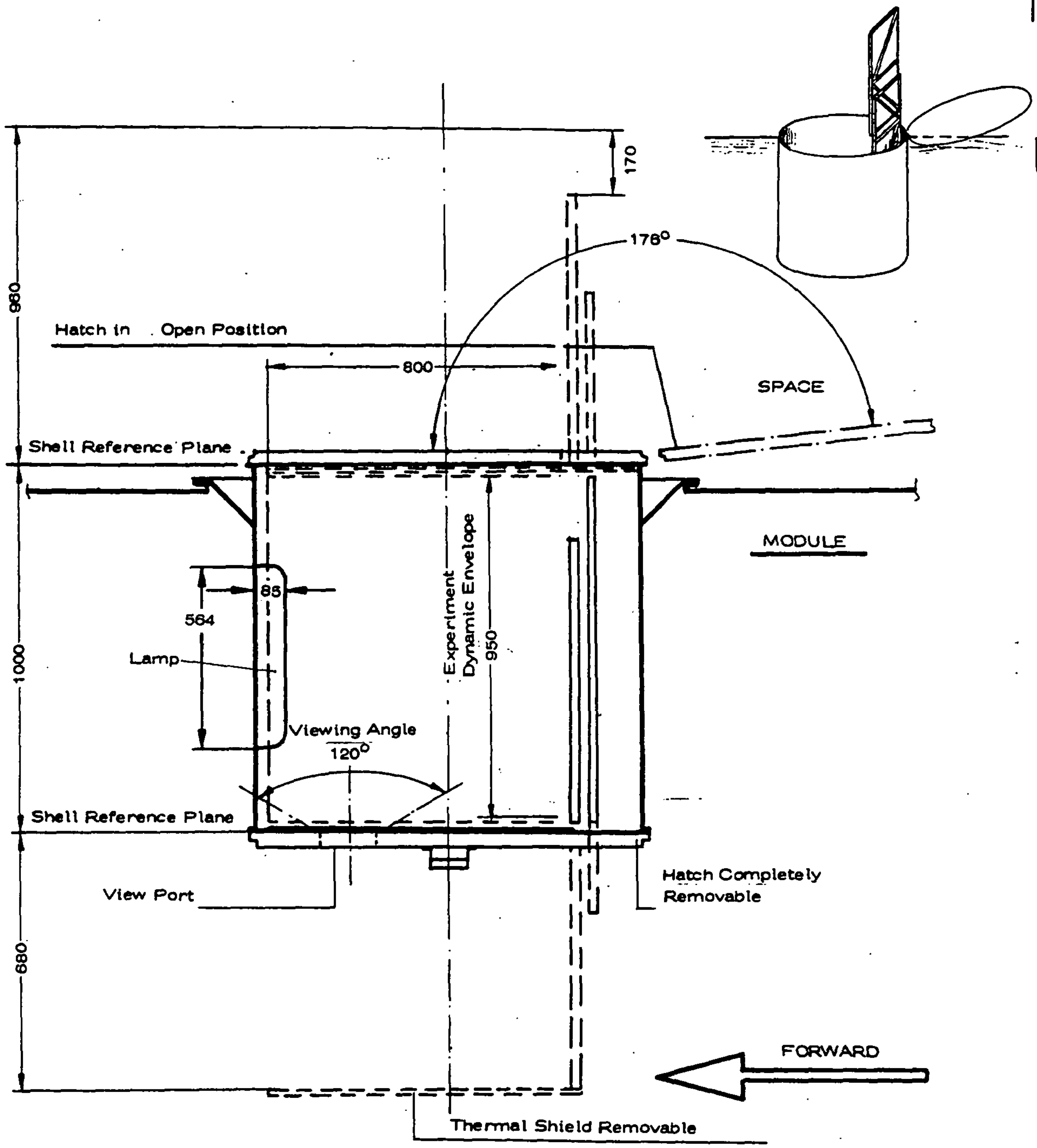

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Figure 4.7-5: Airlock - Experiment Dynamic Envelope

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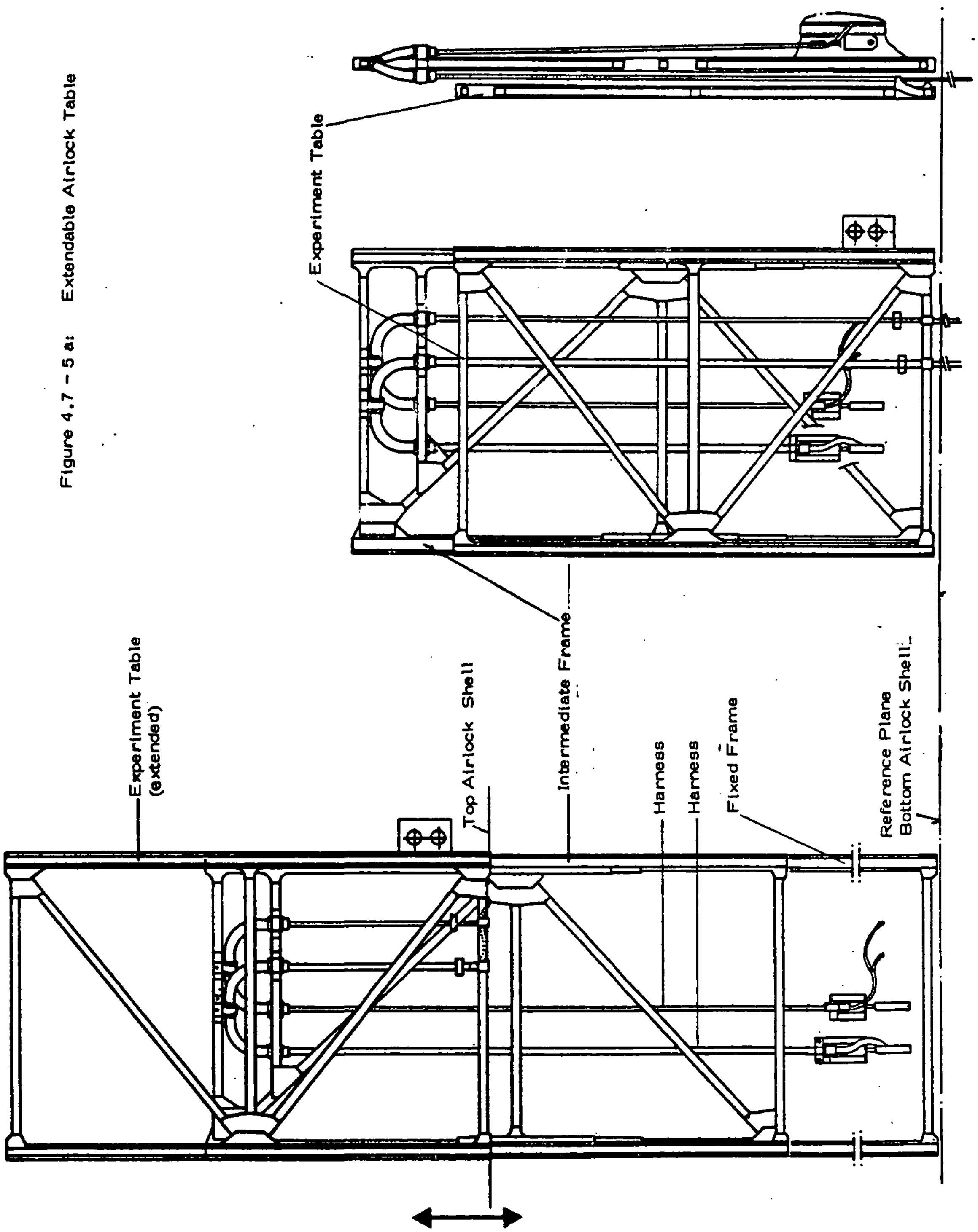

Figure 4.7 – 5 a: Extendable Airlock Table

4.7.1.3 Experiment Table

4.7.1.3.1 Description of Experiment Table

Experiments are normally mounted to the experiment table which is shown in Figures 4.7 – 5a and 4.7 – 6. This consists of an extendable table which has two rails to which the experiment is attached. This table slides over an intermediate frame which, in turn, slides along guide rails attached to the airlock shell. The intermediate frame moves over half the distance of the experiment table and transfers loads from the experiment table to the shell. Flexible electrical harnesses running from the experiment table, via the intermediate frame to airlock shell feedthroughs provide transfer of power and data between the interior of the module and connectors on the experiment table. These harnesses slide through tubes at the end of the intermediate frame.

A manually operated table drive handle is used for table extension. After opening the outer hatch, the experiment table may be extended up to 0.96 m or any intermediate position and then can be locked in that position. It can be seen from Figure 4.7 – 5 that, since the experiment table is shorter than the intermediate frame, an experiment may overhang the table by up to 170 mm so that interference of experiment field of view by the end of the table is minimized. DCN 016

When the inner hatch is removed the experiment table may be extended up to 0.68 m, or any intermediate position, inside the module.

In the retracted position the table can be locked in position by launch lockpins.

4.7.1.3.2 Experiment Mounting

Experiments are mounted to two parallel rails (channel bars) which are attached to each edge of the experiment table (see Figures 4.7–4 and 4.7–6). Attachment is by means of user provided MIO T-bolts and at least four attachment points, two on each rail, are needed. On each rail one bolt must be locked to the rail by means of a pip pin which passes through a hole in the bolt and through a corresponding hole in the rail (these holes are located at intervals along the rail). Experiments may be attached and removed either on the ground or in orbit. The airlock is designed to carry an experiment mass of 100 kg during launch and descent. This limit also applies to an experiment mounted on orbit since, in the case of an emergency descent, there may not be sufficient time to remove the experiment. The experiment can be mounted on the table when it is retracted or when it is extended either inside or outside the module. When experiments with a mass greater than 10 kg are installed on the extended table in 1 g conditions special GSE (not currently supplied by the Spacelab program) must be used to support the mass of the experiment until the table is fully retracted and secured by the launch lockpins.

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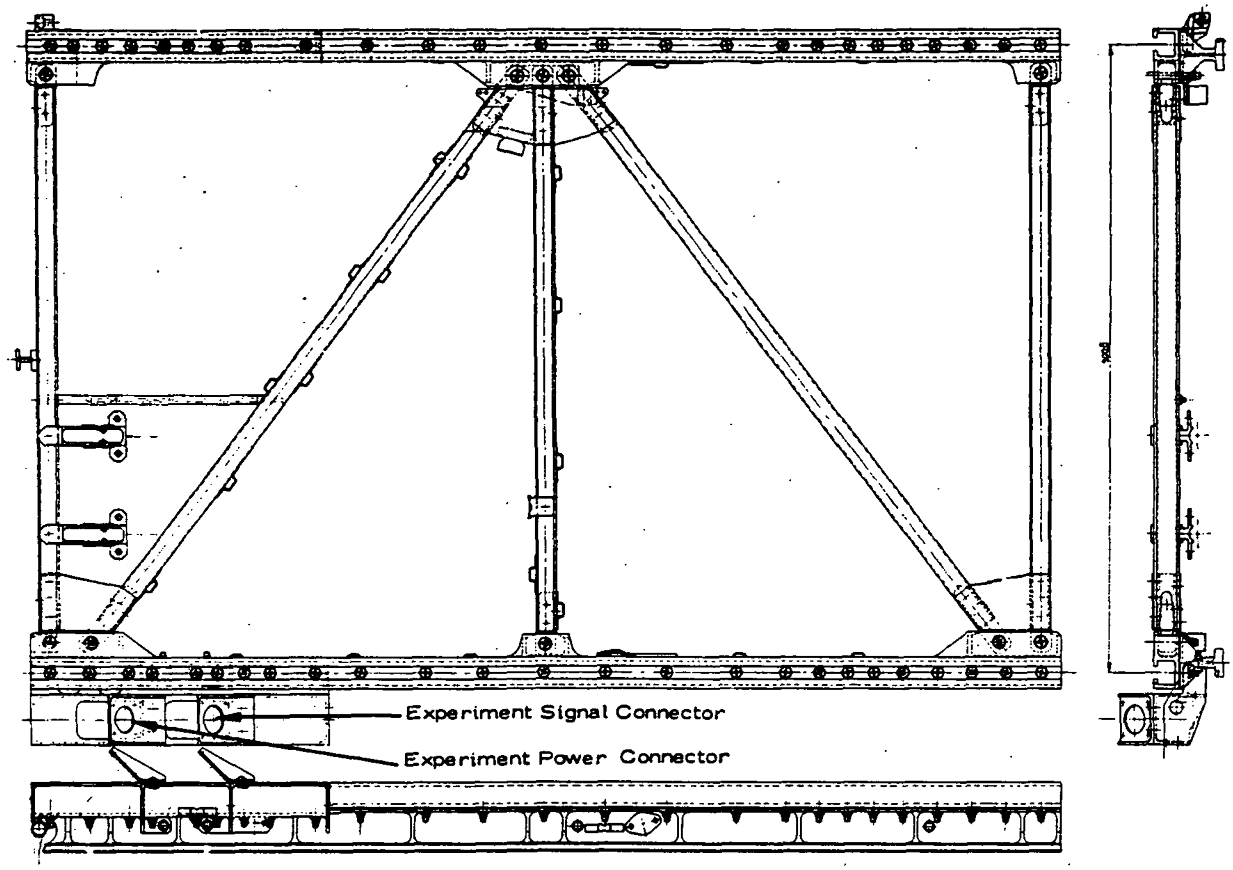

Figure 4.7 - 6 : Assy Experiment Table (Top View)

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4.7.1.4 Power Supply

One experiment dedicated power connector is provided at the platform for

- 28 VDC, protected by a 7 A circuit breaker
- 115 /200 VAC, 400 Hz, 3 Ø, each phase protected by 3 A circuit breakers

The protection and switching circuits are in the EPSP, located in rack No. 4.
For interface details, see Appendix A, Avionics Interface Definition, Section 3.

4.7.1.5 Command and Data Handling for Experiments

Experiment control and data handling for experiment equipment in the airlock can be effected via signal lines which are routed through the airlock shell. The CDMS may be directly utilized (via an experiment RAU mounted inside the airlock or in an experiment rack near to the airlock) or the experiment equipment inside the airlock may be hardwired directly to experiment equipment in a rack).

The airlock contains an experiment dedicated flexible signal harness, which runs from the experiment table via the intermediate platform to the airlock shell feedthroughs and, from there, to a connector bracket (CBS 25) on the module overhead structure. Figure 4.7-5a shows the layout of this harness inside the airlock and Table 4.7 - 1 lists the number and type of cables.

Two alternative harnesses are available but the selection and installation of the required harness can be performed only on the ground. Table 4.7 -1 lists the number and type of cables associated with both harness 1 and harness 2. Harness 1 provides sufficient cables to support the accommodation of an experiment RAU inside the airlock (the necessary cabling from CBS 25 to the CDMS is also provided). DCN 016
If an experiment RAU is used inside the airlock, however, the experimenter is responsible for the mechanical accommodation and the thermal control of the RAU. Alternatively it is possible to remove the RAU and to have available then for experiment usage the full capability of the harness signal lines.

Harness 2 provides suitable cables to enable an experiment inside the airlock to interface with the HRM (but in this case the necessary cabling from CBS 25 to the HRM has to be user provided). If the HRM interface is not used then the relevant cables can be used to interface with experiment equipment outside the airlock. It is not possible to operate an experiment RAU inside the airlock using Harness 2.

Table 4.7 – 1 Experiment Dedicated Flexible Signal Harness

Harness 1				Harness 2			
Number	Cable type	Impedance	needed for RAU	Number	Cable type	Impedance	needed for HRM
2	TSP	75 Ohm		2	TSP	125 Ohm	x
5	TSP	75 Ohm	x	5	TSP	75 Ohm	
6	TSP	–	x	1	TSAC		
1	TS3C	–	x				

DCN 016

For interface details, see Appendix A, Avionics Interface Definition, Section 4.

The two experiment dedicated feedthrough holes (as shown in Figure 4.7 – 4) can also be used for routing experiment utilities. In this case the experimenter has to provide the feedthrough connectors and all cabling. In addition, any guidance support of the utilities inside the airlock to enable proper operation of the experiment table will be part of the payload design. In this context it should be noted that, for operations involving extension of the table into space, the utilities design will have to be such as to enable emergency jettison of the experiment table.

4.7.1.6 Thermal Control

When the inner hatch of the airlock is open the thermal conditions inside the airlock will approximate those of the module interior. Similar conditions will exist when both hatches are closed and the airlock is pressurized (provided that the experiment is not dissipating heat). When the outer hatch is open the thermal conditions for the airlock and the experiment will depend on the particular operating conditions of the mission.

Thermal control of the airlock itself is effected by means of a combination of thermostatically controlled heaters, surface coatings, multilayer insulation and removable thermal shield. Heaters are located on the outer hatch and on the outer hatch seal flange on the airlock shell to maintain satisfactory operation of the seal. Heaters are also located on the experiment table to minimize temperature gradients and distortion of the table. In order to avoid excessive heat transfer between the airlock cavity and space when the outer hatch is open a removable thermal screen (see Figure 4.7-7) is provided to close off the airlock opening. The screen is attached with quick release fasteners to the experiment table on brackets at the outside end for non-extending experiments and at the inward end for extending experiments. It is possible to cut holes in the screen to allow experiment viewing but, in this case, the resultant heat leaks have to be taken care of by the experiment thermal control. DCN 016

The thermal control of the experiment itself will be the responsibility of the experimenter. The airlock structure can absorb only a limited amount of experiment generated heat and the experiment must be thermally insulated at the attachments to the table. Various constraints may apply – such as the need to fully extend the experiment table for passively cooled experiments – but will have to be evaluated for each mission and experiment. For non-extended experiments there is the possibility to use the experiment heat exchanger in the module by running cooling lines through the two experiment dedicated feedthrough holes in the airlock shell (see Figure 4.7 – 4).

When the thermal screen is removed (for example for access to experiments) it can be attached to the inner hatch for stowage (see Figure 4.7 – 2).

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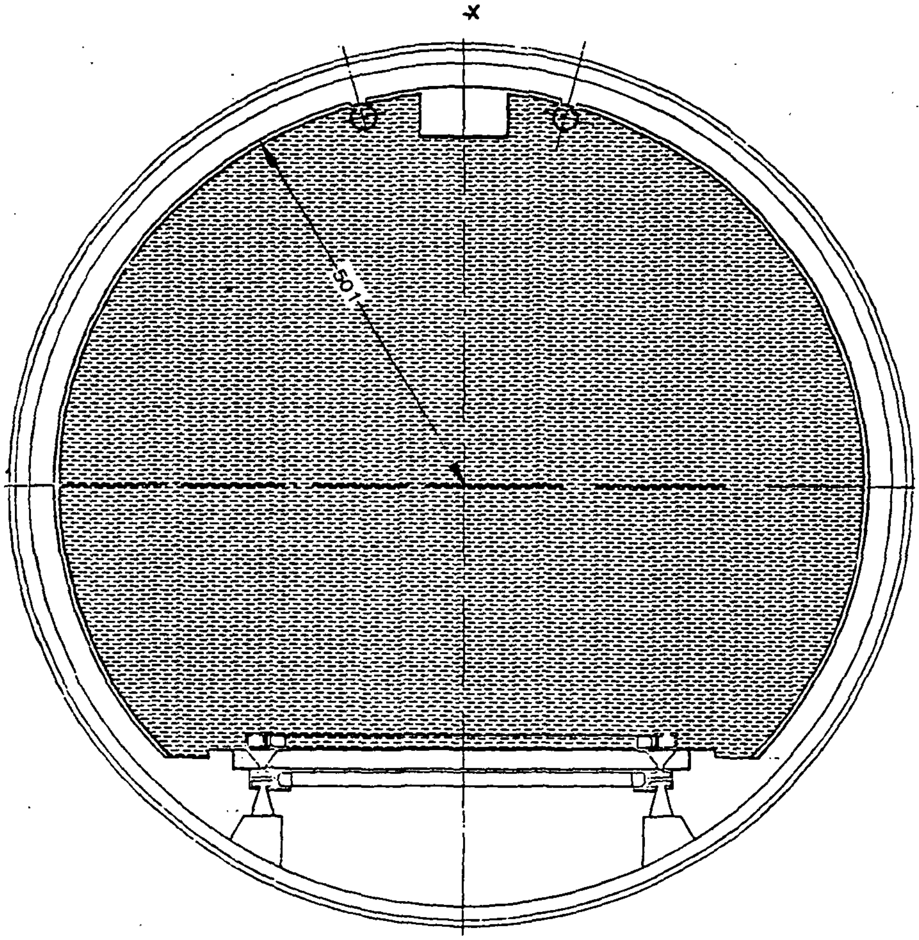

Figure 4.7 – 7: Thermal Screen – Airlock Experiment Table

4.7.1.7 Viewing and Illumination

A 15 cm diameter viewport is provided in the inner hatch as shown in Figure 4.7 – 5 and provides a viewing angle of 120°. The viewport has a cover which is clamped in place by three clips and which, when removed, is stowed on the inner hatch using velcro attachment pads. A lamp inside the airlock provides an illumination intensity of 100 lumen/m^2. The lamp (power consumption 25 W) is controllable from the airlock control panel and at the subsystem control station in the core segment.

4.7.1.8 Airlock Operation

Venting and Re-pressurisation

The pressure in the airlock is controlled by a manual selector vent valve which connects the airlock to the outside of the module, or to a nitrogen supply from the ECS, or to the module atmosphere or which closes off all connections. For venting, the 4-way vent-valve is placed in the venting position. In order to operate the outer hatch the differential pressure between space and airlock must be less than 30 mBar $\pm$ 20 mBar and the time required for venting from 1013 mB to 10 mB is about 10 minutes (Figure 4.7-7a). With the outer hatch open the airlock is exposed to space vacuum but the thermal screen blocks the opening almost completely. Any air leaking from the module will thus escape very slowly and a pressure of 10-4 mBar will probably be the lower limit.

For repressurisation the 4-way vent-valve is placed in the pressurise position. Repressurisation from the nitrogen supply takes about 15 minutes from 0-950 mBar(see Figure 4.7-7a). Equalisation to the module pressure takes only a nominal 1-2 minutes but may have to be delayed until after thermal reconditioning of the airlock and experiment to prevent condensation. Thermal reconditioning from cold case conditions can be expedited by the airlock heaters and by experiment dissipation but it may take several hours before the temperatures of certain parts rise above the dew point. DCN 016

After the inner hatch is removed the module air will mix slowly with the nitrogen in the airlock but diffusion of oxygen may take an hour or more.

Sufficient nitrogen is available from the ECLS to allow a minimum of seven repressurisation cycles per mission. Additional cycles may be possible depending on the overall usage of nitrogen by the ECLS.

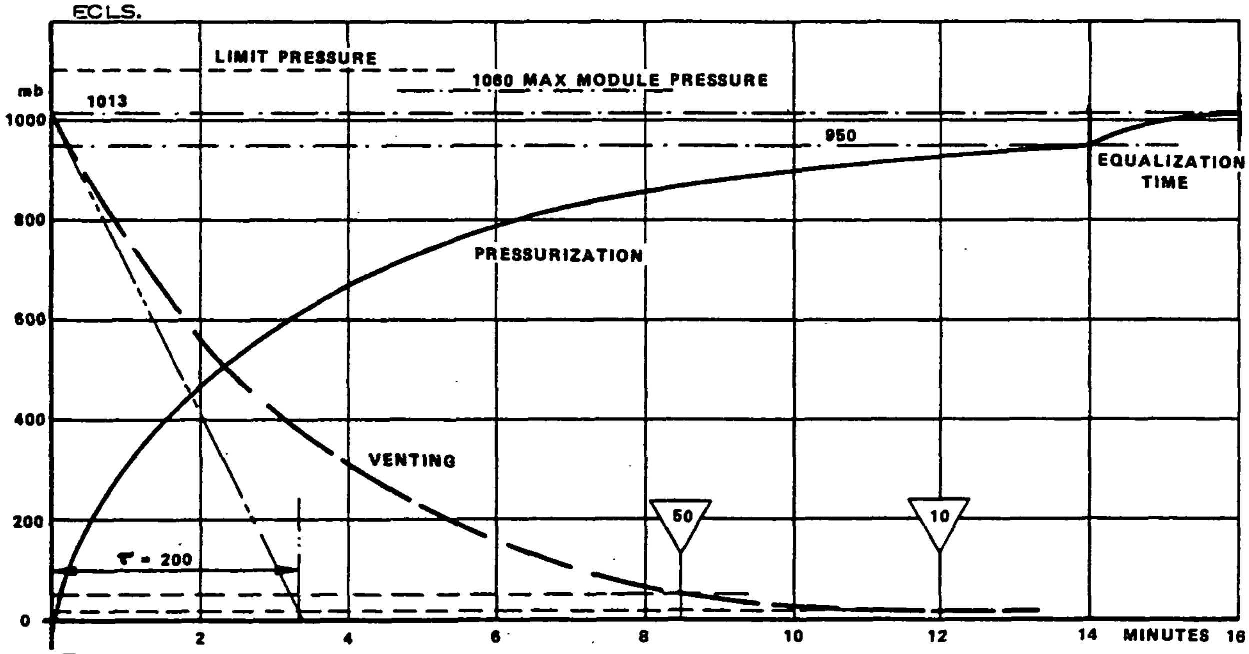

Figure 4.7 - 7a: Pressurization and Vent Times for the Empty Airlock

DCN 016

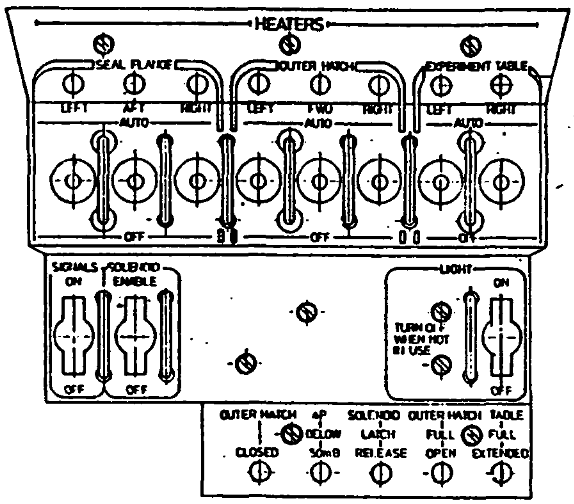

Figure 4.7 - 7b: Layout of Control Panel

Monitoring System

Manual operations on the airlock are monitored by feel and direct observation of handle positions. In addition, certain information is available for display at the airlock or remotely via the CDMS. A direct reading vacuum gauge indicates airlock pressure relative to the module atmosphere. The experiment table mechanism drives a counter which indicates the table position. The control panel on the airlock (see Figure 4.7-7b) has circuit breakers for controlling the airlock lamp and heaters and indicator lights showing the status of outer hatch position, table extended etc. Other data such as temperatures and absolute pressure are available via the CDMS.

Operational Timelines

A simplified example of a possible airlock operational timeline is shown in Table 4.7 - 2. Actual timelines will be highly dependent on thermal reconditioning times and other mission dependent factors.

Table 4.7 - 2 : Airlock Operation

AIRLOCK OPERATION	TIME	
	MIN	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 and secure hatch		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 Airlock		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
Retract Experiment Table		25
Close Outer Hatch		30
Latch Outer Hatch		30
Pressurize Airlock	24	05
Remove & Replace Experiment	1	15

4.7.2 High Quality Window/Viewport Assembly

4.7.2.1 Mechanical Assembly

DCN 030

The high quality window/viewport assembly consists of a viewport and a high quality window mounted in an adapter plate. Figure 4.7-8 shows a conceptual layout of the assembly. The assembly can be mounted in the 1.3 m diameter hole on top of either the core segment or the experiment segment of the module. Details of the viewport are given in Section 4.7.3.

N.B. The high quality window and the adapter plate are NASA furnished equipment (NFE) and detailed design information is currently not available. The high quality window itself is a spare unit of the S 190 A window (left over from the Skylab program) and the information provided in para 4.7.2 is essentially based on performance data available from the Skylab program and from the Martin Marietta Document: Skylab S 190 A Window Utilization Summary Report of 30th May 1975.

The high quality window 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 (Window cross section details in Figure 4.7-9). The window is equipped with a heater system that controls window temperatures to minimize thermal gradients across the glass and to maintain optical performance. The outer surface of the glass has a 40 W electro-conductive film (ECF) heater and two 100 W 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 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.

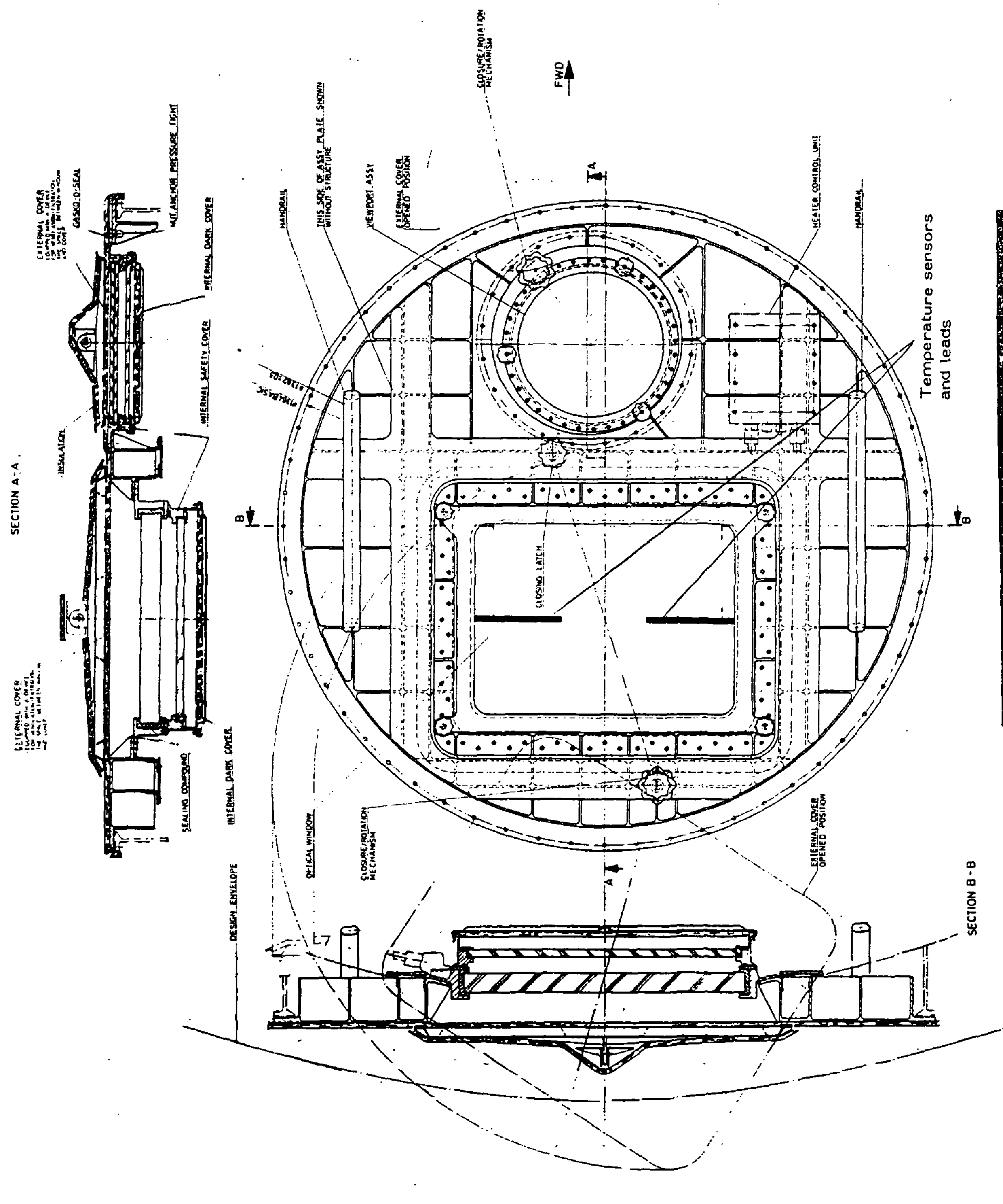

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Figure 4.7 - 8: High Quality Window/Viewport Combination (Conceptual Layout)

DCN 030

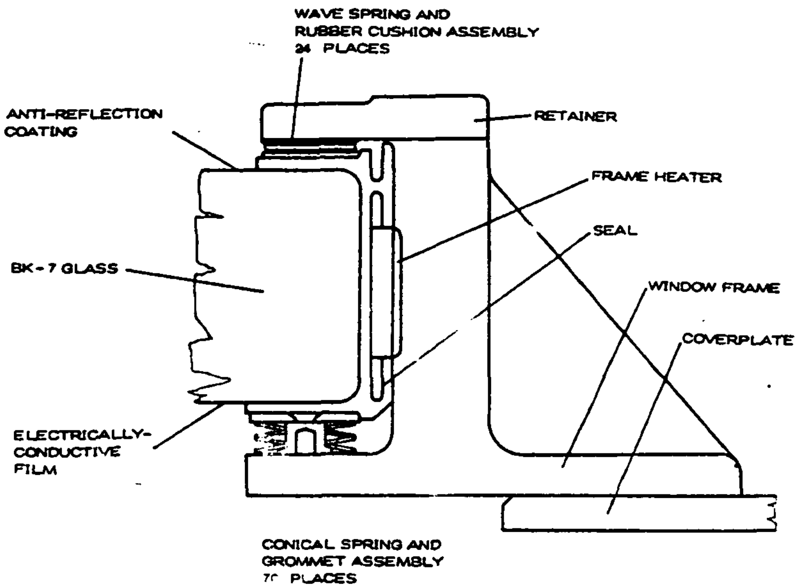

Figure 4.7 – 9: Detail Section Through the Window Frame

4.7.2.2 Optical Performance of Window

Transmissibility

The window transmission characteristics from 300 to 1000 nm are shown in Figure 4.7 - 10.

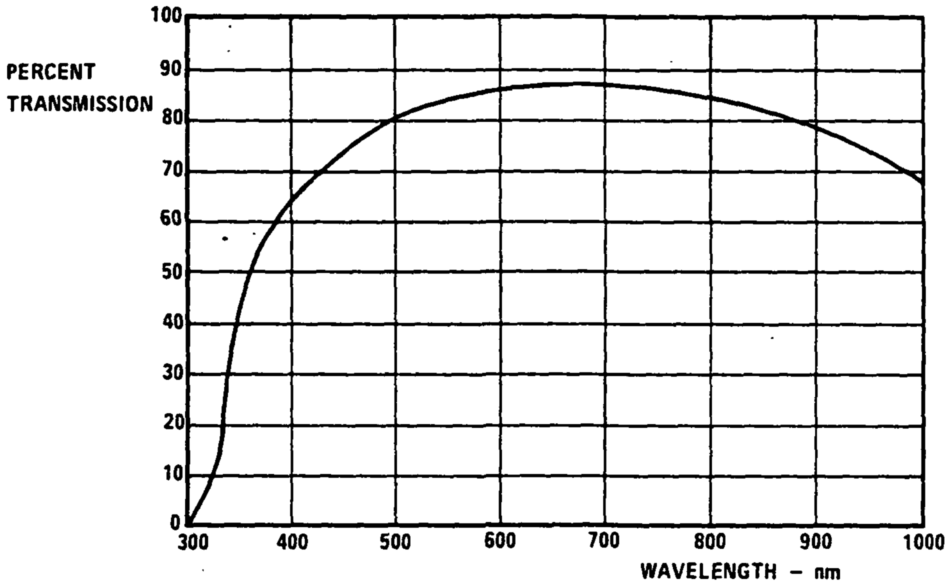

Figure 4.7 - 10: S 190 A Window Transmission

Viewing Area

There are two areas 1x19 cm^2, as shown in Fig. 4.7-8, 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 the performance of the entire window. DCN 030

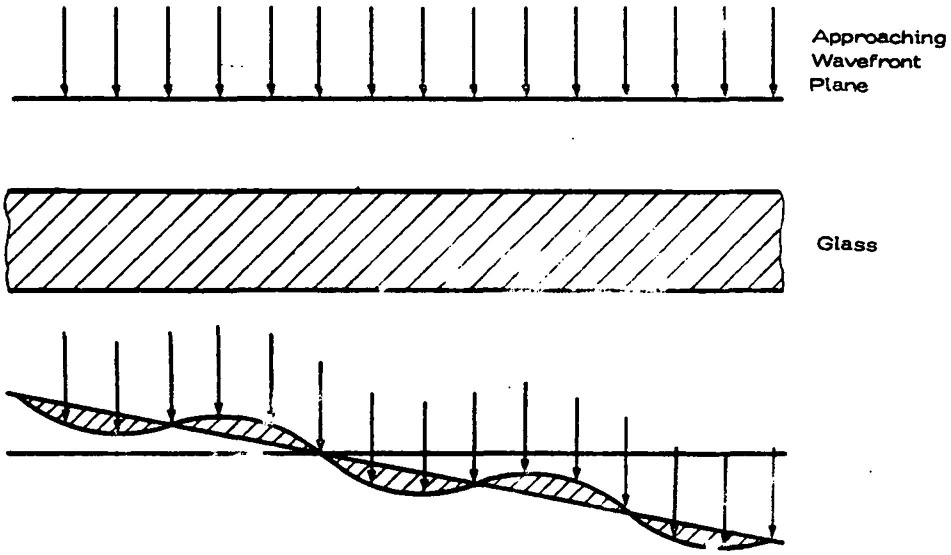

Figure 4.7 - 11: Wavefront Variation Indicating the Mean Deviation and Best Fit Plane After Passing through the Glass

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 4.7-2).

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. Figure 4.7 - 11 illustrates the difference between these two planes.

Table 4.7-3: 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 4.7-3 shows some further optical characteristics of the window.

Table 4.7-4: 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

4.7.2.3 Mounting Provisions

The provisions for mounting cameras and other viewing instruments, as well as permissible 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 for different viewing conditions as the heaters correct for equalization of temperature across the window. Table 4.7 - 5 indicates a preliminary estimate of the power consumption.

Table 4.7-5: 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.

4.7.3 Viewport

4.7.3.1 Mechanical Assembly

Two identical viewports are available for the module. One is permanently located in the aft end cone while the other is CPSE and can be located, as required, either in the high quality window/viewport assembly (see 4.7.2) or in the viewport adapter plate which can be mounted in the 1.3 m diameter hole on top of either the core segment or the experiment segment of the module. Figure 4.7 - 12 shows the layout of the adapter plate.

The viewport provides a space and earth viewing capability for the Spacelab crew and for experiments (e.g. for photography). A time-limited sun viewing capability is also possible. It contains three laminated panes of Triplex "Ten-Twenty" glass of total thickness 26 mm and provides a clear viewing area of 30 cm diameter. Figure 4.7 - 13 shows details of the viewport construction. The viewport glass contains an integral heater film coating to prevent condensation. A manually operated outer cover provides mechanical and thermal protection when the viewport is not in use. A removable inner safety pane of polycarbonate plastic protects the viewport glass against impact from inside the module.

DCN 027

4.7.3.2 Optical Performance of Viewport

Figure 4.7 - 14 shows the transmission curve of the viewport glass (with the inner safety pane removed). The design specification value for optical resolution is that the resolution shall be at least 28 secs of arc over each 50 mm diameter area, for example similar to MIL-G-25871 A (Glass, laminated, aircraft glazing) para 4.4.6.2.2: "Optical Deviation (gunsight area)" or alternatively, or additionally, the window shall meet a test performance requirement for flatness of 7 interference fringes (or less) per inch (25 mm) using monochromatic light of wavelength between 550 mm and 600 mm (see also MIL-G-25871A, para 4.4.7). This resolution applies to at least the central 70 % of the area of the viewport glass.

4.7.3.3 Mounting Provisions

Experiment equipment with a mass up to 25 kg can be attached to the viewport flange using the inner safety pane attachment points (when the inner safety pane is removed). This capability applies for the viewport mounted in the aft end cone, in the viewport adapter plate or in the high quality window/viewport assembly. In addition, the viewport adapter plate contains attachment points (see Figure 4.7 - 12) which can support an experiment mass up to 50 kg (with no mass simultaneously supported by the viewport flange). These attachment points are identical to those of the handrails on the adapter plate so that the handrails and experiment attachments can be rotated 90^{o} from the positions shown in Figure 4.7 - 12 according to mission requirements. The local capabilities defined in this section apply only for experiment equipment which is mounted and removed in orbit although the design allows emergency re-entry and landing with the equipment still mounted.

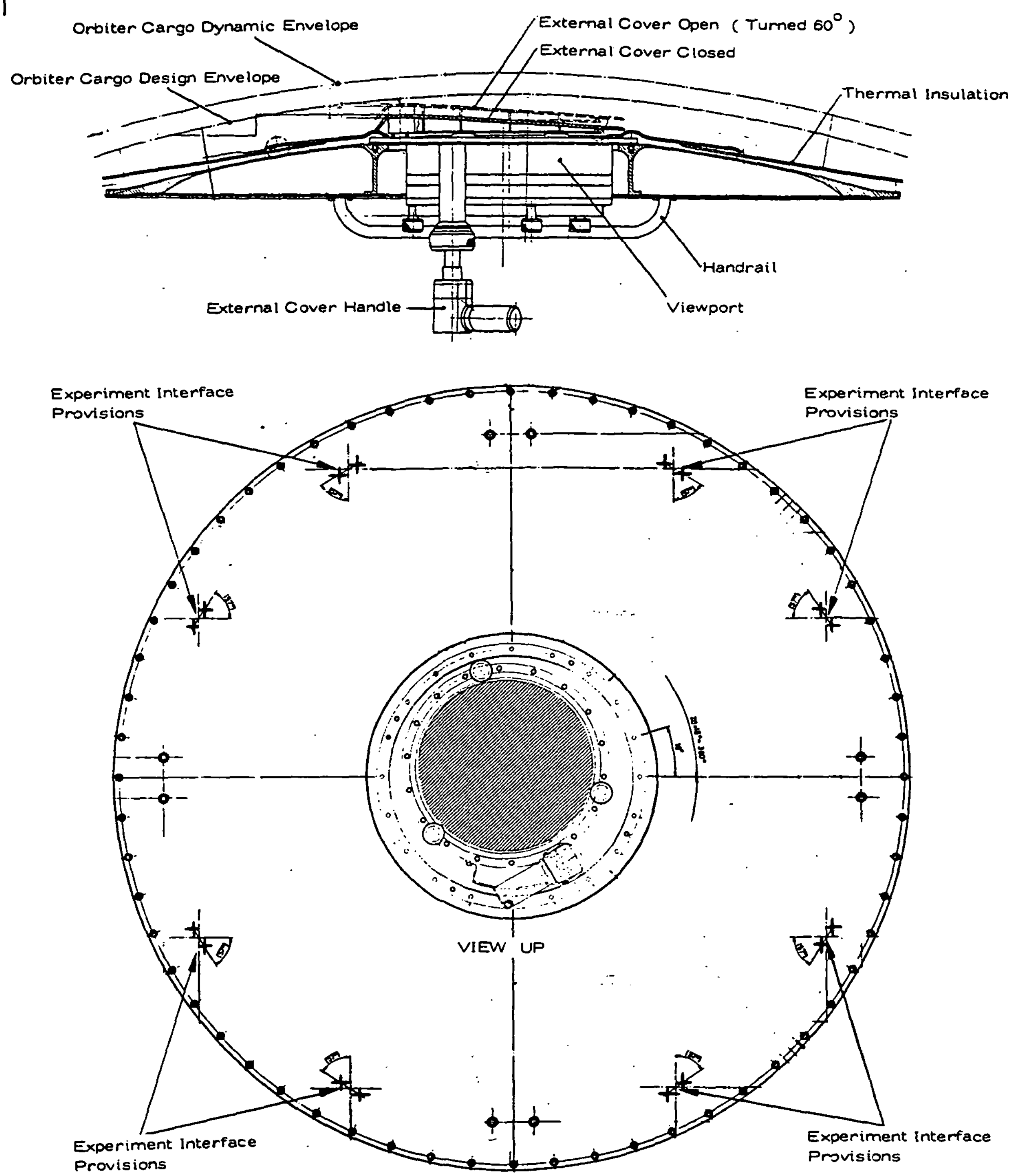

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Figure 4.7 - 12 : Viewport Adapter Plate Assembly

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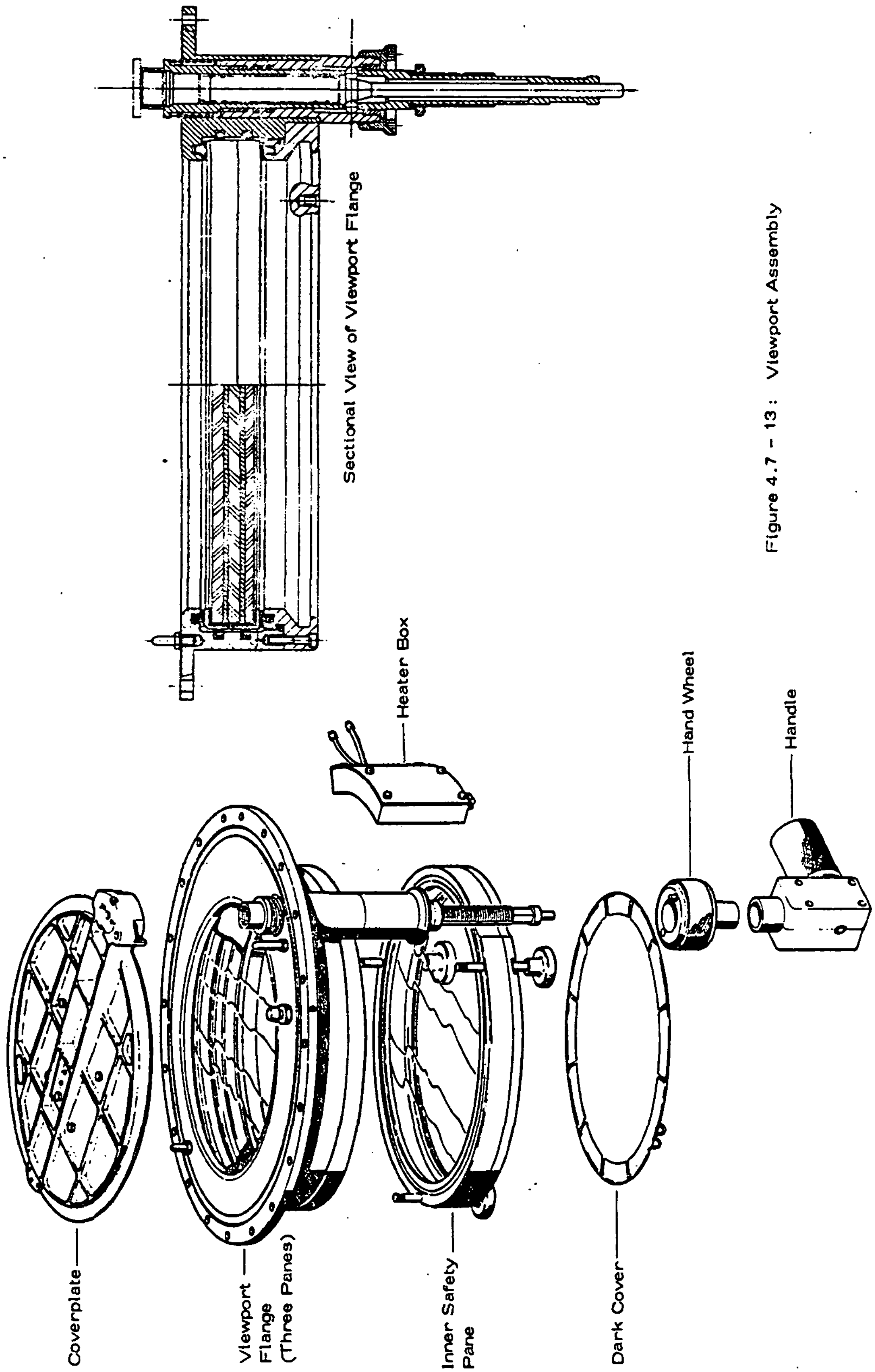
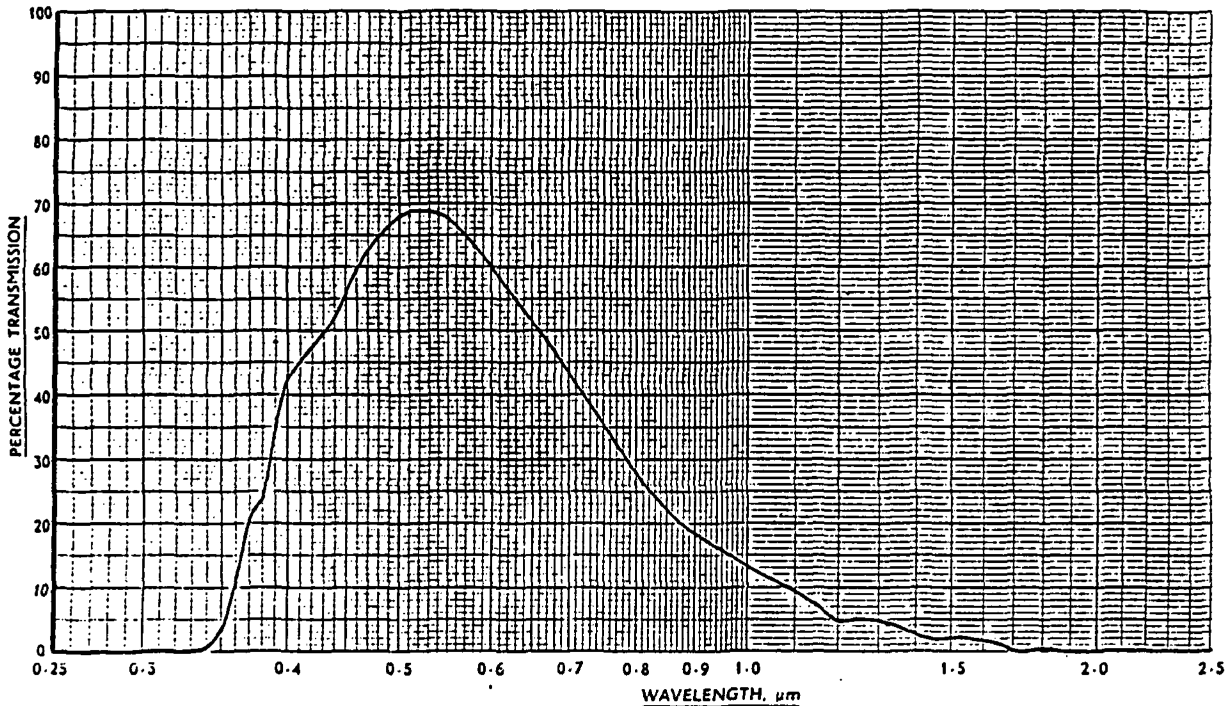


Figure 4.7 - 13 : Viewport Assembly



DCN 027

Figure 4.7 – 14 : Spectral Transmission of Spacelab Viewport

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4.8 Instrument Pointing Subsystem

The Instrument Pointing Subsystem (IPS) will be delivered under the same general terms as the Spacelab and will be available for use on the second and subsequent flights.

