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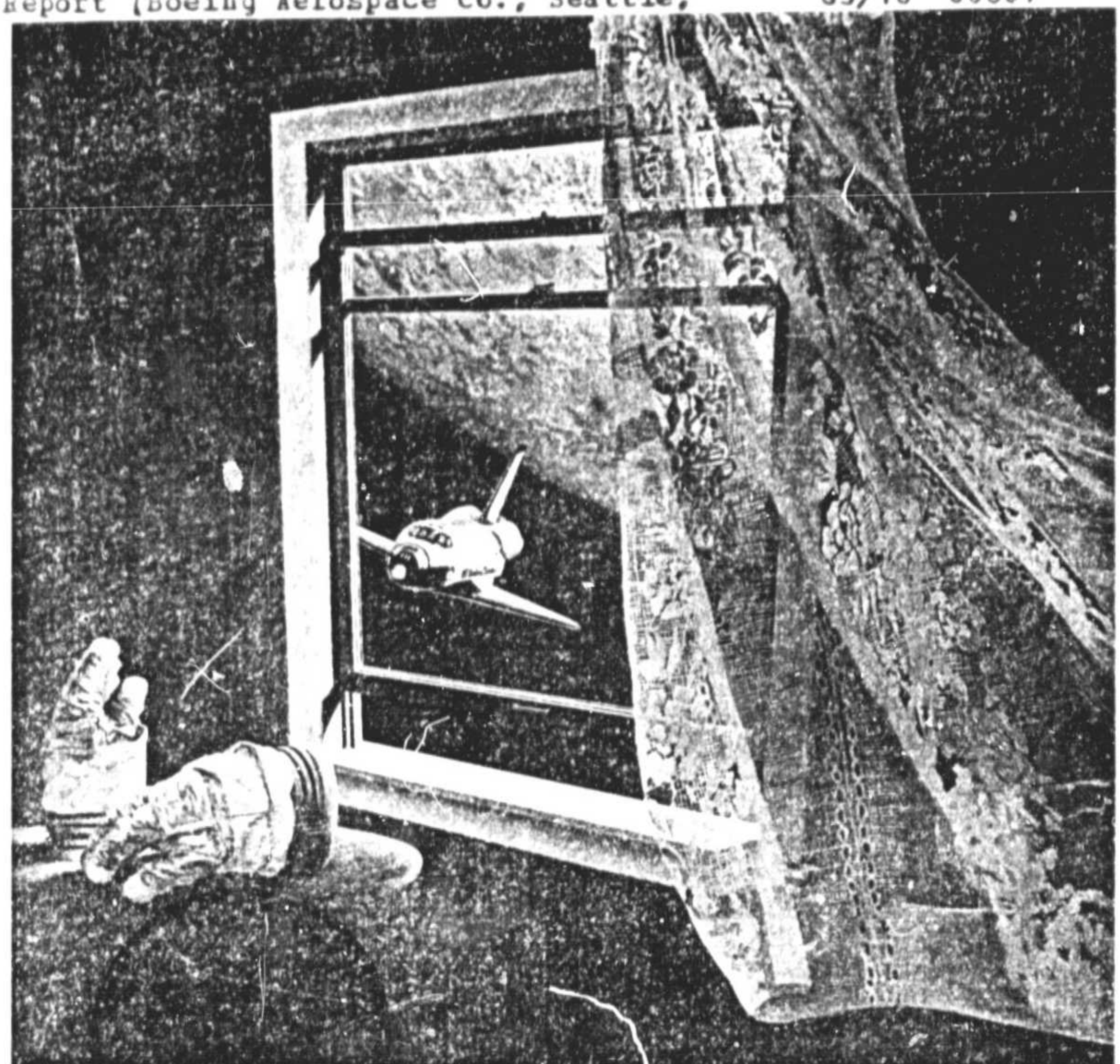
Volume 7-4B
Final Report

not in 7

Data Book
D180-27477-7

Space Station Needs, Attributes, and Architectural Options Study

(NASA-CR-173685) SPACE STATION NEEDS, ATTRIBUTES AND ARCHITECTURAL OPTIONS STUDY. VOLUME 7-4B: DATA BOOK, ARCHITECTURE, TECHNOLOGY AND PROGRAMMATICS, PART B Final Report (Boeing Aerospace Co., Seattle, N84-27800 Unclas G3/18 00807



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Space Station
Needs, Attributes and Architectural Options Study

Contract NASW-3680

D180-27477-7

Final Report

Volume 7 - 4B

Data Book

Architecture, Technology, and Programmatics- Part B

April 21, 1983

for

National Aeronautics and Space Administration

Headquarters

Washington, D. C.

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D180-27477-7

VOLUME 7-4B
DATA BOOK

Architecture, Technology, and Programmatics

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7.4.7 Remote Manipulators - SPAR Aerospace Limited

SPAR-R.1145

ISSUE A

ASSESSMENT OF REMOTE MANIPULATOR ISSUES FOR BOEING SPACE STATION STUDY

APPROVALS

S. Sachdev Manager, System Controls & Analysis Feb 22, 83
B.R. Fuller Space Marketing Feb 25, 83
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REASON FOR RE-ISSUE

DATE January, 1983

AMENDMENTS

NUMBER	REASON FOR AMENDMENT	SIGNATURE	DATE

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PROPRIETARY INFORMATION

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NOTE

Proprietary data has been deleted from this copy of SPAR-R.1145 included in the Volume 7-4.

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1.0 INTRODUCTION

This report presents the tasks performed at Boeing Aerospace Company (BAC), Seattle, Washington, in assessing remote handling aspects of a Space Station. The tasks were performed in response to the Statement of Work attached as Appendix A, over the two week period from January 10 to 21, 1983.

The tasks involved discussions of remote handling operations and future potential developments of space manipulators. In particular, the following subject areas were addressed and are discussed in this report:

(a) SRMS capabilities and characteristics,
(b) potential improvements in SRMS,
(c) space maintainability considerations for RMS,
(d) manipulator development issues: simulation and analysis,
(e) RMS track and base assembly concept for a space station.