4.8.1 IPS Description

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 weight. The overall configuration of IPS with a payload is shown in Figure 4.8-1.

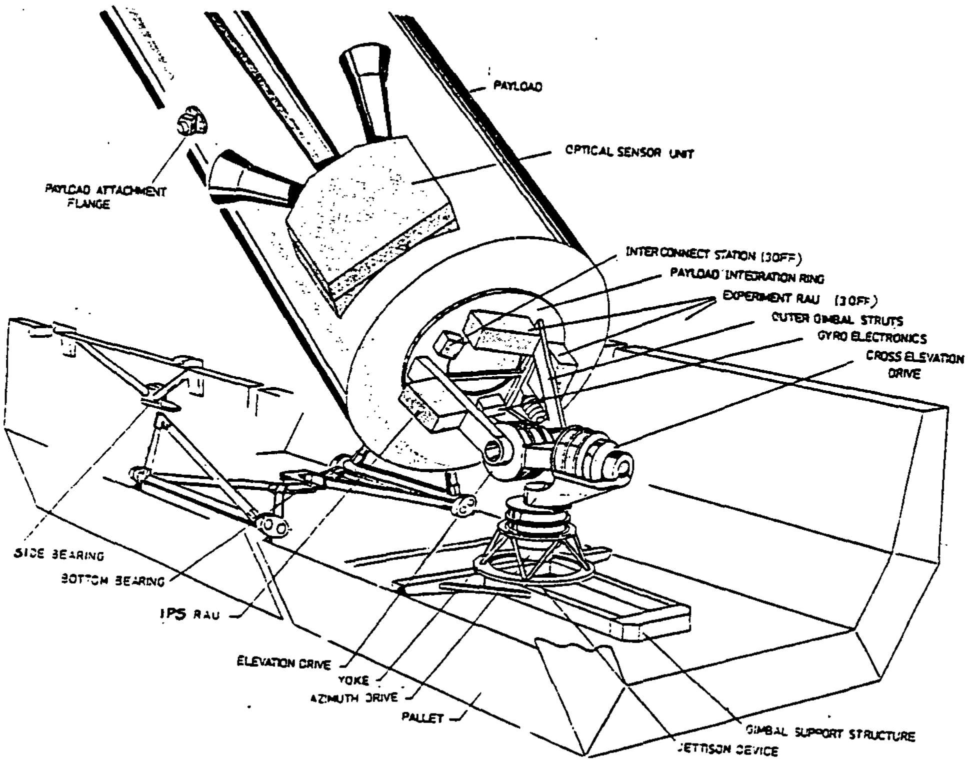

Figure 4.8-1: Instrument Pointing Subsystem.

DCN 013

4.8.1.1 Gimbal System

The Gimbal System (shown in Figure 4.8-2) is attached to the payload when on-orbit, and performs the control maneuvers required by the observation program. During launch and landing the gimbal system and payload are separated, with the payload supported by the IPS Payload Clamp Assembly, as shown in Figure 4.8-3.

When on-orbit, the payload is released from the clamps, attached to the gimbal, and three axis attitude and stability control maintained by rotational torques applied by the three drive assemblies.

The three identical drive units are arranged in such a way that their axes intersect at one point. Each drive assembly employs three wetlubricated ball bearings, two brushless DC-torquers, and two single speed/ multi-speed resolvers. The design of the drive assembly includes a main shaft, an auxiliary shaft and a load by-pass mechanism which allows the loads occurring during ascent and descent to be taken by the assembly housing without the need for additional clamping devices, while at the same time off-loading the ball bearings.

DCN 012

All electrical services for IPS and payload functions are carried across each drive unit by a cable follower device consisting of two flexible cable bundles wound in opposite directions. Except for the power electronics units, the major electronics units of IPS and the payload are mounted on the outer parts of the gimbal structure and on the payload integration ring, as shown in Figure 4.8-1.

The gimbal system includes a jettison device for use in an emergency case in which the payload and/or IPS cannot be retracted to a safe landing configuration. The device imparts no momentum to the jettisoned equipment.

DCN 012

The payload/gimbal separation mechanism necessitates the payload and gimbal to be installed separately and sequentially onto the pallet and then connected together. The design goal for the separation mechanism is to make possible separation of payload and gimbal on-orbit by EVA. This is not a design requirement, however, and it is not the only back-up method to avoid jettisoning in case of an IPS failure.

DCN 012

A replaceable extension column between the jettison device and the gimbal support structure can be changed between missions to adjust the height at the gimbal point of rotation for particular payload requirements. The gimbal system loads are distributed to four pallet hardpoints by the gimbal support structure. This structure incorporates a set of rails for the positioning of the gimbal system.

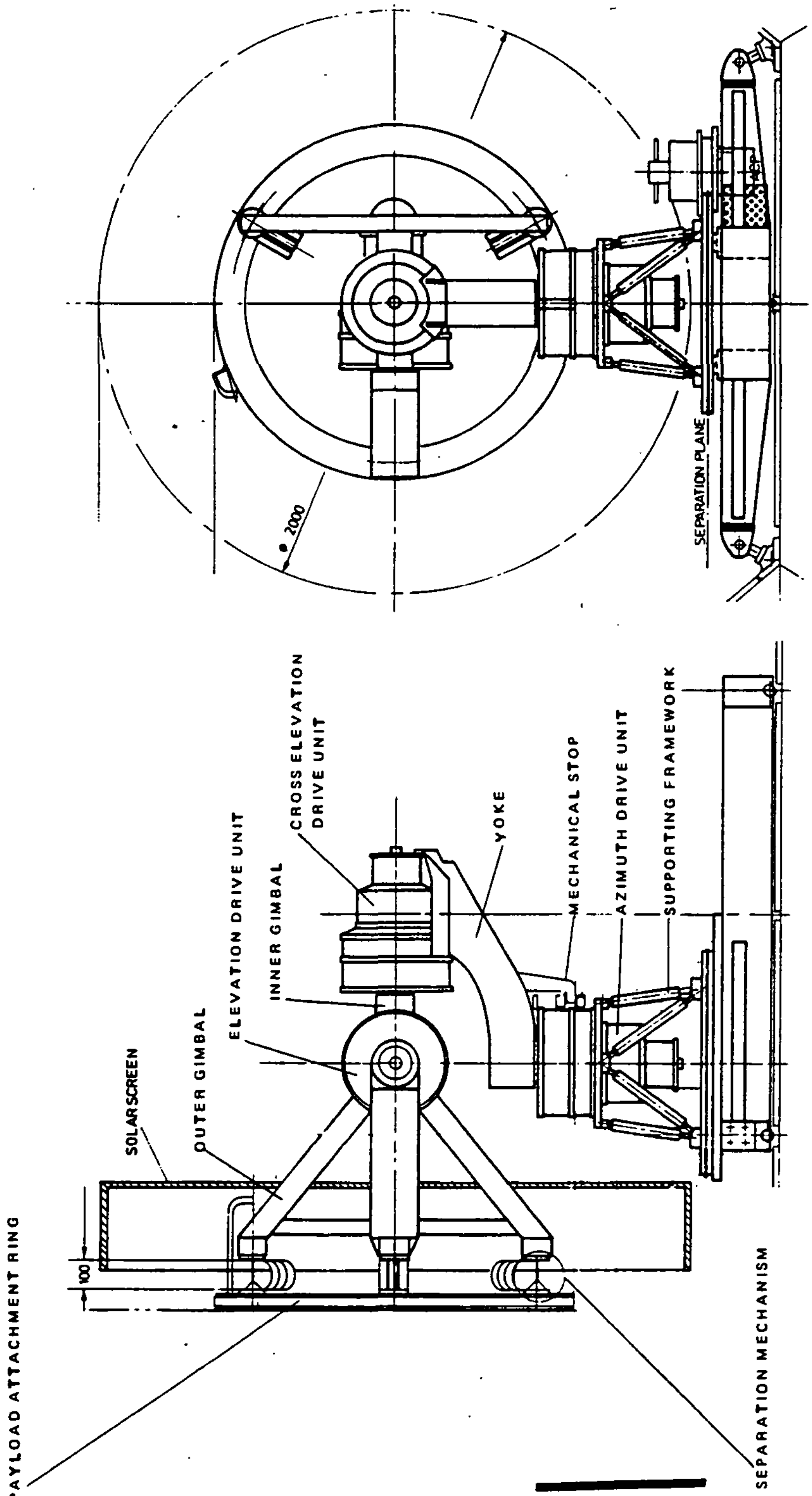

Figure 4.8-2: Gimbal Structure

DCN 013

4.8.1.2 Payload Clamp Assembly

During ascent and descent, the payload is physically separated from the IPS to avoid inputting flight loads to the payload from the IPS. The payload is supported by the Payload Clamp Assembly (PCA) which distributes the flight loads of the payload into the pallet hardpoints (Figure 4.8-3). The PCA is designed such that the directions of the loads induced in the payload are predominantly tangential.

The Payload Clamp Assembly consists of

- o three clamping mechanisms, i.e. three attachment points defining a triangle in the Y_o- Z_o plane.
- o replaceable struts distributing the loads from each clamping mechanism to four pallet hardpoints. These struts will be tailored to fit each payload. The actual dimensions of the struts determine the size of the triangle mentioned above to accommodate payloads between 0.5 m to 3 m diameter.
- DCN 001 o non-replaceable elements distributing the loads from the replaceable struts to the four pallet hardpoints.

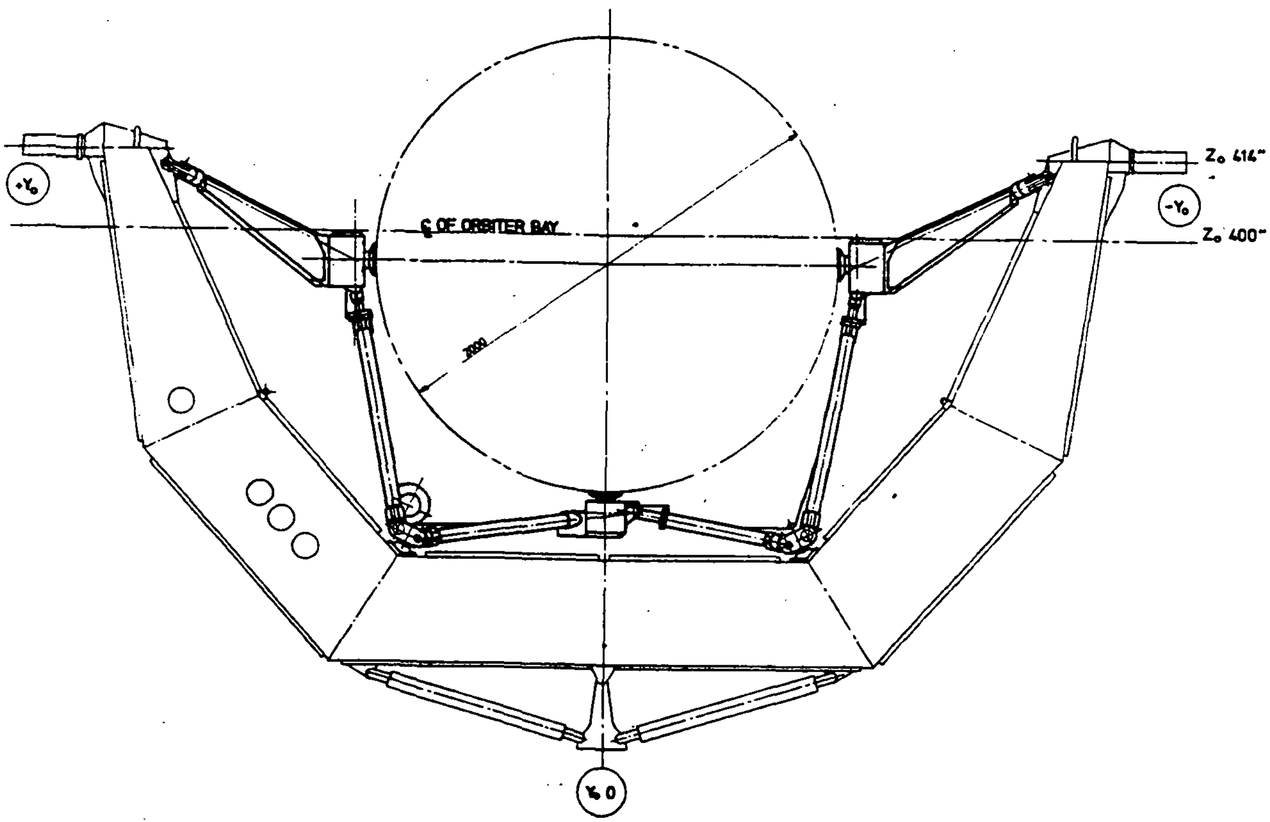

Figure 4.8 - 3: IPS Payload Clamp Assembly

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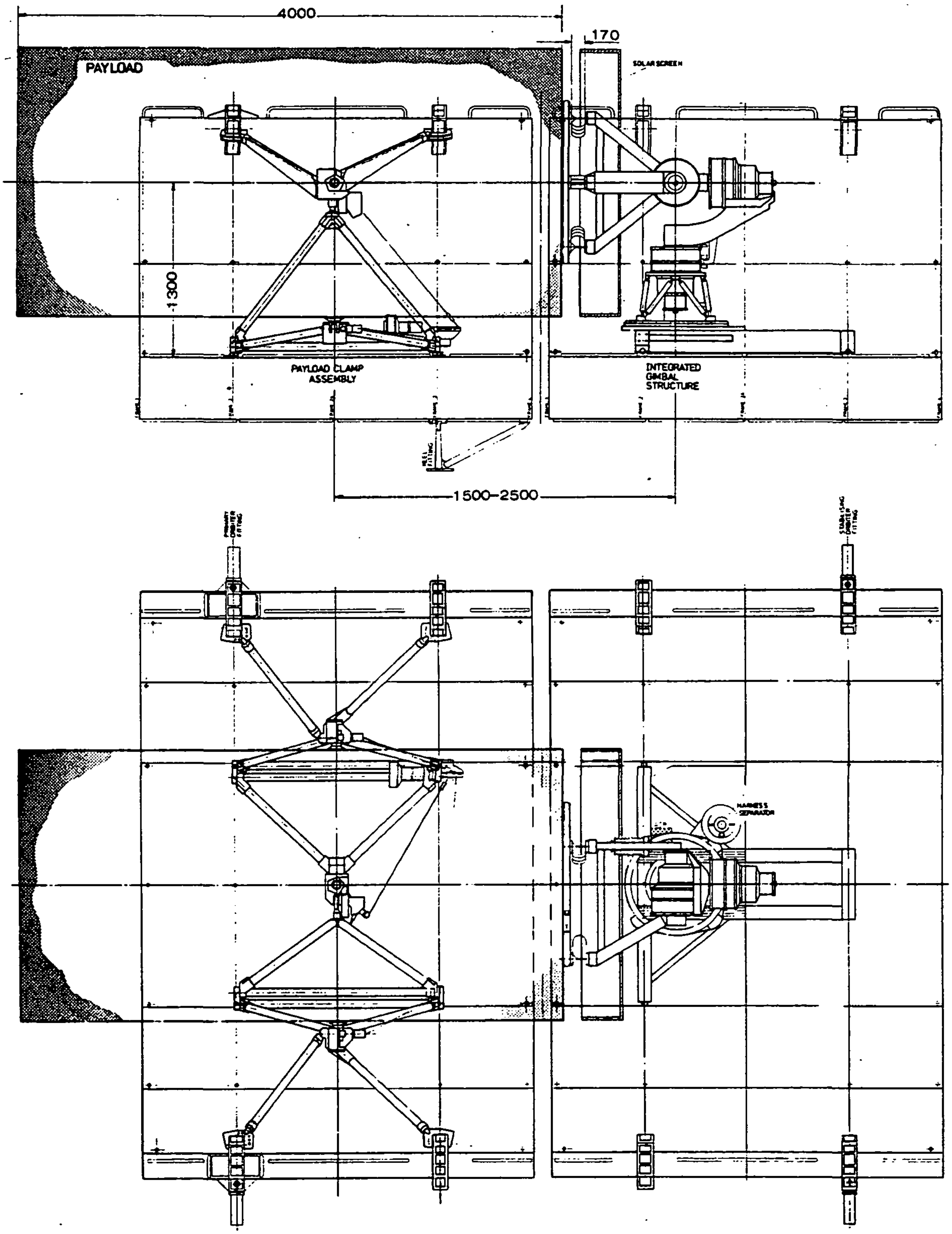

4000
170
PAYLOAD
SOLAR SCREEN
1300
PAYLOAD CLAMP ASSEMBLY
INTEGRATED GIMBAL STRUCTURE
1500-2500

DCN 012

The Payload Clamp Assembly is capable of mounting and distributing the load of a 2000 kg payload into a single unmodified pallet without exceeding safe loading conditions on the basis of compatible payload dimensions and CG location. However, the clamping mechanisms and the non-replaceable elements of the Payload Clamp Assembly are designed for the loads corresponding to a payload mass of 3000 kg with its CG located up to 5 cm radially displaced in the y-z plane from the centre of a 3m diameter circle in the clamps and up to 10 cm displaced in the X direction from the plane of the clamps. In this case, the pallet may require local reinforcement in the location of PCA/pallet attachment points, and the replaceable struts of the PCA must be designed to the loads involved.

The IPS gimbal and control system are capable (within the limitation of 20 Nm torque per axis) of performing all normal pointing functions for payloads of mass greater than 3000 kg, if the payload is supported throughout ascent and descent by a separate clamp system.

4.8.1.3 Electrical System

DCN 012

Overall control of the IPS during normal operations is exercised from the Spacelab control console using the keyboard and display of CDMS and the subsystem computer. Software packages covering all normal IPS functions, from pre-launch check-out through to pre-landing payload retraction, reference guide star tables and the planned observational sequence are all stored in the CDMS. Only the emergency retraction or jettison functions will be initiated from a separate IPS emergency control panel on the Orbiter Aft Flight Deck.

Stability control of the payload is based on rate integrating gyros (RIG) error signals about three axes processed within the IPS mini-processor (DCU) to generate command signals to DC torquers in the three gimbal axes (see Fig. 4.8-4). The RIG package is located on the outer gimbal and hence, aside from distortion or flexures occurring 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 axis for those payloads relying on mechanical alignment. Since mechanical mounting methods do not allow pointing accuracies of the order of a few arc seconds to be achieved, the optical sensors allow for the input of a simulated star image from the payload into the sensor. A correction for the offset from the star-tracker LOS is then made in the software processing of the startracker 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.

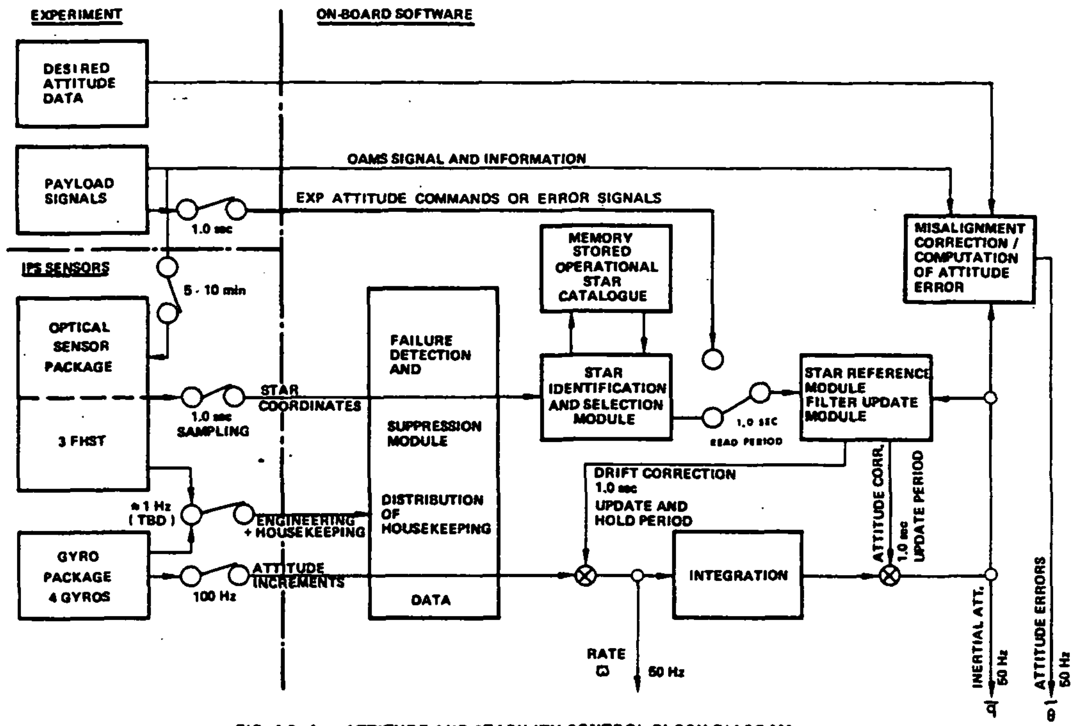


FIG. 4.8-4 : ATTITUDE AND STABILITY CONTROL BLOCK DIAGRAM

DCN 013

4.8.1.4 Mass and Power Budget

Preliminary data of the mass and power budget are presented in Table 4.8 - 1. The mass data apply to an IPS configuration to accommodate a nominal 2000 kg payload.

Table 4.8-1: Mass/Power Budget of IPS

Assembly	No.	Mass (kg)	Power (W) Mean
Gimbal Structure	1	211	0
Drive Assemblies	3	185	48
Thermal Control		25	0-212
Payload Clamp	1	158	0
Attitude Measurement Sensors	1	78	86
Power Electronics	1	54	130
Data Electronics	1	38	88
TOTAL		749	352-564

DCN 013

4.8.1.5 Payload Supporting Services

The IPS will meet the pointing requirements of para. 4.8.2 while providing the following supporting services interfaces across the gimbal system for the use of payloads:

EPDS
- 3 busses for primary DC
  200 W max. cont., 350 W peak each
- 1 bus for experiment essential power
  100 W max. cont.

DCN 012

CDMS
- wiring for 3 experiment RAU's.
- 6 TSP, 125 Ohm impedance,
  for 3 HRM channels up to 16 MB/s
- 2 TSP, 75 Ohm impedance,
  for 1 CCTV channel plus sync
- 1 TSP, 75 Ohm impedance,
  for 1 4.5 MHz analog channel
- 10 pairs flat conductor shielded,
  for Caution and Warning and other
  payload functions

The cables listed above are routed from connector brackets at the payload integration ring across the gimbals to connector brackets at the gimbal support structure on the pallet.

From there a Spacelab harness provides the necessary connections to the relevant Spacelab subsystems for the EPDS lines and the experiment RAU lines. For the remaining signal lines any necessary cabling to Spacelab subsystems or to experiment equipment has to be provided by the experiment/ payload integrator. The functions allocated above to these signal lines are listed only to define their electrical characteristics and the lines can be used for other suitable functions.

4.8.1.6 Flexibility and Growth Potential

In order to accommodate different payload sizes and mounting configurations, the following variations in IPS hardware are possible :

(i) an extension column may be inserted in the gimbal to raise the centre of rotation from 1.3m above the pallet floor (minimum) to any height up to 2.5 m.

(ii) the gimbal may be mounted at any position on the rails of the gimbal support structure.

(iii) the rails of the gimbal support structure are replaceable elements and by using "double length" rails, the gimbal support structure may be positioned in the central region of a pallet. The change of rails is necessary because there are not suitable pallet hard-points on the centre frame of the pallet,

(iv) the gimbal support structure provides attachment points for the payload clamp assembly such that the PCA and gimbal may utilize the same pallet hardpoints or may be mounted separately.

(v) the replacement of struts of the PCA allows different diameter payloads to be accommodated

(vi) using an assymetric strut arrangement, allows the payload CG (clamping plane) location in the X direction to be at unequal distances from the pallet frames supporting the loads.

DCN 012

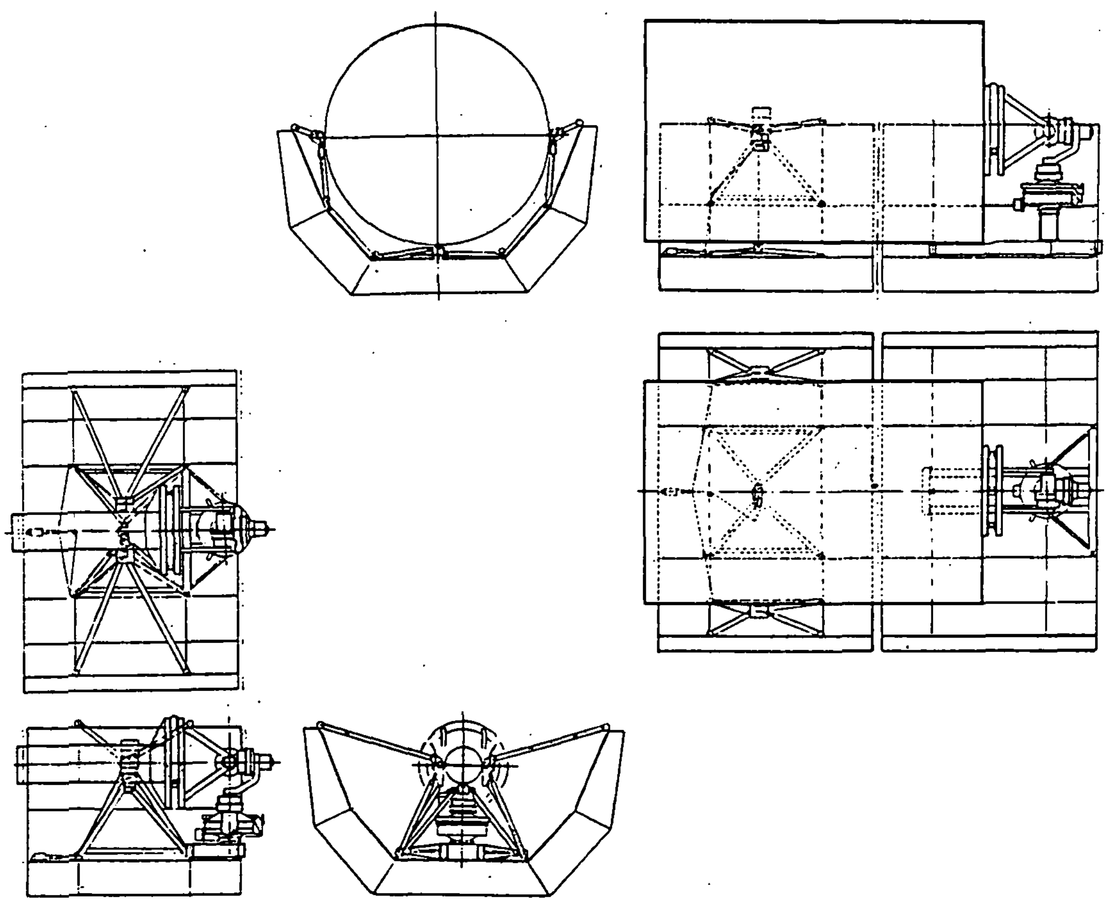


Figure 4.8 - 5: IPS Accommodation of 3 m and 0.5 m Diameter Payloads

Figure 4.8-5 illustrates the accommodation of two quite different payloads, although the asymmetric case (vi above) is not shown. For all missions, it will be necessary that the IPS user verify that the loads induced in all PCA and pallet hardware by his intended configuration of the IPS does not exceed those of the design cases (see 4.8.1.2.)

The optical sensor package includes the capability to have 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.

4.8.2 Pointing and Stabilization

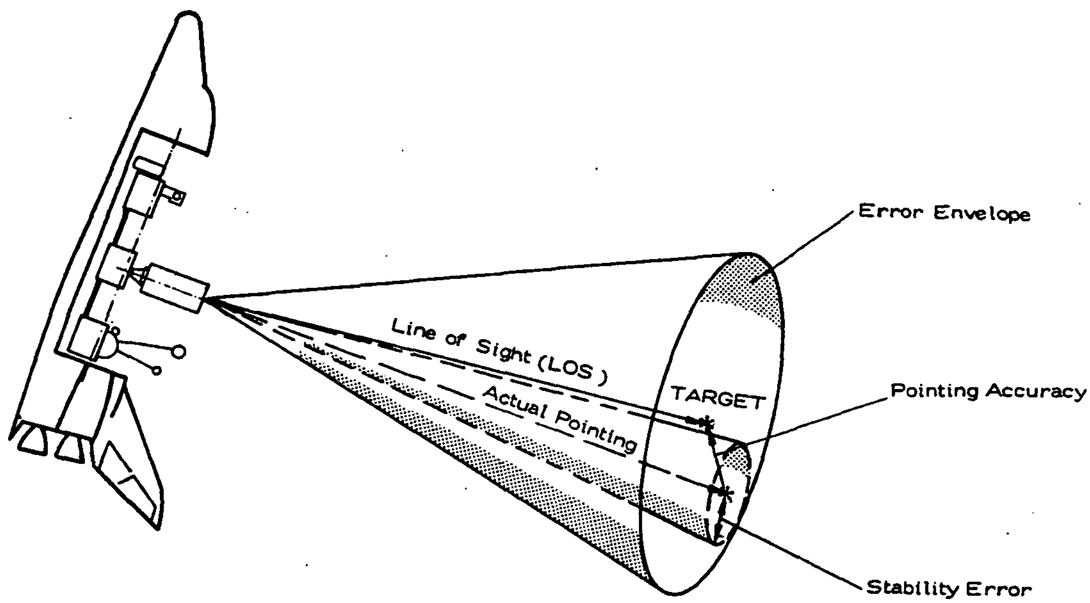
The IPS provides 3-axis attitude control and stabilization for experiments. Since some performance values, for instance the disturbance error, vary with payload physical characteristics, the characteristics of nominal 2000 kg and 200 kg payload are used as design reference payloads except when a requirement specifically states otherwise (Table 4.8-2). 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 pointing accuracy (see Figure 4.8-6) is the long-term attitude error between the desired and actual direction of the LOS. All error sources with time constant greater than five minutes are included within this and the values given represent the root-sum square (RSS) of all the contributing errors.

The quiescent stability error is the short-term attitude variations of the actual LOS about the mean direction (see Figure 4.8-6) for the case of no disturbances from the Orbiter, Spacelab or the payload. All error sources with time constant less than five minutes are included and the values given correspond to that for which the actual error will be less than this value for 67 % of the time. The pointing and stabilization characteristics are summarized in Table 4.8-3 and are presented in more detail in the following paragraphs.

Table 4.8-2: 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
Structural characteristics frequency (TBD mode)	TBD Hz	TBD Hz



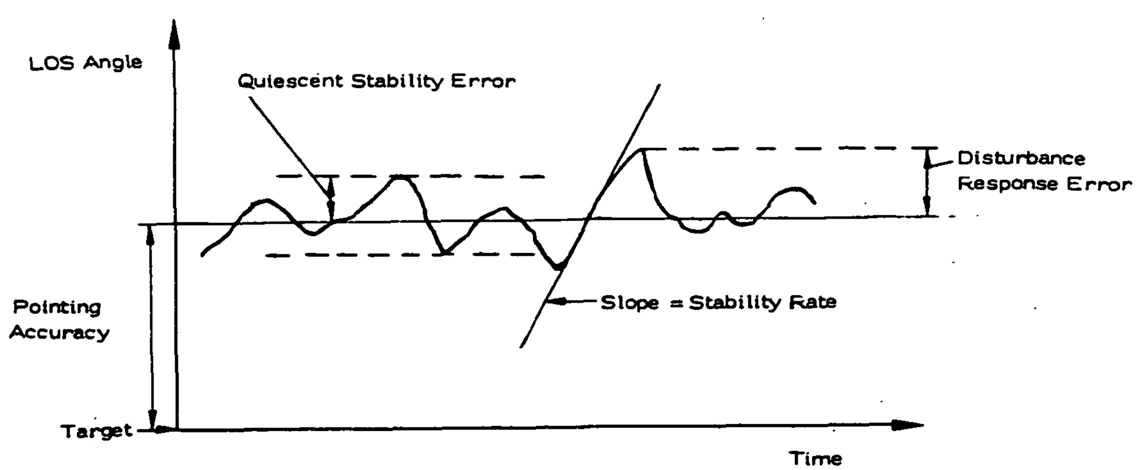


Fig. 4.8 – 6: Error Definitions

4.8.2.1 Pointing Accuracy

Table 4.8 - 3: Pointing and Stability Characteristics

	Requirements	Goals
Pointing Accuracy		
LOS	2 arc sec	0.8 arc sec
ROLL	40 arc sec	15 arc sec
Quiescent Stab.Error		
LOS	1 arc sec	0.33 arc sec
ROLL	3 arc sec	1.6 arc sec
Man Motion Dist.Error		
LOS	3 arc sec (peak)	1 arc sec (peak)
ROLL	10 arc sec (peak)	4 arc sec (peak)
Stability Rate (max.)	1 arc min/sec (peak)	5 arcsec/sec (RMS)
Pointing Range		
LOS	πSter.	N/A
ROLL	πRad.	
Slewing Rate (max.)	2.5 deg/sec	

The pointing accuracy 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.

4.8.2.2 Quiescent Stability Error

The quiescent stability error is 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 200 kg and 2000 kg payloads without disturbances from the Shuttle.

4.8.2.3 Disturbance Response Errors

The disturbance 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 and pointed to any attitude within the LOS range. Since the disturbance error for a given input disturbance varies significantly with IPS location and pointing direction, the disturbance response error values given correspond to worst case values for the 2000 kg payload. For smaller payloads the disturbance response error is larger; for a 200 kg payload the increase is less than a factor 1.7 times.

Man Motion Disturbance

The disturbance error (peak value) due to a standardized man motion disturbance (corresponding to a typical wall push-off by the crew) 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.

Orbiter Limit Cycle Disturbance

The limit cycle errors (peak value) arising in each axis due to Shuttle limit cycle motion of $\pm$ 0.1 degree and 30 m sec duration thruster firing are not greater than those caused by man-motion disturbance, for the same payload and IPS configuration.

4.8.2.4 Stability Rate

During fine pointing, the peak stability rate is less than 60 arc-sec/sec in the two axes perpendicular to the experiment LOS and less than 130 arc-sec/sec about the roll axis for a nominal 2000 kg payload and corresponding to the nominal man motion or the Orbiter limit cycle disturbance.

The design goal for the rms quiescent stability rate is less than 5 arc-sec/sec about the two axes perpendicular to the experiment LOS and 25 arc-sec/sec about the roll axis for a nominal 2000 kg payload.

4.8.2.5 Pointing Range

The IPS has a conical LOS pointing range of at least π steradians without 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. In order to prevent the payload from contacting any surrounding equipment due to error or failure, the IPS contains a redundant system (software in CDMS, hardwired in contingency control electronics) for controlling angular range and rate. This must be adjusted for the configuration of surrounding equipment on each mission. The achievable LOS range and allowable rate in the elevation - and cross-elevation axes are restricted as shown in Figures 4.8 - 7 and 4.8 - 8. The motion about the azimuth axis is restricted in rate to 3°/s and by a mechanical stop to 360° total range. DCN 012

4.8.2.6 Scan and Earth Pointing Modes

Scan motions may be performed under software direction in the "inertially stabilized" mode while maintaining attitude control by the IPS optical sensors, provided that the scan rates do not exceed 3 arc-min/sec and the guide stars used do not leave the sensor fields of view. In this mode the pointing accuracy is degraded by an amount depending on the scan rate up to a maximum of 3 arc-min for a scan rate of 3 arc-min/sec. For scan rates greater than 3 arc-min/sec, the IPS must operate under gyro control only, and the pointing accuracy is degraded by gyro drift and accumulated numerical errors. In both of these modes the stability performance is not significantly affected. DCN 012

Earth scanning or target tracking modes may be software directed but the angular rates involved exclude the use of the optical sensors. The pointing accuracy is therefore as discussed above for the gyro only mode. It is possible for the payload to insert absolute attitude corrections from his own Earth target optical tracker into the IPS control system but this sensor is not provided within the IPS. DCN 012

4.8.2.7 Experiment Control

Pointing control may be exercised by experiment-generated attitude commands and by experiment sensor provided pointing error signals. These data are accepted from one of the experiment RAU's into the RP 5 Data Control Unit.

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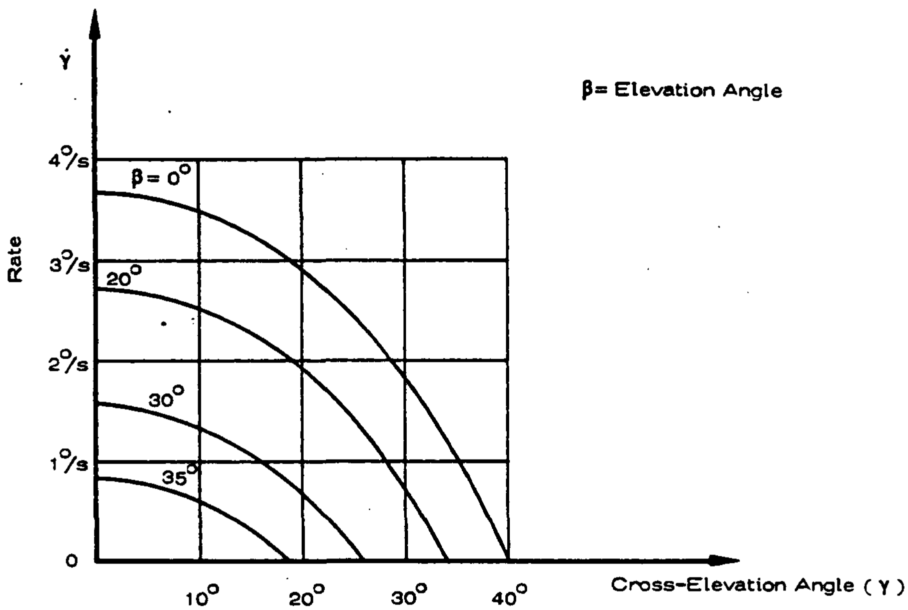

Figure 4.8 - 7: Cross-Elevation Rate/Range Profiles for a Nominal 2000 kg Payload

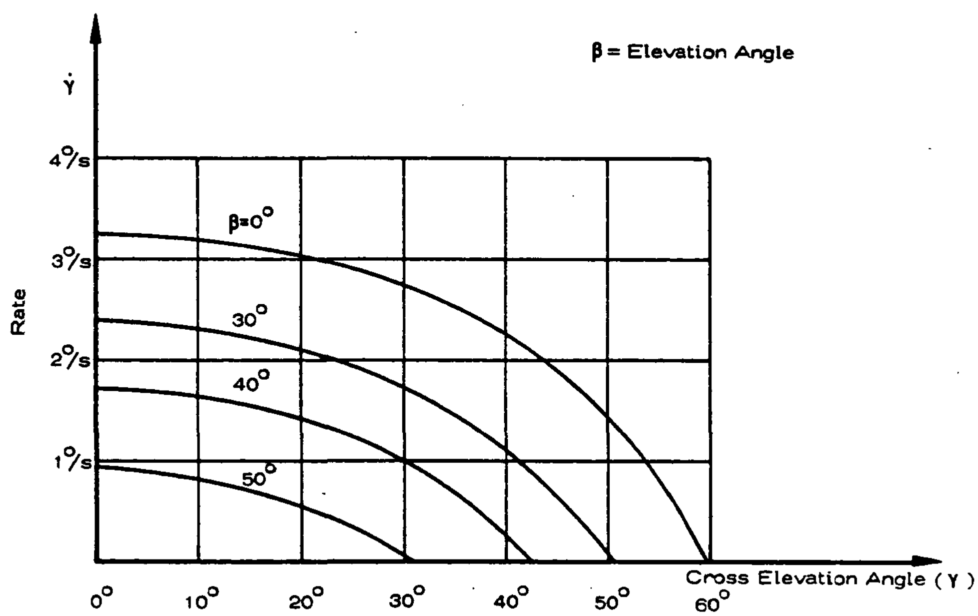

Figure 4.8 - 8a: Cross-Elevation Rate/Range Profiles for a Nominal 200 kg Payload

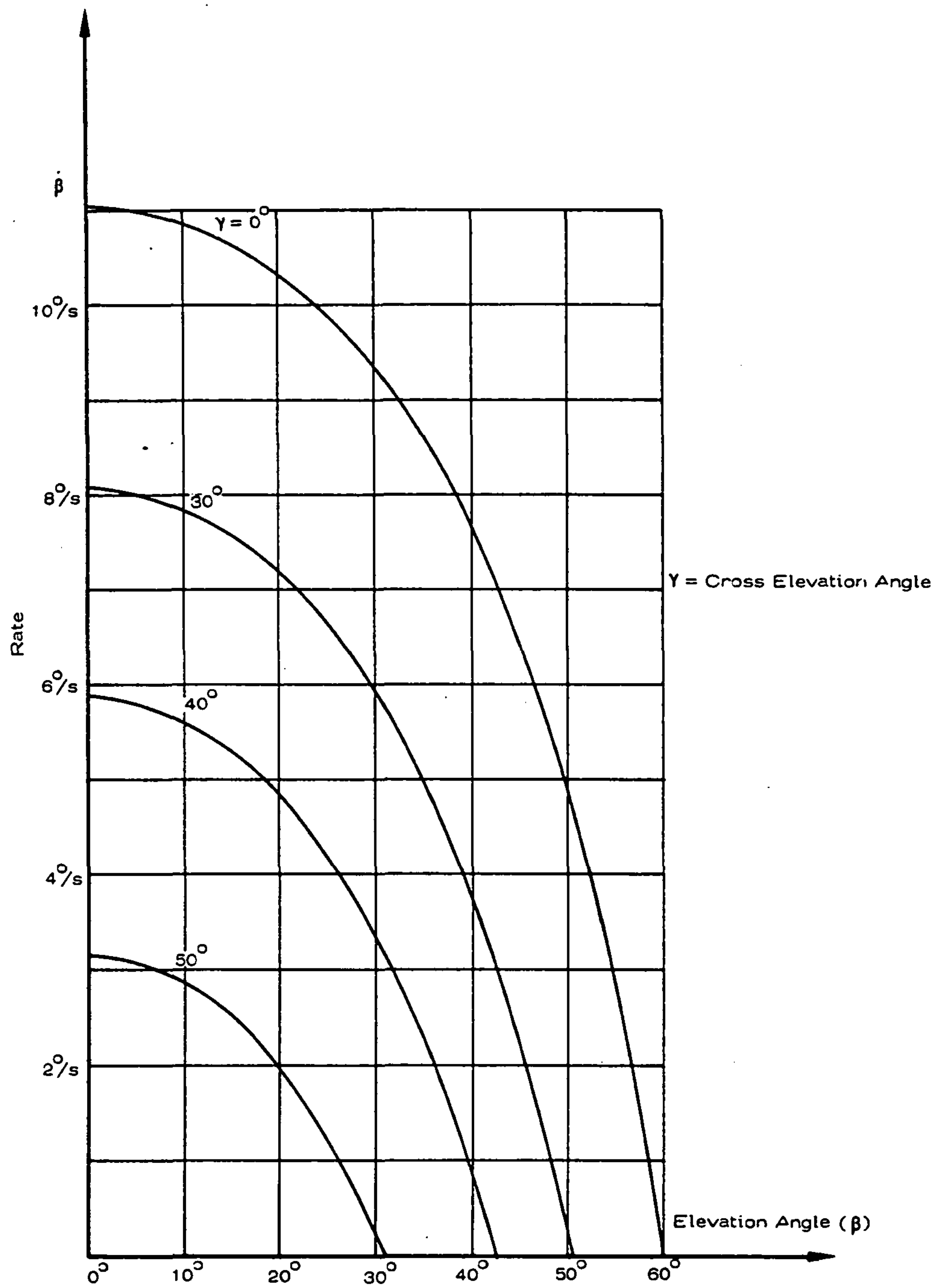

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Figure 4.8 - 8b: Elevation Rate/Range Profiles for a Nominal 200 kg Payload

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4.8.3 IPS Interfaces

4.8.3.1 Spacelab/Orbiter Interfaces

- Software

The IPS software is capable of interfacing via CDMS (S/S computer)with the Orbiter data handling system to allow DCN 012

- 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 and orientation data and timing data. DCN 012

4.8.3.2 Spacelab/Payload Interfaces

- Mechanical Interfaces

The payload interfaces with the gimbal system at three points on the payload attachment ring. DCN 012

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. (Total mass to be carried by payload less than 200 kg)

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- Electrical Interfaces

The IPS provides the interface and the capability to accept experiment generated attitude commands and experiment sensor provided pointing error signals for pointing and alignment purposes

4.8.3.3 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. DCN 012

- 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.

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4.8.3.4 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 retraction being exercised from a separate IPS control panel located on the Orbiter Aft Flight Deck.

4.8.4 Habitability and Cleanliness Requirements

The IPS specific habitability and cleanliness requirements are covered by the general Spacelab system requirements.

4.8.5 Environment

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During orbital operations, the IPS is capable of continuous operation in a completely shadowed configuration. In full solar illumination, some constraints must be imposed on Orbiter attitude, payload shadow configuration or operating time-line, in order to lower component/equipment temperatures (particularily the drive assemblies and RAU's). These constraints have to be evaluated on a case-by-case basis.

4.8.6 Pre-Launch Test and Check-Out

- General Constraints

  Since the IPS is designed to operate in zero gravity, a number of the functions cannot be exercised in the Earth environment. Performance testing of the fully assembled IPS will not be possible prior to each launch. Functional testing which exercises all functions (although not over their complete range) is possible in one-g using IPS GSE but must be performed off-line from Spacelab integration because of the IPS configurations which are required.

- Level IV Integration

  The IPS MGSE supports all the functional tests (see above), and provides for installation of the IPS Gimbal System and the IPS Payload Clamp Assembly (PCA) separately onto pallet elements. Installation of the payload onto the PCA is performed after PCA installation on the pallet. Connections between the Gimbal System and the payload may be made before or after their installation on the pallet. In the former case additional MGSE would be required to handle the connected gimbal system and payload.

  The IPS EGSE provides the capability to functionally test the IPS without the use of any other Spacelab GSE before IPS installation onto the pallet. Use of the EGSE after IPS installation on the pallet is TBD. The IPS EGSE does not verify the experiment-dedicated equipment mounted on the IPS; this must be performed as part of functional testing of the payload. Functional testing of the payload cannot be performed by the IPS EGSE, but may be performed without activating the IPS.

The location and responsibility for level IV integration activities with the IPS will be covered in Section 6.0.

- 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 level III payload integration, augmented by IPS subsystem GSE as required.

- Level II Payload Integration

  After integration of the experiment (including IPS) with the Spacelab, 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. DCN 012

  Functional testing of IPS at this stage of integration is considered desirable within the constraint that only level II 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. No IPS subsystem GSE or Spacelab System GSE is required for this task.

  Functional testing of IPS at this stage of integration is considered 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.

4.8.7 Software

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 monitoring, and for normal on-orbit operations during the different IPS operational modes. DCN 012

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. For development of the IPS, a three-axis IPS simulation will be used, which includes the effects of payload physical and structural characteristics.

4.8.8 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 foreseen.

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. The IPS will be designed to accept steering signals from a hand-controller (joy stick) through a subsystem RAU .

4.8.8.1 Operating Modes

The IPS is capable of being operated in the following modes:

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- stowed mode
- on-orbit monitoring
- acquisition of a reference (guide) star or the sun.

  The probability of achieving 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.
- intertially fixed fine pointing (including solar pointing) for periods of up 90 minutes
- solar off-set pointing over an angular range of at least ± 25 arc-min with respect to the diameter 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 field of view.

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- gimbal hold mode, IPS kept fixed relative to Orbiter coordinates.
- manually controlled slewing using either the CDMS keyboard or hand-controller interfaced to a Subsystem RAU in the SL module or Orbiter AFD.

- 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_o-local vertical orientation (within the IPS torque limitation of 20 N m per axis).

- experiment control, IPS accepting experiment generated attitude commands etc.- see para 4.8.2.7. DCN 012

- emergency retraction into stowed configuration

- emergency jettisoning

4.8.8.2 Contingency Control

The IPS includes 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. Contingency control will provide for: DCN 012

a) payload retraction and locking with positive indications

b) safe jettison of equipment which constitutes a hazard

Provision will be made to perform these contingency control functions from the Orbiter aft flight deck, after contingency evacuation of the Spacelab module has occurred with the IPS left in a normal operating mode, and without use of the CDMS console. DCN 012

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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. Unless otherwise indicated, the environments are applicable to payload equipment wherever it is mounted in the module or on the pallet. Payload equipment in the Orbiter aft flight deck will be exposed to an environment similar to that inside the module except where indicated.

5.1 Mechanical Environment

5.1.1 Flight Environment

5.1.1.1 General Description of Mechanical Environment

Two types of mechanical environments have to be considered:

- static and low frequency transient accelerations
- high frequency random and acoustic excitations

5.1.1.1.1 Static and Low Frequency Transient Accelerations

From the present knowledge of the shuttle dynamic behaviour it can be said that, for all flight events except lift-off and landing, the overall vehicle behaves as a rigid body with a centre of rotation around station Xo 1080. For lift-off and landing current calculations show that significant low frequency transient accelerations will occur simultaneously in three different axes due to wind gust, unsymmetric SRB thrust, unsymmetric landing, descent speed etc. The levels of the accelerations on a specific payload component depends on the response of the Spacelab inside the Orbiter cargo bay and the response of the component/equipment inside Spacelab.

5.1.1.1.2 High Frequency Excitation

- **Random Excitation**

The second major environment is the random vibration. These random vibrations are created by the response of the module shell or the pallet panels to the acoustic noise inside the cargo bay. The vibrations of the shell or pallet panels are transmitted through the support structures (racks, hardpoints etc.) to the payload equipment. The accelerations that the equipment has to withstand depend on its own dynamic behaviour and its location inside Spacelab.

It should be noted that this environment occurs simultaneously with static and low frequency transient lift-off accelerations.

- **Acoustic Noise**

The highest acoustic environment occurs at lift-off and, to a lesser extent, during transonic flight. The external acoustic noise is attenuated first by the orbiter mid fuselage. The cargo bay acoustic noise is further attenuated by the module shell.

5.1.1.2 Sinusoidal Vibration

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 on the payload equipment may be accounted for by a swept sinusoidal vibration environment applied to the equipment at its interface with supporting structure. The sinusoidal environment is an arbitrary environment serving to envelope transient events; it is not additive to any other dynamic or quasi-static induced environment.

- Assembled Payload Level

  Assuming 50 flights, the equivalent environment is as follows:

Frequency Range:	5 - 35 Hz
Level	± 0.25 g 0 to peak
Sweep Rate:	1 Oct/min. (1 sweep up and down)
Axes	3 principal axes

- Equipment/Box Level

  The equipment 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 AT 3 OCT/Min	
Assuming 1 - 10 flights	1 Sweep up and down for each axis	
Assuming 10 - 50 flights	Repeat the sweep for each additional increment of 10 flights (or portions thereof)	

5.1.1.3 Random Vibration

Maximum vibration levels occur during the launch phase and exist for approximately 6 seconds per mission (at lift-off). Equipment within the Spacelab module or on the pallet 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 or pallet structure.

The vibration level for any particular equipment depends on the location, the mounting configuration and the equipment mass. Tables 5 - 2, 5 - 3 and 5 - 4 show the random vibration environment for equipment mounted in various locations in the module, aft flight deck and pallet respectively.

5.1.1.3.1 Aft Flight Deck

The random vibration environment for equipment mounted in the Aft Flight Deck is given in Table 5 - 2 .

Table 5 - 2: Random Vibration Environment for Aft Flight Deck Mounted Equipment

LOCATION	FREQUENCY	LEVEL
Input to equipment mounted in AFD	20 - 150 Hz	+ 6.00 dB/oct
	150 - 900 Hz	0.09 g^2/Hz
	900 - 2000 Hz	- 9.00 dB/oct
	Composite	10.2 g RMS

5.1.1.3.2 Module

The random vibration environment for module mounted equipment is given in Table 5-3.

Table 5 - 3: Random Vibration Environment for Module Mounted Equipment

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LOCATION	FREQUENCY	LEVEL
Input to Racks at Rack / Floor Interface all directions independent of mass loading	20 Hz 20 Hz - 100 Hz 100 Hz - 500 Hz 500 Hz - 2000 Hz 2000 Hz Composite:	0.00008 g^2/Hz + 9 db/oct 0.01 g^2/Hz - 6 db/oct 0.00063 g^2/Hz 2.85 g RMS
Input to Racks at Overhead Support Structure/ Rack Bracket Interface		
X-, Y-Directions independent of mass loading	20 Hz 20 Hz - 180 Hz 180 Hz - 600 Hz 600 Hz - 2000 Hz 2000 Hz Composite:	0.000035 g^2/Hz + 9 db/oct 0.025 g^2/Hz - 9 db/oct 0.00068 g^2/Hz 4.3 g RMS
Z-Direction independent of mass loading	20 Hz 20 Hz - 100 Hz 100 Hz - 600 Hz 600 Hz - 2000 Hz 2000 Hz Composite:	0.00009 g^2Hz + 12 db/oct 0.055 g^2/Hz - 9 db/oct 0.0015 g^2/Hz 5.61 g RMS
Input to Rack Mounted Experiment Equipment at Rack Primary Structure Interface all directions independent of mass loading	20 Hz 20 Hz - 120 Hz 120 Hz - 400 Hz 400 Hz - 2000 Hz 2000 Hz Composite:	0.0046 g^2/Hz + 4 db/oct 0.05 g^2/Hz - 4 db/oct 0.0059 g^2/Hz 6.45 g RMS
Input to Components Mounted on Top Panel 1.3 m Diameter Ring at Ring/Component Interface		
X-, Y-Directions independent of component mass	20 Hz 20 Hz - 120 Hz 120 Hz - 300 Hz 300 Hz - 2000 Hz 2000 Hz Composite:	0.000094 g^2/Hz + 9 db/oct + 0.02 g^2/Hz - 6 db/oct 0.00046 g^2/Hz 3.06 g RMS
Z-Direction dependent on component mass Component Mass ≥ 50 kg but < 150 kg	20 Hz 20 Hz - 150 Hz 150 Hz - 220 Hz 220 Hz - 2000 Hz 2000 Hz Composite:	0.00024 g^2/Hz + 12 db/oct 0.75 g^2/Hz - 10 db/oct 0.00049 g^2/Hz 12.07 g RMS
Component mass ≥ 150 kg but < 250 kg	20 Hz 20 Hz - 128 Hz 128 Hz - 220 Hz 220 Hz - 2000 Hz 2000 Hz Composite:	0.00024 g^2/Hz + 12 db/oct 0.4 g^2/Hz - 10 db/oct 0.00026 g^2/Hz 9.2 g RMS
Component mass ≥ 250 kg	20 Hz 20 Hz - 117 Hz 117 Hz - 220 Hz 220 Hz - 2000 Hz 2000 Hz Composite:	0.00024 g^2/Hz + 12 db/oct 0.28 g^2/Hz - 10 db/oct 0.00018 g^2/Hz 7.86 g RMS

Table 5 - 3: Random Vibration Environment for Module Mounted Equipment (cont'd)

LOCATION	FREQUENCY	LEVEL
Input to Experiment Equipment Attached to Module Overhead Structure at Overhead Structure Interface		
X-, Y-Directions	20 Hz	0.0002 g^2/Hz
independent of equipment mass	20 Hz - 100 Hz	+ 12 db/oct
	100 Hz - 400 Hz	0.12 g^2/Hz
	400 Hz - 2000 Hz	- 6 db/oct
	2000 Hz	0.0049 g^2/Hz
	Composite:	8.78 g RMS
Z-Direction	20 Hz	0.00041 g^2/Hz
independent of equipment mass	20 Hz - 100 Hz	+ 12 db/oct
	100 Hz - 500 Hz	0.25 g^2/Hz
	500 Hz - 2000 Hz	- 6 db/oct
	2000 Hz	0.016 g^2/Hz
	Composite:	14.13 g RMS
Input to Experiment Equipment Mounted on experiment subfloor (not SL baseline) at Subfloor Interface		
X-Direction	20 Hz	0.000036 g^2/Hz
independent of equipment mass	20 Hz - 150 Hz	+ 9 db/oct
	150 Hz - 600 Hz	+ 0.015 g^2/Hz
	600 Hz - 2000 Hz	- 6 db/oct
	2000 Hz	0.0014 g^2/Hz
	Composite:	3.70 g RMS
Y-Direction	20 Hz	0.000040 g^2/Hz
independent of equipment mass	20 Hz - 120 Hz	+ 12 db/oct
	120 Hz - 300 Hz	0.05 g^2/Hz
	500 Hz - 2000 Hz	- 9 db/oct
	2000 Hz	0.00079 g^2/Hz
	Composite:	5.55 g RMS
Z-Direction Dependent on loading conditions within a confined subfloor area of 0.33 m^2		
Z-Direction	20 Hz	0.00016 g^2/Hz
For total equipment mass	20 Hz - 150 Hz	+ 12 db/oct
< 15 kg within 0.33 m^2	150 Hz - 460 Hz	0.5 g^2/Hz
subfloor area and for anti-	460 Hz - 2000 Hz	- 12 db/oct
vibration mounted equipment	2000 Hz	0.0014 g^2/Hz
	Composite:	15.68 g RMS
Z-Direction	20 Hz	0.00012 g^2/Hz
For total equipment mass	20 Hz - 80 Hz	+ 12 db/oct
≥ 15 kg but ≤ 30 kg	80 Hz - 200 Hz	0.03 g^2/Hz
within 0.33 m^2 subfloor area	200 Hz - 300 Hz	+ 9 db/oct
	300 Hz - 688 Hz	0.1 g^2/Hz
	688 Hz - 2000 Hz	- 12 db/oct
	2000 Hz	0.0014 g^2/Hz
	Composite:	8.42 g RMS
Z-Direction	20 Hz	0.000025 g^2/Hz
For total equipment mass	20 Hz - 100 Hz	+ 12 db/oct
≥ 30 kg within 0.33 m^2	100 Hz - 200 Hz	0.015 g^2/Hz
subfloor area	200 Hz - 300 Hz	+ 9 db/oct
	300 Hz - 820 Hz	0.05 g^2/Hz
	820 Hz - 2000 Hz	- 12 db/oct
	2000 Hz	0.0014 g^2Hz
	Composite:	5.30 g RMS

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Table 5 - 3: Random Vibration Environment for Module Mounted Equipment (cont'd)

LOCATION	FREQUENCY	LEVEL
Input to Experiment Equipment Mounted on Module Floor at Floor Interface		
Input to experiments mounted on Floor attach Points X-Direction independent of mass loading	20 Hz 20 Hz - 100 Hz 100 Hz - 500 Hz 500 Hz - 2000 Hz 2000 Hz Composite:	0.0004 g^2/Hz + 6 db/oct 0.01 g^2/Hz - 6 db/oct 0.00063 g^2/Hz 2.84 g RMS
Input to experiments mounted on Floor attach Points Y-Direction independent of mass loading	20 Hz 20 Hz - 60 Hz 60 Hz - 150 Hz 150 Hz - 210 Hz 210 Hz - 500 Hz 500 Hz - 2000 Hz 2000 Hz Composite	0.00085 g^2/Hz + 12 db/oct 0.07 g^2/Hz - 12 db/oct 0.016 g^2/Hz - 6 db/oct 0.001 g^2/Hz 4.46 g RMS
Input to experiments mounted on floor attach points Z-Direction Dependent on experiment and/or equipment mass Input to experiments mounted on floor attach points 1 Mass of equipment < 50 kg/m	20 Hz 20 Hz - 200 Hz 200 Hz - 800 Hz 800 Hz - 2000 Hz 2000 Hz Composite:	0.0001 g^2/Hz + 9 db/oct 0.1 g^2/Hz - 9 db/oct 0.0065 g^2/Hz 9.94 g RMS
Mass of equipment ≥ 50 kg/m but < 100 kg/m	20 Hz 20 Hz - 160 Hz 160 Hz - 800 Hz 800 Hz - 2000 Hz 2000 Hz Composite:	0.0001 g^2/Hz + 9 db/oct 0.05 g^2/Hz - 9 db/oct 0.0032 g^2/Hz 7.13 g RMS
Mass of equipment ≥ 100 kg/m but < 200 kg	20 Hz 20 Hz - 140 Hz 140 Hz - 800 Hz 800 Hz - 2000 Hz 2000 Hz Composite:	0.0001 g^2/Hz + 9 db/oct 0.033 g^2/Hz - 9 db/oct 0.0021 g^2/Hz 5.83 g RMS
Mass of equipment ≥ 200 kg/m	20 Hz 20 Hz - 120 Hz 120 Hz - 800 Hz 800 Hz - 2000 Hz 2000 Hz Composite:	0.0001 g^2/Hz + 9 db/oct 0.02 g^2/Hz - 9 db/oct 0.0013 g^2/Hz 4.58 g RMS
Input to Experiments mounted on Floor Attach Points Floor panel unloaded	20 Hz 20 Hz - 80 Hz 80 Hz - 300 Hz 300 Hz - 400 Hz 400 Hz - 800 Hz 800 Hz - 2000 Hz 2000 Hz Composite:	0.0003 g^2/Hz + 15 db/oct 0.3 g^2/Hz - 12 db/oct 0.1 g^2/Hz - 9 db/oct 0.0065 g^2/Hz 12.71 g RMS

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Table 5 - 3: Random Vibration Environment for Module Mounted Equipment (cont'd)

LOCATION	FREQUENCY	LEVEL
Input to experiment mounted on the airlock experiment table (independent of mass loading)		
X-, Y-Directions	20 - 33 Hz	0.0025 g^2/Hz
	33 - 76 Hz	+ 9 db/oct
	76 - 260 Hz	0.03 g^2/Hz
	260 - 2000 Hz	- 6 db/oct
	2000 Hz	0.00051 g^2/Hz
	Composite:	3.60 g RMS
Z-Direction	20 Hz	0.0023 g^2/Hz
	20 - 80 Hz	+ 12 db/oct
	80 - 220 Hz	0.6 g^2/Hz
	220 - 2000 Hz	- 10 db/oct
	2000 Hz	0.00038 g^2/Hz
	Composite:	12.2 g RMS

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5.1.1.3.3. Pallet

The random vibration environment for pallet mounted equipment is given in Table 5-4.

Table 5 - 4 : Random Vibration Environment for Pallet Mounted Equipment

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LOCATION	FREQUENCY	LEVEL
Input to Experiments mounted on Pallet Hard Points at Hardpoint interface		
All directions Independent of mass loading	20 Hz 20 Hz - 150 Hz 150 Hz - 600 Hz 600 Hz - 2000 Hz 2000 Hz Composite:	0.00024 g^2/Hz + 9 db/oct 0.1 g^2/Hz - 9 db/oct 0.0027 g^2/Hz 8.72 g RMS
Input to Experiments mounted on Pallet Panels at Panel interface		
Out of plane motion Panel loaded by < 8 kg	20 Hz 20 Hz - 150 Hz 150 Hz - 250 Hz 250 Hz - 2000 Hz 2000 Hz Composite:	0.0098 g^2/Hz + 12 db/oct 30 g^2/Hz - 6 db/oct 0.47 g^2/Hz 102.27 g RMS
Out of plane motion Panel loaded by < 10 kg/m^2 or ≥ 8 kg but < 15 kg per panel	20 Hz 20 Hz - 80 Hz 80 Hz - 450 Hz 450 Hz - 2000 Hz 2000 Hz Composite:	0.0098 g^2/Hz + 12 db/oct 2.5 g^2/Hz - 9 db/oct 0.028 g^2/Hz 34.5 g RMS
Out of plane motion Panel loaded by ≤ 25 kg/m^2 or ≥ 15 kg but < 30 kg per panel	20 Hz 20 Hz - 45 Hz 45 Hz - 120 Hz 120 Hz - 150 Hz 150 Hz - 700 Hz 700 Hz - 2000 Hz 2000 Hz Composite:	0.0098 g^2/Hz + 12 db/oct 0.25 g^2/Hz + 15 db/oct 0.7 g^2/Hz - 9 db/oct 0.03 g^2/Hz 25.5 g RMS
Out of plane motion Panel loaded by < 50 kg/m^2 or ≥ 30 kg per panel	20 Hz 20 Hz - 25 Hz 25 Hz - 77 Hz 77 Hz - 150 Hz 150 Hz - 700 Hz 700 Hz - 2000 Hz 2000 Hz Composite:	0.0098 g^2/Hz + 12 db/oct 0.025 g^2/Hz + 15 db/oct 0.7 g^2/Hz - 9 db/oct 0.03 g^2/Hz 24.9 g RMS

Table 5 - 4: Random Vibration Environment for Pallet Mounted Equipment

LOCATION	FREQUENCY	LEVEL
In plane motion Independent of mass loading	20 Hz 20 Hz - 150 Hz 150 Hz - 550 Hz 550 Hz - 2000 Hz 2000 Hz Composite:	0.00003 g^2/Hz + 12 db/oct 0.1 g^2/Hz - 12 db/oct 0.0006 g^2/Hz 7.82 g RMS
Input to Experiments mounted on Pallet Cold Plates at Cold plate interface		
Out of plane motion Cold plate loaded by ≥ 10 kg but < 20 kg	20 Hz 20 Hz - 135 Hz 135 Hz - 450 Hz 450 Hz - 2000 Hz 2000 Hz Composite:	0.00054 g^2/Hz + 12 db/oct 1.0 g^2/Hz - 9 db/oct 0.011 g^2/Hz 23.7 g RMS
Out of plane motion Cold plate loaded by ≥ 20 kg but < 40 kg	20 Hz 20 Hz - 120 Hz 120 Hz - 450 Hz 450 Hz - 2000 Hz 2000 Hz Composite:	0.00054 g^2/Hz + 12 db/oct 0.7 g^2/Hz - 9 db/oct 0.008 g^2/Hz 19.9 g RMS
Out of plane motion Cold plate loaded by ≥ 40 kg	20 Hz 20 Hz - 110 Hz 110 Hz - 450 Hz 450 Hz - 2000 Hz 2000 Hz Composite:	0.00054 g^2/Hz + 12 db/oct 0.5 g^2/Hz - 9 db/oct 0.0057 g^2/Hz 17 g RMS
In plane motion Cold plate loaded by ≥ 10 kg but < 20 kg	20 Hz 20 Hz - 150 Hz 150 Hz - 400 Hz 400 Hz - 2000 Hz 2000 Hz Composite:	0.0001 g^2/Hz + 12 db/oct 0.35 g^2/Hz - 12 db/oct 0.0006 g^2/Hz 12.0 g RMS
In plane motion Cold plate loaded by ≥ 20 kg but < 40 kg	20 Hz 20 Hz - 150 Hz 150 Hz - 400 Hz 400 Hz - 2000 Hz 2000 Hz Composite:	0.00008 g^2/Hz - 12 db/oct 0.24 g^2/Hz - 12 db/oct 0.0004 g^2/Hz 9.96 g RMS
In plane motion Cold plate loaded by ≥ 40 kg	20 Hz 20 Hz - 150 Hz 150 Hz - 400 Hz 400 Hz - 2000 Hz 2000 Hz Composite:	0.00005 g^2/Hz + 12 db/oct 0.15 g^2/Hz - 12 db/oct 0.0002 g^2/Hz 7.82 g RMS

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5.1.1.4 Acoustic Noise

The acoustic sound pressure field inside the cargo bay can be considered a reverberant one. The spectrum varies during launch and ascent with respect to level and shape. The maximum acoustic level occurs for a few seconds during main engine run up and launch and the time history of the overall sound pressure level in the Orbiter cargo bay is given in Figure 5 - 1. The acoustic spectrum in the Orbiter cargo bay is given in Figure 5 - 2 and Table 5 - 5. The overall acoustic level (OAL) is related to the sound pressure level (SPL) in each frequency band by the expression:

$$\text{OAL (dB)} = 10 \ \log_{10} \quad \Sigma \ e^{\frac{SPL}{10} \log_{10} 10}$$

Aerodynamic noise during entry is significantly less than the levels during lift-off through ascent

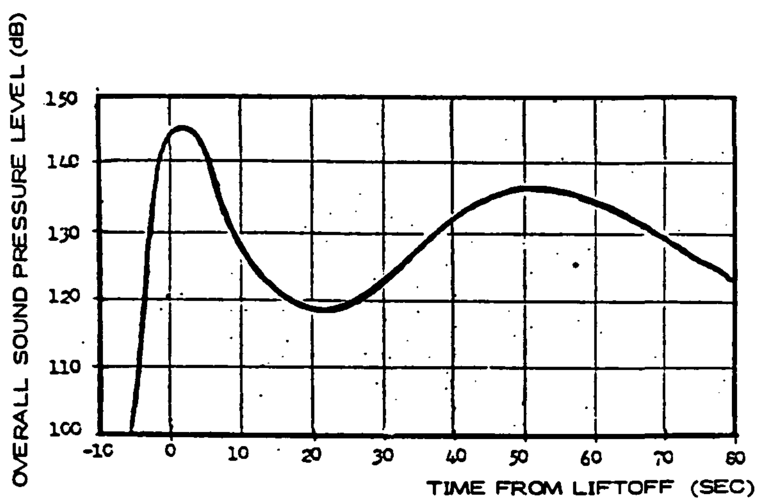

Figure 5 - 1: Orbiter Cargo Bay
Internal Acoustic Time History

Table 5 – 5: Orbiter Cargo Bay Internal Acoustic Noise

1/3 Octave Band Center Frequency (Hz)	Sound Pressure Level (dB) re. 2×10^{-5} N/m^2 Lift-Off	Sound Pressure Level (dB) re. 2×10^{-5} N/m^2 Aeronoise
31.5	120.0	112.0
40.0	122.0	114.0
50.0	124.0	116.0
63.0	126.0	118.0
80.0	128.0	120.0
100.0	130.0	122.0
125.0	132.0	124.0
160.0	133.0	125.0
200.0	134.5	126.5
250.0	135.0	127.0
320.0	135.0	127.0
400.0	135.0	127.0
500.0	135.0	127.0
530.0	133.0	125.0
830.0	131.0	123.0
1,000.0	127.0	119.0
1,250.0	126.0	118.0
1,600.0	125.0	117.0
2,000.0	123.0	115.0
2,500.0	121.0	113.0
3,200.0	119.0	111.0
4,000.0	117.0	109.0
5,000.0	115.0	107.0
6,300.0	113.0	105.0
8,000.0	110.0	102.0
10,000.0	109.0	101.0
OAL	145.0	137.0

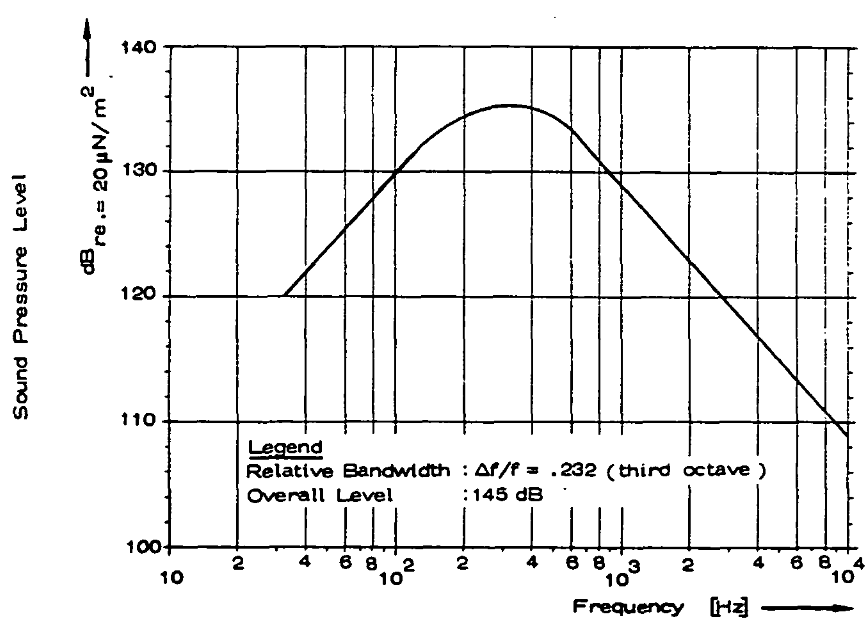

Figure 5 – 2 : Analytical Prediction of the Acoustic Spectrum in the Orbiter Cargo Bay

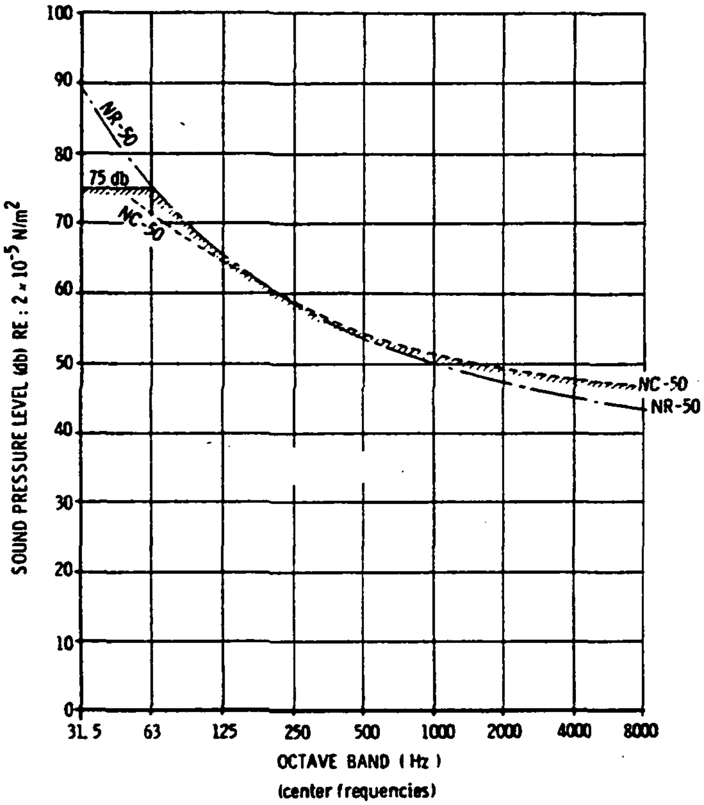

Figure 5 - 3: Acoustic Noise Inside the Module (On Orbit)

5.1.1.4.1 Module

The acoustic level inside the module during lift-off and ascent is attenuated by the module shell and is given in Table 5 - 6.

The acoustic noise level on-orbit results from the operation of Spacelab pumps, fans etc.
The maximum noise level in the module working area, measured at crew member head position at the standard work bench, during orbital operation will not exceed level NR-50 of the International Organization of Standardization (ISO) Noise Rating, or NC-50 of the United States Noise Criteria Standard, whichever is higher, except that the noise levels in the octave bands of 63 Hz and below are limited to a maximum of 75 dB.
These noise levels are shown in Figure 5 - 3.

Table 5 - 6: Module Internal Acoustic Noise Spectra (Lift-off, Ascent)

1/3 Octave Band Center Frequency (Hz)	Sound Pressure Level (dB) re. 2×10^{-5} N/m^2 Lift-Off	Sound Pressure Level (dB) re. 2×10^{-5} N/m^2 Aeronoise
31.5	104	96
40.0	107	99
50.0	110	102
63.0	113	105
80.0	116	108
100.0	119	111
125.0	121.5	113.5
160.0	123	115
200.0	125	117
250.0	126	118
320.0	126.5	118.5
400.0	125.5	117.5
500.0	124.5	116.5
630.0	121	113
800.0	118	110
1000.0	112.5	104.5
1250.0	109.5	101.5
1600.0	106.5	98.5
2000.0	102.5	94.5
2500.0	99	91
3200.0	97	89
4000.0	95	87
5000.0	93	85
6300.0	91	83
8000.0	89	81
10000.0	88	80
OAL	134	126

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5.1.1.4.2 Aft Flight Deck

The acoustic level inside the Orbiter aft flight deck is attenuated by the surrounding structure and is given in Table 5 - 7.

Table 5 - 7: Aft Flight Deck Acoustic Environment

1/3 Octave Band Center Frequency (Hz)	Sound Pressure Level - dB re. 2×10^{-5} N/m^2	
	Liftoff	Transonic
31.5	129.0	121.0
40.0	128.5	120.5
50.0	128.5	120.5
63.0	128.0	120.0
80.0	126.5	118.5
100.0	125.0	117.0
125.0	122.5	114.5
160.0	120.0	112.0
200.0	116.0	108.0
250.0	112.0	104.0
315.0	108.0	100.0
400.0	103.0	95.0
500.0	98.0	90.0
630.0	94.5	86.5
800.0	90.0	82.0
1000.0	85.0	77.0
1250.0	81.0	73.0
1600.0	77.0	69.0
2000.0	72.0	64.0
OAL	136.0	128.0

The acoustic noise level on-orbit is TBD.

5.1.1.4.3 Pallet

The acoustic level for equipment mounted at any location on the pallet is given in Table 5-5.

5.1.1.5 Shock

5.1.1.5.1 Module

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 (see 5.1.2).

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. Maximum crew-induced design limit loads are given in Table 5 - 8 . DCN 001

Table 5 - 8: Crew-Induced Limit Loads

Crew System or Structure	Type of Load	Limit Load	Direction of Load	Allowable Deflection (mm)
Tether	Concentrated load, pull (tension)	550 N (125 lbs)	Any direction	N/A
Tether attach point	Concentrated load, pull (tension)	550 N (125 lbs)	Any direction	N/A
Handholds/Handrails	Concentrated load on most critical 5 cm of member to be grasped	550 N (125 lbs)	Any direction	13
Foot restraint (each)	Concentrated load, pull	445 N (100 lbs)	Any possible direction	N/A
	Torsion	200 Nm (125 ft lbs)	Torsion vector normal to floor	
Levers, handles, operating wheels	Push or pull concentrated on most extreme tip or edge Lateral handles force	200 N (45 lbs) 110 N (25 lbs)	Any possible direction	N/A
Small knobs	Twist (torsion)	15 Nm (11 ft lbs)	Any possible direction	N/A
Cabinets, and any normally exposed equipment	Concentrated load-applied by flat round surface with an area of 20 $\pm$ 1.5 cm^2	550 N (125 lbs)	Any direction	N/A

5.1.1.5.2 Pallet

On-orbit, 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 5 mm 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 should be capable of operating normally after such loads have occurred. For missions in which EVA would only occur as an emergency activity, pallet-mounted equipment need only be capable of surviving these loads intact.

5.1.1.6 Acceleration

5.1.1.6.1 Nominal Mission/Emergency Sequence

Limit load tables envelope maximum low frequency accelerations corresponding to the Spacelab response to known Orbiter forcing functions.
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 are being established by means of coupled Shuttle/Spacelab dynamic analysis.
The maximum expected accelerations (limit load factors) for payloads during ascent/descent are given in Table 5 - 9.

Table 5 - 9: Payload Limit Linear Accelerations(g's) and Angular Accelerations (RAD/SEC^2)

Condition	$\ddot{X}_o$	$\ddot{Y}_o$	$\ddot{Z}_o$	$\ddot{\Phi}$	$\ddot{\theta}$	$\ddot{\psi}$
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	± 0.5	± 0.1
Orbiter Max. Load	- 3.0 ± 0.26	± 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

DCN 041 Note 1: X, Y, Z are defined in Figure 2 - 1

Note 2: The linear and angular accelerations quoted for each case have to be superimposed.

Note 3: Nominal center of rotation of Orbiter: X_o = 1080"; y_o = 0"; z_o = 400"

DCN 041 Note 4: The angular accelerations are referred to the nominal center of rotation

The accelerations during lift-off and landing will vary according to the payload location within Spacelab since the overall Shuttle/Spacelab combination does not behave as a rigid body during these mission phases (see 5.1.1.1.1.). Preliminary estimates (resulting from coupled analyses) of these accelerations are available for some conditions.

Table 5 - 10 shows estimated levels in the module for payload equipments which have resonant frequencies above 25 Hz, when mounted directly to module primary structure, or, above 35 Hz when attached to secondary structure such as a rack, airlock or on the center aisle floor.

Table 5 - 10: Module-mounted Payload Limit Linear Accelerations (g's) During Lift-off and Landing

Acceleration / condition	X (g)	Y (g)	Z (g)
Lift-off	+ 0.1 - 3.0	± 3.3	+ 3.3 - 2.7
Landing	+ 4.0 - 3.5	± 5.0	+ 7.6 - 4.5

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Note: The linear accelerations quoted for each condition have to be superimposed and combined with the random loads for design loads, as given for the various locations in Appendix B - Structure Interface Definition.

Table 5 - 11 shows estimated levels for pallet payload equipments which have resonant frequencies above 25 Hz, when attached directly to pallet primary structure, or, above 35 Hz when attached to secondary structures.

Table 5 - 11: Pallet-mounted Payload Limit Linear Accelerations (g's) and Angular Accelerations (rad/s^2) During Lift-off and Landing

Acceleration / condition	$\ddot{X}$ (g)	$\ddot{Y}$ (g)	$\ddot{Z}$ (g)	$\ddot{\Phi}$ rad/s^2	$\ddot{\Theta}$ rad/s^2	$\ddot{\Psi}$ rad/s^2
Lift-Off	+ 2.11 - 4.3	± 1.4	+ 5.5 - 6.1	± 1.9	± 5.8	± 0.7
Landing	± 4.0	± 1.0	+ 6.6 - 4.0	± 3.4	± 6.0	± 1.2

Notes: Linear and angular accelerations have to be superimposed and combined with the random loads for design loads, as given for the various locations in Appendix B - Structure Interface Definition.
The nominal center of rotation shall be taken at the payload CG.

Steady state emergency landing accelerations are given in Table 5 – 12. 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). Orbiter crew compartment interior emergency landing accelerations are also included on this table, being applicable to mounting structure for equipment in the AFD.

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Emergency landing accelerations act separately and are ultimate. The longitudinal accelerations are directed within a cone of 20° 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 emergency landing. For emergency landing acceleration loads, however, para 7.2.3, Experiment Integrity, is applicable.

Table 5 – 12: Ultimate Emergency Landing Accelerations

	TOTAL ACCELERATIONS (g)					
Directions	+ X	– X	+ Y	– Y	+ Z	– Z
Emergency Landing Loads	4.5	1.50	1.50	1.50	4.5	2.0
Emergency Landing Orbiter cabin interior	20.0	3.30	3.30	3.30	10.0	4.4

DCN 015

Accelerations to act independently

5.1.1.6.2 On-Orbit Maneuvers

During normal Orbiter attitude-control activities, thrusting of the Orbiter RCS will cause slight acceleration to be exerted on payload equipment depending on its location with respect to the center of rotation. Values are given in Table 5 - 13 for the RCS thrusters. The values shown are based on an Orbiter prior to the deorbit burn with a 14 515 kg (32 000 lb) cargo.

All three angular accelerations may occur simultaneously and the linear acceleration at any location in Spacelab may be calculated based on the distance from the Orbiter's center of gravity, the location of which will vary to some extent with the particular payload weight distribution. The typical location of the Orbiter's center of gravity without Spacelab and its payload is given in para 2.2.7.

Table 5 - 13: Orbiter RCS Maximum Acceleration Levels (Reference: ICD-2-1900) DCN 041

Direction	Translational, m/sec^2 (ft/sec^2) 1)					Rotational, degrees/sec^2 2)			
RCS System	$+\ddot{x}_o$	$-\ddot{x}_o$	$\pm\ddot{y}_o$	$+\ddot{z}_o$	$-\ddot{z}_o$	$\pm\ddot{\phi}$	$+\ddot{\theta}$	$-\ddot{\theta}$	$\pm\ddot{\psi}$
Primary Thruster	0.18 (0.6)	0.16 (0.5)	0.22 (0.7)	0.4 (1.3)	0.34 (1.1)	1.2	1.4	1.5	0.8
Vernier Thrusters	0	0	0.0021 (0.0070)	0	0.0024 (0.0080)	0.04	0.03	0.02	0.02

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 .

5.1.1.6.3 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.

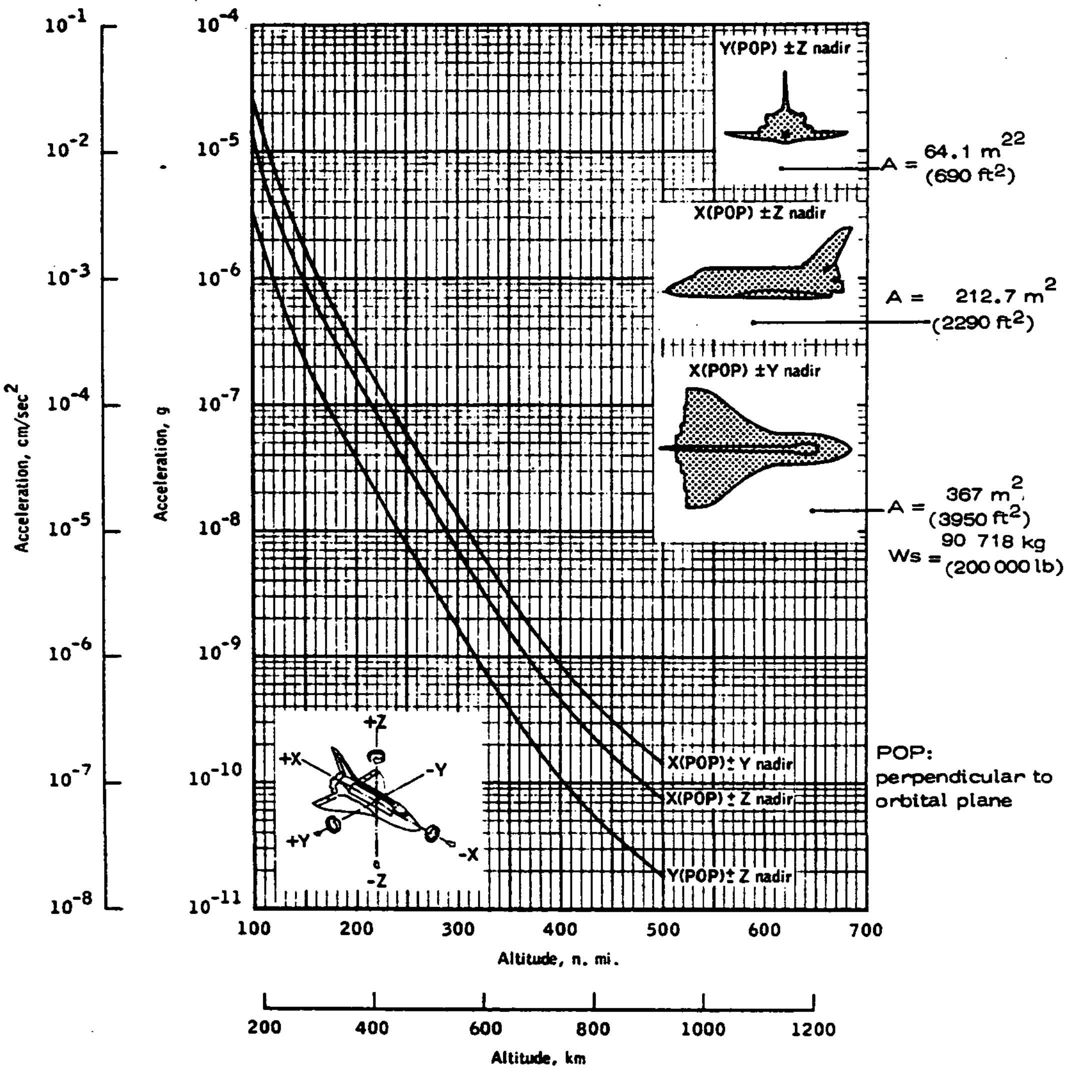

Figure 5 - 4 Effects Of Atmospheric Drag On The Orbiter
(Ref.: Vol. XIV, Rev. E, Figure 3 - 15)

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5.1.2 Ground Environment

This section describes the environments to which experiment equipment may be subjected during transportation, integration, installation and pre-launch check-out activities.