2.0 SHUTTLE REMOTE MANIPULATOR SYSTEM CAPABILITIES AND CHARACTERISTICS

The Shuttle Remote Manipulator System (RMS) is designed primarily for the deployment and retrieval of payloads from the orbiter.

The manipulator arm of the RMS is a 50 ft. long tubular structure (Figure 2-1), consisting of six rotational degrees of freedom. The joints correspond to the Degrees of Freedom (DoF's) of a human arm. The arm is attached at one end to the Manipulator Positioning Mechanism (MPM) of the orbiter, and the other end carries an end effector which is designed to capture and release payloads. The joints near the MPM, called "shoulder joints", provide two DoF's and the joints near the end effector called "wrist joints", provide three DoF's. An elbow joint in the middle of the arm provides one DoF. The shoulder and elbow provide three dimensional translations of the end effector and the wrist joints provide three dimensional rotations of the end effector.

2.1 Design Capabilities

The Shuttle RMS has been designed to provide the following capabilities:

(a) Deploy, or return without release, payloads weighing up to 65,000 lbs. with 60 ft. length and 15 ft. diameter.

(b) Capture and retrieve up to 32,000 lb. free-flying max-dimension payload whose relative rate with respect to the shoulder is 0.1 ft/sec. and 0.1 deg/sec. about each axis.

(c) Position the end effector relative to the shoulder attach point to within ±2 inches and ±1 degree, in the Automatic mode.

(d) Release payloads up to 65,000 lbs. and maximum envelope, within 5 degrees attitude error with tip-off rates less than 0.015 deg/sec.

(e) Maintain end effector rate accuracies of ±0.03 ft/sec. and ±0.09 deg/sec. in the manual augmented mode.

(f) Stop the end effector within 2 ft. under all loading conditions.

(g) Provide the maximum rates at the end effector as follows:

Load	Rate Limits ft/sec.	deg/sec.
Unloaded	2	4.76
Loaded (32,000 lb.)	0.2	0.476
Loaded (65,000 lb.)	0.1	0.238

(h) Provide the following minimum force/moment capability at the end effector:

i) combined 12 lbs. shear force and 160 lb-ft. bending moment,
ii) torque about the end effector longitudinal axis (roll) of 230 lb-ft.

(i) Provide two arm (port and starboard) serial operation capability.

(j) Provide fail-safe operation.

2.2 Design Characteristics

The components of the shuttle RMS are shown schematically in Figure 2-2. System block diagrams are shown in Figure 2-3 and Figure 2-4. Mass, power and size data for RMS subassemblies are given in Table 2-1.

The mechanical arm of the RMS consists of the following major elements:

(a) mechanical arm assembly,
(b) end effector,
(c) thermal protection system,
(d) CCTV system.

The mechanical arm assembly consists of the six electromechanical joints and two arm booms which provide structural connection between the shoulder, elbow and wrist joints. The upper arm boom connects the shoulder pitch and elbow pitch joints, and the lower arm boom connects the elbow pitch joint to wrist pitch joint. The booms are of thin-wall graphite-epoxy construction and the joint structures are metallic. A thin kevlar-skin honeycomb bumper system provides protection to the booms against impact damage. The key performance characteristics of the joints are given in Table 2-2.

The end effector is the element which physically interfaces with a payload. The standard end effector has the capability to attach to a payload and to release it on command from an operator. It is a snare type device and has a standardized interface with a payload. The payload side of the interface is formed by a grapple fixture which is attached to the payload. The snare design feature enables the standard end effector to capture the grapple shaft of the grapple fixture (and, thus, the payload) when it is offset by up to 4 inches from the end effector centerline and is misaligned by up to 15° in pitch/yaw and up to 10° in roll. It also enables the payload to be released with minimal release-impulse. Operations of the end effector and grapple fixture are shown in Figure 2-5.

The standard end effector is equipped with a 51 pin electrical connector (only 25 pins are available for use at present), which can be used to transfer power and signals to the payload.

However, the payload needs to be provided with the Electrical Flight Grapple Fixture (EFGF) to permit this electrical interface. This electrical grapple fixture features a linkage mechanism to engage connectors using motive force provided by the end effector. Electrical connections can be made through 25 pins. The other end of these wires terminate in the aft flight deck and are usable at a payload panel.

The thermal protection system of the mechanical arm consists of thermostatically controlled heaters in the joints and electronics compartments and multilayer thermal insulation blankets covered with Beta cloth on the exterior of the joints and arm booms. Specific areas of the electronics compartment are free of insulation and are painted white to provide radiating areas for heat dissipating components.

The CCTV system includes a CCTV camera and light located on the wrist-roll joint next to the end effector, as shown in Figure 2-6. The camera has a remotely controlled zoom lens with focal length variation from 18 mm to 108 mm, and corresponding fields-of-view of 30° (48° diagonal) to 5° (8.5° diagonal). The near focus distance is 31.5 inches (0.8 meters) from the front of the lens. The viewing light has power of 200 watts (28 VDC) and provides a 40° included angle beam.

There is a provision for a kittable CCTV camera with pan and tilt at the elbow joint, as shown in Figure 2-7. The camera is identical to the wrist camera. The pan and tilt unit is remotely controlled from the aft deck and provides ±70° tilt and ±170° pan capability.

2.3 Operating Modes

The shuttle RMS can be operated in any one of the following modes:

(a) manual augmented,
(b) automatic,
(c) single joint control,
(d) direct drive control,
(e) back-up drive control.

The manual augmented mode is the most frequently used method of operating the arm. The operator uses the two hand controllers to "fly" the end effector. Using the translational hand controller (THC) he or she commands the end effector linearly in any direction while the Rotational Hand Controller (RHC) is used to command angular motions (roll, pitch or yaw). Hand controller deflections result in proportional rates of the end effector. The operator has a choice of commanding in the orbiter coordinate system, the end effector coordinate system, or a payload-based coordinate system. The hand controllers are shown in Figure 2-8 and Figure 2-9.

The automatic mode, as the name implies, allows the operator to call up any one of the twenty preprogrammed trajectories which are executed in a "hands-off" manner. In addition, the operator can enter into the Shuttle General Purpose Computer (GPC), via the keyboard, desired start and end-point coordinates for the arm to "fly" from and to, automatically.

The single joint mode is computer-assisted and enables the operator to move the arm in a controlled, joint-by-joint manner, using switches on the Display and Control (D&C) panel. The D&C panel, shown in Figure 2-10 and Figure 2-11, has digital displays for joint angles and rates. The joints not commanded are held in position-hold by the GPC. This mode is used mainly to unstow and restow the arm from and to its retention latches, as well as to drive the arm away from joint travel limits, when required.

The direct drive mode is a hard-wired mode which by-passes the Manipulator Controller Interface Unit (MCIU) and GPC and gives the operator the capability of driving the arm joint-by-joint in a contingency. The uncommanded joints are held in position by their respective brakes. Joint angle and rate data may or may not be available in this mode, depending on what system fault necessitates selection of the direct drive mode.

The back-up drive mode is similar to the direct drive mode but is accessed and implemented by a back-up system which is separate from the prime system used for other operating modes (with the exception of the final gear train). This mode is used when all modes in the prime channel become unavailable due to malfunctions. It fulfills the SRMS fail-safe requirement and allows the operator to place the SRMS in a safe position on a joint-by-joint basis (with no display information) for either payload separation, arm latching or arm separation.

2.4 Software

The shuttle RMS software resides in the GPC where it translates the operator commands, transmitted to the GPC from the D&C panel, into commands required for the RMS hardware operation. The software also performs the task of monitoring the hardware and RMS status. Interface with RMS hardware is via the data bus between the MCIU and the GPC.

The RMS software subsystem is non-redundant and is divided into fifteen principal functions which perform the required mathematical and logical operations. The software modes provided can be classified as either requested modes (manual, automatic, single or test) or non-requested modes (suspend, idle or temperature monitor). The requested modes are those for which a position exists on the mode-select switch of the D&C panel. The software is in suspend mode whenever communication with the MCIU is shut down. Idle mode is executed when either hardware safeing is in progress, brakes are being applied, software stopping has been ordered, or a test mode has been selected on the D&C panel. Temperature-monitor mode is executed when no arm has been selected ("powered").

The executive function (EXEC) performs the operating mode initialization and permits mode transitions when various prerequisites are met. Depending on the software mode, EXEC calls a specific set of principal functions in a particular sequence in order to control and monitor arm motion.

The RMS software occupies 13K 32-bit words in the GPC memory and is executed at 12.5 Hz. The MCIU features an 8-bit microprocessor with 4K bytes of ROM and 1K bytes of RAM. The MCIU processng is executed at 25 Hz.

2.5 Interfaces

Description and requirements of interfaces between the SRMS and payloads, and the SRMS and the orbiter are given in the following documents (NASA document numbers are given in parenthesis):

(a) RMS/Payload Interface Control Document - SPAR-RMS-ICD.019 (NASA-ICD-2-06001),

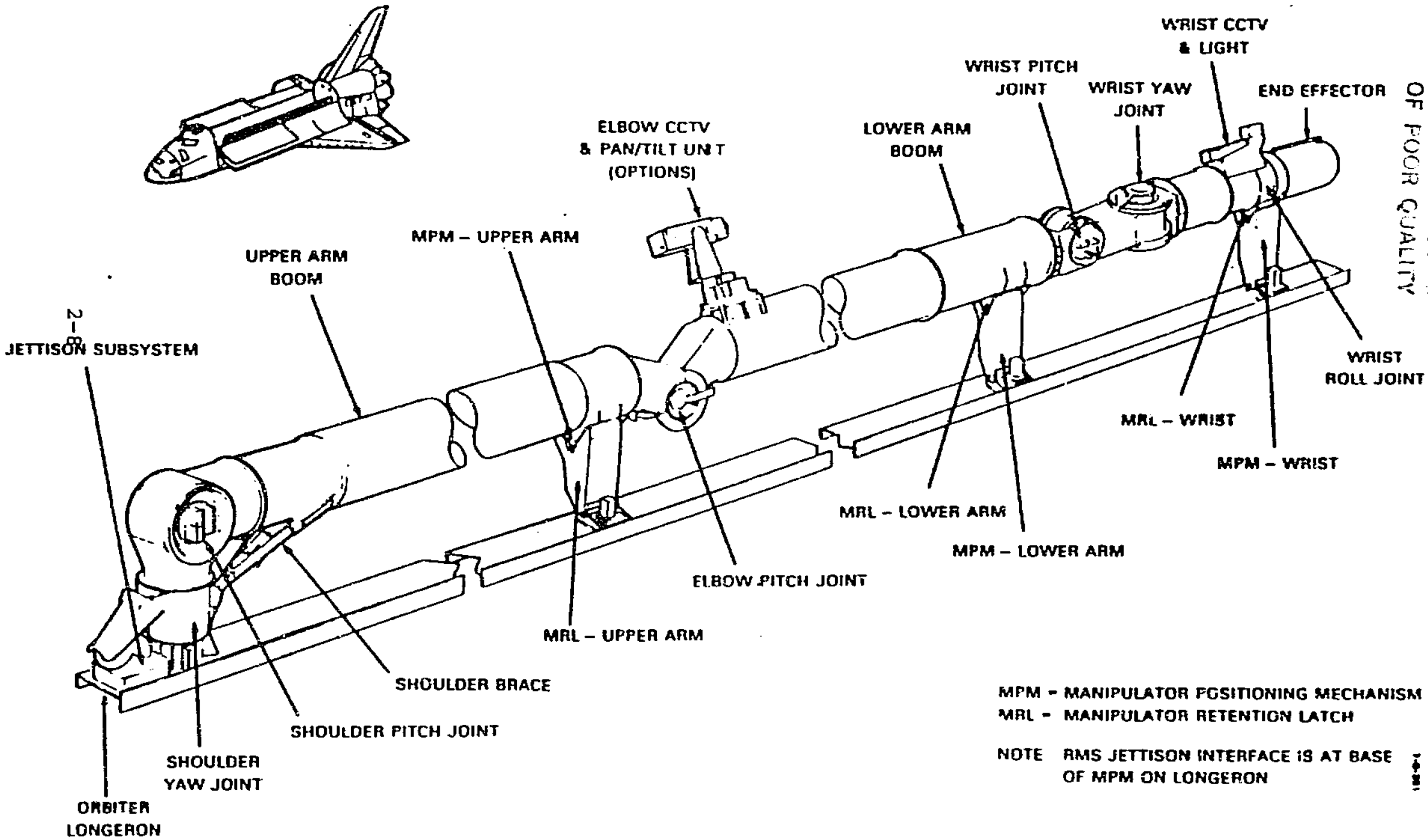
(b) Manipulator/Orbiter Physical Interface Control Document - SPAR-RMS-ICD.026 (NASA-ICD-3-0018-01),