During transportation and handling of experiment flight hardware, ground support equipment may be provided which controls dynamic loads imposed on flight hardware to be lower than those specified for flight operations in paragraph 5.1.1.

For typical operations limit acceleration factors are depicted in Table 5 -14.

Table 5 - 14: Limit Acceleration Factors (g´s) for Ground Handling, Road, Air and Barge Operations
(Ref.: ECP 50514)

Operation	$\ddot{x}_L$	$\ddot{y}_L$	$\ddot{z}_L$	Remarks
• **Handling**				
- Dolly	± 1.0	± 0.75	1.0 ± 0.5	NASA SRB Loads
- Hoisting	0.0	0.0	1.0 ± 0.33	Within 20° Half Cone
- Orbiter Mate	± 0.5	± 0.5	1.0 ± 1.0	NASA SP 8057 1)
• **Truck**				
- Shocks	± 0.7	± 0.8	1.0 ± 1.3	Attenuated by MGSE 40 MPH Top Speed
• **Air**				
- Continuous Vibration	± 0.8	± 0.9	1.0 ± 0.7	Attenuated by MGSE Equivalent Limit of C5A Vibration
• **Barge**				
- Slamming	0.0	0.0	1.0 ± 0.8	NASA SP 8077 2)
- Waves	± 0.3	± 0.5	1.0 ± 0.6	

1)"Structural Design Criteria Applicable to a Space Shuttle"
2)"Transportation and Handling Loads"

Note :

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.

5.2 Thermal Environment

5.2.1 Flight Environment

The thermal environment for payloads during any particular mission is highly dependent on both the mission characteristics (in particular the orbiter attitude) and the composition and operation of the payload itself. In this section will be given a range of thermal environments which lie between extremes of temperature which represent the limits of operation of Spacelab. A preliminary definition of the accommodated payload and the particular mission profile is then needed to arrive at a more precise knowledge of the actual range of temperatures to be experienced by any experiment equipment.

5.2.1.1 Module

During on-orbit operations, the temperature of module mounted payload equipment will be controlled primarily by the forced flow of air in the equipment racks (avionics loop) and the cabin area (cabin air loop). Experiment equipment which is cold plate mounted, or which utilises the experiment heat exchanger via a secondary coolant loop, will be additionally controlled by the temperature of the module water loop and details of these thermal interfaces are contained in section 4.6.

a) On-Orbit Cabin Air Temperature

The cabin air temperature may be adjusted within the range 18 ^{o} to 27 ^{o}C and will be controlled automatically to within $\pm$ 1^{o}C. Constraints on this temperature range may be imposed during certain attitudes because of the effect of heat leakage between the module and the external environment.

b) Ascent/Descent Phase Cabin Temperature

During ascent, descent and landing, the cabin air temperature depends on the extremes of the external temperatures. The estimated temperature limits for these periods are given in Table 5 - 15.

Table 5-15: Cabin Temperature Limits During Ascent/Descent
(Preliminary Results of Thermal Analysis)

OPERATIONAL PHASE	Tmin. (^{o} C)	Tmax. (^{o} C)
Launch/Ascent	5	50
Re-Entry	10	50
Post Landing	10	50

c) Avionics Loop Temperature

Equipment installed inside the avionics racks is cooled by forced air with the following temperature characteristics (estimated values) as shown in Table 5 - 16:

Table 5 - 16: Avionic Rack Forced Air Cooling (Estimated)

Draw-Through Cooling	Maximum Inlet Temperature:	35° C
	Nominal Inlet Temperature:	22° C
	Minimum Inlet Temperature:	10° C
Surface Cooling	Maximum Temperature:	35° C
	Nominal Temperature:	22° C
	Minimum Temperature:	10° C

d) Surface Temperature

Under normal operations, the mean radiant temperature of the habitable module interior will not exceed $30\ ^{\circ}$C and the temperature of any inside wall surface of possible crew contact will not exceed $45\ ^{\circ}$C unless protective guards are provided.

No external equipment surfaces, whether reachable or not, will be cooler than the dew point temperature of the module atmosphere.

5.2.1.2 Airlock

a) Inner Hatch Open

During airlock operations with the inner hatch open the temperature inside the airlock will be similar to that in the module as defined in para 5.2.1.1.

b) Both Hatches Closed

Following closure of the inner hatch and with the outer hatch closed the temperature inside the airlock (assuming no heat dissipation from an experiment) will be similar to that in the module as defined in para 5.2.1.1. The airlock temperature when the outer hatch is re-closed after experiment exposure to Space will depend on the thermal capacity and temperature of the experiment. Repressurisation of the airlock will help to stabilize the temperature in the airlock to approximate to module interior conditions before opening the inner hatch to avoid problems of condensation or temporary overload of the cabin air heat rejection capability.

c) Outer Hatch Open

With the outer hatch open the experiment will be exposed to the space environment specified in para 5.2.1.4.2.2. For extended experiments the radiative environment will be additionally influenced by the thermal properties of the Spacelab surfaces (module exterior, outer hatch, airlock thermal screen etc) in the vicinity of the experiment.

5.2.1.3 Aft Flight Deck

The temperature of payload equipment mounted in the Orbiter aft flight deck will be controlled by the forced flow of cabin air through the equipment. The air inlet temperature will be within the range 18 to 27 °C except for entry.

5.2.1.4 Pallet

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. This section essentially covers only the air temperature environment and the radiator environment. The conductive environment for equipment which interfaces with pallet cold plates is described in para 4.6.

5.2.1.4.1 Launch /Landing 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 temperature 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 - 17.

Table 5 - 17: Orbiter Cargo Bay Air Temperature

CONDITION	Tmin. (°C)	Tmax. (°C)
Purge on Launch Pad	7*	38
Launch/Ascent (estimated values)	7*	38
Re-Entry and Landing (estimated values)	Temperature profile as defined in Figure 5 - 5	

* estimated values

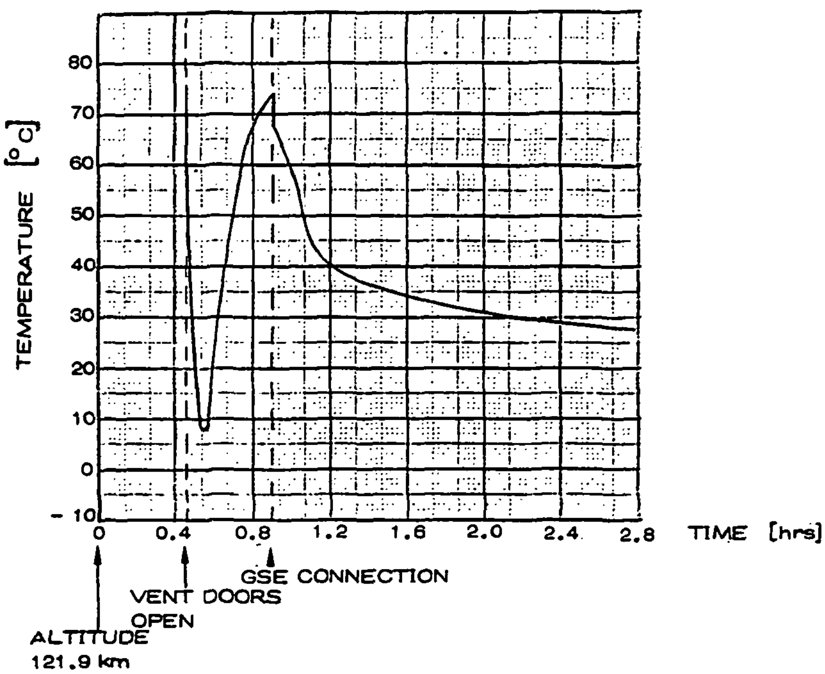


Figure 5-5: Estimated Temperature Profile of the Cargo Bay Air During Descent (Reference: Vol XIV, Rev D, Figure 4-19).

5.2.1.4.2 On-Orbit Conditions

5.2.1.4.2.1 Orbiter Cargo Bay Temperatures

While the Orbiter cargo bay doors are closed, the relative coupling with all exposed payload 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 mission phases are given in Table 5 - 18.

Table 5 - 18: Orbiter Cargo Bay Wall Temperature

CONDITION	MINIMUM	MAXIMUM
Prelaunch	+ 5° C	+ 38° C
Launch	+ 5° C	+ 50° C
On-Orbit (doors closed)	TBD	TBD
Entry and postlanding	-73° C	+ 93.5° C

5.2.1.4.2.2 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 - 19, the Orbiter attitude, the equipment shadow/illumination configuration and the thermo-optical properties.

Table 5 - 19: Space Thermal Environment

Environmental Parameter	Unit	
Solar Radiation	W/m^2 $(Btu/h\text{-}ft^2)$	1400 (444)
Earth Global Albedo	Percent (%) of Solar Radiation	30
Earth Thermal Radiation	W/m^2 $(Btu/hr\text{-}ft^2)$	243 (77)
Space Sink Temperature	K	0
Orbit Altitude	km (naut. miles)	454 (240)

5.2.1.4.2.3 Thermo-Optical Properties of Spacelab/Orbiter External Surfaces

The radiative environment for pallet-mounted payloads is also determined by the thermo-optical properties of Spacelab and Orbiter surfaces in the vicinity of the payload equipment. Tables 5-20 and 5 - 21 show the thermo-optical properties of Spacelab and the Orbiter, respectively.

The thermo-optical properties of Spacelab and Orbiter surfaces, which are essential for the radiation environment of pallet-mounted payloads, are summerised in the subsequent Tables 5-20 and 5-21.

Table 5-20: Thermo-Optical Properties of Spacelab External Surfaces.

Surface Description	Surface Material	α New	α Degraded	ε New	ε Degraded	ρ Diff. %	ρ Spec. %
Pallet Top Cover	Kapton Side of Gold Kapton	0.32	TBD	0.67	TBD		
Pallet Inner Panel	Typical White Paint	0.25	0.33	0.90	0.9		
Pallet Outer Panels	Typical White Paint	0.25	0.33	0.90	0.9		
Pallet Forward End Frame	Typical White Paint	0.25	0.33	0.90	0.9		
Pallet Aft End Frame	Typical White Paint	0.25	0.33	0.90	0.9		
Pallet Sill	Typical White Paint	0.25	0.33	0.90	0.9		
Pallet Attachment	Typical White Paint	0.20	TBD	0.90	TBD		
Module Hard Cover	NOT APPLICABLE						
Module Soft Cover	TBD	0.22	0.33	0.78			

Table 5 – 21: Thermo-Optical Properties of Orbiter Surfaces

(Ref.: ICD-2-05201)

Surface Description	Design Criterion	Surface Material	α New	α Degr.	ε New	ε Degr.	ρ Diff. %	ρ Spec. %
P/L Bay Liner	$\alpha_s/\varepsilon \leq .4$ $\varepsilon \geq .8$	Teflon Coated Glass Cloth	.22	.36	.9	.9	≈ 99	None
Fwd Bulkhead	$\alpha_s/\varepsilon \leq .4$ $\varepsilon \geq .8$	Teflon Coated Glass Cloth	.22	.36	.9	.9	≈ 99	None
Aft Bulkhead	$\alpha_s/\varepsilon \leq .4$ $\varepsilon \geq .8$	Teflon Coated Glass Cloth	.22	.36	.9	.9	≈ 99	None
Radiator Concave Surface	N/A	Silver-Coated Teflon	.08	.10 to .11	.8	.8	1 – 4	96 – 99
Radiator Convex Surface	N/A	Silver-Coated Teflon	.08	.10 to .11	.8	.8	1 – 4	96 – 99
P/L Bay Doors Concave Surface	Fwd N/A Aft $\alpha_s/\varepsilon \leq .4$ $\varepsilon \geq .8$	Teflon Coated Glass Cloth	.08 .32	.10 to .11 .36	.8 .9	.8 .9	1 – 4 ≈ 99	96 – 99 None
P/L Bay Doors Convex Surface	$\alpha_s/\varepsilon = 2$ to .4 $\varepsilon \geq = .8$	FRSI/LRSI[1]	.16/.16	.32 /.32	≤.8/≤.8	≤ .8 / ≤ .8	N/A	N/A
Fuselage Mid Section Sides	$\alpha_s/\varepsilon = .2$ to .4 $\varepsilon \geq = .8$	FRSI/LRSI[1]	.16/.16	.32 /.32	≤.8/≤.8	≤ .8 / ≤ .8	N/A	N/A
Wing Upper Surface	$\alpha_s/\varepsilon = .2$ to .4 $\varepsilon \geq = .8$	FRSI/LRSI[1]	.16/.16	.32 /.32	≥.8/≥.8	≥ .8 ≥ .8	N/A	N/A
Wing Lower Surface	$\alpha_s/\varepsilon \geq = .7$ to 1.0 $\varepsilon \geq .8$	HRSI[2]	.8	.56	≥ .8	≥ .8	N/A	N/A
Bottom of Orbiter	$\alpha_s/\varepsilon \geq = .7$ to 1.0 $\varepsilon \geq .8$	HRSI[2]	.8	.56	≥ .8	≥ .8	N/A	N/A
SPACELAB Tunnel	$\alpha_s/\varepsilon \leq .4$ $\varepsilon \geq .8$	Teflon Coated Glass	.22	.36	.9	.9	≈ 99	None
Orbiter Tunnel Adapter	$\alpha_s/\varepsilon \leq .4$ $\varepsilon \geq .8$	Teflon Coated Glass	.22	.36	.9	.9	≈ 99	None
SPACELAB Tunnel Airlock	$\alpha_s/\varepsilon \leq .4$ $\varepsilon \geq .8$	Teflon Coated Glass	.22	.36	.9	.9	≈ 99	None

1) Felt re-usable surface insulation/low temperature re-usable surface insulation

2) High temperature re-usable surface insulation

5.2.1.4.2.4 Pallet Surface Temperatures

The pallet surface temperatures are highly dependent on the mission profile. In particular, the on-orbit and re-entry temperatures will depend on the Orbiter attitude. Table 5 - 22 shows the extremes of temperatures which result from hot and cold case conditions. Operational constraints on Orbiter attitude hold may be planned - on a mission dependent basis - to reduce the temperature excursions on a particular mission.

Table 5 - 22: Temperature Range of Pallet Structure Surface
(Preliminary Results of Thermal Analysis)

OPERATIONAL PHASE	T min. (° C)	T max. (° C)
Launch / Ascent	+ 4.5	+ 65.5
Orbit doors closed	- 50	+ 70
Orbit doors open	- 150	+ 120
Re-Entry	- 140	+ 110
Post Landing	- 110	+ 120

5.2.2 Ground Environment

The temperatures encountered by payload equipment during transportation, integration and pre/post launch shuttle operations are described in para 5.3.2.

5.3 Atmospheric Environment

5.3.1 Flight Environment

5.3.1.1 Module

5.3.1.1.1 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 shown in Figure 5 - 6. Estimates were made for both adiabatic and isothermal conditions and the actual conditions for the module atmosphere will be somewhere in between the two curves.

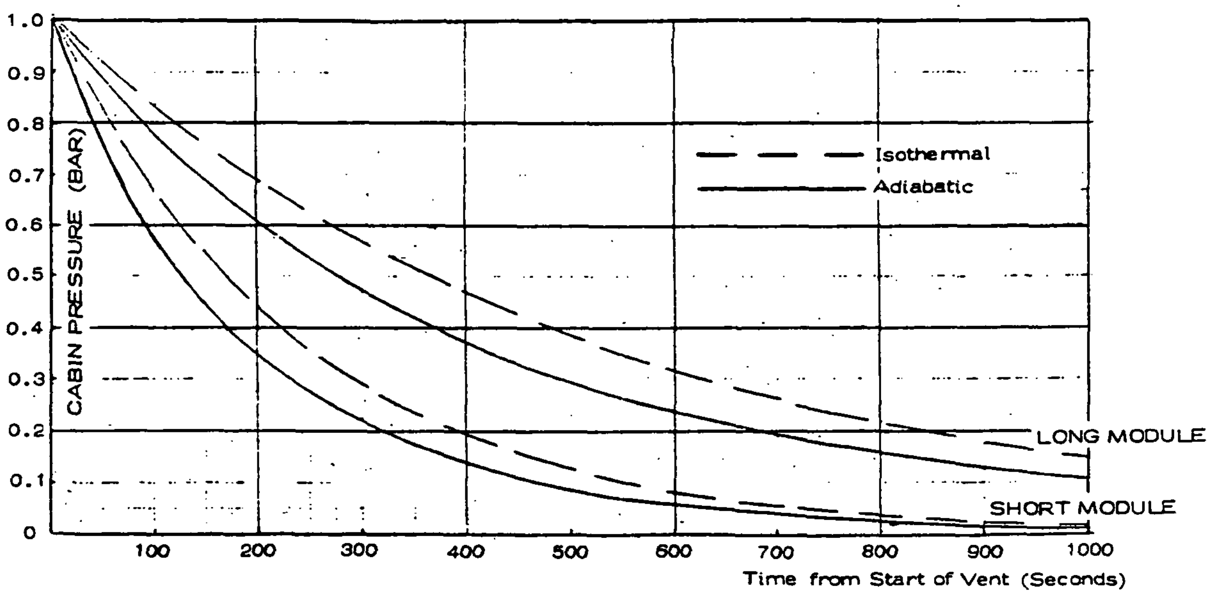


Figure 5-6: Pressure Profile for Emergency Depressurization of the Module

Table 5 - 23: Module Atmosphere Gas Composition

Cabin Pressure	
O_2/N_2 total pressure:	1.013 $\pm$ 0.013 bar
O_2 partial pressure:	0.220 $\pm$ 0.017 bar
CO_2 partial pressure:	0.0067 bar (nominal)
	0.01 bar (maximum for normal operations)

5.3.1.1.2 Atmospheric Composition

The gaseous composition of the module atmosphere is given in Table 5 - 23 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 - 7. The cabin atmosphere design requirements are depicted 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.

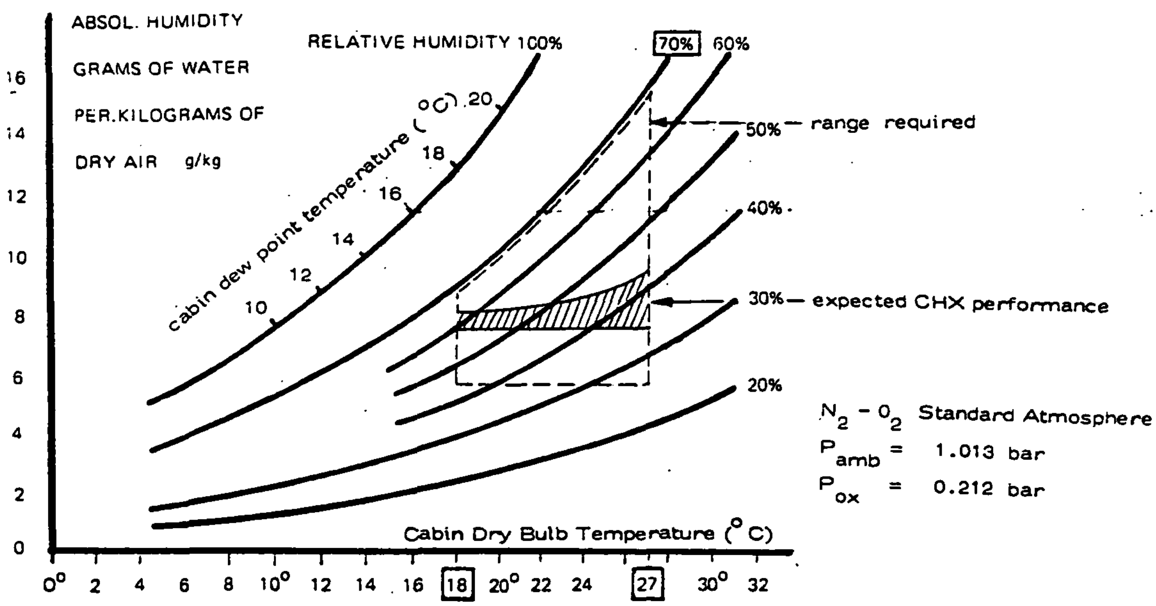


Figure 5 - 7: Crew Thermal Comfort Criteria

5.3.1.1.3 Airlock Atmosphere

During airlock operations with the inner hatch open the atmosphere inside the airlock is the same as in the module.
During airlock depressurization (with both hatches closed) the maximum rate of depressurization is 6.6×10^{-3} bar/sec. The detailed pressure history is TBD. After depressurization, and with the outer hatch open, the atmospheric conditions will be as described in para 5.3.1.2.2.
Repressurization of the airlock is performed by filling with dry nitrogen at a rate not exceeding 6.6×10^{-3} bar/sec. The detailed pressure history is TBD and may be significantly affected by experiment thermal capacity and temperature.

5.3.1.2 Pallet

5.3.1.2.1 Launch Sequence

The Orbiter cargo bay is vented during the launch and entry phases and operates unpressurized during the orbital phase of the mission. Figures 5 - 8 and 5 - 9 define the cargo bay pressure history during ascent.

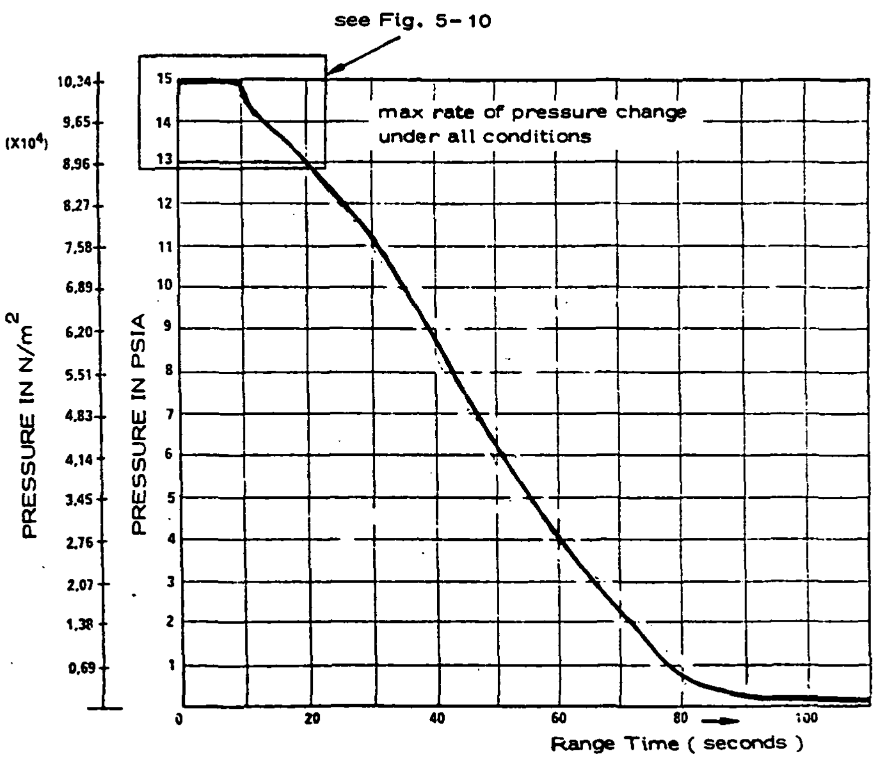
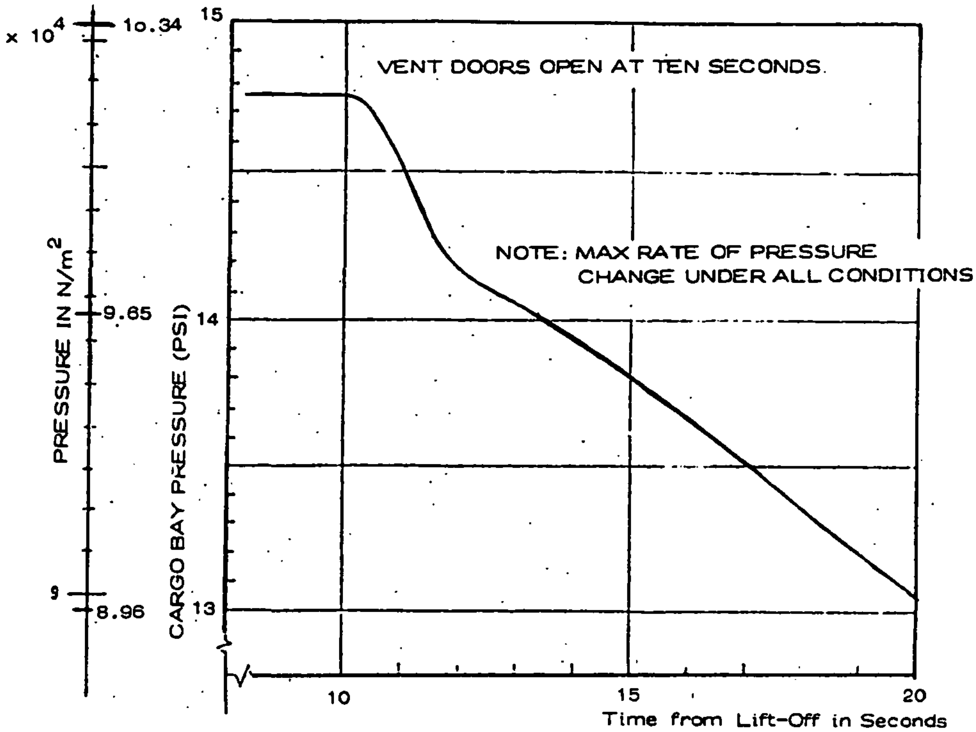


Figure 5-8: Orbiter Cargo Bay Internal Pressure History During Ascent

(Reference: Vol XIV, Rev. E, Figure 4-2)

DCN 041



(from ICD - 2 0320)

Figure 5 - 9: Orbiter Cargo Bay Internal Pressure History During Ascent/Lift-Off Detail

5.3.1.2.2 On-Orbit

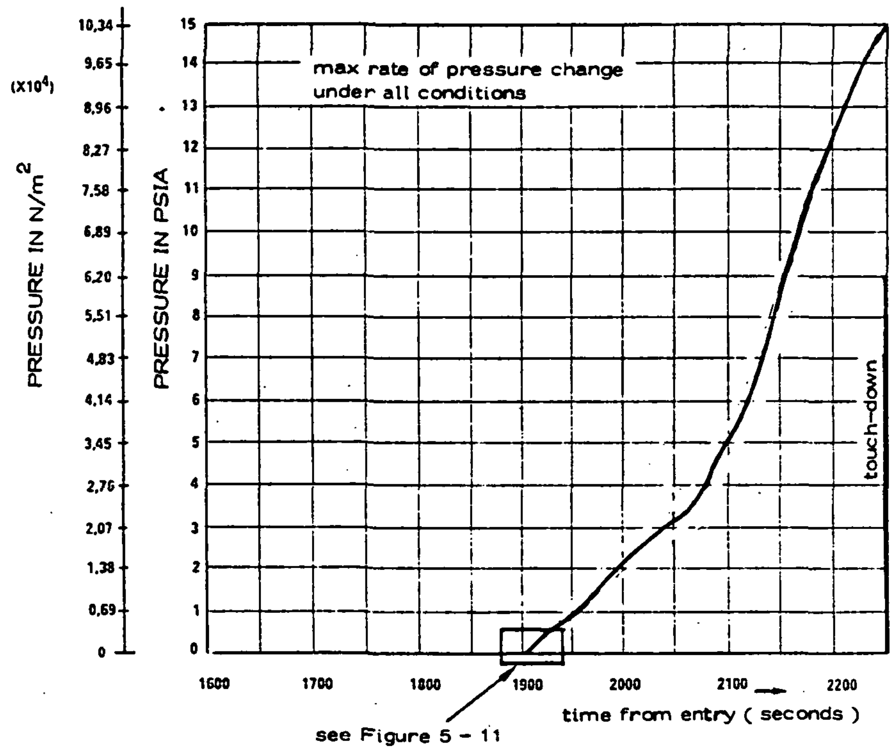
Respective values are given in Table 5 - 23.

Table 5 - 24: Atmospheric Pressure Variation with Altitude

Altitude	Max P (bar)	Min P (bar)
130 km	1.8×10^{-8}	1.06×10^{-8}
185 km	2.9×10^{-9}	6.3×10^{-10}
926 km	2.9×10^{-12}	6.3×10^{-14}
2,222 km	Approx. 1.33×10^{-14} in either case	

5.3.1.2.3 Re-Entry Sequence

During re-entry and landing the cargo bay pressure will be increased to the landing site pressure by using filtered atmospheric air (35 microns absolute). Figures 5 - 10 and 5 - 11 define the cargo bay pressure history during reentry.



DCN 041

Figure 5-10: Orbiter Cargo Bay Internal Pressure History During Entry for all Flight Modes

DCN 041

(Reference: Vol XIV, Rev. E, Figure 4-3)

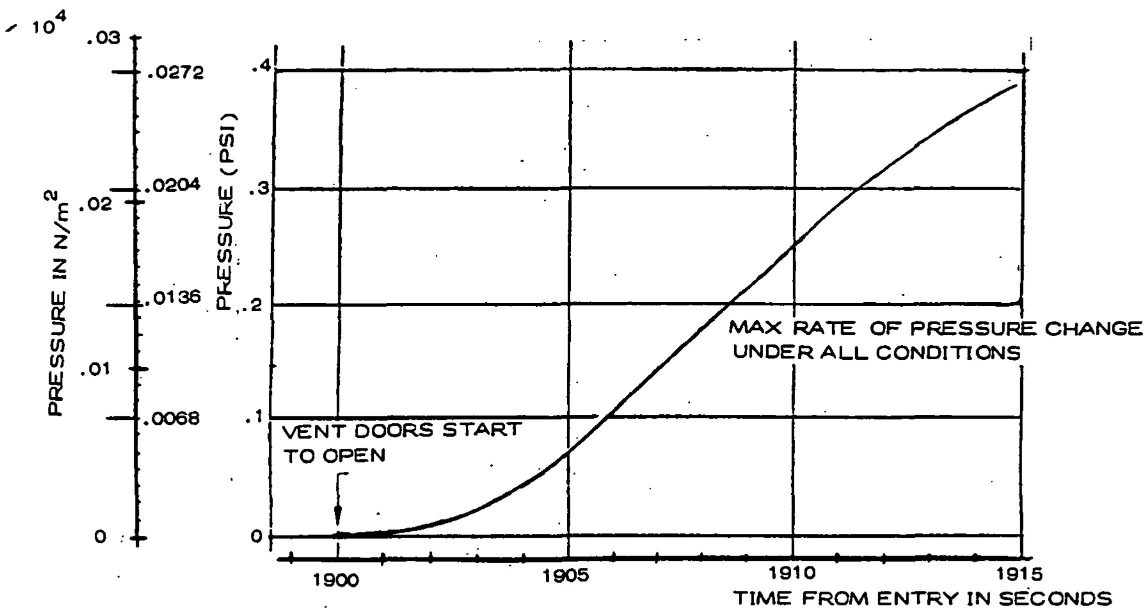


Figure 5 - 11: Cargo Bay Pressure Profile Detail
Vent Doors Opening During Entry

5.3.2 Ground Environment

5.3.2.1 Integration Operations

The worst-case environmental conditions enveloping all Spacelab processing facilities in the USA are as follows:

DCN 045

Temperature:	+ 18 to + 25° C
Humidity:	30 % - 60 % R.H.
Pressure:	ambient to ambient + 1 mb
Cleanliness:	Visibly clean

These conditions apply to the offline Spacelab processing facilities at Kennedy Space Center (KSC), Vandenberg Air Force Base (VAFB), Johnson Space Center (JSC), Marshall Space Flight Center (MSFC), Goddard Space Flight Center (GSFC), Langley Va., and experimenter sites. The online facilities (Spacelab installed in Orbiter) at KSC and at the VAFB will also have the same environmental conditions.

5.3.2.2 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^{o} C)(65 - 85^{o} F) controlled to $\pm$ 1^{o}C ($\pm$ 2^{o}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.

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^{o} C (70 $\pm$ 5^{o} F)
- Cleanliness: Nominally class 100 guaranteed class 5000 (HEPA filtered)
- Humidity: 45 % $\pm$ 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 - 38^{o} C (45 - 100^{o} F) controlled to $\pm$ 1^{o} C ($\pm$ 2^{o} F) of desired setting, according to ICD-2-05201 (Table 3.2.4.1-1)
- 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^{o} and 38^{o} C (45 - 100^{o} F) controlled to $\pm 1^{o}$ C ($\pm 2^{o}$ F), according to ICD-2-05201.
- 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 to 38^{o} C (45 - 100^{o} F) controlled to $\pm 1^{o}$ C ($\pm 2^{o}$ F) of the desired setting, according to ICD-2-05201.
- 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

5.3.2.3 During Transportation

During transportation of flight hardware the transported article is protected by transportation equipment. The atmospheric conditions for the transported article are dependent 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 conditions.

Temperature:	- 10 to + 55^{o}C
Humidity:	10 to 90 % RH for all Spacelab flight hardware.
Pressure:	Sea Level Pressure to 0.27 bar

5.3.2.4 Terrestrial Environment

Except under emergency conditions payload equipment integrated with Spacelab 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.

5.4 Electromagnetic Environment

5.4.1 Electrical Environment

5.4.1.1 Radiated Emissions

5.4.1.1.1 Orbiter Emissions

Shuttle contributions to the payload electro-magnetic environment will be limited to levels shown in Figures 5-12 and 5-13 with the exception of the levels associated with Orbiter-installed transmitters as specified in Table 5-25. The levels apply only with the cargo bay doors open.

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.

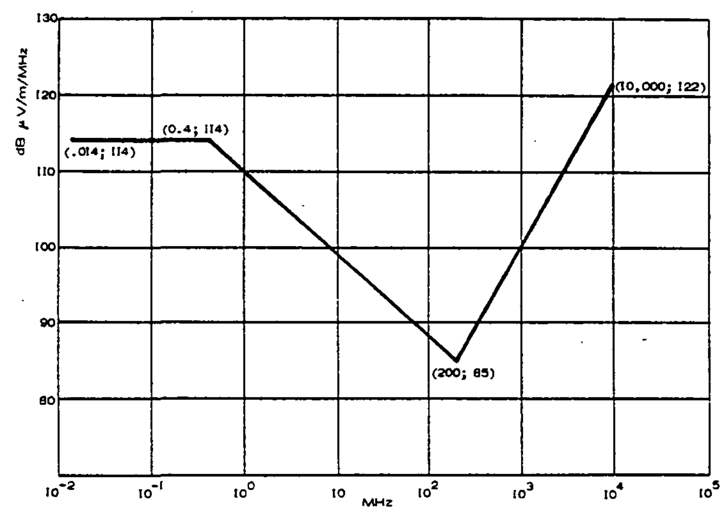


Figure 5 - 12: Shuttle - Produced Cargo Bay Radiated Broadband Emissions

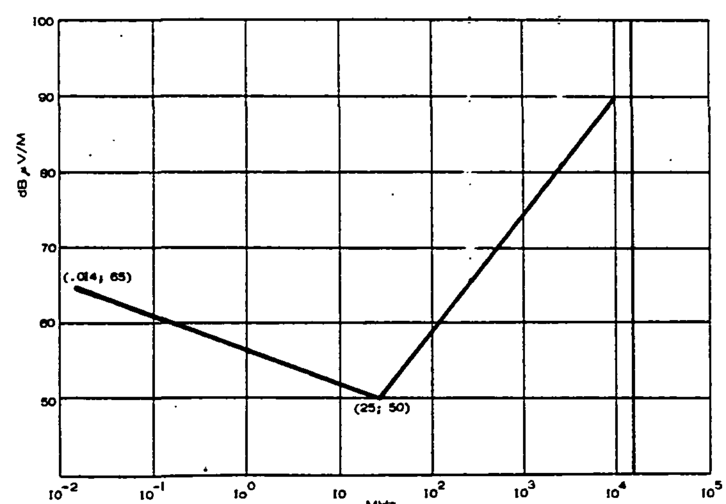


Figure 5 - 13: Shuttle - Produced Cargo Bay Radiated Narrowband Emissions

Table 5-25: Orbiter Transmitter - Produced Electric Fields
(Ref.: ICD-2-05301, ECP 73-001)

Transmitter	Frequency	Field Strength $Y_o = 0$ $Z_o = 460$		
		$X_o = 702$	$X_o = 942$	$X_o = 1182$
S-Band Hemi	2000 - 2300 MHz	4 V/m	2 V/m	1.2 V/m
S-Band Quad	2200 - 2300 MHz	8 V/m	5 V/m	3.5 V/m
S-Band Payload	2000 - 2200 MHz	3 V/m	1 V/m	0.6 V/m
Ku Band (1)	13 - 15 GHz	45 V/m	23 V/m	15 V/m
EVA Back Pack (2)	250 - 300 MHz	4 V/m	4 V/m	4 V/m

(1) All values are average field intensities in first side lobe (4° off boresight). Peak values for radar are TBD. Tracked payload maximum field intensity is 20 V/M average.

(2) EVA backpack values apply 1 meter from backpack.

Figure 5 - 14 shows the location of the Orbiter antennas

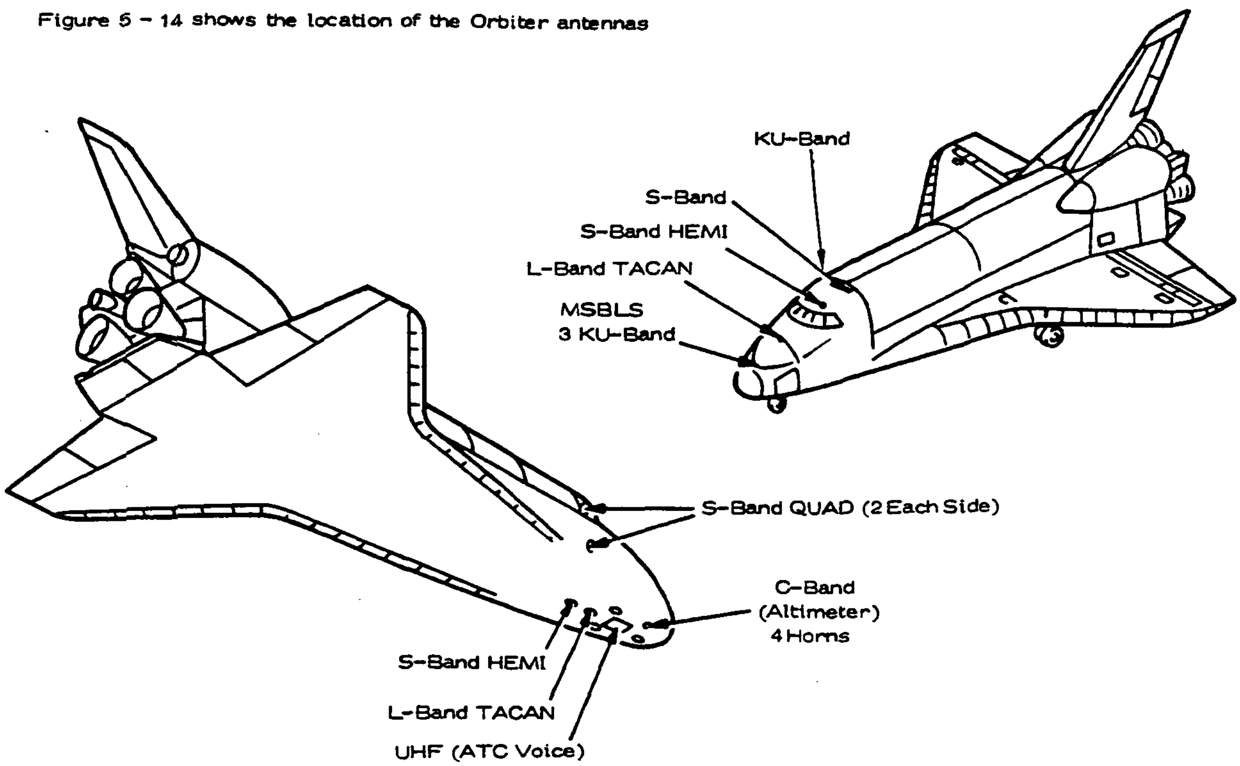


Figure 5 - 14: Orbiter Antenna Locations

5.4.1.1.2 Spacelab Emissions

RF emissions radiated by the Spacelab and individually installed Aft Flight Deck equipments are controlled to the specification limits given in Figures 5 - 15 and 5 - 16. Limits apportioned to individual experiment equipments are listed in Section 7, Figure 7 - 12.

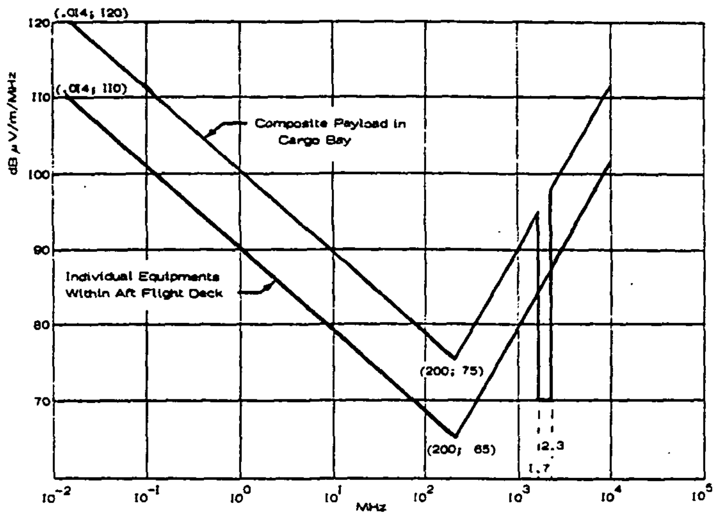


Figure 5 - 15: Spacelab/Experiments Allowable Radiated Broadband Emissions

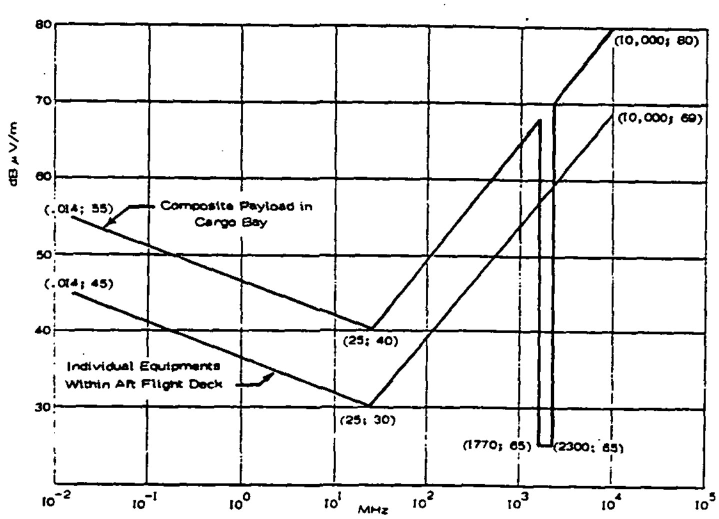


Figure 5 - 16: Spacelab/Experiments Allowable Radiated Narrowband Emissions

5.4.1.1.3 Composite Payload Environment

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).

Pallet mounted equipment can be expected to encounter the full Orbiter and Spacelab emission levels due to the absence of any module shielding.

5.4.1.1.4 Launch Environment

PIRN JSC 3028 gives the environment external to the Orbiter, but Orbiter attenuation data for payloads is not yet available to give direct values for fields impinging on Spacelab and payloads on pallets.

5.4.1.2 Conducted Emissions

The composite conducted interference environment seen by the Spacelab payloads reflects the operation of the EPDS and the primary DC power source (Orbiter fuel cell(s) or GSE) in the following modes

- Normal on-orbit conditions – where the primary DC power source is a dedicated Orbiter fuel cell. In this case the power bus characteristics will be determined principally by Spacelab emission levels, although other Spacelab payloads may add to these levels.

- Ascent/descent or back-up power operating conditions – where the primary DC power source is an Orbiter fuel cell which is shared with Orbiter loads in parallel with Spacelab. In this case the power bus characteristics will be determined principally by the Orbiter emission levels.

- Ground operations – where the primary DC power source is GSE. In this case the power bus characteristics will be determined principally by the GSE emission levels.

The composite environment which encompasses all these conditions may be represented by the following test signals at the payload interface injected between:

a) DC power positive and power return
b) DC power return and structure

Where both current and voltage levels are indicated whichever level is reached first is intended to be the limit.