(c) Manipulator/Orbiter Thermal Interface - SPAR-RMS-ICD.015 (NASA-ICD-3-0018-06),

(d) Manipulator/Orbiter Electrical Interface - SPAR-RMS-ICD.021 (NASA-ICD-3-0018-02),

(e) Manipulator Displays & Controls/Orbiter Interface Control Document - SPAR-RMS-ICD.028 (NASA-ICD-3-0018-03).

FIG 2-1 MECHANICAL ARM — STOWED POSITION & GENERAL ARRANGEMENT



MPM - MANIPULATOR POSITIONING MECHANISM
MRL - MANIPULATOR RETENTION LATCH

NOTE RMS JETTISON INTERFACE IS AT BASE OF MPM ON LONGERON

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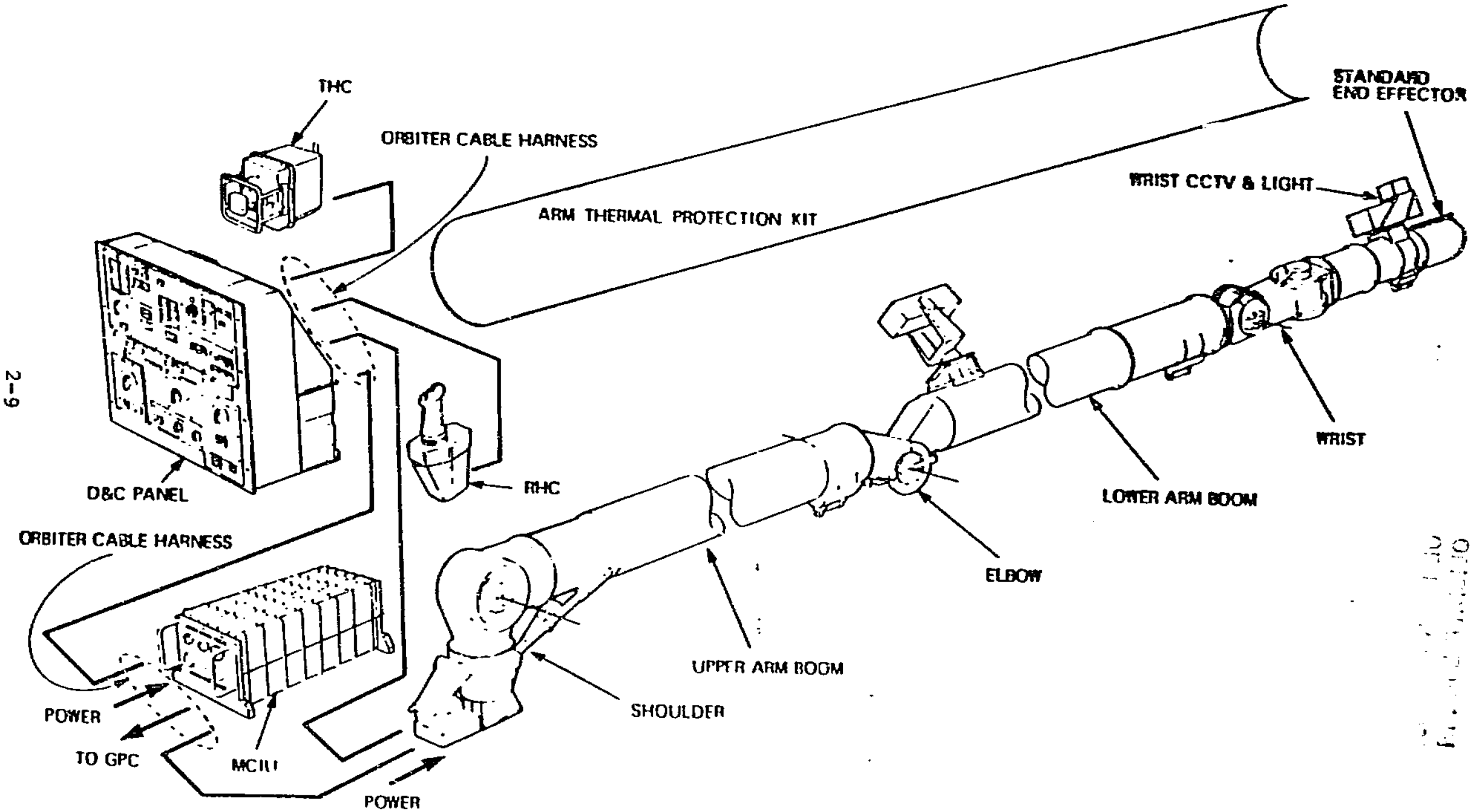