5.4.1.2.1 Narrowband Noise

A swept sine wave
- Amplitude: see Figure 5 - 18
- Frequency: 30 Hz - 400 MHz
- Sweep rate: 1 decade/min

Note: the characteristics of the Spacelab AC power bus are described in Appendix A - Avionics Interface Definition

5.4.1.2.2 Broadband Noise and Spikes

A swept square wave

•	Amplitude:	see Figure 5 - 17
•	Frequency:	30 Hz - 400 MHz p.r.f.
•	Mark/Space ratio:	1 to 1
•	Rise time:	= 10 n s
•	Fall time:	= 10 n s
•	Sweep rate:	1 decade/min

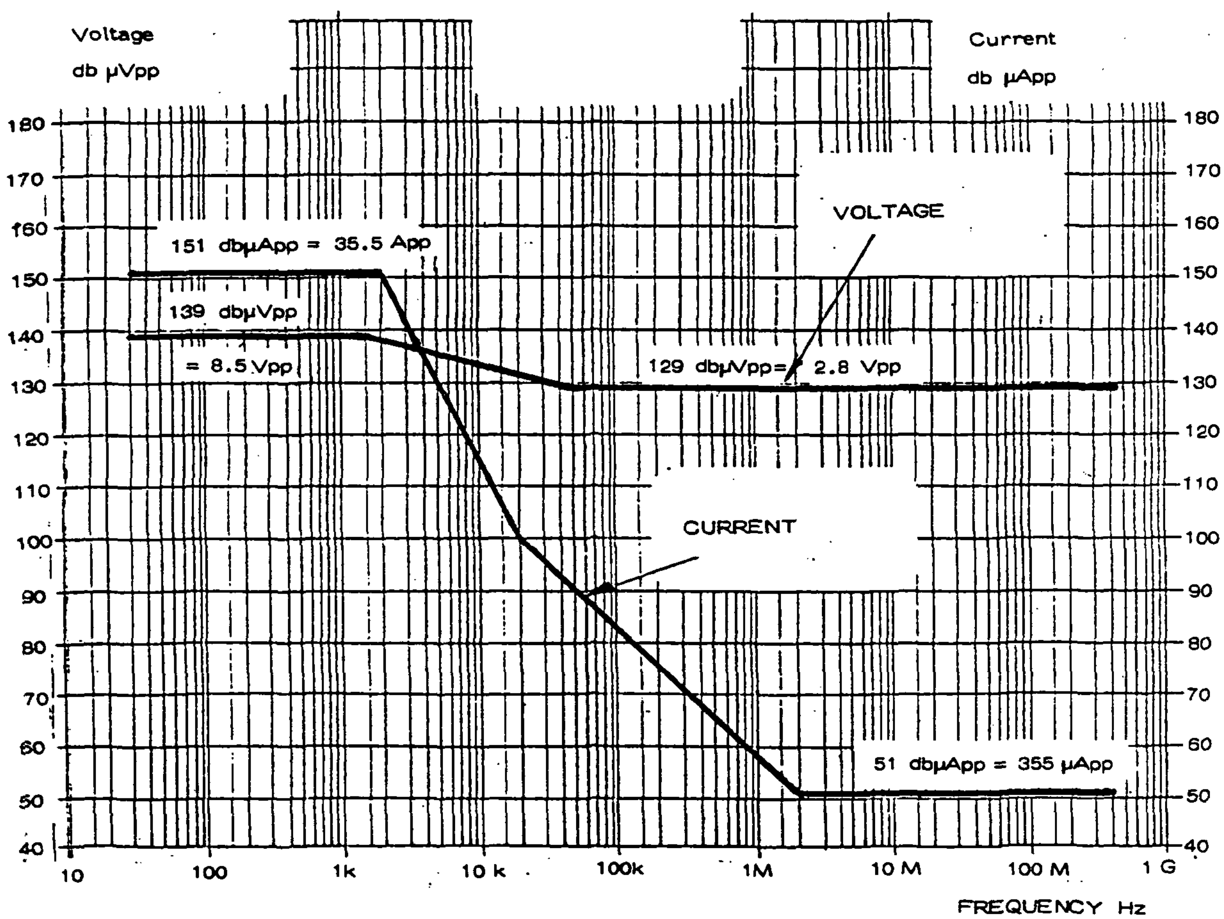
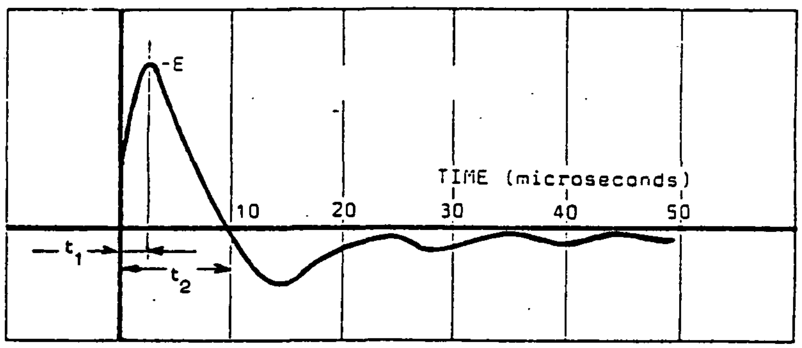


Figure 5 - 17: Conducted Noise Levels

5.4.1.2.3 Transients

- Amplitude: 56 V peak
- Frequency: = 6 pulses per minute
- Test duration: 30 minutes
- Pulse polarity: positive and negative
- Transient shape: see Figure 5 – 18



E = 2 times line voltage or 1oo volts, whichever is less

$t_1 \geq$ 2 microseconds

$t_2 =$ 1o microseconds

Figure 5 – 18: Conducted Transient Waveform

5.4.1.3 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. The current best estimate of the transient magnetic field in the Orbiter cargo bay due to lightning strikes is a triangular waveform with a peak amplitude of approximately 70 amperes per meter, a rise time of 2 microseconds, and a decay time of 100 microseconds.

5.4.1.4 Electrical Surface Properties: TBD

5.4.2 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 have been estimated to reach 170 dB pT, but are not controlled by present specifications.

Editorial note: The above AC magnetic field values are still under review by NASA. They apparently are caused by ripple currents on the DC powerline to the Orbiter Aft Engine Compartment which then returns the current through structure back to the fuel cells.

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 1 pico-Tesla in accordance with Shuttle specifications. No requirements have been imposed to control magnetic fields below 30 Hz.

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. Payload equipment which is located in certain areas may therefore 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: 146 dB above a pico-Tesla at 30 Hz decreasing linearly to 80 dB above a pico-Tesla at 50 kHz for module installed equipments, and 146 dB above a pico-Tesla (30 Hz to 2 kHz), falling 40 dB per decade to 50 kHz for pallet-mounted equipment.

5.5 Cleanliness and Contamination

5.5.1 Flight Environment

5.5.1.1 Module

Prior to installation into the Spacelab module itself or into rack sections, all equipment surfaces will be cleaned to a visibly 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. Trace contaminant levels in the cabin air are reduced by means of an atmosphere scrubber located in the tunnel.

5.5.1.2 Pallet

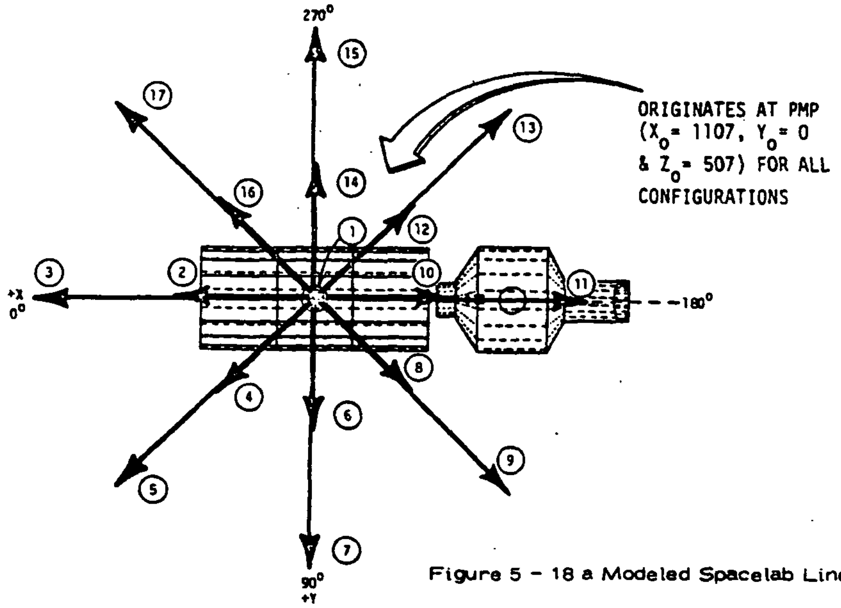
The contamination aspects of the Orbiter are described in para 2.2.8.

Contamination produced by Spacelab itself (outgassing, air leakage etc.) on orbit has been assessed analytically and the results of this assessment are documented in Martin Marietta Final Report MCR - 77 - 105 of September 30, 1977 - Spacelab Contamination Assessment. Some of the more significant predicted values are presented below.

5.5.1.2.1 Molecular Number Column Density (NCD) Predictions

Predictions have been made for seventeen fixed lines-of-sight which encompass the 120° conical volume centered around the + Z axes and originating at X_0 = 1107, Y_0 = 0 and Z_0 = 507 (see Figure 5-18). These predictions are presented in Table 5 - 26 for three modelled Spacelab configurations. Values

DCN 005 are given for nonmetallic materials outgassing, early desorption at 10 hours of vacuum exposure and cabin atmosphere leakage. Outgassing products consist mainly of hydrocarbons whereas early desorption and leakage components consist mainly of CO_2, H_2O, N_2 and O_2.



LEGEND

Line-of-Sight	Degrees Off +Z	Degrees Off +X
1	0	0
2	30	0
3	60	0
4	30	45
5	60	45
6	30	90
7	60	90
8	30	135
9	60	135
10	30	180
11	60	180
12	30	225
13	60	225
14	30	270
15	60	270
16	30	315
17	60	315

Figure 5 - 18 a Modeled Spacelab Lines-of-Sight

Table 5 - 26: Spacelab Molecular Number Column Density Predictions

SOURCE/CONFIG. / LINE-OF-SIGHT	NUMBER COLUMN DENSITY (molecules·cm^{-2})							
	OUTGASSING			EARLY DESORPTION			LEAKAGE	
	LMOP	SMTP	FIVP	LMOP	SMTP	FIVP	LMOP	SMTP
1	1.9E8*	2.8E8	1.3E8	3.7E12	1.8E12	2.1E11	2.6E12	1.4E12
2	1.6E8	2.4E8	1.2E8	3.0E12	1.5E12	1.9E11	2.2E12	1.1E12
3	1.6E8	2.2E8	1.1E8	2.7E12	1.4E12	1.7E11	1.9E12	9.9E11
4	1.7E8	2.5E8	1.3E8	3.3E12	1.6E12	2.0E11	2.3E12	1.2E12
5	1.6E10	1.5E9	1.2E8	3.4E12	1.5E12	1.8E11	2.1E12	1.0E12
6	1.3E9	5.2E8	1.3E8	3.9E12	1.8E12	2.1E11	2.6E12	1.4E12
7	3.4E10	3.9E9	1.2E8	5.1E12	2.0E12	1.8E11	2.6E12	1.3E12
8	7.2E8	6.6E8	1.4E8	4,8E12	2.3E12	2.2E11	3.1E12	1.7E12
9	3.7E10	7.4E9	1.5E8	7.5E12	3.4E12	2.3E11	3.7E12	2.2E12
10	1.1E9	9.6E8	1.5E8	5.1E12	2.6E12	2.3E11	3.3E12	1.9E12
11	1.1E10	2.7E9	1.7E8	8.4E12	4.4E12	2.7E11	4.5E12	3.3E12
12	6.9E8	6.5E8	1.4E8	4.4E12	2.2E12	2.2E11	3.1E12	1.7E12
13	3.7E10	7.3E9	1.4E8	6.4E12	3.0E12	2.2E11	3.7E12	2.2E12
14	1.3E9	5.1E8	1.3E8	3.5E12	1.7E12	2.0E11	2.6E12	1.4E12
15	3.4E10	3.9E9	1.1E8	4.4E12	1.7E12	1.7E11	2.6E12	1.3E12
16	1.6E8	2.4E8	1.2E8	3.1E12	1.5E12	1.9E11	2.3E12	1.2E12
17	1.6E10	1.5E9	1.1E8	3.1E12	1.4E12	1.7E11	2.1E12	1.0E12

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*1.9E8 = 1.9×10^8

LMOP = Long module one pallet configuration
SMTP = Short module three pallet configuration
FIVP = 15 meter pallet configuration

The predicted NCD levels for outgassing and leakage will remain relatively constant throughout a Spacelab mission, however, the early desorption NCD levels will decrease rapidly as the early desorption rate decays with time or vacuum exposure. The primary contamination threats from early desorption will, therefore, be limited to the initial on-orbit phases of a given mission.

5.5.1.2.2 Molecular Return Flux Predictions

For most Spacelab payloads, the primary transport mechanism of the major contaminant sources will be the return flux resulting from contaminant molecular collisions with the ambient atmosphere flux. Direct line-of-sight and self-scattering return flux transport were evaluated and deemed negligible under the major Spacelab source conditions. All major Spacelab sources were evaluated for maximum return flux (i.e.; ambient drag vector perpendicular to surface of interest) to a 2π steradian field-of-view surface located at the PMP. The worst case orbital altitudes were considered for each source modeled (i.e., early desorption and leakage at 200 km and outgassing at 250 km) and medium solar activity was assumed. The results are presented in Table 5 - 27.

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Table 5 - 27: Spacelab Molecular Return Flux Predictions

SOURCE/ ALTITUDE CONFIGURATION	MAXIMUM RETURN FLUX-2π sr SURFACE (molecules$\cdot cm^{-2} \cdot s^{-1}$)		
	OUTGASSING AT 250 km	EARLY DESORPTION AT 200 km	LEAKAGE AT 200 km
LMOP	8.7E11	5.0E14	4.1E14
SMTP	1.6E11	2.4E14	2.1E14
FIVP.	1.4E10	2.4E13	———

5.5.2 Ground Environment

See para 5.3.2

5.6 Radiation Environment

5.6.1 External Environment

The natural radiation environment at the expected Orbiter altitudes consists of

a) galactic cosmic radiation,
b) geomagnetically trapped radiation, and
c) solar flare particle events.

This environment may be defined by establishing a description of the particle flux as a function of energy, species, and location (time and space). The following data are derived from NASA TMX 64627, Space and Planetary Environment Criteria Guideline for use in Space Vehicle Development, 1971.

a. Galactic Cosmic Radiation

Composition: 85 % protons
13 % alpha particles
2 % heavier nuclei

Energy Range: 10^7 to 10^{19} electron volts, (predominantly 10^9 to 10^{13})

Integrated Yearly Rate: Approx. 1×10^8 protons/cm^2

Integrated Yearly Dose: Approx. 4 to 10 rads.

b. Trapped Radiation

The earth's magnetic field provides the mechanism which traps charged particles (electrons and protons) in belts around the earth.

The inner radiation belt has a maximum omnidirectional particle flux at the equator at an altitude of

- approximately 1800 km for electrons (10^8 electrons/cm^{2}s > 0.5 MeV)
- approximately 3500 km for protons (10^4 to 10^5 protons/cm^{2}s > 38 MeV)

A detailed description of the environment is contained in NASA TM X - 73358, Rev. 1, Charged Particle Radiation Environment for the Spacelab and other missions in Low Earth Orbits, Nov. 76. DCN 006

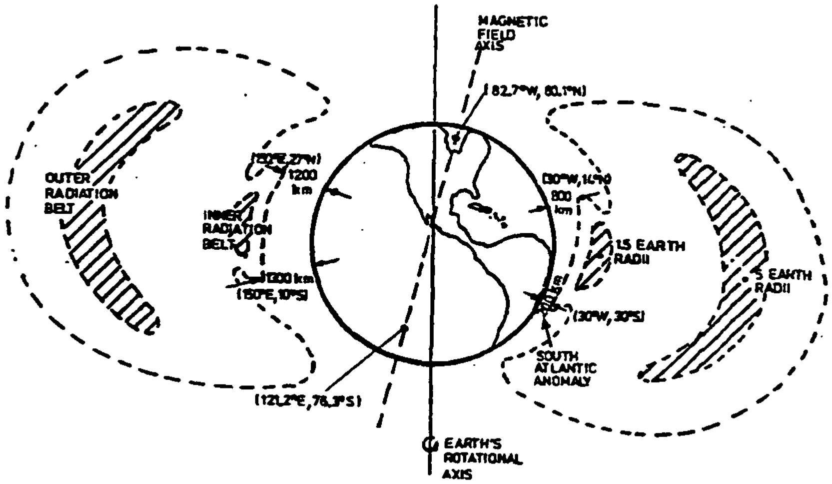


Figure 5 – 19: Cross Section Through the Trapped Radiation Belts

Most of the particle flux encountered by Spacelab at low orbit altitudes (200 – 400 km) will result from the South Atlantic Anomaly which is shown in Figure 5 – 20.

Figure 5 – 19 is in the plane of the 30° W geographical meridian and illustrates relative altitudes at which the South Atlantic Anomaly particles mirror.

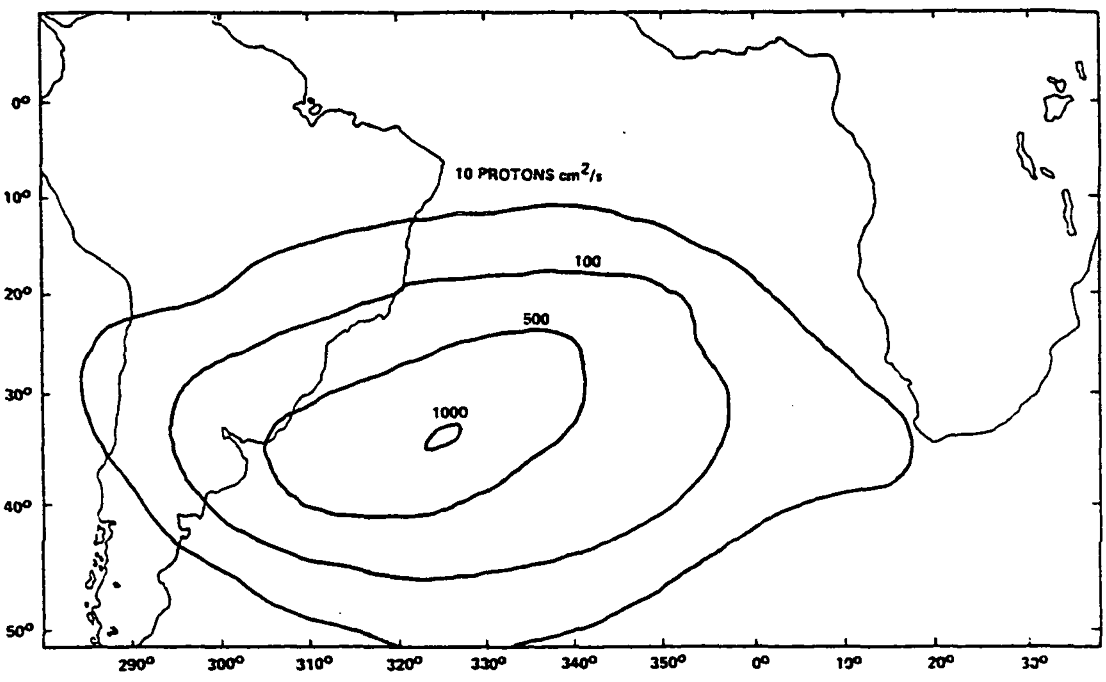


Figure 5 – 20: Proton Isoflux Contours for Energies Above 34 MeV in the South Atlantic Anomaly

c. Solar Particle Events

Solar particle events are the emission of charged particles from distributed regions on the sun during solar flares. They are composed of energetic protons and alpha particles that occur sporadically and last for several days.

Details of the actual radiation levels for the various possible Spacelab orbits are contained in NASA Technical Memorandum TMX-64936.

5.6.2 Internal Environment

The radiation levels experienced by payload equipment will depend on the shielding effectiveness of Spacelab and Orbiter structure. Figures 5 - 21 and 5 - 22 show the shielding properties (in g/cm^2 equivalent aluminium) of the Orbiter and the Long Module.

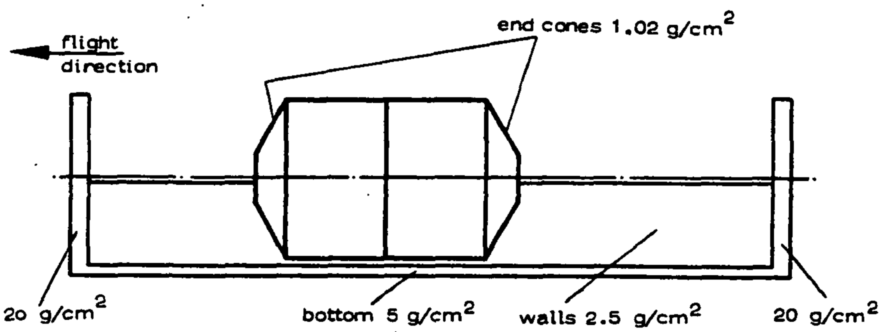
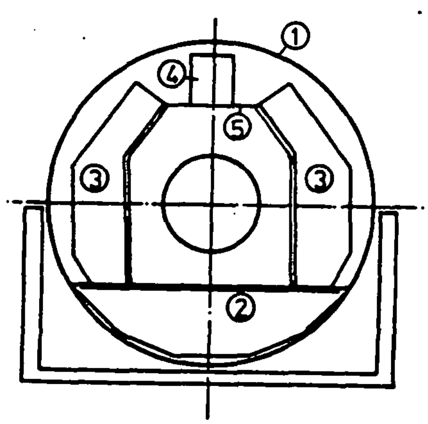


Figure 5-21: Orbiter Shielding Properties With Long Module



1 shell 0.45 g/cm^2
2 floor 0.23 g/cm^2
subfloor 0.2 g/cm^2
3 rack front side 0.57 g/cm^2
rear side 0.23 g/cm^2
4 container, walls 0.23 g/cm^2
5 cover of overhead structure 0.23 g/cm^2

Figure 5-22: Module Shielding Properties

Radiation models defined for Spacelab and analyses of the dose rates for various orbits have been made. Figure 5 - 23 shows the estimated daily dose (excluding solar flare contributions) at the center of the Long Module using the environment data from NASA TMX-73358, Rev. 1, Nov. 76.

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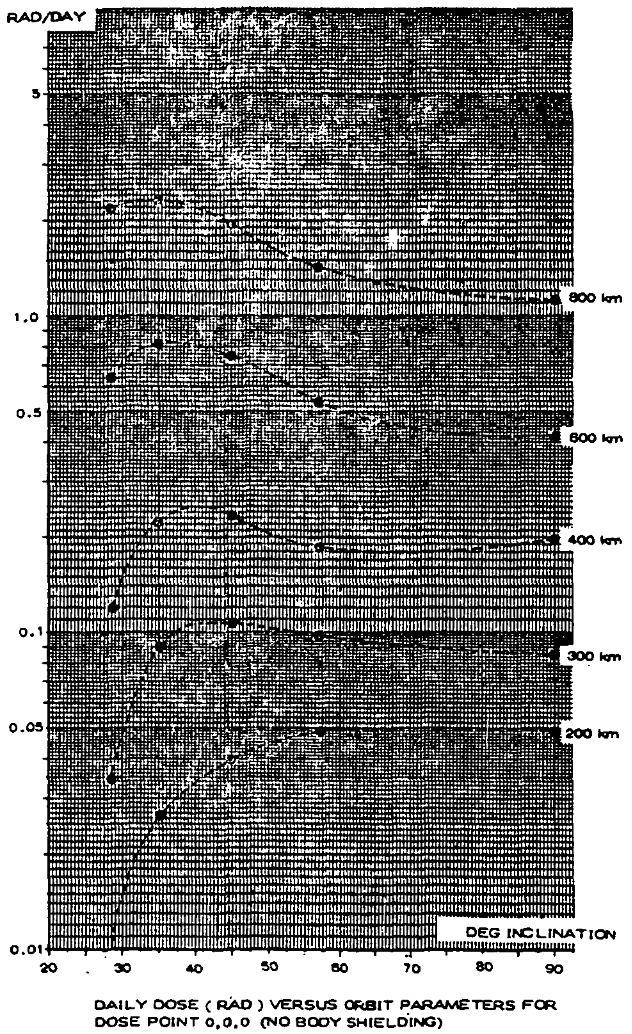

Figure 5 - 23: Daily Dose (Rad Versus Orbit Parameters) at the Center of the Long Module

5.7 Meteoroid Environment

Payload equipment mounted on the pallet or in the airlock (with the outer hatch open) will be subjected to meteoroid impacts during the time in orbit when the cargo bay 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 $10^{-6} \leq m \leq 10^{0}$ $\quad \log Nt = -14.37 - 1.213 \log m$

(2) For $10^{-12} \leq m \leq 10^{-6}$ $\quad \log Nt = -14.339 - 1.584 \log m - 0.063 (\log m)^2$

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.

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Section 6 - OPERATIONS

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LIST OF FIGURES

6. OPERATIONS

This Section discusses general boundaries and constraints for payload integration concepts, mechanical and electrical ground support equipment as tools for payload integration, ground operations (mainly Shuttle turn-around) and flight operations.

6.1 Payload Integration

The ground operations concept defines the general boundaries and constraints within which the payload integration approach has to be developed.

Typically, these constraints which are important to the development of a payload integration concept are:

- Individual payloads (single or groups of experiments) are assumed to be qualification/acceptance tested prior to meeting Spacelab flight hardware in the ground processing flow.
- Once the Spacelab ground processing flow is joined, the emphasis is placed on physical integration of the individual payloads into a total Spacelab payload and then into a complete Spacelab flight configuration. In general, payload testing is restricted to interface verification (experiment to experiment and payload to Spacelab subsystems) and an overall Spacelab mission simulation.
- The only payload dedicated operations currently planned in the baseline ground processing flow are essential calibration/re-calibration and servicing activities.

The complete payload integration process forms part of the Spacelab off-line and on-line flows as well as the Shuttle turnaround flow. The interaction between these flows is so arranged that there is a maximum decoupling between them, thus enabling each flow to "turnaround" in a timeframe which supports the overall STS objective of maximum hardware utilization.

The ground operational processing of Spacelab and Spacelab payloads is based on four levels of integration, the definition and scope of which are as follows:

<u>Level IV Integration</u>

- Integration and checkout of experiment equipment with individual experiment mounting elements (e.g. racks, rack & floor assemblies and pallet segments). DCN 024

  Level IV Integration is the lowest level of payload integration defined which involves basic SL hardware, and may be performed at sites remote from the launch site(s) within or outside the U.S.A.

  Because Level IV Integration involves Spacelab flight hardware the environment of the Level IV integration site has to satisfy the cleanliness requirements for at least standard class 100 K according to Federal Standard 209 B, April 24, 1974, Clean Room and Work Station Requirements for Controlled Environment.

The scope of activities which may be performed at Level IV is primarily dependent upon

- The type and complexity of the experiments involved
- The SL accommodations required by the experiments, i.e. the type, number and mix of Spacelab experiment mounting elements required (racks, pallets, etc.).
- The test facilities and equipment available at the Level IV site and provided by either the experimenter and/or the payload integrator.
- The Level IV activities will also be designed to satisfy Spacelab requirements for verification of interfaces made at Level IV.

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The final level of hardware integration achieved at Level IV is somewhat dependent on the location of the particular sites insofar as the Level IV integrated hardware has subsequently to be transported to the primary launch site (KSC) by one of the assigned SL transportation modes.

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Current NASA planning is to pre-configure individual experiment racks, rack & floor assemblies and pallets with appropriate SL subsystem/experiment interfacing hardware (RAU's, power distribution boxes/panels, cold plates, etc.) at the primary launch site (KSC) prior to shipping these to the Level IV integration sites. This preconfiguring will be consistent with the Level IV activities planned for the particular experiments to be integrated. In certain cases, specific pieces of SL subsystem/experiment interfacing hardware will be provided separately from the primary structure. This will provide the capability to integrate those pieces of hardware with the experiment hardware prior to mounting of the experiment on the pallet or rack.

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Preconfigured experiment racks and/or pallets will be shipped to the Level IV integration sites with the MGSE required for handling and subsequent transportation back to the primary launch site.

Current NASA planning calls for the Level IV integrator and/or experimenters to provide all test equipment/GSE to functionally test/validate the experiments after installation and to validate hardware/software interfaces with the subsystem experiment interfacing hardware.

Level III Integration

- Combination, integration and checkout of all experiment mounting elements (e.g. racks, rack sets and pallet segments) with experiment equipment already installed.

  Level III integration will normally be performed in the O & C building at the primary launch site (KSC) on receipt of the Level IV integrated individual experiment racks, rack & floor assemblies and pallets.

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  The scope of Level III activities to be performed in the O & C building is currently restricted to assembly of the already integrated experiment racks and/or pallets into rack/floor sets and pallet trains, followed by mechanical integration of these with the already checked out basic SL into a complete Spacelab flight configuration.

Functional Level III testing of the integrated Spacelab payload in the O & C building is not currently planned prior to the integration of this with the basic Spacelab on board subsystems.

The possibility of performing full functional "SL off line" Level III testing of the integrated SL payload prior to mechanical integration with the basic SL, either at the launch site or elsewhere, is dependent on the particular integrated SL payload configuration, and the facilities/GSE made available by the payload integrator/experimenter to perform the required integration activities.

Level II Integration

- 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 II integration will normally be performed in the O & C building at the primary launch site (KSC) following the mechanical integration of the Spacelab payload and basic Spacelab into a complete Spacelab flight configuration.

The scope of Level II activities, as currently planned, is primarily orientated towards functional verification of Spacelab payload hardware and software interfaces to the on-board subsystems, followed by an overall systems check and an abbreviated mission simulation.

During Level II testing Spacelab/Orbiter, interfaces will be simulated to allow limited "end to end" testing prior to Spacelab installation in the Orbiter. DCN 018

Level I Integration

- Integration and checkout of the Spacelab and its payloads with the Shuttle Orbiter, including the necessary pre-installation testing with simulated interfaces.

Level I integration for KSC launches will be performed in the OPF at KSC & Level I integration for WTR launches will be performed in the OMCF at VAFB after transportation of the integrated Spacelab from the primary launch site (KSC).

The scope of Level I activities is limited to functional verification of Spacelab/Orbiter hardware and software interface, followed by a final Orbiter Integrated Test to allow limited "end to end" testing to Orbiter stacking with other Shuttle elements

6.2 Mechanical Ground Support Equipment

6.2.1 General Overview

The Mechanical Ground Support Equipment (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.

The Mechanical Ground Support Equipment consists of the operational means used for the handling, transportation, servicing, alignment, and environmental protection of the assembled Spacelab and modular elements. with or without payloads installed in the case of racks and pallets. The modular Spacelab requires a flexible MGSE capability to accommodate all possible Spacelab configurations.

Turnaround scheduling is aided by the ability to perform "SL off-line" integration of groups of experiments in experiment racks and/or on pallets. Experiments can be inserted into the module or mated with the igloo after all subsystem functions for the experiment packages 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.

To provide flexibility, the MGSE incorporates individual supports for each major Spacelab element (core module, experiment module and pallet segments). Module-to-module or pallet-to-pallet supports can be interlocked to accommodate the various assembled groupings. In addition, a family of handling and assembly aids is provided for the igloo, racks, floor structure, the utility support structures, airlock, aft flight deck equipment, and insulation.

MGSE servicing equipment fills, drains and leak checks the fluid loops of the thermal control system.

Access platform workstands, transportation covers and shipping platforms are also provided. Access devices are foreseen to guide and support men and equipment as they enter the Spacelab module interior when in either the horizontal or vertical position, the latter after the Spacelab has been "stacked" on the Shuttle Orbiter. (Late access)

A more detailed description is given below of MGSE items which are of interest to the user.

6.2.2 Transportation Support

6.2.2.1 Experiment Rack Handling

A "rack handling and transport kit" is provided for handling individual single and double experiment racks. The kit consists of a "rack support structure" which comprises a hoisting structure and a floor stand (Figure 6-1 a) and a transport container (Figure 6-1 b). There are two sizes of the kit - one for a single rack and one for a double rack - and sufficient kits are provided for handling all the experiment racks for a long module. As shown in Table 6-1 b the transport container allows the accommodation of a fully equipped rack with protrusions within the envelope defined in Sect. 4.1.

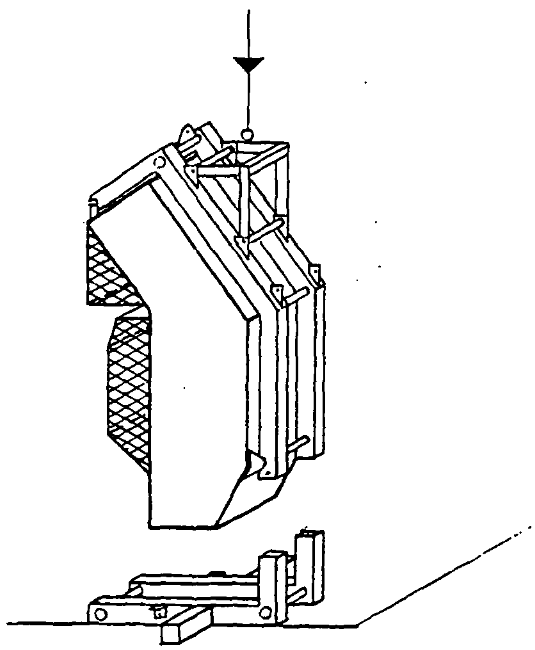

Figure 6 - 1 a: Rack Support Structure

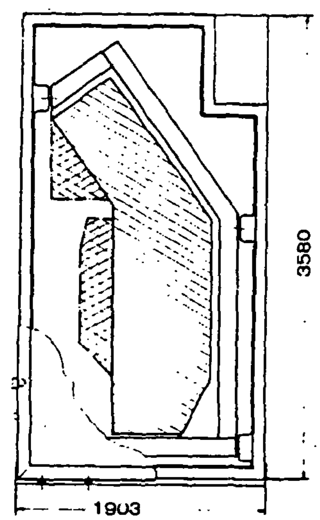

Figure 6 - 1b : Rack Handling and Transport Kit

The empty rack is delivered attached to the Rack Support Structure to the User (Level IV) Integration Site The transport container is sized to accommodate both units. The experiment rack is removed from the container and placed in a vertical position, as shown in Figure 6 – 2. Stand-alone capability is ensured by the Rack Support Structure and experiment equipment may be integrated into the rack with the support structure in place. The design of the Rack Support Structure allows for free access to the front side of the racks and allows the removal of the rear panels. The integrated rack is then returned to the transport container together with the support structure.

For integration with the module floor the rack and support structure are removed from the transport container and placed in the vertical position. The hoisting structure is then decoupled from the floor stand and the rack can be placed onto the module floor as shown in Figure 6 – 3. Before removing the hoisting structure the rack is attached to the floor and to the "racks and floor transport braces kit" which is also shown in Figure 6 – 3. The hoisting structure can then be removed.

DCN 024 The mass of the rack support structure is approximately 500 kg for a single rack and 600 kg for a double rack. The mass of the transport container is approximately 900 kg for a single rack and 1200 kg for a double rack.

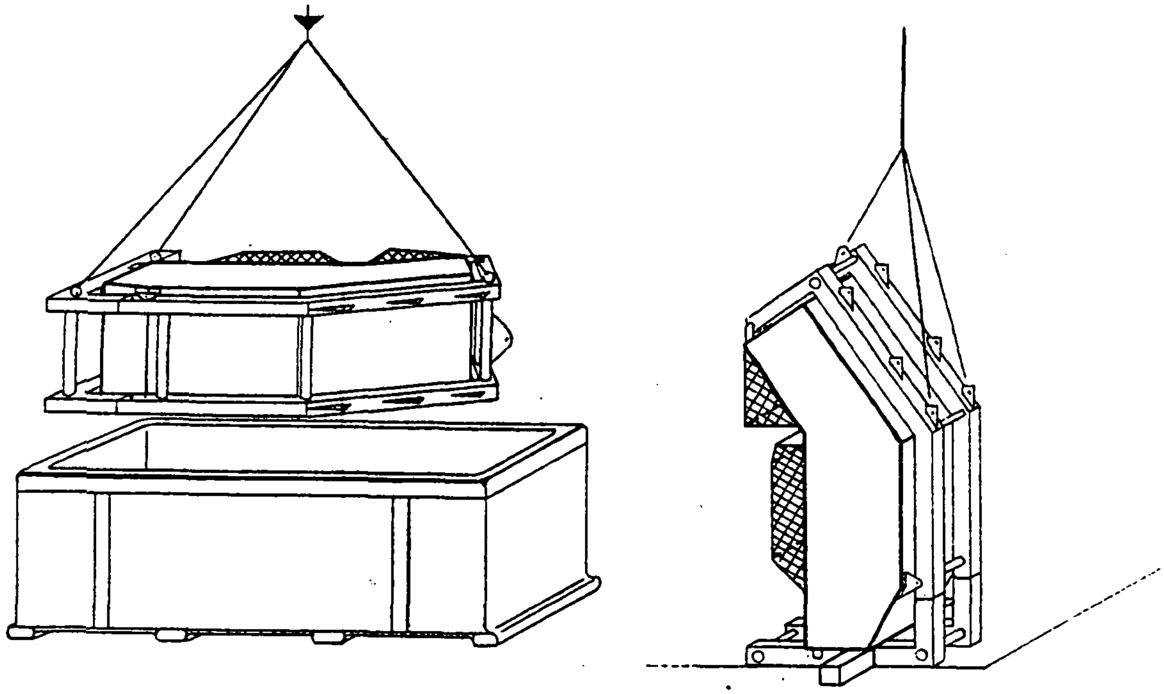

Figure 6 – 3: Removal of the Experiment Rack from the Container

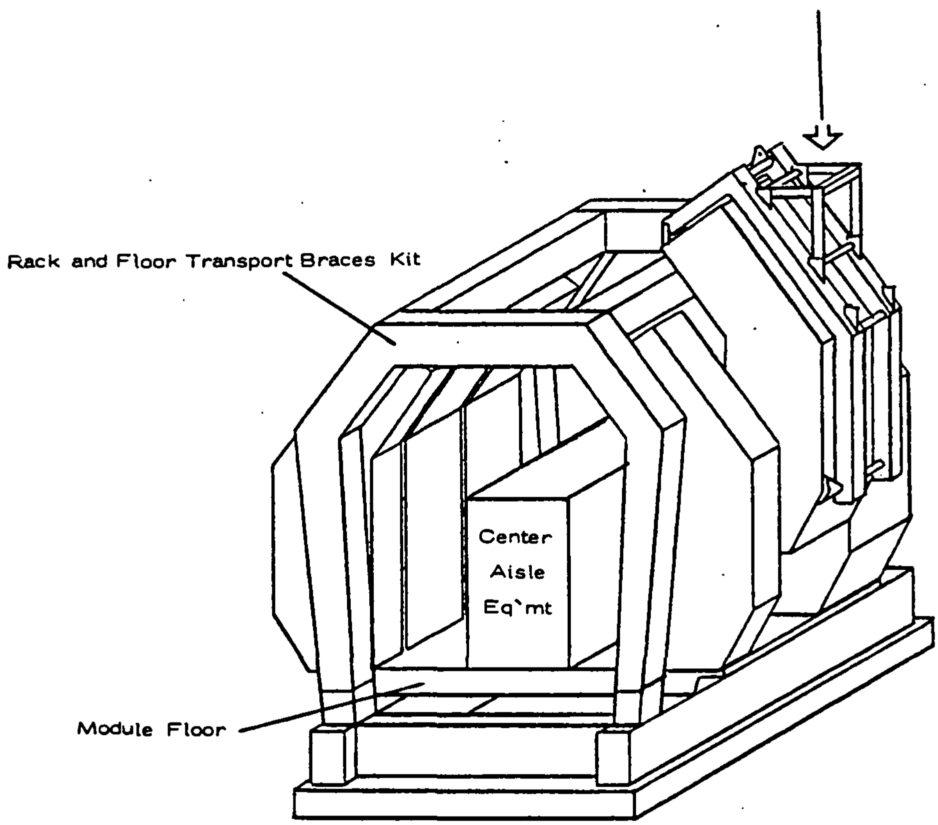

Figure 6-3 Rack / Floor Integration

When a complete floor mounted payload is assembled, rack and floor assembly combined with payload can be transported as a single unit by means of the "rack and floor transport braces kit", which provides access to the front and rear side of the racks, to the main floor, and to equipment located at the center aisle.

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6.2.2.2 Pallet Segment Transportation

Pallet transport cages and pallet transport covers are provided in two different sizes:

- the single cage to handle a single pallet without overhanging payload and without pallet utility support structure.

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- the double/triple cage to transport a three segment pallet train without overhanging payload or a two segment pallet train with overhanging payload. It should be noted that the location of the attachments for the two and three segment pallet trains does not allow transportation of a single pallet (e.g. with overhanging payload).

The masses of the transportation MGSE are given in Table 6 - 1a.

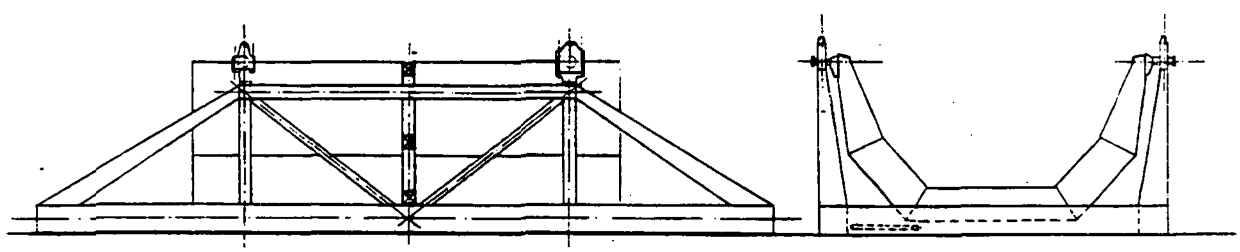

Figure 6 - 4 : Pallet Transport Double/Triple Cage

Table 6 - 1a Pallet Transportation GSE Masses

MGSE Item	(kg) Mass for single pallet	Mass (kg) for double/triple pallet train
Cover for		
empty pallet	815	-
C5A extension	300	2500
Orbiter extension	500	1200
Pallet transport cage	3850	10000
Attenuation Frame	2800	7500
Desiccants	400	1200
Trunnion handling fittings	256	256
Tie-downs	32	TBD

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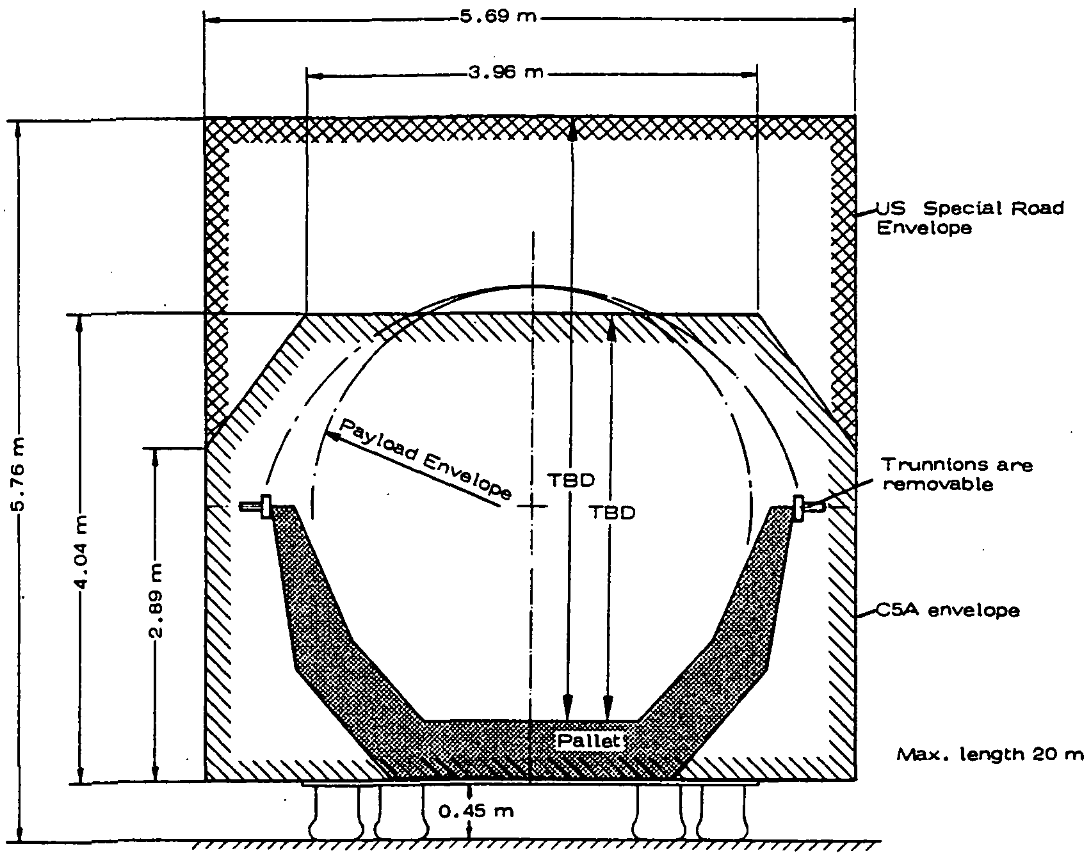

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Note: US Road Envelope (H=4.12 m; W=3.66 m) allows for athwart transportation of single pallet segments .