FIG. 2-2 SRMS COMPONENTS

GPC - GENERAL PURPOSE COMPUTER
MIA - MULTIPLEXER INTERFACE ADAPTOR
MCIU - MANIPULATOR CONTROLLER INTERFACE UNIT
ABE - ARM BASED ELECTRONICS
DSKY - DISPLAY AND KEYBOARD ENTRY

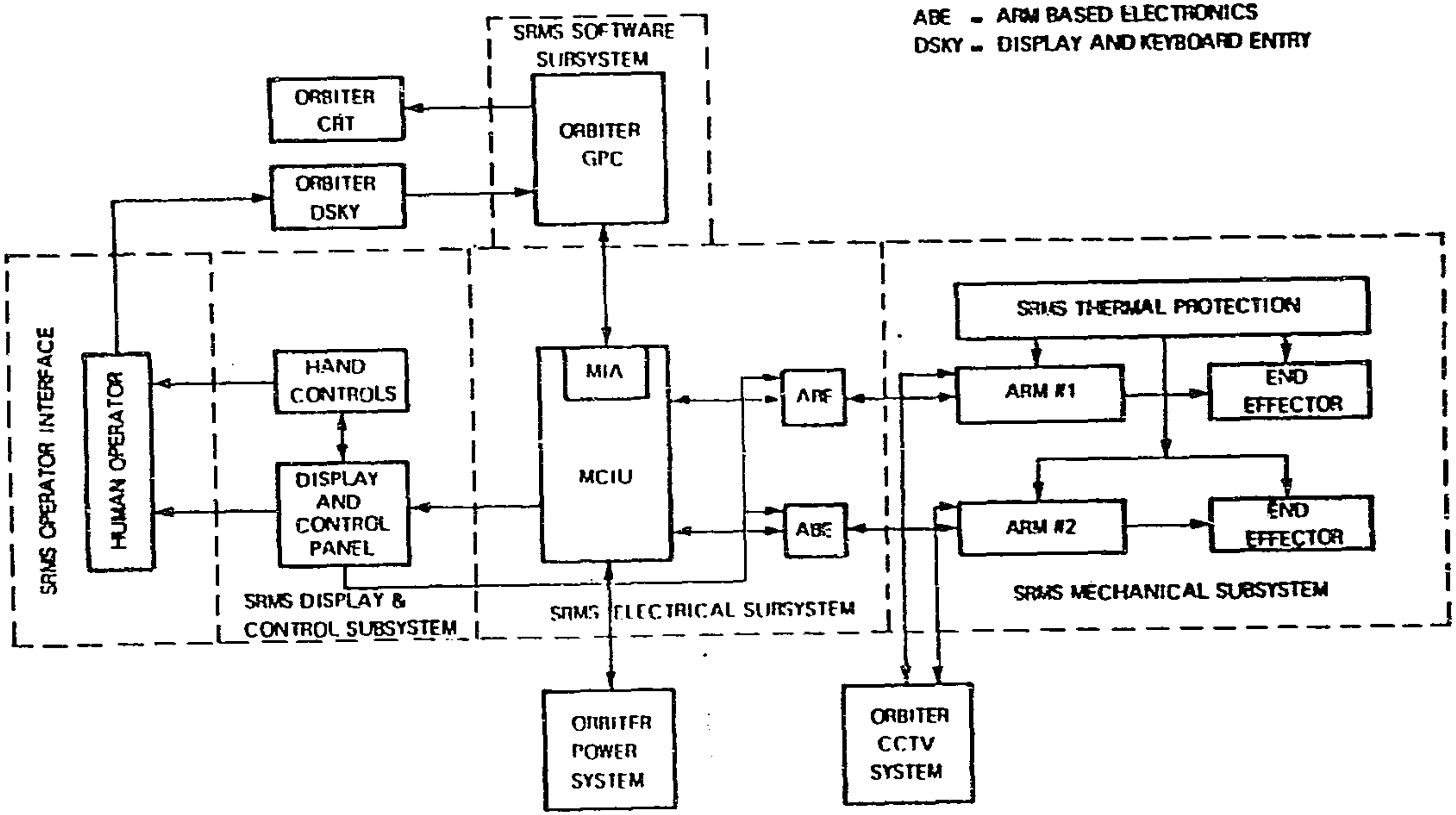

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FIG. 2-3 SIMPLIFIED SYSTEM BLOCK DIAGRAM