Figure 6 - 5: Specified Highway / C 5 A Cargo Envelope for Pallet / Experiment Transportation

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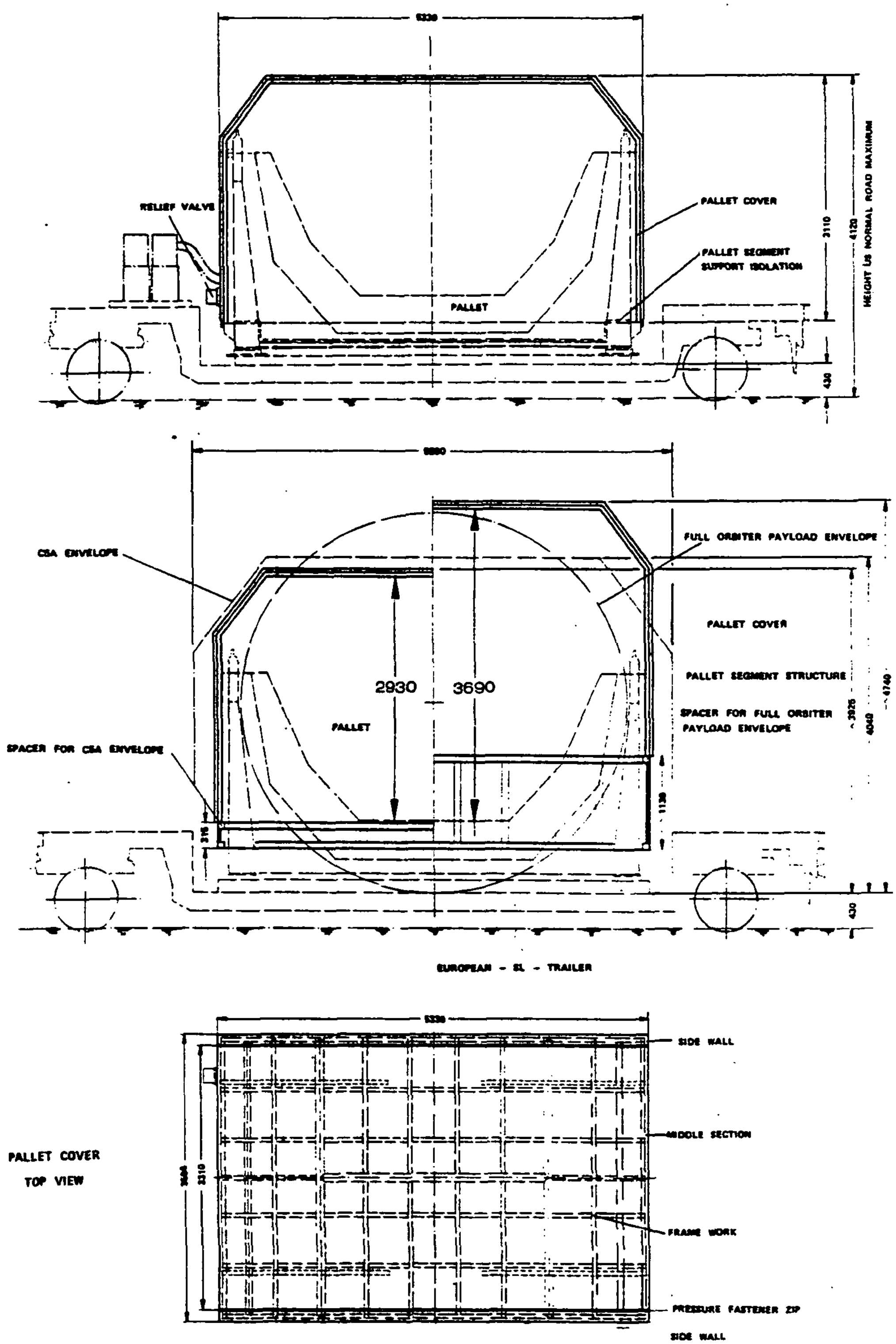

Figure 6 - 6: Pallet Transport Cover

6.2.3 Integration Support

6.2.3.1 Floor/Rack Train Roll-In

The integrated rack and floor assembly is inserted into the module by means of the racks and floor installation/removal braces kit. This kit, when attached to the floor and the racks, supports the racks when the racks and floor transport kit is removed. After attaching the racks to the overhead structure of the module, and after attaching the main floor to the main floor support structure, the racks and floor installation /removal braces kit will be removed.

The racks and floor installation /removal braces kit is also used to add or remove a rack in later operational phases outside the module.

The operational sequence of inserting the rack and floor assembly into the module is shown in Figure 6 - 7

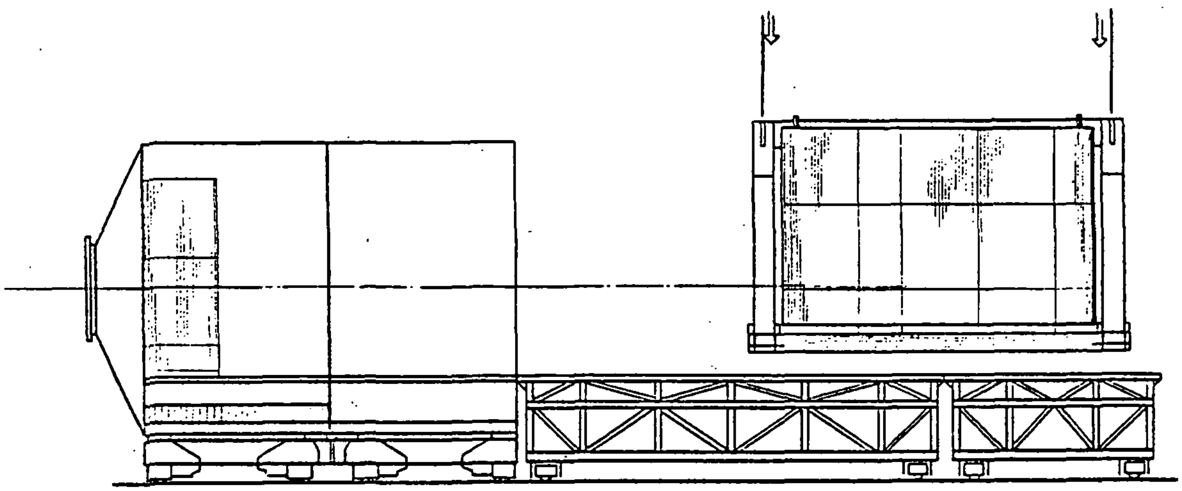

Figure 6 - 7 a: Placing of Floor / Rack Assembly on Support Structure

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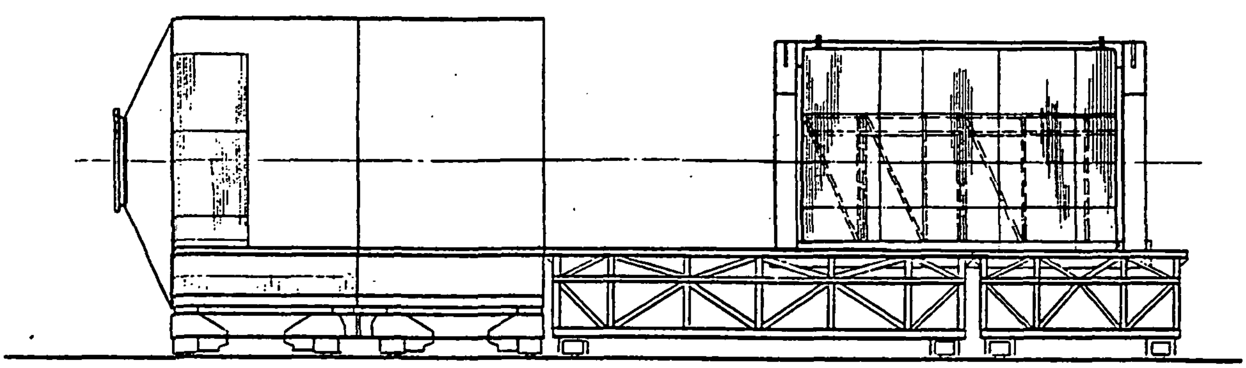


Figure 6 - 7 b Installation of Racks and Floor Installation / Removal Braces Kit

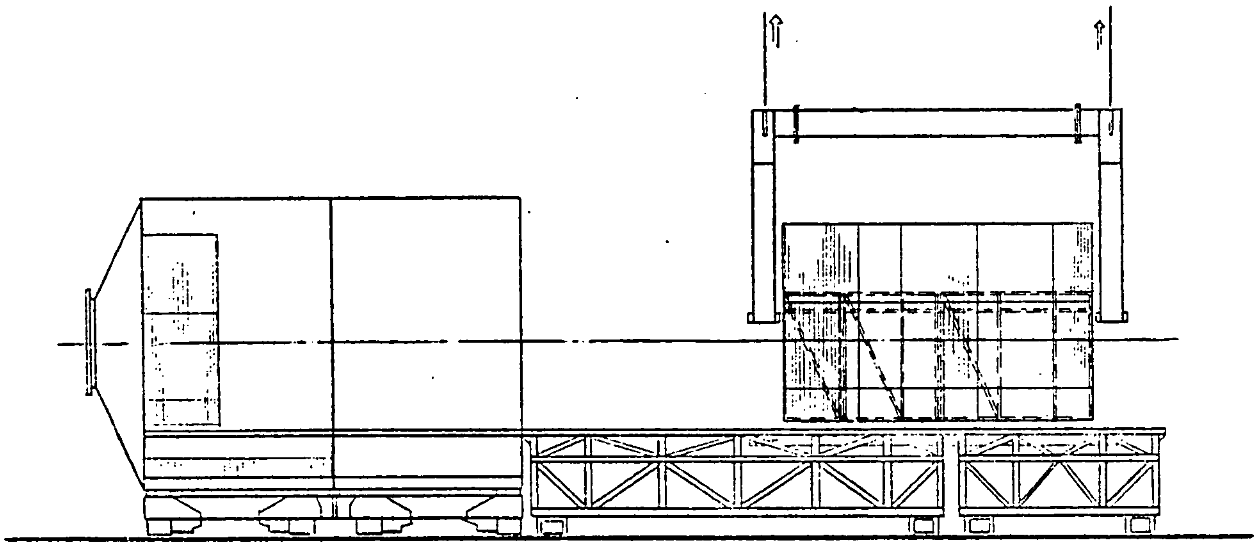


Figure 6 - 7 c: Removal of Racks and Floor Transport Kit

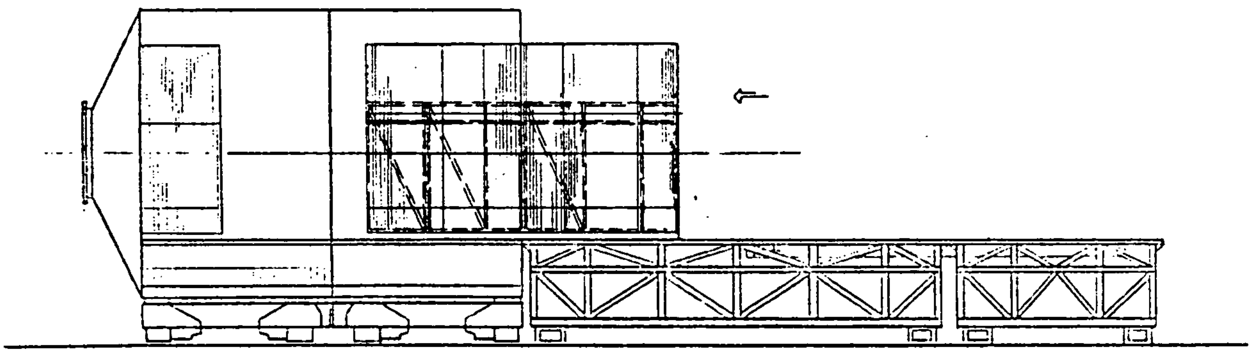


Figure 6 - 7 d : Roll-In of Floor / Rack Assembly

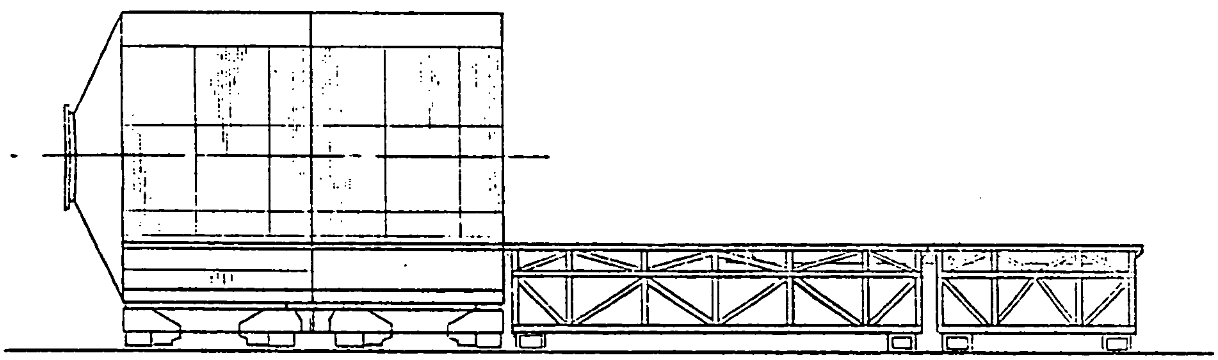


Figure 6 - 7 e Rack and Floor Installation/Removal Braces Kit Removed

As shown in Figure 6 - 8 the clearance of the racks and floor installation/removal braces kits does not interfere with the envelope allocated for center aisle equipment (see Section 3) and allows for limited protrusions of rack mounted equipment into the center aisle.

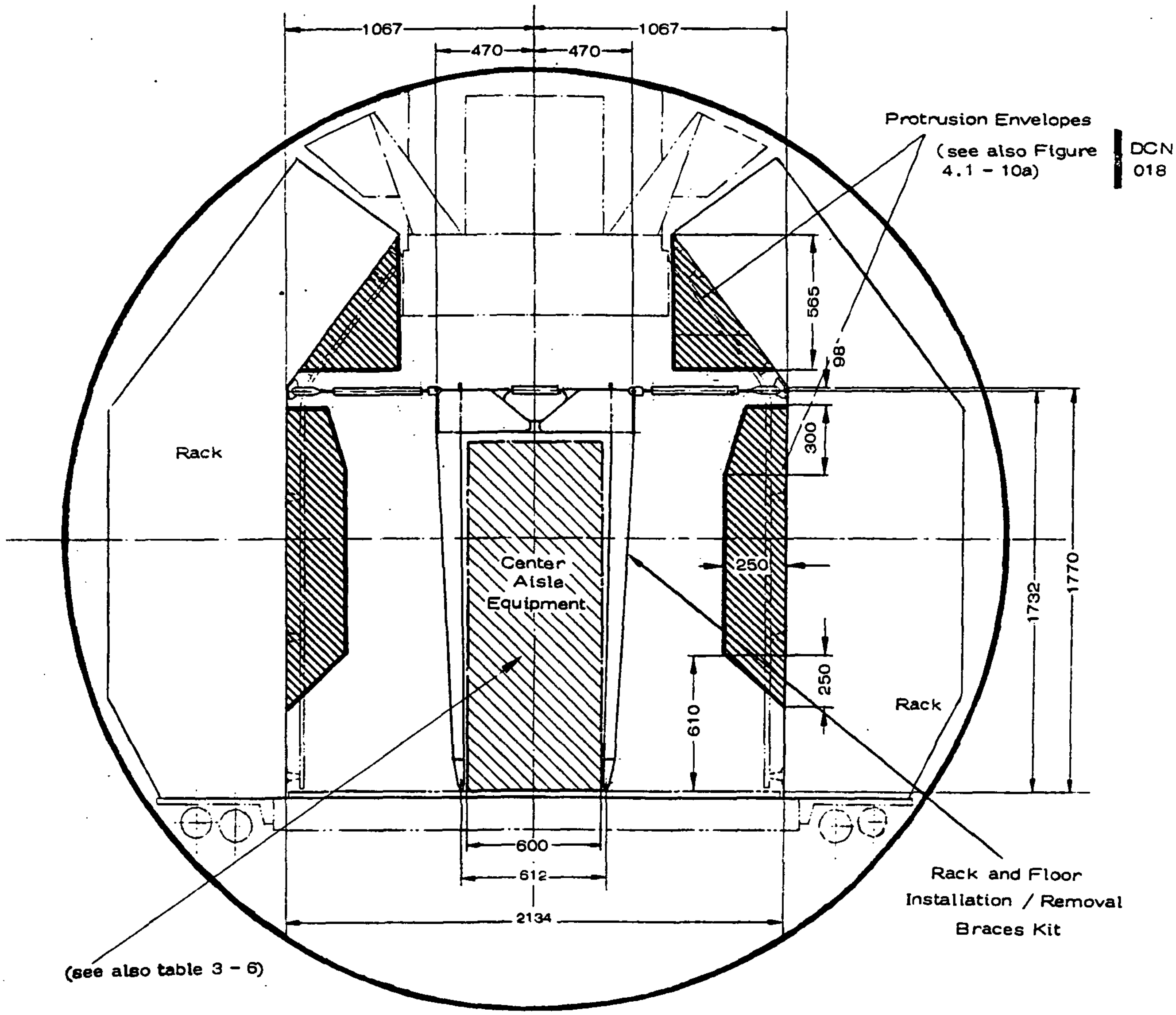


Figure 6 - 8 : Racks and Floor Installation / Removal Braces Kit Clearance

6.2.3.2 Spacelab Configuration Assembling

To provide 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, the utility bridges, airlocks, aft flight deck equipment, insulation, and other subsystem units. Figure 6 - 9 gives examples of typical integration support equipment provided.

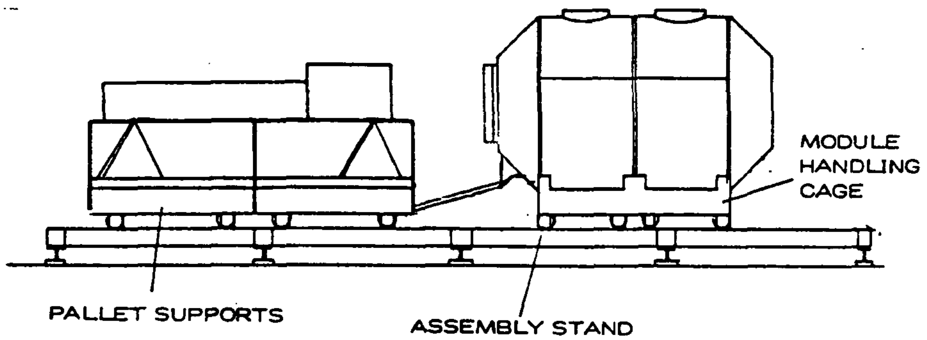


Figure 6 - 9 Matched-Rail Assembly Stand

The MGSE provides for servicing equipment to fill, drain, and leak check the fluid loops of the Spacelab 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 occurred. After each mission, it will similarly be verified whether or not environmental degradation has precluded its further use. Figure 6 - 10 gives an overview of the servicing equipment.

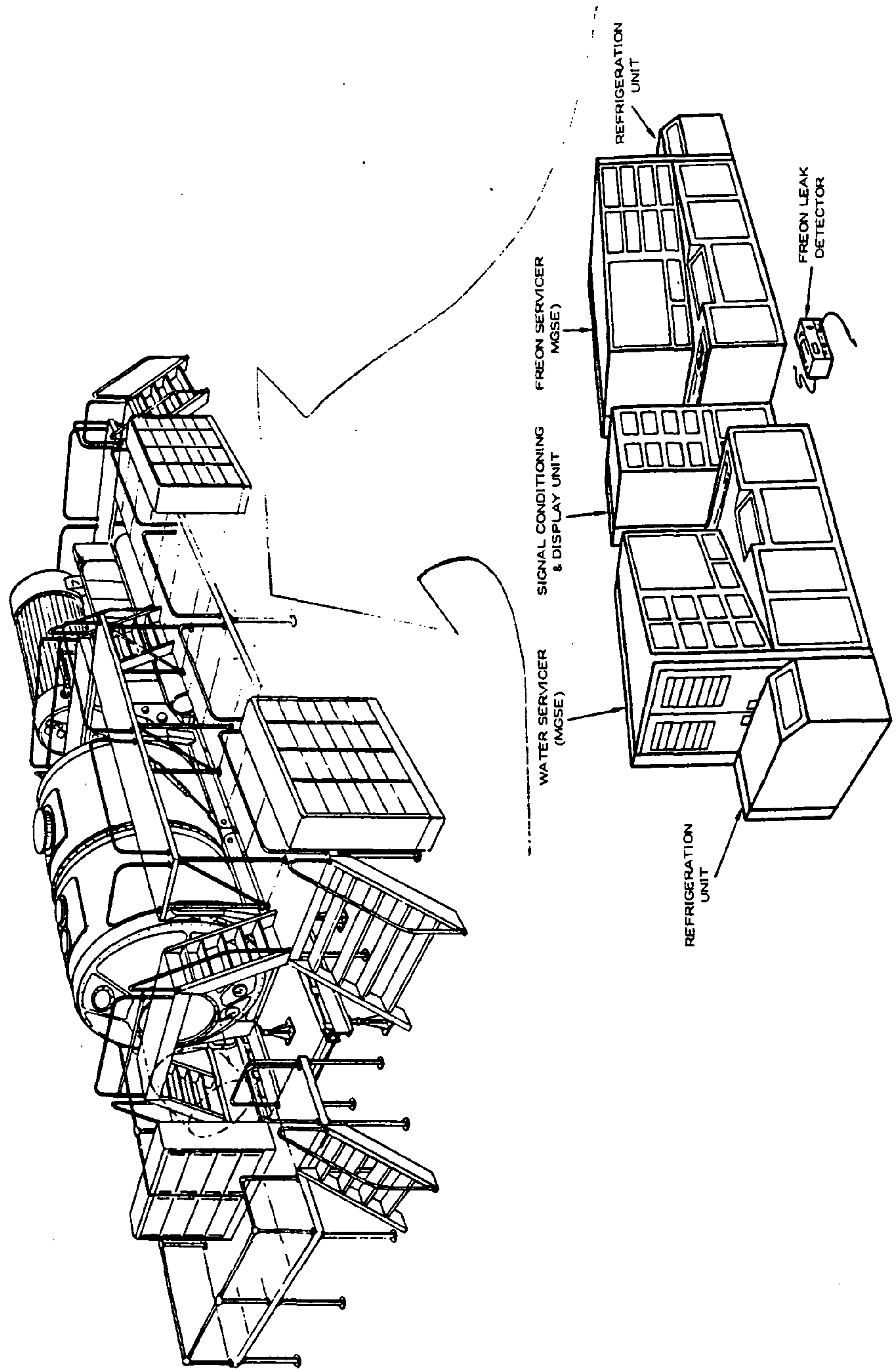


Figure 6 - 10 Servicing Equipment Configuration

6.2.4 Access Support

The MGSE access equipment is designed to support access to Spacelab subsystems and to payload integrated in the module. 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.

For late access, when Spacelab is on the launch pad in a vertical position, with the Orbiter cargo bay doors open, the MGSE provides a Vertical Access Kit. The hardware concept presently under discussion is shown in Figure 6 - 10. Major elements of the Vertical Access Kit are:

1. Main Beam
2. Main Beam Support
3. Main Beam Platforms
4. Transverse Beam
5. Sectional Ladders
6. Fold-Down Ladder Platforms
7. Support Equipment - Hoist, Equipment Dolly, Lights, Breathing Air, Rescue Kit, etc.

DCN 024 Access is only possible through the core segment CPSE opening. This opening will contain a viewport assembly, a high quality window/ viewport assembly or a closure plate.

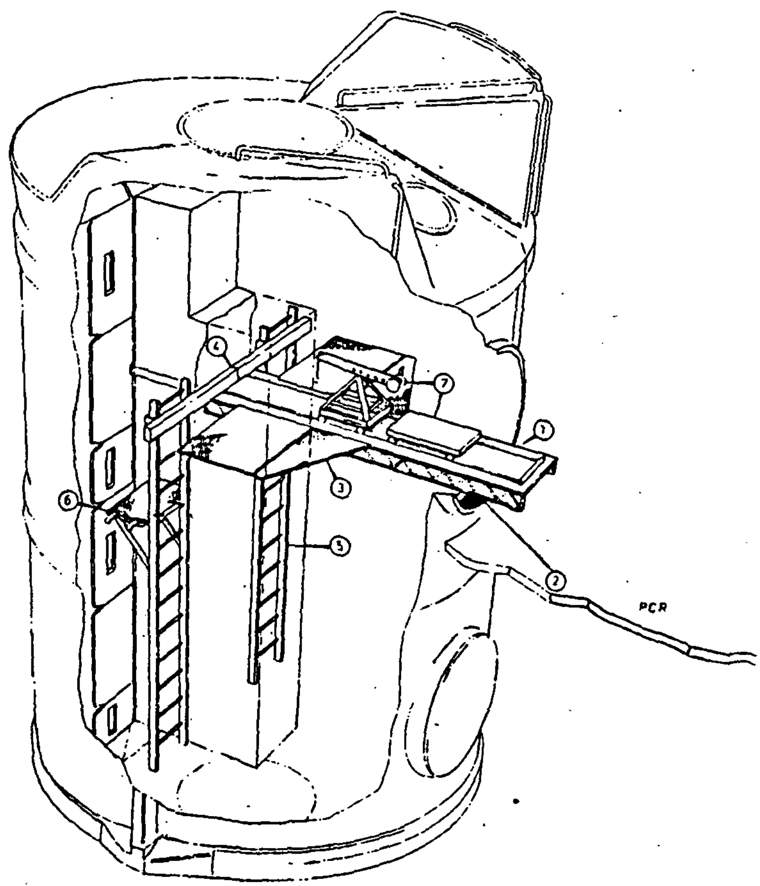
Late access into the module may be required for subsystem servicing and maintenance and for experiment installation, servicing and maintenance. These may be normal planned operations or contingency operations.

Expected experiment needs for late access are as follows:

- Biological specimens (rats, primates, etc.) will be installed as late as possible as they require continuous servicing and controlled environment.
- Experiments sensitive to electrical power or servicing interruption will be installed or finally adjusted on the launch pad.
- Experiments may require contingency corrective maintenance.

Preliminary information on the limited clearance available through the CPSE opening is given in Figure 6 - 12. In addition the clearance will be degraded by the MGSE vertical access structure itself. The overhead structure of the module can be removed, but this will reduce the capability to accommodate ceiling containers.

It should be observed that the vertical access kit, especially the main beam, may reduce the envelope available for center aisle equipment.



Access Kit Elements

1. Main Beam
2. Main Beam Support
3. Main Beam Platforms
4. Transverse Beam
5. Sectional Ladders
6. Fold-Down Ladder Platforms
7. Support Equipment- Hoist. Equipment Dolly, Lights, Breathing Air, Rescue Kit, etc.

Figure 6 - 11 Vertical Access Support

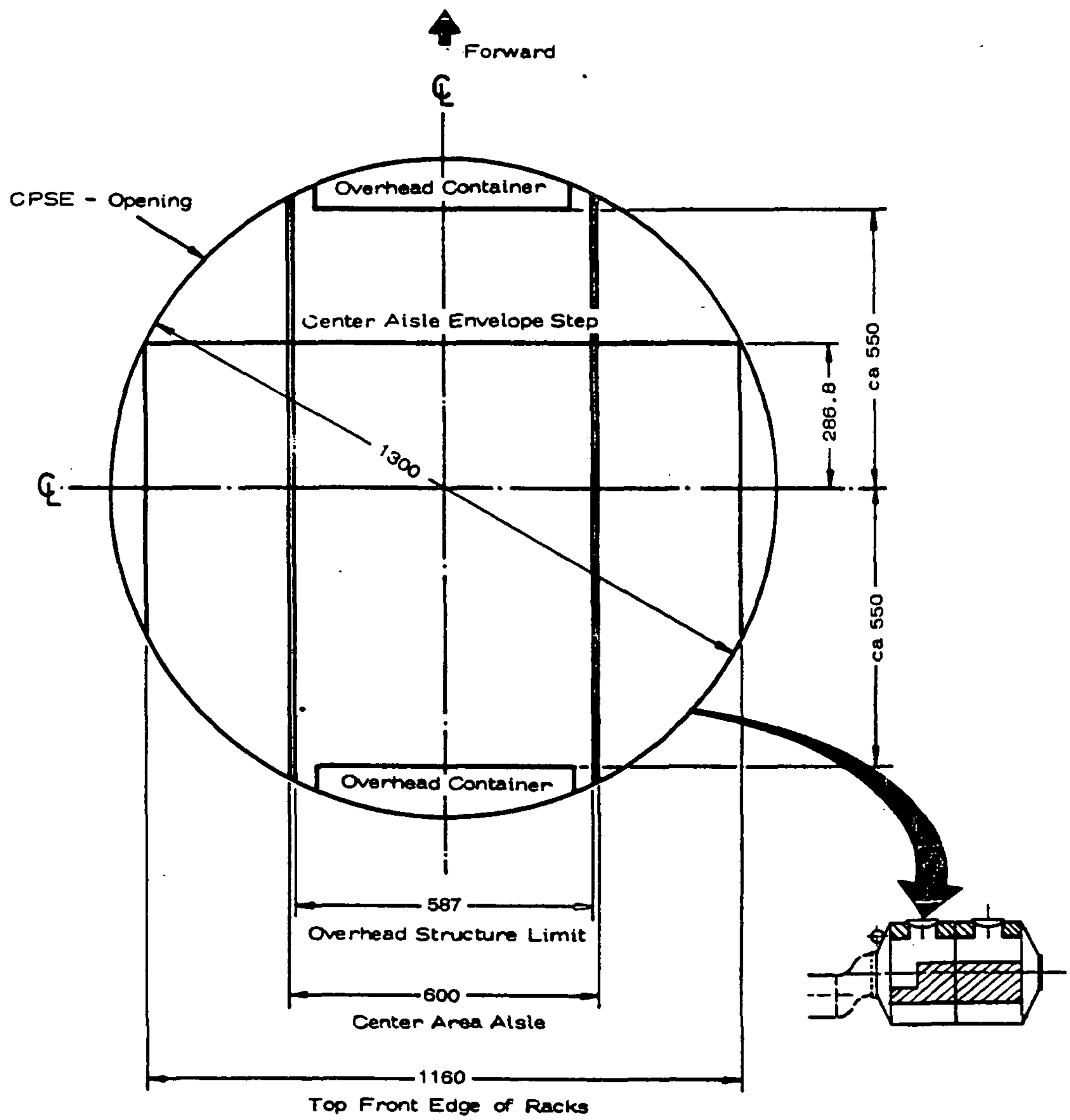


Figure 6 - 12: Clearances for Vertical Access (View Down)

6.3 Electrical Ground Support Equipment

6.3.1 General

The Electrical Ground Support Equipment (EGSE) design is based on the use of computer controlled Automatic Test Equipment (ATE) augmented by the Orbiter Interface Adapter (OIA) and a Ground Power Unit (GPU). It is designed to support the Spacelab and its payloads during the integration Levels II and III (see para 6.4). The primary purpose is to determine whether the Spacelab subsystems are operating within their design limits. In addition, EGSE equipment supports experiment integration and final verification in conjunction with special experiment provided GSE. DCN 018

Overall test control is implemented via the EGSE computer using the checkout software. 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.

6.3.1.1 Automatic Test Equipment (ATE)

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.

DCN 018 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 most malfunctions in the Spacelab subsystems to the LRU (when used in conjunction with the CDMS). Identification of experiment malfunction is supported but depends on the test points provided within the experiments and on the software loaded in the ATE computer. The ATE block diagram is shown in Figure 6 – 13.

6.3.1.2 Orbiter Interface Adapter (OIA)

DCN 018 The OIA acts as the primary EGSE to Spacelab interface during all test phases when the Spacelab is outside the Orbiter cargo bay. It simulates Spacelab related electrical Orbiter resources. Functions not provided by the OIA, but which are generated in the ATE, are routed through the OIA. The OIA hooks up directly to the Spacelab feedthrough plates in the forward end cone of the module and Igloo, respectively.

The OIA includes a simulated Orbiter aft flight deck (AFD), via which Spacelab as well as experiment AFD equipment electrical interfaces are represented.

Fig. 6 – 14 is an overall OIA block diagram with Spacelab AFD equipment included.

6.3.1.3 Ground Power Unit (GPU)

The ground power unit provides all electrical power to spacelab during all test phases when the Spacelab is outside the Oribiter. The GPU represents a full simulation of the Oribiter power interfaces to Spacelab and to Spacelab AFD equipment. This includes a simulation of the fuel cell characteristices as well as
DCN 018 protection, switching and distribution.

The GPU hooks up directly to the Spacelab feed through plates of the core segment or the igloo and to the AFD equipment installed in the OIA.

Figure 6 – 14 shows the block diagram of the GPU

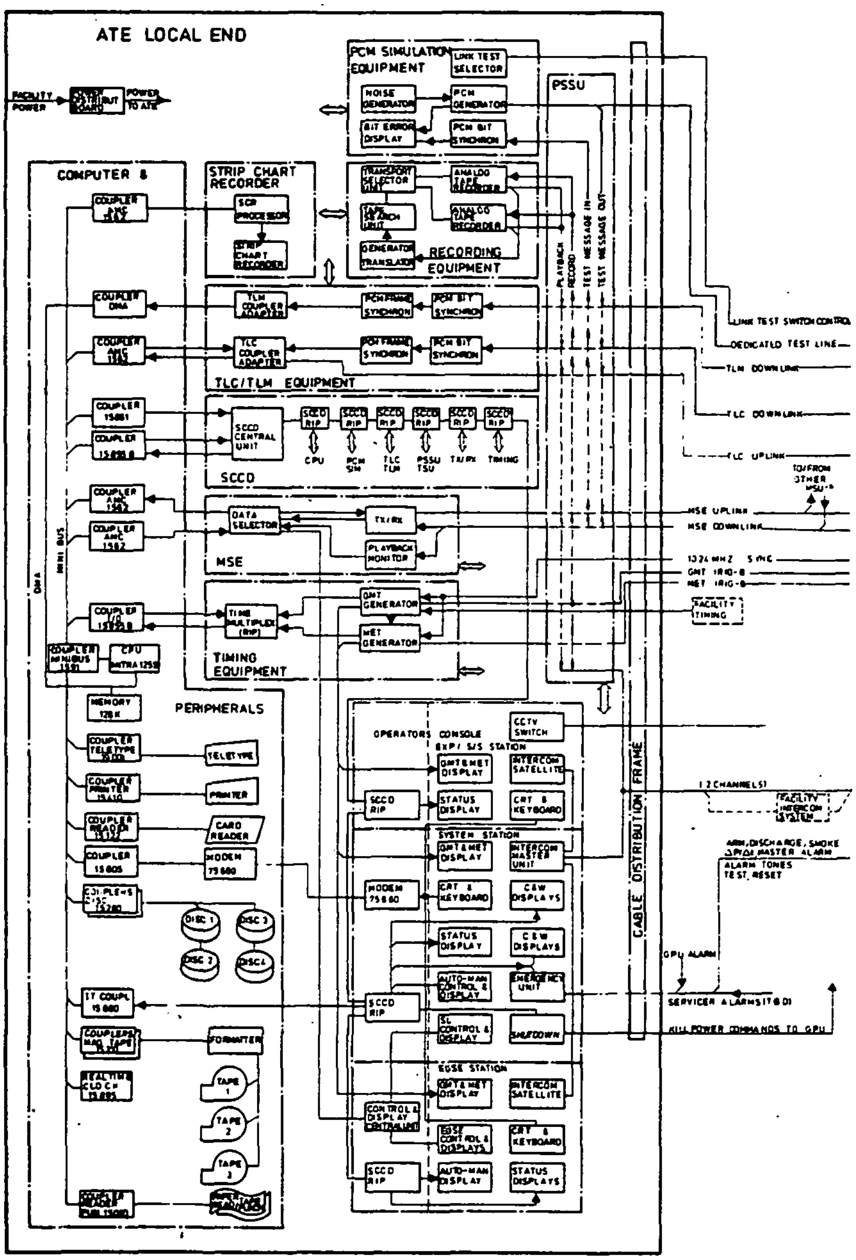


Figure 6 - 13: Automatic Test Equipment

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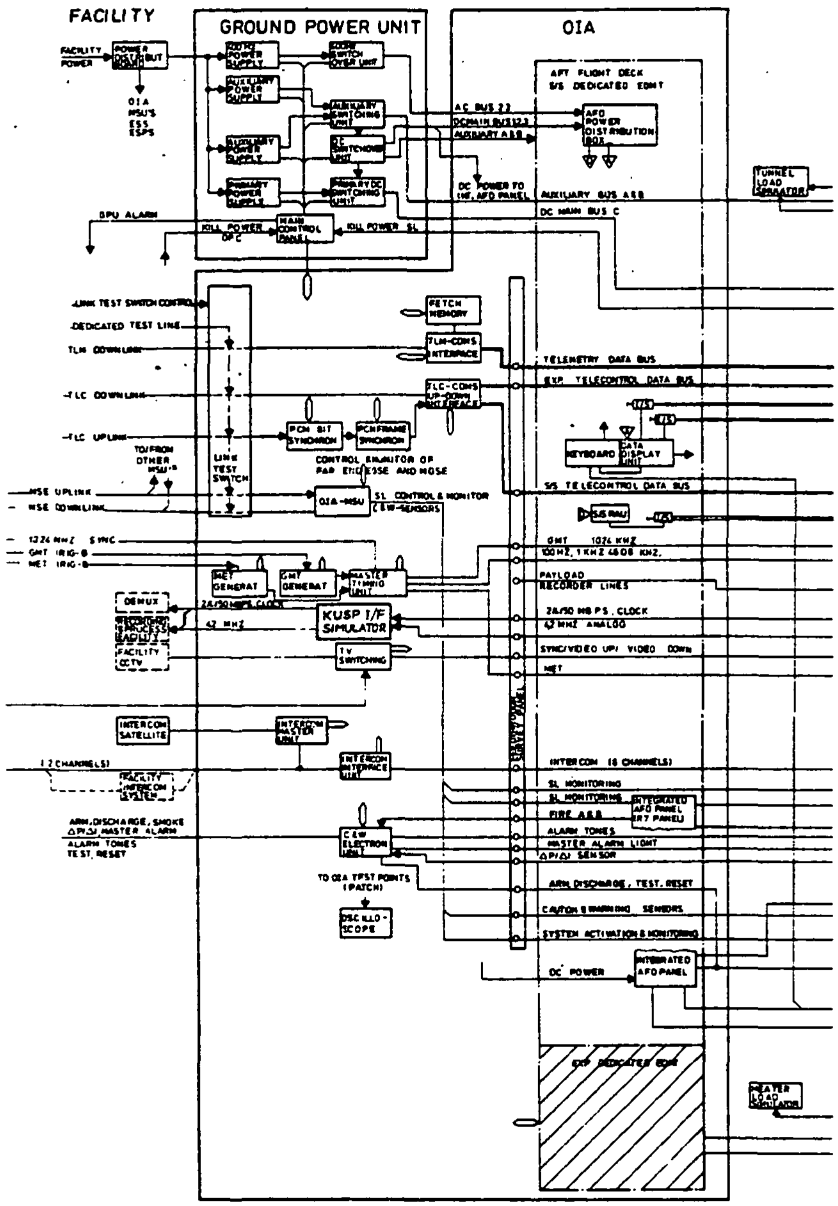

Figure 6 - 14: Orbiter Interface Adapter

6.3.2 Utilization of EGSE

In support of the payload integration and test EGSE is used in various configurations for testing of Spacelab and its payload at the following levels of integration.

6.3.2.1 Level IV Integration

The EGSE is not specifically designed to support Level IV activities; however, this support can be provided to the extent possible and practicable with existing EGSE hardware.

6.3.2.2 Level III Integration

The integration into an experiment train (and its verification) of racks / pallet segments integrated with experiments will take place within the Level III activities.
The current scope of Level III activities is restricted to mechanical integration. Functional Level III testing of the integrated Spacelab payload is not currently planned prior to integration of the payload with the basic Spacelab on-board subsystems.

6.3.2.3 Level II Integration

The assembly of the experiment train with Spacelab flight support subsystem elements (module or igloo) will take place within the Level II activitites. The applicable test configurations are presented in Figure 6 - 15.

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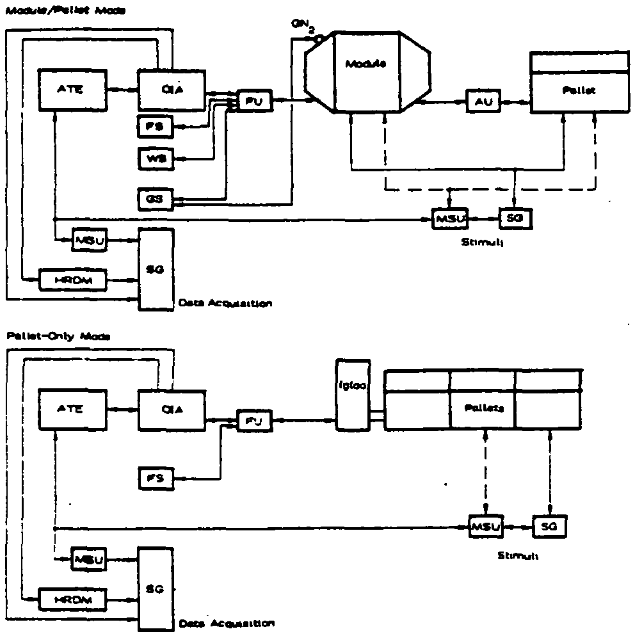

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AU	-	Aft Utilities
FS	-	Freon Servicer (GSE)
FU	-	Forward Utilities (GSE)
GS	-	Gas Component Test Stand
SG	-	Special Experiment GSE (experiment provided)
WS	-	Water Servicer (GSE)

Figure 6 - 15: Level II Test Configurations

In the above test configurations, the experiments interface with the EGSE through the normal experiment to Spacelab and Spacelab to Orbiter channels; 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 availablity of suitable checkout software from experimenters.

In the post-flight phase EGSE supports Level II contingency activities in test configurations as described before.

Figure 6 - 16 illustrates the Spacelab Assembly Stand with associated EGSE. It encompasses the Local End EGSE (ATE, operator consoles, computer etc.) and the Far End GSE (the ground power unit (GPU) and OIA). The Local End EGSE is physically separated from the actual assembly stand.

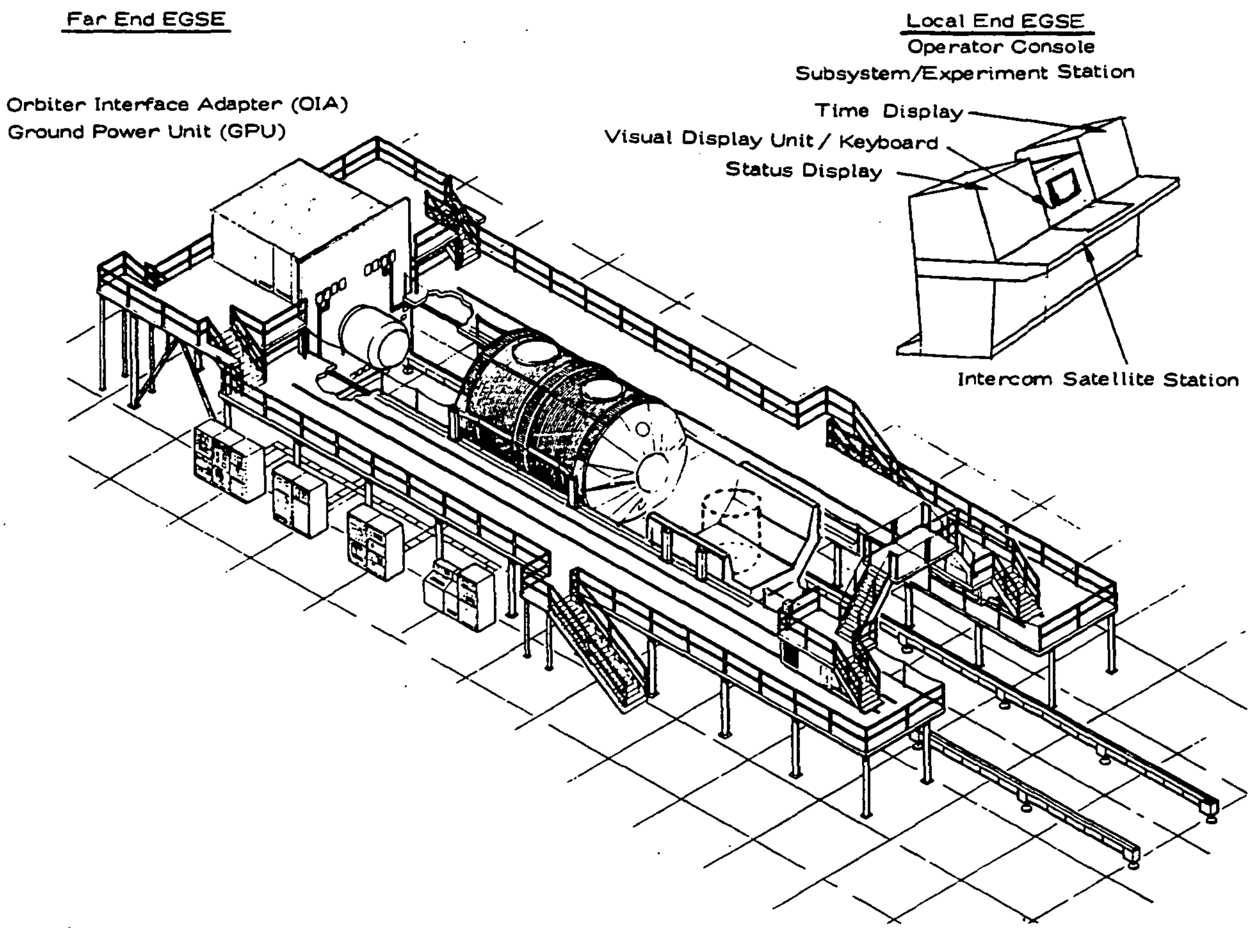

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Figure 6 - 16: Spacelab Assembly Stand With Associated EGSE

6.3.3 Experiment Checkout

6.3.3.1 Concept

Experiment checkout during Payload Integration can make use of the SL Subsystems installed in the Core Segment, Experiment Segment and on Pallets, of SL EGSE and of Experiment GSE if required (See Figure 6-17). However, experiment checkout is a payload function and its accomplishment during Level III and Level II integration will require prior approval from the Spacelab Program.

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The basic checkout task is performed by the Spacelab CDMS Experiment Computer. This task includes:

- Command Execution
- Data Acquisition
- Data Monitoring/Display
- Telemetry Formatting (for PCMMU link)

The operator interface to the CDMS experiment computer during the C/O phase is via the S/L DDU and Keyboard or via the ATE operators console. The ATE provides an experimenter dedicated console station communicating with the CDMS experiment computer via the simulated serial MDM link.

In order to coordinate tasks performed by the experiment computer and the experiment GSE, the ATE computer provides a hardware interface to the experiment GSE. This interface is provided by means of the Measurement and Stimuli Units of the SL EGSE. The MSU characteristics are described in para 6.3.3.2.