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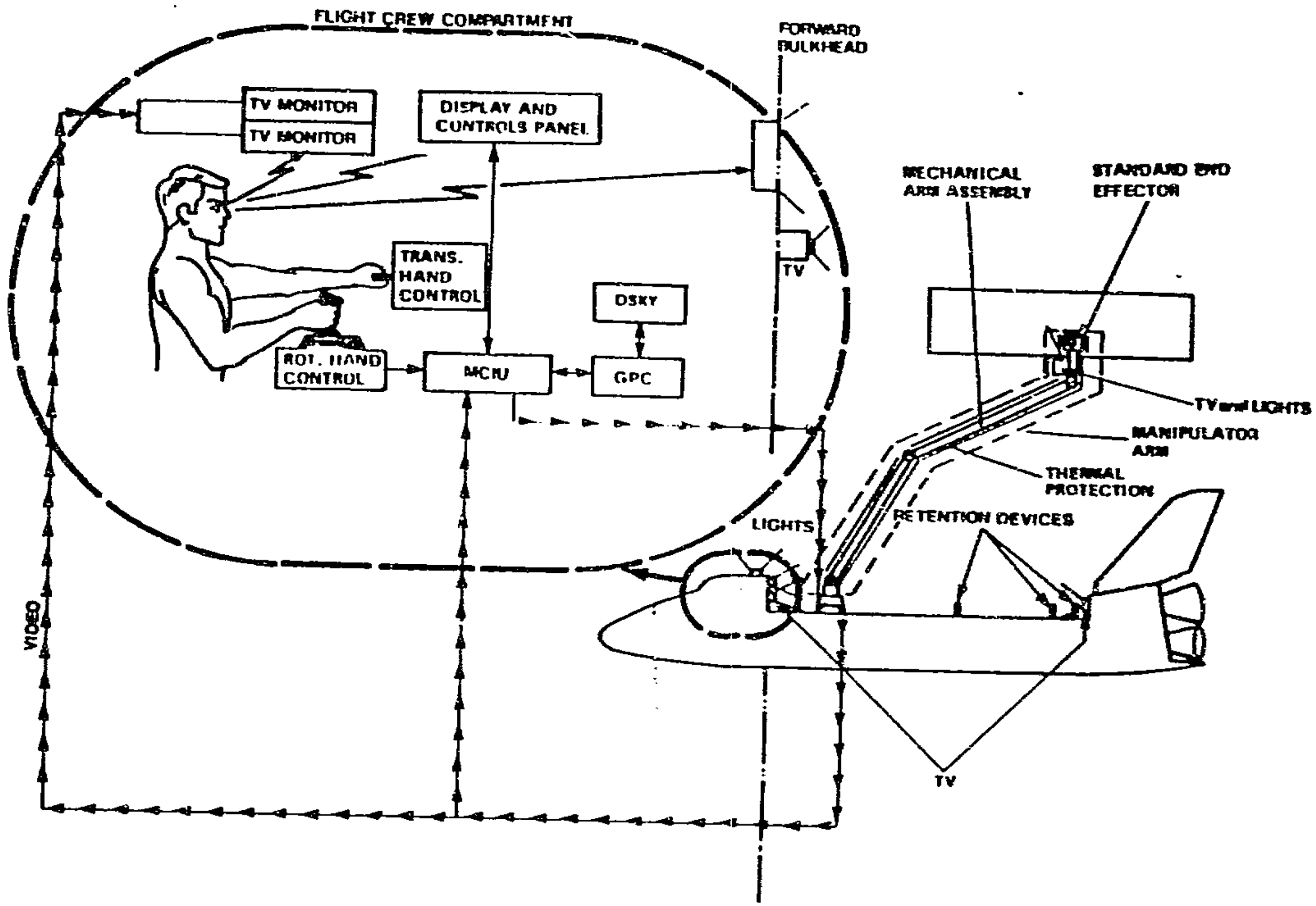