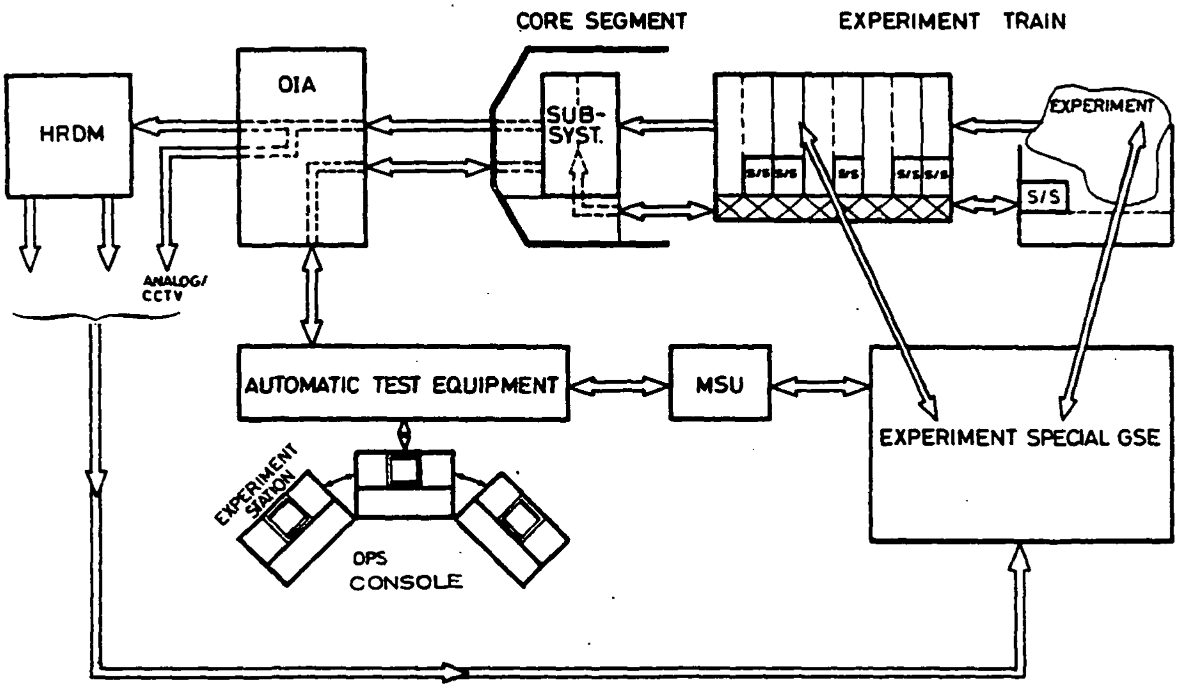

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Figure 6 - 17: Experiment Checkout

The software residing in the ATE computer for payload checkout supervision will have to be written by the Payload Integration contractor. Basic features and constraints are described in para 6.3.3.2.

Experiment GSE will be required for checkout to perform the twofold task -

- stimulation of experiments
- processing and assessment of scientific data resulting from the stimulation.

The scientific data can be acquired via the HRM link and via the PCMMU link.
In the HRM link data are acquired by the on board HRM and then hardwire transmitted to, and demultiplexed by, the HRDM. The HRDM, being part of the Spacelab EGSE, interfaces directly with the experiment GSE and no constraints on data rates are imposed compared with the capability in flight.
In the PCMMU link the experiment data are acquired (via the PCMMU simulated in the OIA) by the ATE using payload provided software resident in the ATE computer. The maximum transfer rate into the ATE is 600 kb/s. However, the further transfer of these data, from the ATE computer via the MSU to the experiment GSE, is constrained to about 256 kbs. DCN 018

In addition to the task of coordinating experiment/experiment GSE and Spacelab subsystem operations the Spacelab EGSE can also serve as a link between experiment GSE and the experiment itself (instead of a direct hardwired link).

As shown in Figure 6 - 18, the checkout of the HRM links is a typical example of such a configuration.

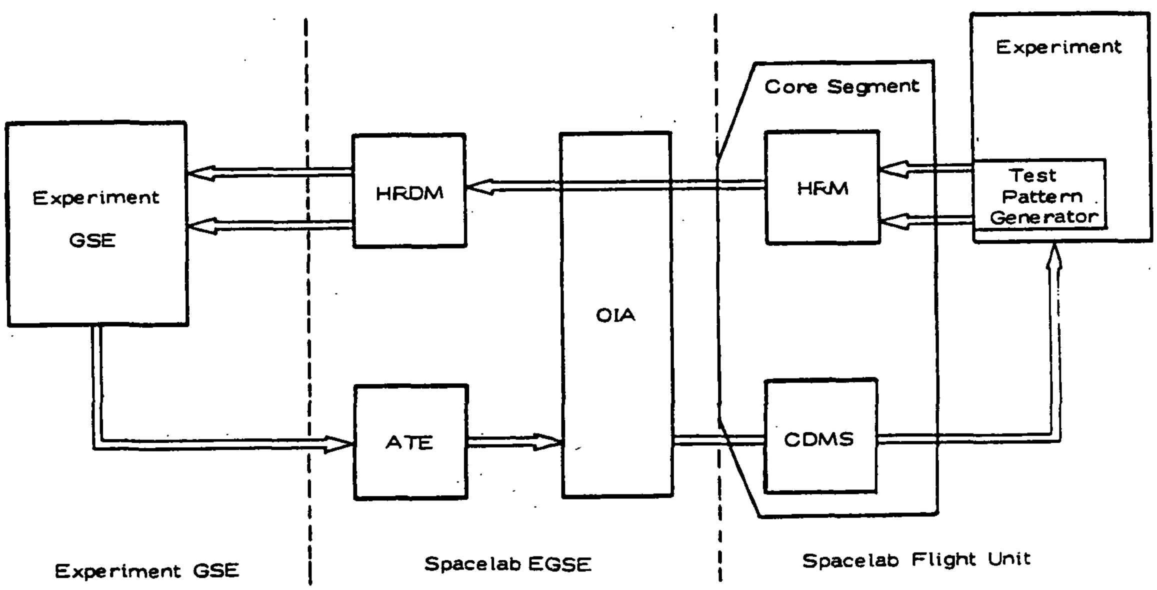

Figure 6 - 18: Checkout of HRM Links

6.3.3.2 MSU Characteristics

A functional block diagram of the MSU is shown in Figure 6 – 19.
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 trans-
DCN 018 ferred via this bus assuming that suitable software is provided. The following two principal operating modes are possible:

- software controlled
- manual access via keyboard and CRT data display (software controlled)

Each MSU provides the following user input and output channels for experiments:

- Measurement

  56 Analog Inputs (11 bit resolution)
  40 Discrete Inputs
  3 Digital Inputs (12 bit parallel)

- Stimuli

  28 Analog Outputs
  40 Discrete Outputs
  3 Digital Outputs (12 bit parallel)

For details see Avionics Interface Definition Appendix A

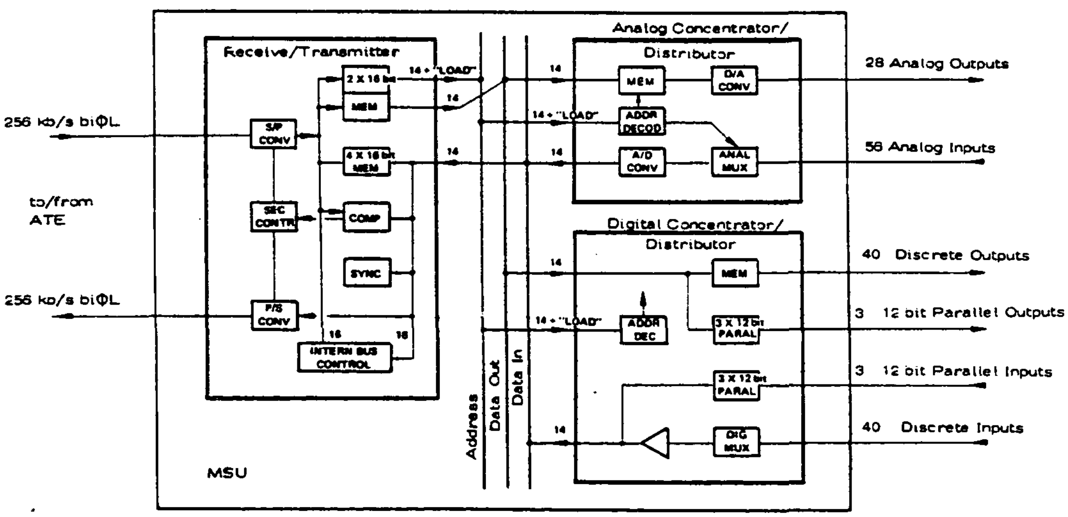

Figure 6 – 19: Functional Block Diagram of MSU

6.3.3.3 ATE Software

A software package for experiment checkout in the ATE computer has to be user provided. The present estimate of core memory size for such an applications package is 10 K words. This package has to interface with the Ground Computer Operating System (GCOS) via interpreters. It is assumed that the applications package will be written in GOAL. This will allow a Spacelab baseline ATE console interface for the operator and the use of the keyboard and checkout interpreter package developed for the ATE computer.

Typical tasks for this software will be:

- Activation/deactivation and control of experiments via CDMS experiment computer.
- Activation/deactivation and control of experiment GSE via MSU.
- Monitoring and processing of selected telemetric parameters (acquired via simulated PCMMU link)
- Print-out of test results on ATE printer.
- Strip charts of selected parameters on ATE SCR.
- Step-by-step checkout with the ATE console operation in the loop.
- Acquisition of experiment GSE events with correlation to experiment events/parameters reactions.
- Execution of commands from experiment GSE.

It must be noted that scientific data acquired by the SL HRM do not enter the ATE computer. Such data have to be recorded/processed by experiment GSE.

6.4 Ground Operations

Ground operations are those activities directly associated with the pre-mission physical preparation of Spacelab and experiment hardware for the flight, and the post mission activities associated with Spacelab and experiment recovery, Spacelab maintenance and reconfiguration.

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.

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, and those Spacelab unique maintenance and checkout activities which occur prior to arrival of Level IV integrated payload elements at the O & C Building.

The data presented include

- o Groundrules and Assumptions
- o Ground Operations Processing activities
- o Standard NASA Spacelab Facilities

The overall ground operations flow is depicted in Figure 6 - 20 .

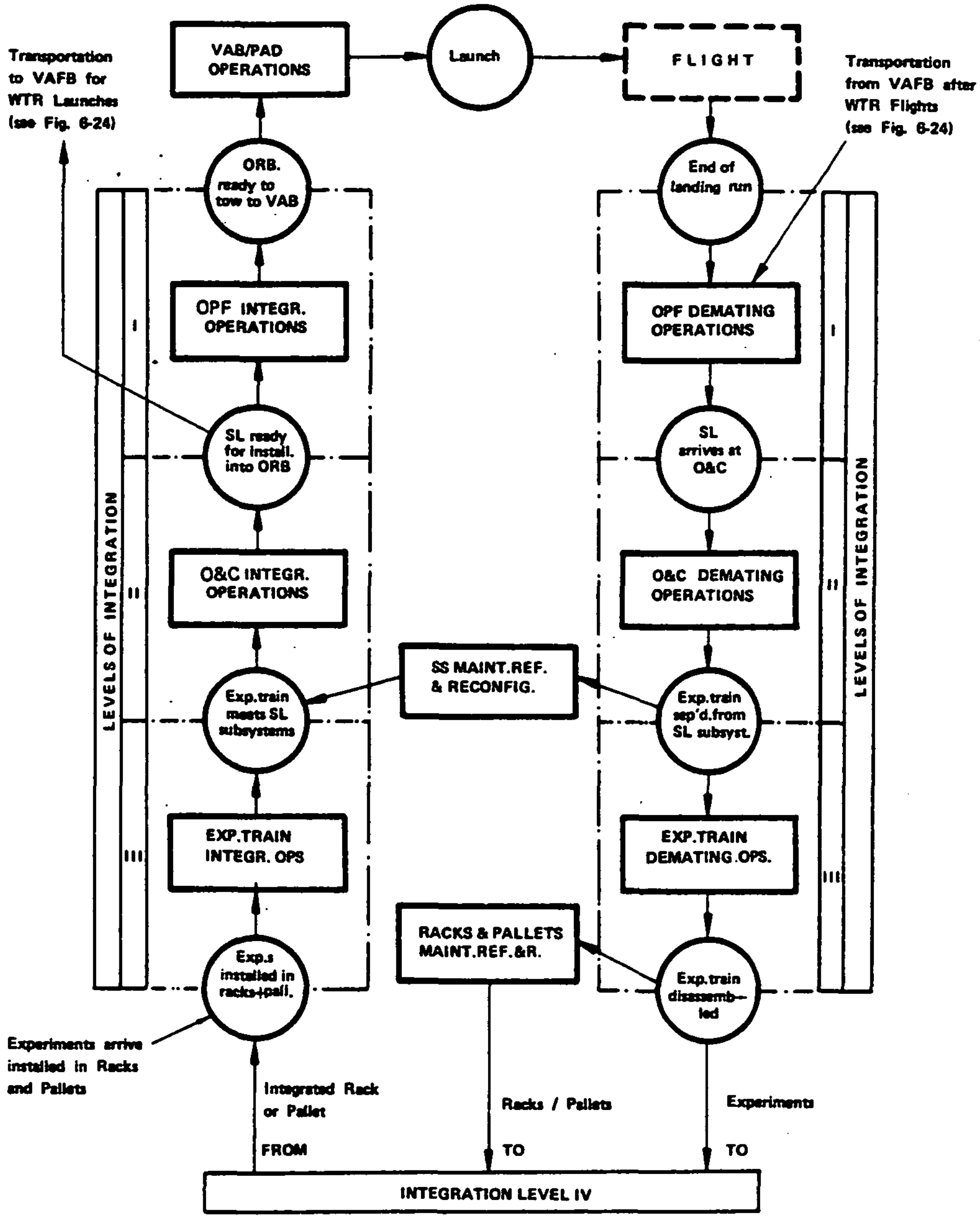

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Figure 6 - 20 Overall Ground Operations Flow (KSC Launch)

6.4.1 Groundrules and Assumptions

The ground operations concept is based on the following groundrules and assumptions:

The ground operation processing of Spacelab and of its payloads is performed at different integration levels, as described in para 6.1.

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.

- Spacelab processing activities considerations and constraints include:

  - 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 operations in a vertical position. As a goal planned or contingency access to experiments will be possible up to 11 hours before launch. Access to the interior of the Spacelab during pad operations will be limited to functions which are time-critical and, therefore, cannot be performed earlier in the ground flow. 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 15 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.
  - The post mission refurbishment of Spacelab hardware will 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.

6.4.2 Ground Operations Processing Activities

The Spacelab ground operations philosophy is influenced by two primary factors of the Spacelab program, these are: (1) The reusable nature of the Spacelab carrier elements and....

(2) The Spacelab traffic model.

To minimize Spacelab program costs and associated costs 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 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.

The Spacelab operational phase Ground Operations encompass those operations associated with the normal turnaround processing of fully operational Spacelab flight hardware. Figure 6 - 21 provides a summary of the basic ground processing flow applicable to the operational phase.

6.4.2.1 Functional Flow

The baseline ground processing is given in Figure 6 - 21
The activities are described in detail in items (1) thru (12).

The timeline (Figure 6-22) identifies the sequence constraints and typical time allocations for each block activity shown on the functional flow block diagram.

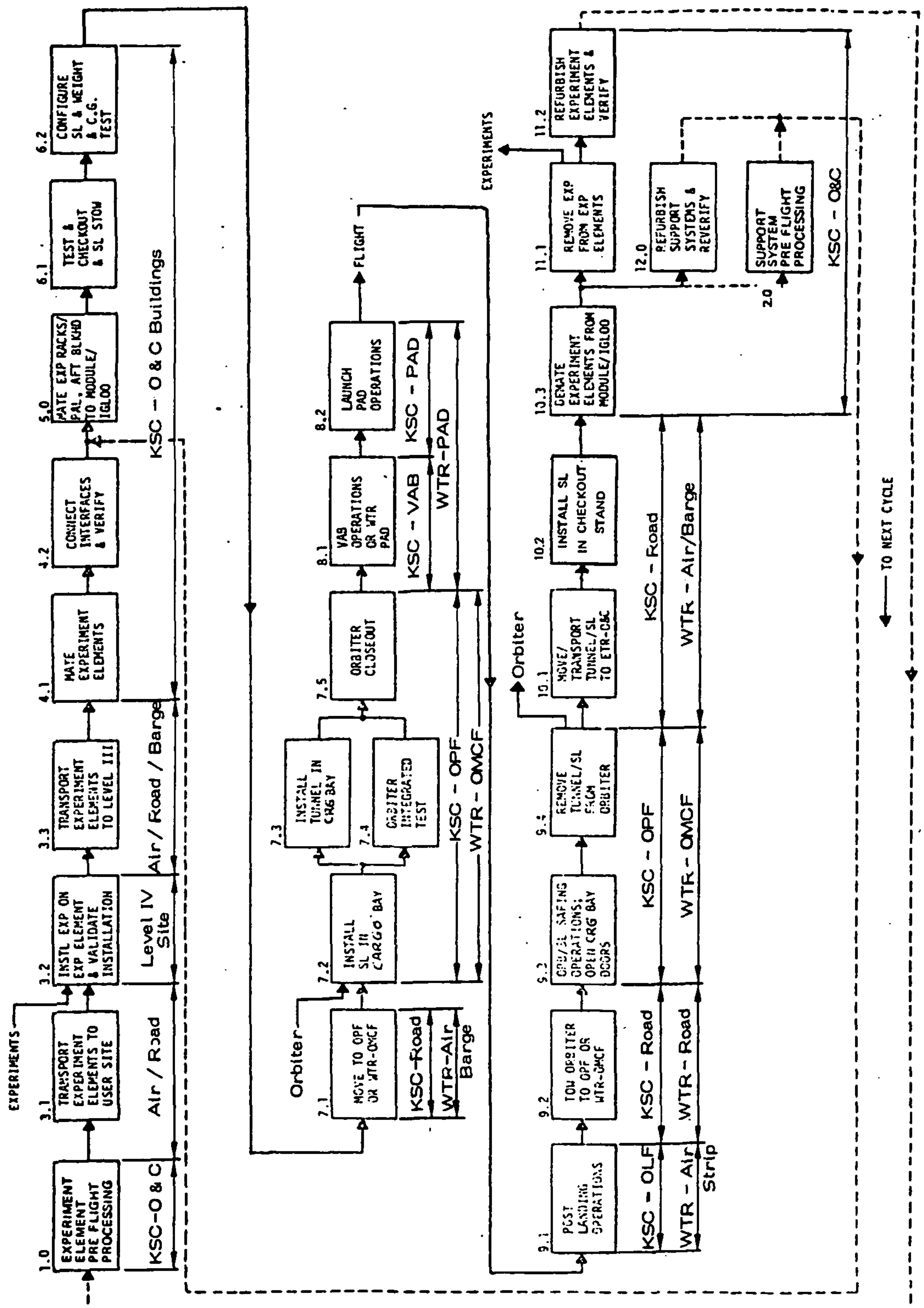

Figure 6 - 21: Ground Processing Flow - Operational Phase

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This Figure is TBD

Figure 6 - 22: Operational Phase Timeline, Spacelab Ground Operations

Activity Descriptions* (see Figure 6 - 21)

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 core segment and experiment segment 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
- Set-up rack cooling airflows.
- Perform special experiment tests
- Prep for transportation

(4) Level III integration

- Receiving inspection
- Install experiment elements in rack stands / pallet stands
- Mate pallet segments, connect interfaces and verify
- Mate rack sets, connect interfaces and verify
- Final checks of combined racks airflow settings.

(5) Level II integration

- Inspect and install flight hardware in check out (C/O) stand
- Mate and verify GSE interfaces
- Install rack and floor assembly in module and perform bonding checks
- Connect subsystem and experiment interfaces and verify
- Connect pallet/module interface harness
- Mate aft end cone to module

*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) equipment in EGSE and connect interfaces
- Payload pre-power verification
- Module/pallet/rack/experiment functional interface verification
- Simulated Orbiter mission sequence test
- Air entrainment checks
- Post test securing
- Remove AFD equipment from EGSE
- Flight crew equipment stowage
- Spacelab internal shakedown inspection
- Spacelab close-out / preparations to move
- Spacelab external shakedown inspection

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(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 AFD equipment in Orbiter cabin and verify interfaces
- Provide access into Spacelab
- Orbiter integrated test - verify Spacelab/Orbiter functional interfaces
- Install tunnel in Orbiter and leak check interfaces
- Perform Spacelab closeout inspection
- Stow time-critical items in Spacelab
- Remove all non-flight protective covers
- Post test secure and Orbiter checkout
- Remove all access equipment
- Configure Spacelab cabin switches for launch

- Close cargo bay doors
- Activate cargo 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 cargo bay purge
- Open cargo bay doors
- Remove or cover experiments, as required
- Install Spacelab access equipment (if required)
- Demate and remove tunnel
- Demate and remove Spacelab
- Remove AFD equipment
- Prep Spacelab hardware for move or transport DCN 018
- 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
- Spacelab entry preparations
- Remove experiment specimens and data
- Spacelab interior inspection
- Flight crew equipment destowage

- Remove pallets
- Aft end cone demate
- Disconnect and remove rack and floor assembly

(11) Experiment elements disassembly and refurbishment

- Install experiment elements in rack stands / pallet stands
- Disconnect and remove experiments from experiment elements
- Prep experiments equipment for shipment to user
- Control, monitor and verify shipping environment
- Disassemble rack and floor assembly
- Perform maintenance, refurbish, and reverify experiment elements, harnesses, etc.

(12) Support system refurbishment

- Perform maintenance, refurbish and reverify core/experiment segments, igloo, aft end cone, CPSE, tunnel, AFD equipment and harnesses as applicable

6.4.2.2 Vertical Payload Removal

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 cargo bay as an integrated package and to position on transporter
- Spacelab is in deactivated status

6.4.2.3 User Involvement

Figure 6-23 shows the typical user involvement in the Spacelab ground operations processing and the associated function in which in the item is used.

COORDINATE FUNCTIONS	APPLICATION: Planning Phase	Processing Activities (Ref. Fig. 6-21): 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 - 23: User Involvement in Spacelab Ground Processing

6.4.2.4 Vandenberg Air Force Base (VAFB) Launch/Landing Operations (Western Test Range)

Ground operations will provide the capability of supporting a Spacelab launch from VAFB. Major requirements and constraints are as follows:

- The Spacelab and its integrated payload arrives pre-assembled at VAFB 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) as applicable.

Figure 6-24 is a flow diagram isolating the unique hardware processing activities and sequence necessary to support a VAFB launch/landing option. This flow is the alternate processing flow for VAFB and would substitute for items 7.1 thru 9.4 on Figure 6 - 21.

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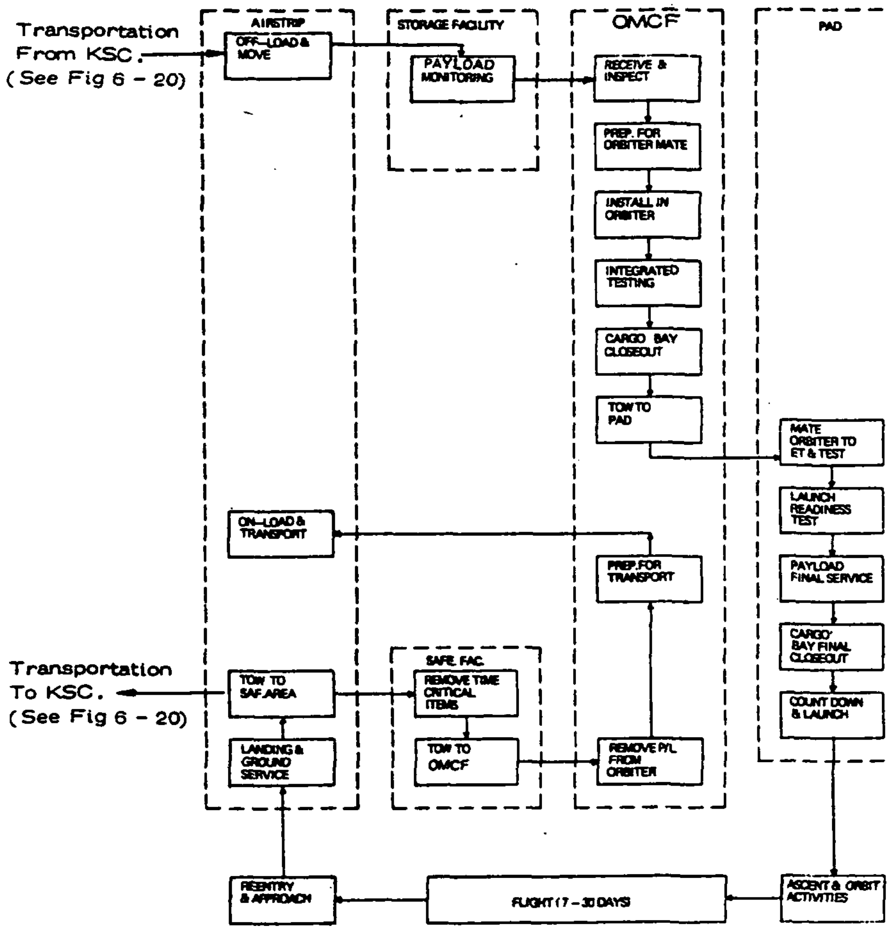

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Figure 6 - 24: VAFB Launch /Landing Processing Flow

6.4.2.5 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).

6.4.2.6 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.

6.4.3 Standard Facilities for Spacelab and Spacelab Payloads

Beside the Spacelab MGSE and EGSE described in section 6.2 and 6.3 KSC and VAFB facilities are available 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.

DCN 018 This subsection identifies the payload involvement at KSC, VAFB and Level IV experiment integration sites to support planned NASA payload prelaunch and post mission ground processing operations.

6.4.3.1 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 within the flow. 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 and pyrotechnic loading.
- Pad/PCR - Access to the exterior or 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.

Table 6 - 1 Main Facility Characteristics

Facility	Primary Uses	Environment
Operations and Checkout Building (O & C Bldg)	1) Spacelab refurbishment 2) Spacelab Processing 3) Horizontal cargo integration	see section 5.3.2
Orbiter Processing Facility (OPF)	1) Orbiter refurbishment 2) Payload installation and interface verification	see section 5.3.2
Vehicle Assembly Building (VAB)	Shuttle assembly	see para 5.3.2.2.
Payload Changeout Room (PCR)	Vertical payload changeout	see section 5.3.2
Launch pad	1) Shuttle Launch 2) Payload installation and interface verification	not controlled

6.4.3.2 Facility Characteristics (NASA)

The main characteristics of the Spacelab NASA facilities are described in Table 6 - 1 . The services available to Spacelab payloads are:

a) Spacelab Staging

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 III/II Integration

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.

c) Level I Integration DCN 018

Fluids services will be provided for Spacelab through umbilical connections. Fluids that are payload peculiar (such as unique gases) must be supplied by the user. Launch facility will provide the capability of topping off fluids to operational levels and pressures (payload service).

6.5 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 6.5.1 is a synopsis of the planned Spacelab operations/capabilities and service.

Subsection 6.5.2 consists of the Groundrules and Guidelines under which Spacelab Flight Operations has been developed.

DCN 018 Subsection 6.5.3 summarizes the concept for the User's flight operations utilization of the Spacelab.

6.5.1 Operational Capabilities Description

6.5.1.1 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 and the degree of POCC control will be determined by the requirements and constraints of the individual flights.

6.5.1.2 Flight Phases

a) Ascent/Descent

During ascent, launch through orbital insertion, descent and 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. These holding functions will be limited to power and cooling only, within the constraints of the Orbiter resources and the partially active Spacelab subsystem consumption DCN 018

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.

6.5.1.3 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 operations of payloads conducted on board and will advise and consent in ground control operations. 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 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 community 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.

6.5.2 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 6.5.3.

- Payload mission requirements will be 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.

6.5.3 Mission 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.

6.5.3.1 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 optimally 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.

6.5.3.2 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, academic, 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.

6.5.3.3 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.

6.5.4 Payload Flight Operations Concept

A mature payload operations concept has not yet been developed within the Spacelab Program. However, certain characteristics of the payload sequencing and data management system are inherently evident, and can be developed in the form of guidelines for experimenter use of Spacelab capabilities.

o Role of the Payload Specialist

The Spacelab operational environment is unique in comparison with unmanned satellites in that an intelligent operator is available to provide local on-orbit control and monitoring of an Experiment. The Payload Specialist (PS) will be able to communicate with an Experiment either directly (via manual controls), or via the keyboard /DDU facility of the CDMS. Maximum use should be made of the intelligence, training and judgement of the Payload Specialist when deciding upon the characteristics and location of a data management task.

o Role of the CDMS

The CDMS experiment computer should be regarded as the primary facility for real-time data, processing in support of adaptive (man-interactive)experiment management. The computer combines the capabilities of medium processing power and fast reaction time with a dynamic command and display operator interface. Typical tasks for the experiment computer include experiment health and status checking, storage and execution of command sequences, and quick look data processing to verify correct scientific operations. The latter is particularly important where mainstream experiment data rates are so high that final processing must be performed on the ground via the HRM link.

o Role of the POCC

The POCC will generally provide full fidelity data processing and sequence generation capabilities in support of ongoing experiment operations. The PS should be considered as a real time extension of the POCC in so far as he will rely on POCC services to efficiently execute the experiment mission profile. Because of delays in the relay of data to the ground, the POCC will not normally have an "online" function. The POCC will therefore utilize its large computational power to provide non-real time or archival processing, as well as generation and validation of changes to the ongoing mission profile.

The POCC may also require major support from the MOCC for the determination of supporting data such as ephemeris, attitude or consumables management .

In order to clarify the interrelationship between the Payload Specialist, the CDMS, and the POCC, the overall system can be considered as a set of various control loops (see Fig. 6 - 25).
In this approach the loops can be characterized in a systematic manner by reaction time and data processing power. Although the processing power is not easily quotable, it is obvious that the data processing power increases with increasing reaction time (see Figure 6 - 26).

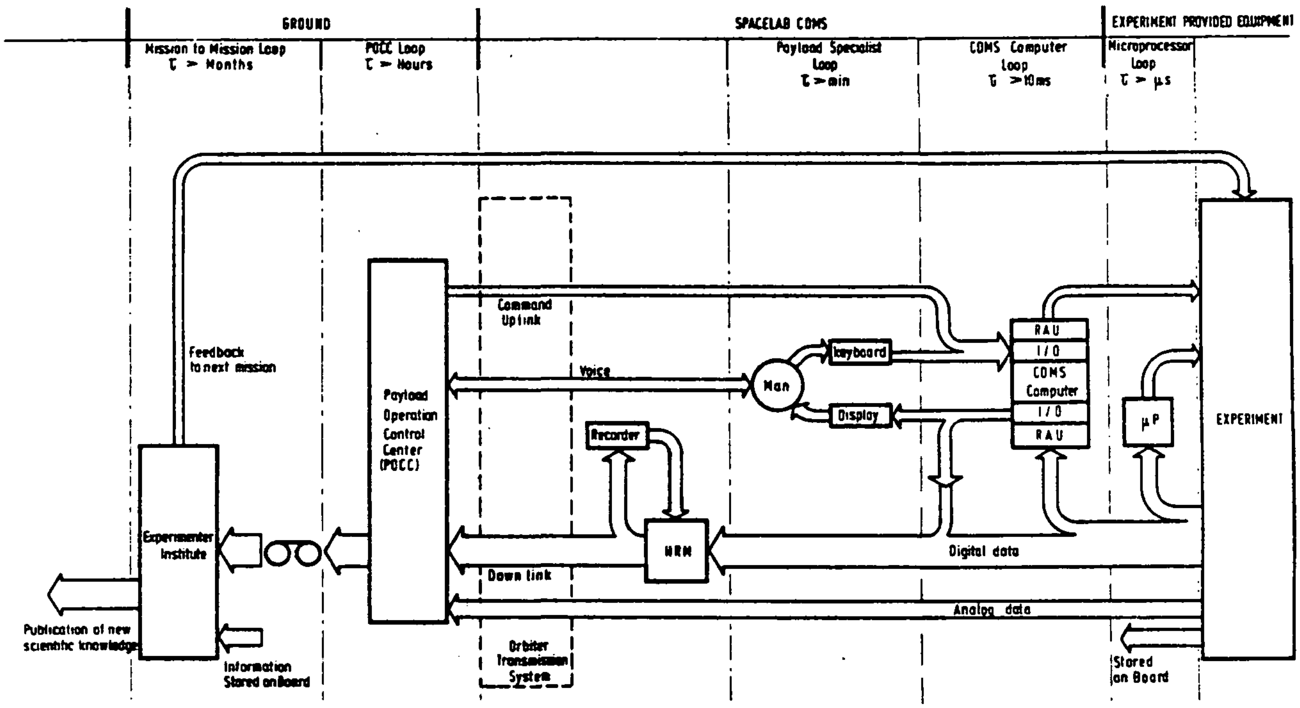

Figure 6 - 25 Payload Control Loops

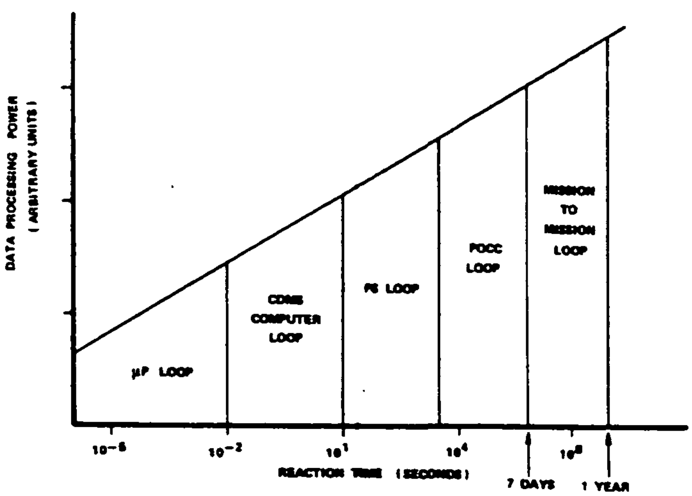

Figure 6 - 26:
Data Processing Power
Versus Reaction Time

In particular, the following loops can be identified:

- The fastest loop is the Microprocessor loop with a reaction time in the order of microseconds. Such a loop is part of the experimenter provided equipment, interfacing with Spacelab CDMS only for parameter interchange.
- The CDMS computer loop, if employing the synchronous GML, has a reaction time of some 10 milliseconds.
  This loop is entirely Spacelab provided. The RAU's provide the interface to experiments at all standard locations.
- The payload specialist loop is a loop with a reaction time in the order of minutes.
  The payload specialist is the man in the loop, normally interfacing with experiments via keyboard and display unit and CDMS and, if more appropriate, directly with experiment equipment.
- Because of the TDRSS non coverage periods the POCC loop, in the worst case, has a reaction time in the order of hours. The communication return from the POCC to the experiments is via the command uplink as well as the voice link to the payload specialist, i.e. beside the task of being the man in the loop, the payload specialist can be considered as an extension of the POCC.
- Not shown are non-time-critical Spacelab payload data processing facilities that are under discussion for installation at GSFC and in Europe as European Payload Operation Control Centers. These facilities are linked more closely to the experimenter site rather than to the POCC. The increased data processing power again has to be paid for by a longer reaction time, which is in the range between a few hours and a few days.
- The last loop is the mission to mission loop. Because of the high Spacelab mission frequency, this loop can still be considered as a closed loop. The finally evaluated scientific data form the basis for an improve ment of the experimenter's equipment and, after establishing the cricial events, for a more effective use. Depending on mission planning, the time between the reflights of particular experiments may be some months to years.

Most experiments, to a certain extent, will employ all these control loops together. The scientific objectives of a specific experiment will determine which loop has to carry the main control task rather than what loop will be used exclusively. For example, an experiment which bases its data processing on a built-in microprocessor will have an interface to the CDMS for housekeeping data; the payload specialist has to know whether it is still healthy and working properly, and the POCC certainly wants to know whether the scientific return is valid.

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Section 7 – DESIGN REQUIREMENTS FOR EXPERIMENTS

DCN 044

LIST OF TABLES

LIST OF FIGURES

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). Any deviations from the requirements of this Section will be considered only following a formal request from an experimenter to NASA/ESA.

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. Where specific requirements are not presented, the following documents may be used as a source of design guidelines

ESA Document EWP 1073

Preliminary mechanical design guidelines for Spacelab experiments

NASA Document EL 34 (77-34)

Dynamic environmental design and test criteria for MSFC Spacelab experiments DCN 007

ESA Document EWP 1133

Spacelab Payload Damage

Tolerance Design Guidelines

Experiment equipment shall be designed so that the package integrity and load carrying capability of structural mounting provisions fulfill the following requirements:

- The yield factor of safety shall be 1.4, against limit load conditions defined in Section 5.1
- Reflyable structural elements shall show
  - a fatigue life of four (4) times the design life and
  - containing assumed defects in sensitive areas, limit load capability and safe crack growth life of four (4) times an inspection interval (safe life design) or after failure of the primary member, a remaining fatigue life of the assembly of four (4) times an inspection inverval (fail-safe design)
- Experiment equipment used for a single mission or disassembled and inspected prior to each mission
  - has to be safe even when containing undetected defects in sensitive areas. The size of the undetected defect is defined by the applied inspection technique or
  - shall be designed by fail-safe principle, i.e. after failure of primary load path, the remaining structure shall be able to withstand the limit loads defined in Section 5.1.
- Experiment equipment shall be designed so that, when subjected to the emergency landing environment specified in Table 5 - 12 there shall be no hazard to personnel or prevention of egress from the Orbiter.

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7.2.4 Extension, Ejection, Deployment and Capture

7.2.4.1 Emergency Retraction and Ejection

Experiment equipment shall be designed to meet the emergency retraction and ejection requirements as stated in paragraph 8.1 .

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 General

The requirements of this section apply to the design of all experiment equipment that has a man-machine interface. Where specific requirements are not presented or referenced the following documents may be used as design guides:

MSFC - STD - 512 A
MIL - STD - 1472 B
MSF - 07 387

DCN 007

7.2.5.2 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 shall be made captive.

7.2.5.3 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.

7.2.5.4 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.

7.2.5.5 Corners, Edges and Protrusions

The following requirements apply to experiment equipment which is accessible to crew members:

Exposed Edges (see Figure 7-1)

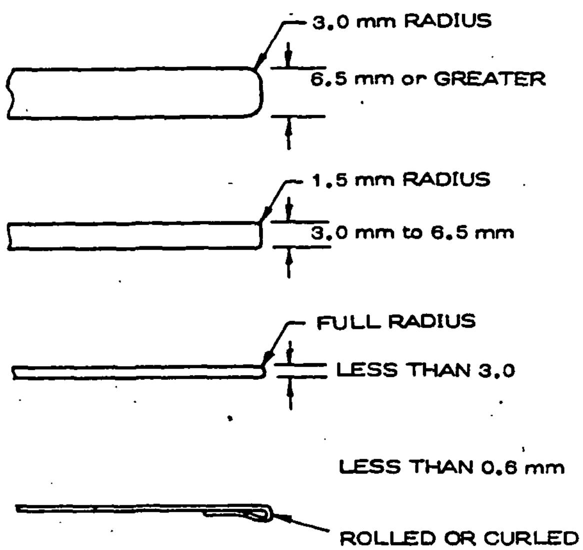


Figure 7-1: Exposed Edges Design Criteria

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 curled or bent.

Exposed Corners (see Figure 7-2)

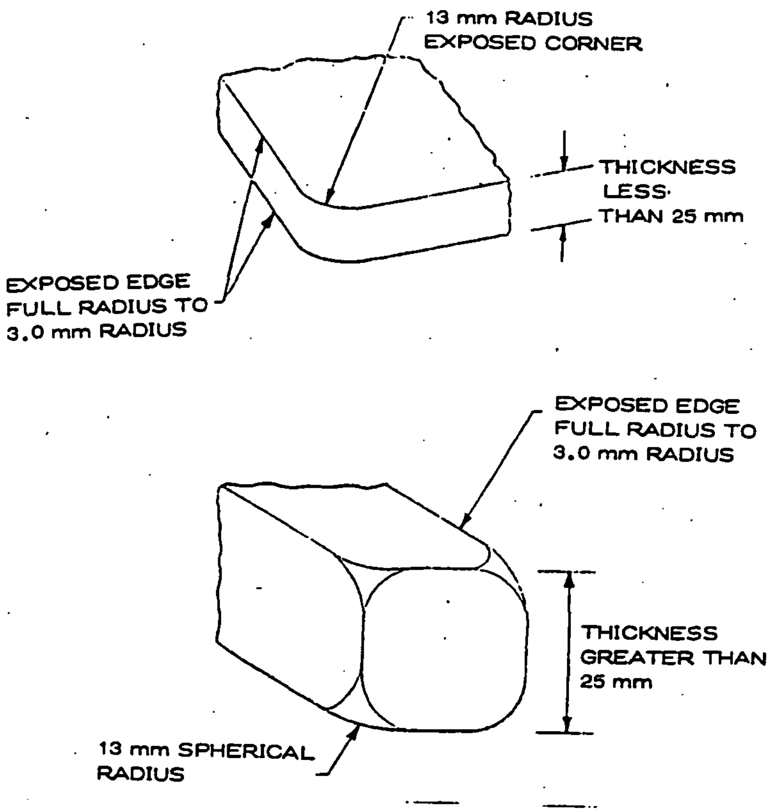


Figure 7-2: Exposed Corners Design Criteria

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

7.2.5.6 Area Closures

Experiment equipment shall be designed to prevent loose equipment such as small tools, screws, bolts, nuts, fuses, etc. from drifting into inaccessible areas. An inaccessible area shall be defined as any area with an opening that will accept a looseand floating object of 10mm in diameter which cannot be retrieved or captured by using a retrieval tool and/or a crewman reaching his hand and forearm into the area.

7.2.5.7 Crew Applied Loads

All experiment equipment that has a potential interface with the crew for operation, use or impact (whether inadvertent or not) shall be designed to withstand the crew applied loads of Table 5 - 8 without surface penetration or hazardous failure.

7.2.5.8 Controls and Displays

DCN 007 A minimum set of requirements relating to safety aspects of controls and displays are given here. More comprehensive criteria are contained in MSFC-STD-512 A and MIL-STD-1472 B.

Switches whose inadvertent activation may cause personnel injury or damage to Spacelab or whose location may pose a potential source of injury to personnel shall be provided with suitable guards or shall be recessed.

Emergency controls and displays which communicate requirements for immediate action to prevent hazards to personnel or Spacelab shall be conspicuously located.

Controls shall be located so as to have finger or glove clearance between controls and adjacent hardware.

DCN 007 Experiment front panels shall have a 12 x 27 mm (minimum) clear area reserved for attachment of locator code decal in upper panel area, but preferably in the upper left corner.

7.2.5.9 Electrical Safety

Protection from shock to the crewman shall be provided during use of experimental and medical equipment that attaches directly to the crewman.

The crew shall be protected from static electric shock due to static charge buildup in metallic and non-metallic materials of experiment equipment.

Experiment wire bundles and cables must be supported and physically separated from lines containing flammable liquids, gases and oxygen and associated equipment.

All experiment electrical connectors, plugs and receptacles shall be designed to prevent incorrect connection with other accessible connectors, plugs or receptacles where such connection would result in a hazardous condition.

Experiment wire harness installation shall utilize routing and attachment techniques which would preclude physically mismating connectors where such mismating would result in a hazardous condition.

Portable experiment electrical equipment shall have integral power switches.

Protective covers or caps shall be placed over electrical plugs and receptacles whenever they are not connected to the mating part. Restraint shall be provided for all protective covers.

7.2.5.10 Labels for Caution and Warning and Emergency Use Items

Caution and warning markings shall be used to indicate potentially undesirable conditions arising from the use of experiment equipment. The marking shall indicate the type of hazard and the action which would cause or prevent its occurrence, if it is not obvious. The marking shall be located in a position which permits sufficient opportunity for the crew to avoid the hazard.

Experiment items which are designated for emergency use, e.g., repair kits, emergency lighting, fire extinguishers, etc., shall display a unique marking, either on the item or adjacent to it. The marking shall consist of the nomenclature EMERGENCY USE surrounded by diagonal yellow and black stripes. If the item is located within a storage container, the diagonal striping shall also be applied to the door of the container. The title of the emergency items shall be incorporated on the container marking instead of the words EMERGENCY USE.

7.2.5.11 Crew/Equipment Interface for EVA: TBD

7.3 Thermal Interface 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.

DCN 007 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 - 118° C.

7.4 Electrical Power Interface Requirements

Experiment equipment shall be designed for compatibility with the Spacelab power and energy capabilities described in Sections 3. Experiment equipment of Spacelab shall interface with the EPDS only at standard interfaces as defined in Appendix A , Avionics Interface Definition .

At each EPDS interface, an experiment shall provide an ON/OFF power switch. These switches shall be capable of simultaneous make/brake of all hot lines and return lines between the outlet and the experiment. The switches shall be accessible for manual operation in orbit or shall be remotely controllable via CDMS.

The experiment shall provide a fuse for each hot line upstream of the associated experiment switch. The fuse characteristics shall be such that it blows before the associated circuit breaker uptstream in the EPDS trips.

The size of all wiring between experiment fuses and EPDS outlets shall be compatible with the rating of the relevant EPDS circuit breaker (see Table 7 - 3).

Table 7 - 3: Maximum Design Current Versus AWG Wire Size

WIRE DIAMETER (mm)	AWG*	MAX. DESIGN CURRENT (A) FOR A BUNDLE	
		SPACE ENVIRONMENT	PRESSURIZED ENVIRONMENT
8.25	0	75	150
6.54	2	50	110
5.19	4	35	80
4.16	6	25	60
3.26	8	20	45
2.59	10	14.5	33
2.05	12	11	23
1.63	14	8	17
1.29	16	6	13
1.02	18	4.8	10
0.812	20	3.8	7.5
0.644	22	2.7	5.0
0.511	24	2.0	2.5

* AWG: American Wire Gauge

7.5 Command and Data Handling Interface Requirements

Experiment equipment shall be designed for compatibility with the Spacelab CDMS capabilities and characteristics described in Sections 4.4 and shall interface with the CDMS only at the standard interfaces defined in Appendix A, Avionics Interface Definition.

7.6 GSE Interface 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

A minimum set of EMC requirements relating to safety and compatibility with Spacelab are given here. Where specific requirements are not presented, the following document may be used as a source of design guidelines :

DCN 002

ESA/SPICE Specification
- GEN-RE-003: Electromagnetic Compatibility
Requirements for Spacelab Payloads -

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 7.7.2.2.5.