FIG. 2-4 SHUTTLE REMOTE MANIPULATOR SYSTEM

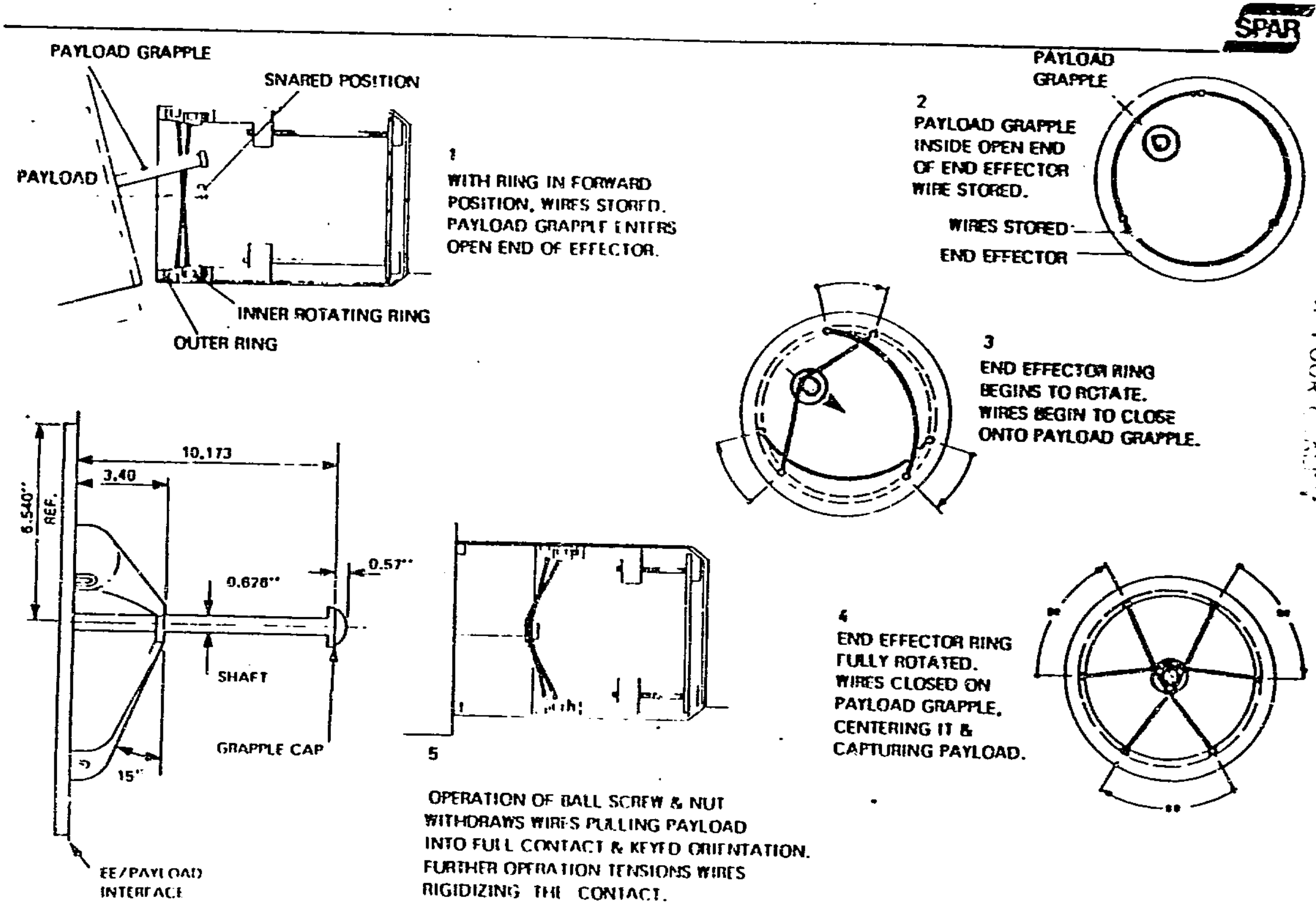

FIG. 2-5 SNARE E/E – CAPTURE AND RIGIDIZE SEQUENCE

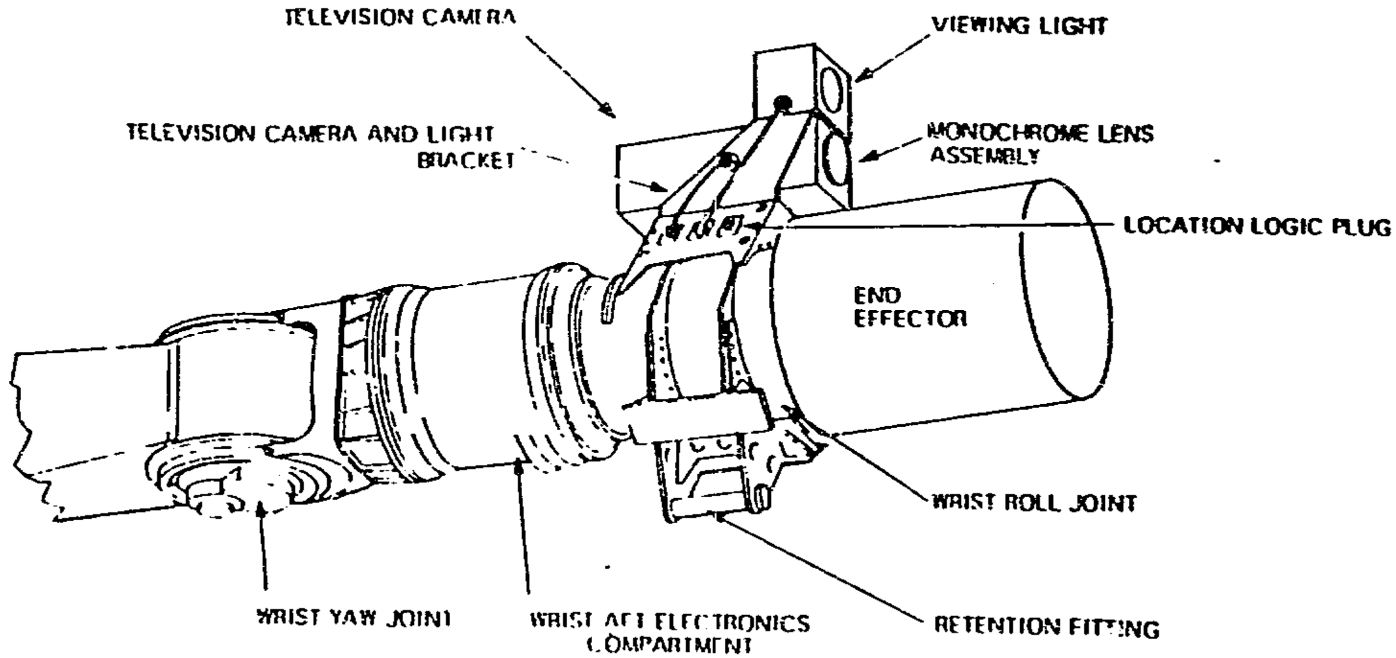

FIG. 2-6 WRIST ROLL MOUNTED RMS CLOSED CIRCUIT TELEVISION CAMERA AND LIGHT SUBASSEMBLY

SPAR