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

All experiment equipment connected to one EPDB outlet (including via EPSP, Airlock power connector, etc.) shall maintain a DC isolation of at least 1 Meg ohm in parallel with a stray capacitance of less than 1 nF

- between the power hotlines as well as the power return lines and structure for DC and AC power
- between DC and AC power lines

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 as long as transformer isolation is included in the power line, 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, airlock connectors, or IPS connectors) when assuming as the load of the noise source a bus impedance of the characteristics shown in Figure 7 – 3, shall be within the following limits:

a) narrowband 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.

e) inrush transients shall be less than:

$\left.\begin{array}{l}\pm\ 28\ \text{Vp in amplitude}\\ 10\,\mu\ \text{sec in duration}\end{array}\right\}$ or equivalent Voltsecond product (= 280×10^{-6} Vs)

shape as defined in Figure 7 – 5

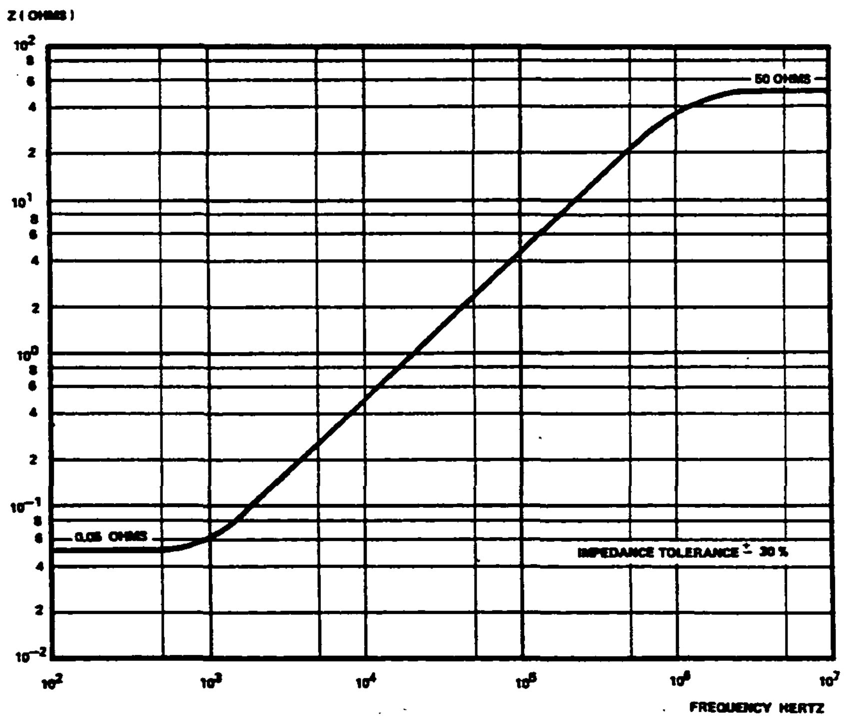


Figure 7 – 3: 28 V DC Bus Impedance Characteristics

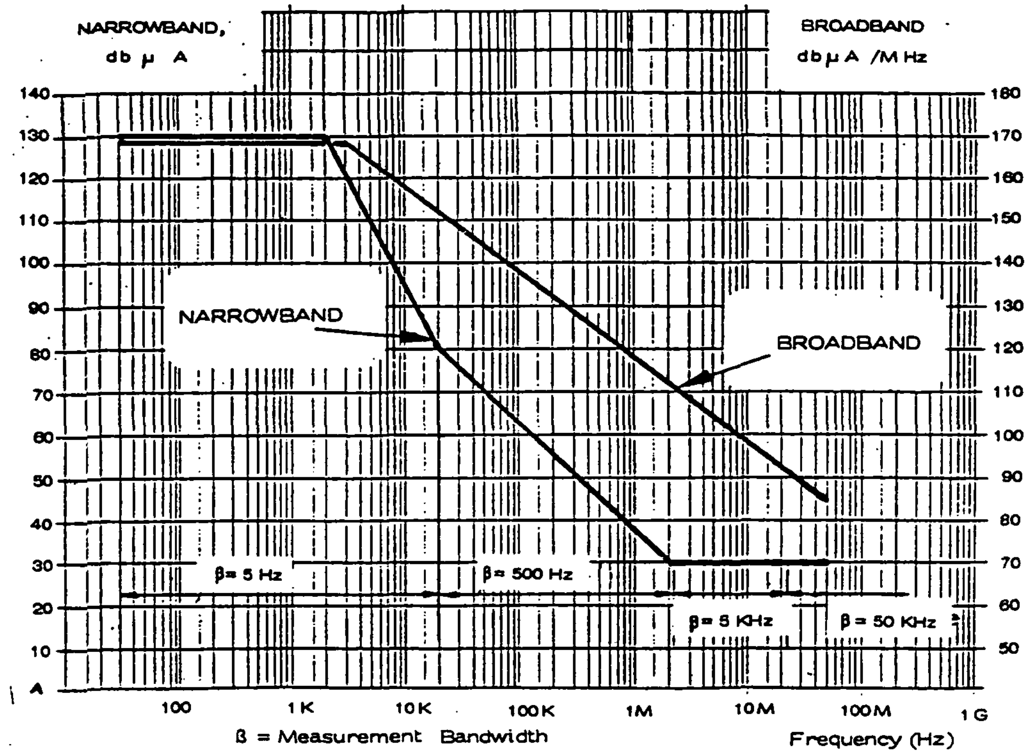
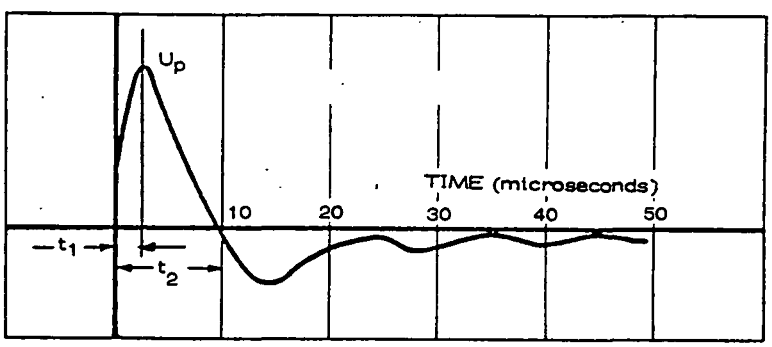


Figure 7 – 4 : Conducted Noise Limits for DC Power Lines



U_p = peak amplitude

t_1 = 2 microseconds

t_2 = 10 microseconds

Figure 7 – 5: Conducted Inrush Transient Waveform

(2) Experiment 400 Hz Power Bus

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, airlock connectors, or IPS connectors) when assuming as the load of the noise source a bus impedance of the characteristics shown in Figure 7 - 6, shall be within the following limits:

a) narrowband spectral conducted current: as shown in Figure 7-7.

b) broadband 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 shall be less than 50 Vpp

e) inrush transients shall be less than:

$\left.\begin{array}{l}\pm 60 \text{ Vp in amplitude} \\ 10\ \mu\text{ s in duration}\end{array}\right\}$ or equivalent Voltsecond product ($= 600 \times 10^{-6}$ Vs)

shape as defined in Figure 7-5

- This Figure is TBD -

Figure 7 - 6 : 400 Hz Bus Impedance Characteristics

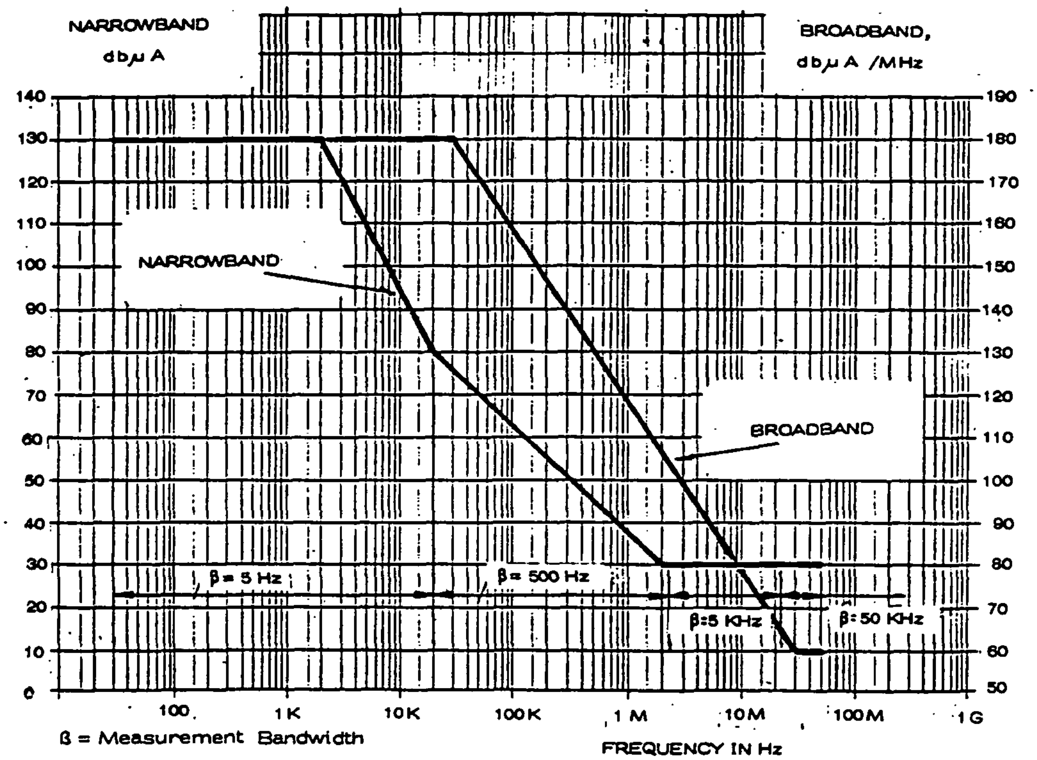


Figure 7 - 7: Conducted Noise Limits for AC Power Lines

(3) RAU Flexible Input Lines Used as Analog Input

Within the band 30 Hz - 50 MHz, the differential noise appearing (at the RAU connector) on the positive or return flexible input lines, when assuming as the load of the noise source an RAU input impedance of 500 k Ohms in parallel with 200 pF, 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 domain conducted voltage ripple and spikes shall be less than 40 m Vpp

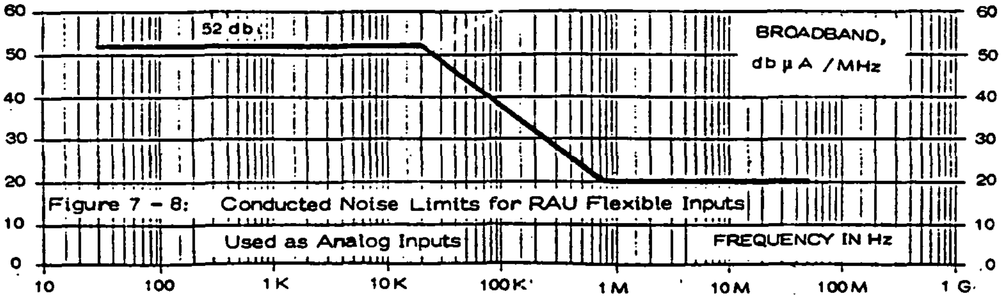


Figure 7 - 8: Conducted Noise Limits for RAU Flexible Inputs Used as Analog Inputs

(4) RAU Flexible Input Lines Used as Discrete Inputs

Within the band 30 Hz - 50 MHz, the differential noise appearing (at the RAU connector) on the positive or return flexible input lines, when assuming as the load of the noise source an RAU input impedance of 500 k Ohms in parallel with 200 pF, shall be within the following limits:

a) narrow band sprectral conducted current: not required

b) broad band spectral conducted current: as shown in Figure 7 - 9.
These levels shall apply for digital lines in both the logic "1" and "0" state.

c) time domain conducted current ripple and spikes: not required

d) time domain conducted voltage ripple and spikes:
Signal level plus the sum of DC plus AC ripple and spikes shall be for

LOGIC '1' within + 2.5 V to + 6.0 V
LOGIC '0' within + 2.0 V to - 0.5 V

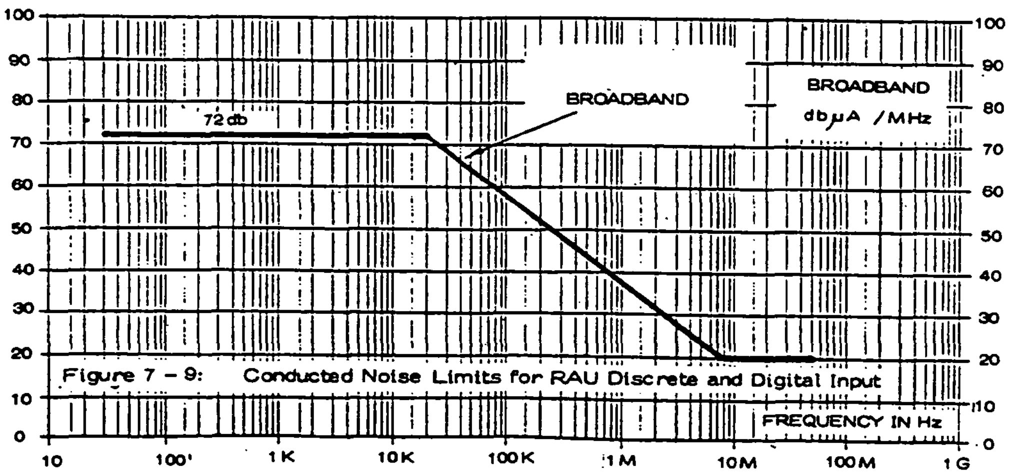


Figure 7 - 9: Conducted Noise Limits for RAU Discrete and Digital Input

(5) RAU Serial PCM Data, Clock, and Request Lines

Within the band 30 Hz - 50 MHz, the differential noise appearing (at the RAU connector) on the positive or return line, when assuming as the load of the noise source an RAU input impedance of 75 k Ohms, 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 - 9

c) time domain conducted current ripple and spikes: not required

d) time domain conducted voltage ripple and spikes

Differential Signal level plus the sum of DC plus AC ripple and spike shall be within 2.0 V to 5.0 V.

(6) RAU Output Lines UTC and UTC Update

TBD

(7) RAU Output Lines Serial PCM Commands and Clock

Signal level plus sum of DC plus AC ripple and spike shall be compatible with the TBD overvoltage range.

(8) Orbiter CCTV and 4.5 MHz Analog Channel Lines

TBD

(9) High Rate Multiplexer Experiment and Direct Access Channel Lines

TBD

7.7.2.2.3.2 Common – Mode Conducted Emissions

These are the emissions appearing between the positive and return line together, and the chassis.

(1) Experiment DC Power Bus

Within the band 30 Hz – 50 MHz, the common mode noise appearing on the experiment DC power bus (at the outlets of the experiment power distribution boxes, switching panels, top or aft airlock connectors, or IPS connectors) shall be within the following limits:

a) narrowband spectral conducted current: not required

b) broadband spectral conducted current: as shown in Figure 7-10

c) time domain conducted current ripple and spikes shall be less than 10 mApp

d) time domain conducted voltage ripple and spikes shall be less than 500 mVpp

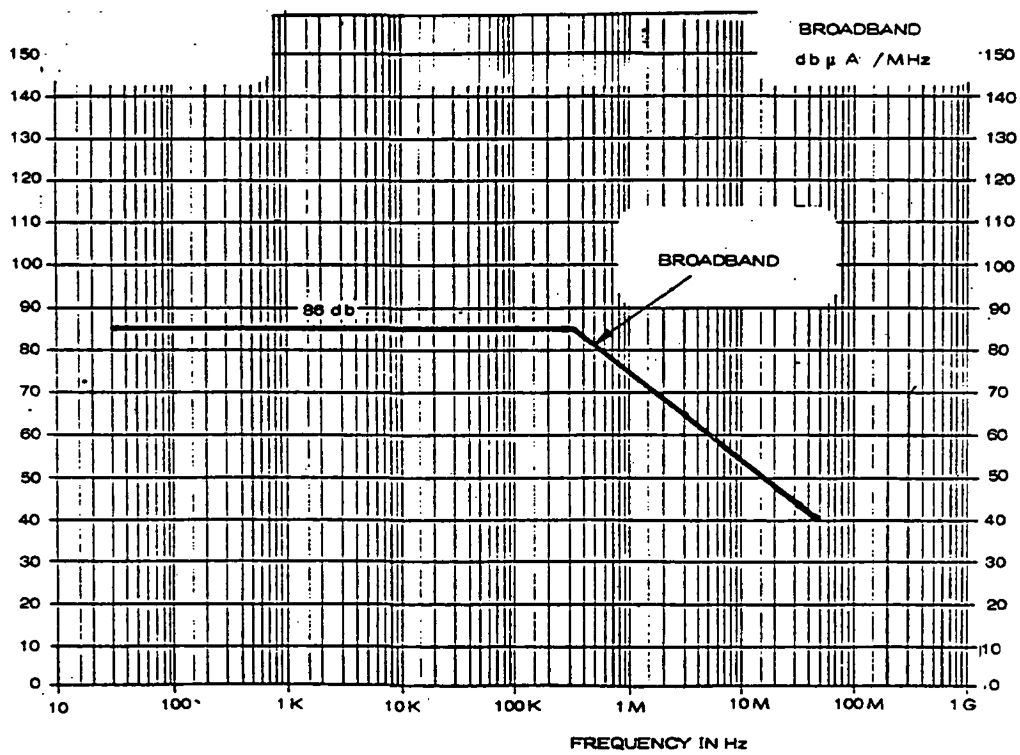


Figure 7-10: Common Mode Conducted Noise Limits for DC Power Lines

(2) Experiment 400 Hz Power Bus

Within the band 30 Hz – 50 MHz, the common mode noise appearing on the 400 Hz power lines (at the outlets on the experiment power distribution boxes, switching panels, top or aft airlock connectors, or IPS connectors) shall be within the following limits:

a) narrowband spectral conducted current: not required

b) broadband spectral conducted current: as shown in Figure 7-11

c) time domain conducted current ripple and spikes shall be less than 10 mApp

d) time domain conducted voltage ripple and spikes shall be less than 500 mVpp

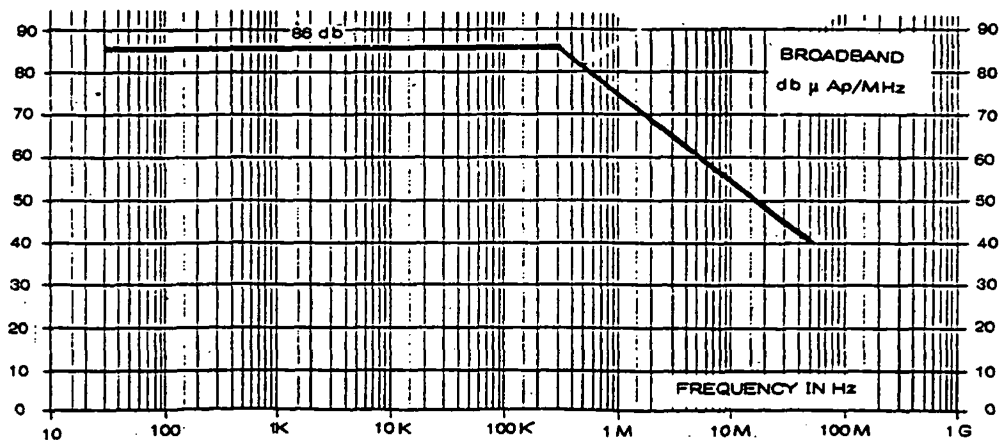


Figure 7 - 11: Common Mode Conducted Noise Limits for AC Power Lines

(3) RAU Input Lines (Flexible Inputs and Serial PCM Data Channel)

Due to the experiment ground requirements no direct common mode conducted noise requirements are stipulated for these lines.

(4) RAU Output Lines (ON/OFF Commands, Serial PCM Command Channels, and UTC/UTC Update)

Due to the experiment input circuit isolation requirements no direct common mode conducted noise requirements are stipulated for these lines.

(5) Orbiter CCTV and 4.5 MHz Analog Channel Lines

TBD

(6) High Rate Multiplexer Experiment and Direct Access Channel Lines:

TBD

7.7.2.2.4 Static ("DC") Magnetic Field Emissions

Spacelab and the Orbiter do not presently impose any requirements on payload static magtic fields.

7.7.2.2.5 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 - 12 measurement bandwidths, β as defined on the Figure.

b) broadband spectral: not required.

The levels shall apply for both vertical and horizontal polarization.

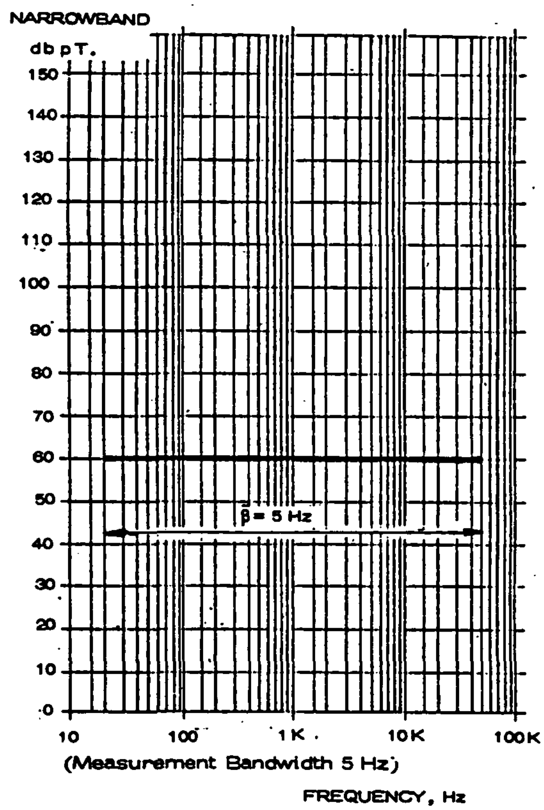


Figure 7-12: AC Magnetic Field Emission Limits

7.7.2.2.6 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 7-13 measurement bandwidths, β, as defined on the Figure

b) broadband spectral: as shown in Figure 7-13

The levels shall apply for both vertical and horizontal polarization.

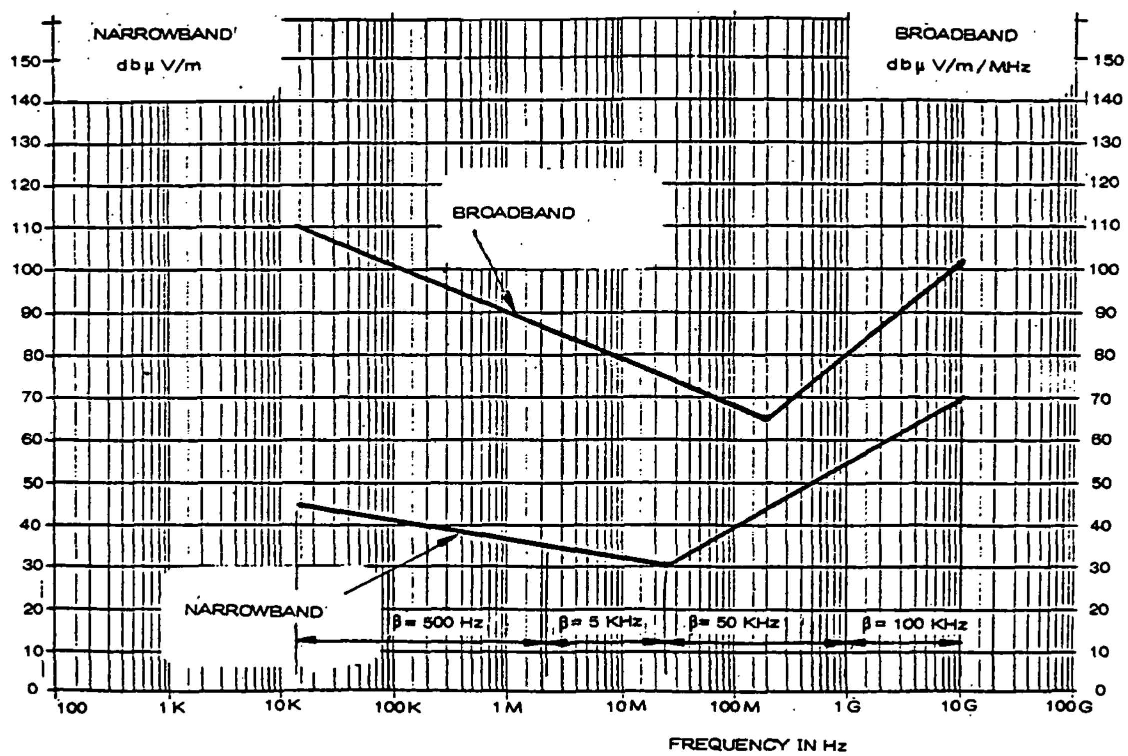


Figure 7 - 13: AC Electric Field Emission Limits

7.7.2.3 Ionizing Radiation

TBD

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) Experiments 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) The experiment should provide the capability to allow easy verification of equipment status and experiment activity to the operator.

(12) Experiments shall be designed to require no physical access on the launch pad unless it is absolutely necessary to achieve experiment objectives.

(13) Experiments shall be designed to require physical access not earlier than 45 hours after landing when the Orbiter is in the O&C Building. If required to achieve experiment objectives, earlier access may be possible to experiments depending on their location:

- o Access to equipment in the aft flight deck may be about 20 min. after landing.
- o Access to the module may be about 30 min. after landing (depending on tunnel design and safety regulations)
- o Access to pallet mounted equipment may be about 15 hours after landing (after cargo bay door opening in the OPF).

7.10 Material Control Requirements

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.

(4) corrosion or material incompatibility 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 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 intolerable 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 outlined here shall not be applied for hardware to be mounted in the Orbiter Aft Flight Deck, see para. 7.10.2.1).

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. 408 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 PSS-09-QRM-02 T or NASA-specification JSC-SP-R-0022 DCN 007

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. Although not directly applicable to black box outgassing testing, useful 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
- Polyvinyl chloride (PVC, e.g. wire insulation, wrapping)
- Carcinogenic or toxic materials

The approval of radioactive sources is covered in Section 8, Safety, para 8.3.24 .

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.

- Shatterable or flaking materials
- Beryllium and beryllium alloys
- Cadmium
- Zinc

The use of magnetic materials shall be minimized as far as possible. If their use cannot be avoided then the type, quantity and location 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 applications which are normal in electronic circuits).

7.10.10 Stress Corrosion Cracking Materials

Materials which are susceptible to stress corrosion cracking shall not be used in payload structure or support bracketry.
MSFC-SPEC - 522 "Design Criteria for Controlling Stress Corrosion Cracking", shall be used as a reference for such structural design.

7.10.11 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).

DCN 007

- 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."

7.11 Contamination and Cleanliness Requirements

7.11.1 General Contamination Control Requirements

Experiment equipment shall be designed to minimize or contain the generation of loose particulate matter and liquid or gaseous contamination which may be detrimental to Spacelab operation or crew safety.

Sensitive experiment equipment which needs an operational environment which is cleaner than the environment specified in Section 5 shall provide the necessary protective covers, purging equipment etc.

7.11.2 Surface Cleanliness

Experiment equipment exterior surfaces shall be free from visible contamination such as scale, particles, rust, dirt, dust, grease, oil, water and other foreign materials when examined under white light of 540 - 1600 lumens/m^2 and from a distance of 0.3 to 0.6 m.

Experiment equipment exterior and accessible interior surfaces shall be designed for easy cleanability.

7.11.3 Contamination Inside the Module and/or Orbiter

7.11.3.1 Gaseous Contamination

Experiment equipment shall be designed so that its contribution to the gaseous contamination of the air in habitable compartments, when combined with the contributions from Spacelab/Orbiter equipment and other experiment equipment, shall not cause the maximum allowable concentration of total organics to exceed TBD or the individual maximum allowable concentrations of each of the specified trace contaminants in Table TBD to exceed the limits specified in that table and shall not be the source of obnoxious odours. However, experiments that offgass the following contaminants shall not be allowed in the habitable environment :

- o Methyl chloroform
- o Chloroform
- o Carbon tetrachloride
- o Trichloroethylene

7.11.3.2 Particulate Contamination

Experiment equipment shall be designed so that its contribution to the particulate contamination of the air in habitable compartments, when combined with the contributions from Spacelab/Orbiter equipment and other experiment equipment, shall not cause the resultant total particulate contamination to exceed TBD.

7.11.3.3 Microbiological Contamination

Experiment equipment which contains micro-organisms which may come into contact with air supplied from the habitable atmosphere shall be provided with a filtration system which removes or destroys the micro-organisms before returning the air to the habitable atmosphere.

7.11.4 Contamination External to the Module and/or Orbiter - TBD

7.12 Caution and Warning Design Safety Requirements

The following design safety requirements shall be specifically applied for payload installed and provided C/W sensors and associated circuitry:

The caution and warning system shall monitor all safety critical parameters and any payload equipment functions that have the potential of endangering the crew.

All payload safety critical functions shall be capable of being monitored by a C&W system.

Payload C/W elements and their function shall be considered safety critical.

All safety critical payload equipment will be fully qualified by test.

All C&W system sensors shall fail in such a manner that a signal input will be initiated to the C&W system, resulting in a C&W alarm.

Warning signals and their application shall be designed to minimize the probability of wrong signals or of improper personnel reaction to the signal.

Special identification and adequate protection shall be provided for safety critical switches.

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An inhibit capability shall be provided in each C&W sensor circuit to allow isolation of a single malfunctioning sensor and permit normal operation of all other remaining sensing units.

Redundant safety critical electrical circuits shall not be routed through the same connector and shall be separated from each other.

Safety critical and non safety critical circuitry shall be isolated from each other to prevent failure propagation. All sensors used for inputs to systems other than C&W which are also used by the C&W system shall be isolated such that a failure in the other system will not affect the caution and warning system.

Back up power sources for safety critical functions shall be provided.
All sensors for all parameters monitored by the C&W system shall be independently powered to prevent loss of hazard indication due to power failure of a monitored system.

Provision shall be made to perform end-to-end check of critical caution and warning system functional paths up to sensor interface; whenever possible, provision shall be made to check the sensor function, range and/or sensitivity.

Safety Critical control circuits shall be easily accessible for verification without creating a hazardous condition.

Verification that the safety critical redundant hardware is functional shall be provided.

Safety critical electrical components shall be protected against the effects of liquid leakage, moisture, condensation, vibration, arcing contacts and corona.

Equipment which is required to control or perform safety critical functions when the module is depressurized to vacuum or partial vacuum shall be designed for operation under vacuum conditions or partial vacuum conditions respectively.

Equipment which is required to control or perform safety critical functions after module repressurization shall be designed to survive depressurization, vacuum and repressurization without degradation in performance after repressurization.

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It shall be noted that the listed requirements are those which are unique to caution and warning design. Normal design safety requirements as listed in Section 8 are also applicable.

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Section 8 - SAFETY

DCN 008

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8. SAFETY

This section reflects the principles and intentions of NASA Headquarters Office of Space Flight Document "Safety Policy and Requirements for Payloads using the Space Transportation System" which are applicable to experiments to be flown on Spacelab missions. Additional safety information is contained in JSC 11123 "Space Transportation System Payload Safety Guidelines Handbook" which provides safety recommendations for experiment design and operation. Any deviations from the safety requirements will be considered only following a request from an experimenter to NASA / ESA .

8.1 Safety Requirements

Safety, as applied to experiments flown on Spacelab missions, is intended to protect personnel, the Spacelab, Spacelab Payloads and the Orbiter from experiment related hazards. Implicit within this, is the requirement to assure the retention of the capability for the safe recovery of the Orbiter/Spacelab and the crew.

- Experiments extended outside the Orbiter cargo bay envelope, except when mounted on the IPS or on the airlock experiment table, or outside the Spacelab airlock envelope defined in para 4.7.5 must have a capability for emergency ejection and/or retraction. This capability shall be provided by a dedicated system capable of control from the Orbiter and optionally from the module . DCN 008
- Experiments must have the capability of being returned to a safe or inert status at the termination of the experiment operations, including emergency shut-down provisions in the case of hazardous conditions.
- Experiments and experiment materials which are used or stored in the Spacelab Module shall be subject to the Spacelab materials requirements (Reference Section 7.10).

8.2 Hazard Reduction

In order to ensure that experiment originated hazards are identified and eliminated or controlled, each experiment design and its operational concept shall be analysed in a systematic manner. The analysis shall identify the hazard source, its possible effects, the actions taken to eliminate or control the hazard, and whether the hazard is considered to be eliminated, controlled, or uncontrolled. DCN 008

In order to eliminate or control hazards, the following sequence or combination of requirements shall be implemented:

8.2.1 Design for Minimum Hazard

The major goal throughout experiment design shall be to ensure inherent safety through the selection of appropriate design features and through the elimination of hazard sources within the experiment. This shall also include damage control and containment, and the isolation of potential hazards. DCN 008

8.2.2 Safety Devices

Hazards which cannot be eliminated through design selection shall be reduced and made controllable through the use of safety devices as part of the experiment.

8.2.3 Warning Devices

Where it is not possible to preclude the existence, or occurrence of a known hazard, devices shall be included in the design for the timely detection of the condition and the generation of an adequate warning signal coupled with automatic or manual contingency procedures designed to ensure that the hazard cannot become uncontrolled.

Warning signals and corrective actions may be displayed and processed either by experiment data processing command equipment, or by the Spacelab Caution and Warning subsystem. Use of the Spacelab C & W System will depend on the corrective action or contingency procedures to be implemented.

8.2.4 Special Procedures

Where it is not possible to reduce the magnitude of an existing or potential hazard by design, or the use of safety and warning devices, special procedures shall be developed to counter hazardous conditions for the enhancement of ground and flight crew safety.

8.2.5 Residual Hazards

Hazards which remain after the application of the hazard reduction sequence and control action (para. 8.2.1 thru 8.2.4) are residual risks. These shall be identified and the rationale for acceptance provided.

8.3 Provisions Against Hazards

The following design requirements are directed towards reducing hazards inherent or incidental to experiments at all times during integration, checkout, launch preparation and launch, orbital, landing and post landing operations as appropriate:

8.3.1 Hazard Indication

Instrumentation shall be adequate to provide timely indication of hazardous out-of-tolerance conditions and provision made to correct such conditions prior to the condition becoming a hazard to the crew, the Orbiter or Spacelab.

8.3.2 Module Decompression

All experiment equipment within the habitable areas of Spacelab shall be structurally capable of withstanding decompression of the Spacelab without hazardous failure (e.g. Release of toxic, corrosive or flammable materials, or explosion caused by module pressure drop).

8.3.3 Rapid Evacuation

Experiments requiring the presence of personnel in the Orbiter Cargo Bay while on the ground shall not preclude rapid evacuation from the Cargo Bay in the event of an emergency.

8.3.4 Hazardous Material Control

Hazardous materials are subject to the material control as defined in para 7.10.

8.3.5 Hazardous Material within the Module

When the use of toxic, corrosive and /or flammable materials has been accepted (Ref. paragraph 7.10.11) these materials shall be stored and used such that failure of the primary container shall not release the material, or any chemical reaction products, into the module atmosphere. DCN 001

Provision shall be made for the safe collection and storage of used or spent materials considering also their possible chemical or physical interaction.

8.3.6 Fluid Release

Hazardous fluids shall not be released into the Cargo 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 Cargo Bay may be permitted under some conditons.

8.3.7 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 VIII, 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.

8.3.8 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.

8.3.9 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.

*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

8.3.10 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.

8.3.11 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.

8.3.12 Rotating Machinery

Rotating machinery must be protected by suitable guards. Where machinery is highly stressed, containment for possible failure must be provided.

8.3.13 Corners and Protrusions

Exposed sharp corners, edges and protrusions shall be avoided (see 7.2.5.5).

8.3.14 Material Shattering

Restrictions for frangible materials are control requirements within paragraph 7.10.

8.3.15 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.

8.3.16 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.

8.3.17 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.

8.3.18 Experiment Grounding

Experiment grounding shall be such as to preclude electrical discharge hazards and shocks.

8.3.19 Accidental Swich Actuation

DCN 008 All switches, which-if accidentally actuated - could cause personnel injury or damage to hardware, shall be recessed or otherwise protected.

8.3.20 Emergency Switch Off

A rapid means of switching off power under emergency conditions shall be provided.

8.3.21 Lightning Strikes

DCN 008 Experiment equipment shall not cause personnel injury or damage to Spacelab, Spacelab Payload equipment or the Orbiter in case of lightning strikes.

8.3.22 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".

8.3.23 Radiation Sources

Experiments that contain radioactive materials or contain equipment that generates 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 Interagency Agreement 1052.72 A

8.3.24 Biological Specimens

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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.

8.3.25 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.

8.3.26 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.

8.3.27 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.

8.3.28 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.

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9 MISCELLANEOUS

9.1 Abbreviations and Acronyms

AC	Alternating Current
ACCU	Audio Central Control Unit
ADC	Analog Digital Converter
AFB	Air Force Base
AFD	Orbiter Aft Flight Deck
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
CEI	Contract End Item
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

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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	Greenwich Mean Time
GN&C	Guidance, Navigation and Control
GOAL	Ground Operations Aerospace Language
GPC	Orbiter General Purpose Computer
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
HAL	Huston Aeronautic Language
HDRR	High Data Rate Recorder
HEPA	High Efficiency Particle Air-Filter
HOL	High Order Language
HPI	High Performance Insulation
HRM	High Rate Multiplexer
H/W	Hardware
HX	Heat Exchanger

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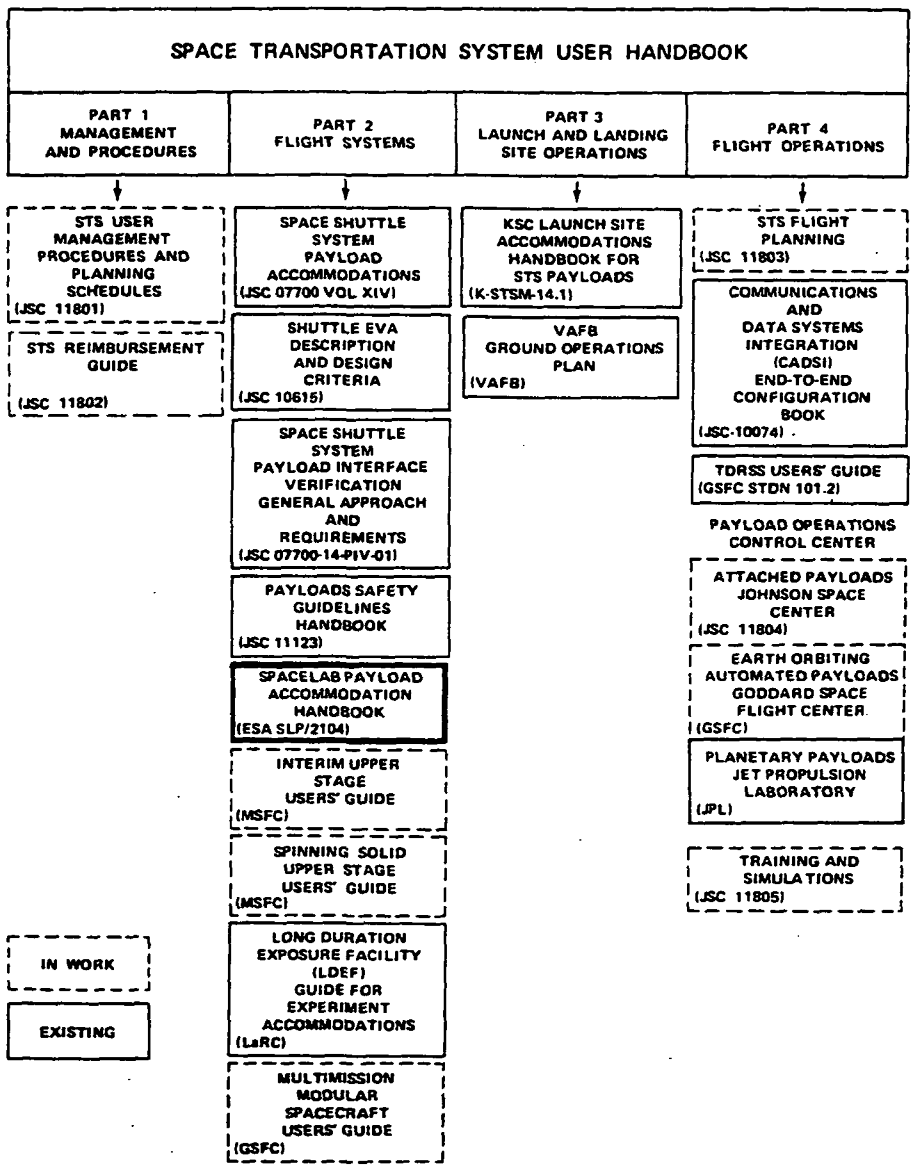
ICS	Interpretive Computer Simulator
ICD	Interface Control Document
ICMS	Intercom Master Station
ICRS	Intercom Remote Station
IECS	Igloo Environmental Control Subsystem
IMU	Inertial Measurement Unit
INSTR.WD	Instruction Word
INV	Inverter
I/O	Input/Output
IOBPS	Input Output Box and Peripheral Simulator
IOU	Input Output Unit
IPS	Instrument Pointing Subsystem
IS	Interconnecting Station
IVA	Intra-Vehicular Activity
JSC	Johnson Space Center
JSLWG	Joint Spacelab Working Group
LSU	Remote Loudspeaker Unit
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 DCN 001
MMU	Mass Memory Unit
MSE	Measuring & Stimuli Equipment
MSFC	Marshall Space Flight Center
MSU	Measurement and Stimuli Unit
MTU	Master Time Unit
MUSS	Module Utility Support Structure
NASA	National Aeronautics and Space Administration
N/C	Numerical Control
NCD	Number Column Density
NFE	NASA Furnished Equipment
NRZ-L	No Return to Zero-Level Defined
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
DCN 001	PLSS	Portable Life Support Subsystem
	POCC	Payload Operations Control Center
DCN 001	POP	Perpendicular to Orbit Plane
	PPR	Payload Preparation Room
	PSA	Pressure Switch Assembly
	PSS	Payload Specialist Station
	PTCS	Passive Thermal Control Subsystem
	PTT	Press-to-Talk
	P&M	Process and Materials
	PUSS	Pallet Utility Support Structure
	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
	STS	Space Transportation System
	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
DCN 001	VAB	Vehicle Assembly Building
	VAFB	Vandenberg Air Force Base
	VCM	Volatile Condensible Materials
	VDC	Volts Direct Current
	WTR	Western Test Range

9.2 Relationship of Spacelab Payload Accommodation Handbook to the Overall Space Transportation System Documentation

The Spacelab Payload Accommodation Handbook is one component of the user related documentation of the Space Transportation System as listed below.



9.3 Reference Documents

The following list contains the major documents which are referenced in SPAH

SP 1001	Spacelab User's Guide
JSC 07700, Volume XIV, Revision D	Space Shuttle System Payload Accommodations
SD 72-SH-0120-47	Space Shuttle Orbiter Mass Properties Status Report
ICD-2-05101	Shuttle Vehicle / Spacelab Structural /Mechanical Interfaces
ICD-2-05201	Shuttle Vehicle / Spacelab ECS / Thermal Interfaces
ICD-2-05301	Shuttle Vehicle / Spacelab Avionics Interfaces
MIL-STD-189	Racks Electrical Equipment, 19 inch, and Associated Panels
MIL-STD-1472 B	Human Engineering Design Criteria for Military Systems, Equipments and Facilities
MIL-STD-1512	Electro Explosive Subsystems, Electrically Initiated Design Requirements and Test Methods
JSC 8060	Pyrotechnic Specification, Space Shuttle System
JSC 11123	Space Transportation System Payload Safety Guidelines Handbook
ARINC 404 A	Air Transport Equipment Cases and Racking
RS 170	Electrical Performance Standard - Monochrome Television Studio Facilities, Electronic Industries Association
RS 330	Electrical Performance Standards - Closed Circuit Television Camera 525/60 Interlaced 2 : 1, Electronic Industries Association
ISO R 1996	Accoustic Assessment of Noise with respect to Community Response
FED-STD-209 B	Clean Room and Work Station Requirements, Controlled Environment
MSFC 40 A 99005	Spacelab GSE Allocation and Requirements Plan
NASA TMX 64627	Space and Planetary Environmental Criteria Guidelines for use in Space Vehicle Development
NASA TMX 64936	Charged Particle Radiation Environment for the Spacelab and other Missions in Low Earth Orbit
MSFC-STD-512	Man/System Requirements for Weightless Environments
MSC 07387	Crew Stations Specification
NASA - TBD	Dynamic Environmental Design and Test Criteria for MSFC Spacelab Experiments
ESA-EWP 1073	Preliminary Mechanical Design Guidelines for Spacelab Experiments

ESA/SPICE GEN-RE-003	Electromagnetic Compatibility Requirements for Spacelab Payloads	DCN 002
JSC-SP-R-0022	Vacuum Stability Requirements of Polymeric Materials for Spacecraft Application	
ESA PSS-09-QRM-02 T	A Screening Test Method employing Thermal Vacuum for the selection of Materials to be used in Space	
NHB 8060 IA	Flammability, Odor and Offgassing Requirements and Test Procedures for Materials in Environments that support Combustion	
		DCN 001
NASA	Safety Policy and Requirements for Payload using the Space Transportation System	
MSFC-SPEC-522	Design Criteria for Controlling Stress Corrosion Cracking	DCN 001
NMI 1052.72 A	NASA/ERDA Interagency Agreement - Isotope Snap Device for NASA Space Vehicles	
NSS HP 1740.1	NASA Aerospace Pressure Vessel Safety Standard	
ASME	Boiler and Pressure Vessel Code	
MCR-77-105	Spacelab Contamination Assessment	DCN 002
ICD-2-19001	Space Shuttle Interface Control Document "Shuttle Orbiter/Cargo Standard Interfaces	
ESA-EWP 1133	Spacelab Payload Damage Tolerance Design Guidelines	
JSC-08934 Volume II RevA	Shuttle Operational Data Book	
ED-2002-1771	Skylab S 190 A Window Utilization Summary Report	DCN 030
MIL-G-25871A	Glass, Laminated, Aircraft, Glazing	

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