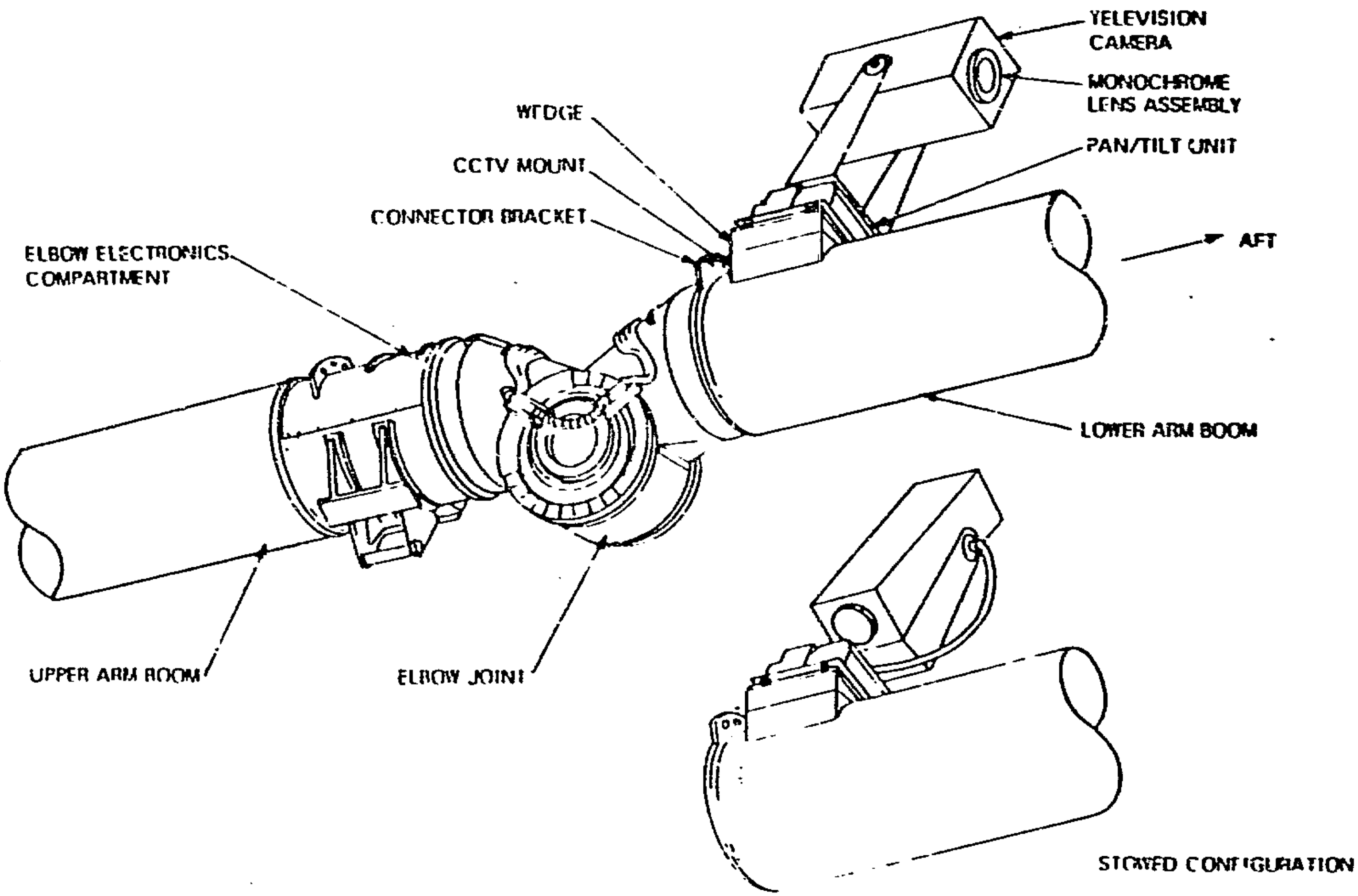

FIG. 2-7 ELBOW MOUNTED RMS CLOSED CIRCUIT TELEVISION CAMERA (CCTV) AND PAN/TILT SUBASSEMBLY

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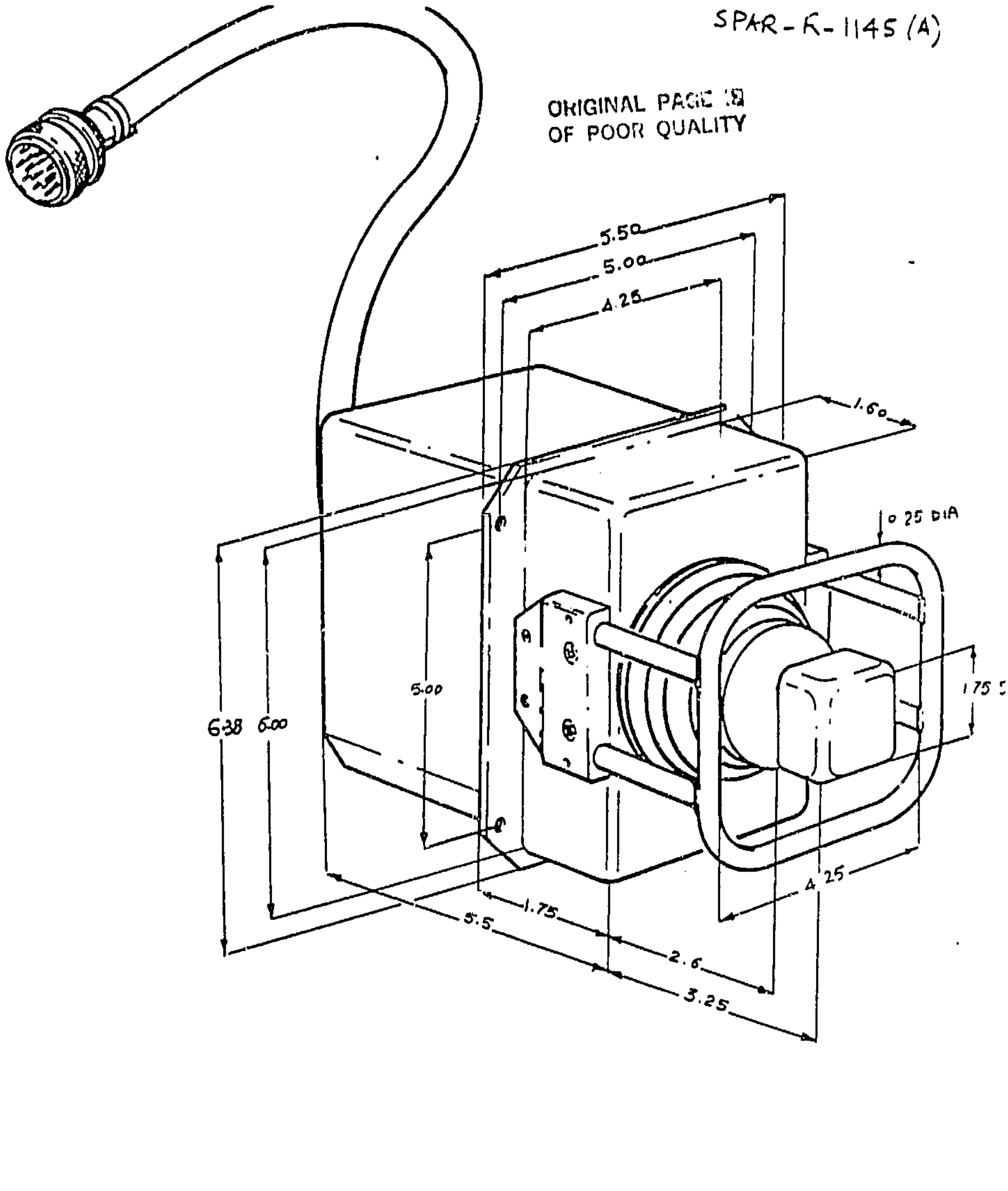

FIG 2.8: TRANSLATIONAL HAND CONTROLLER (THC)

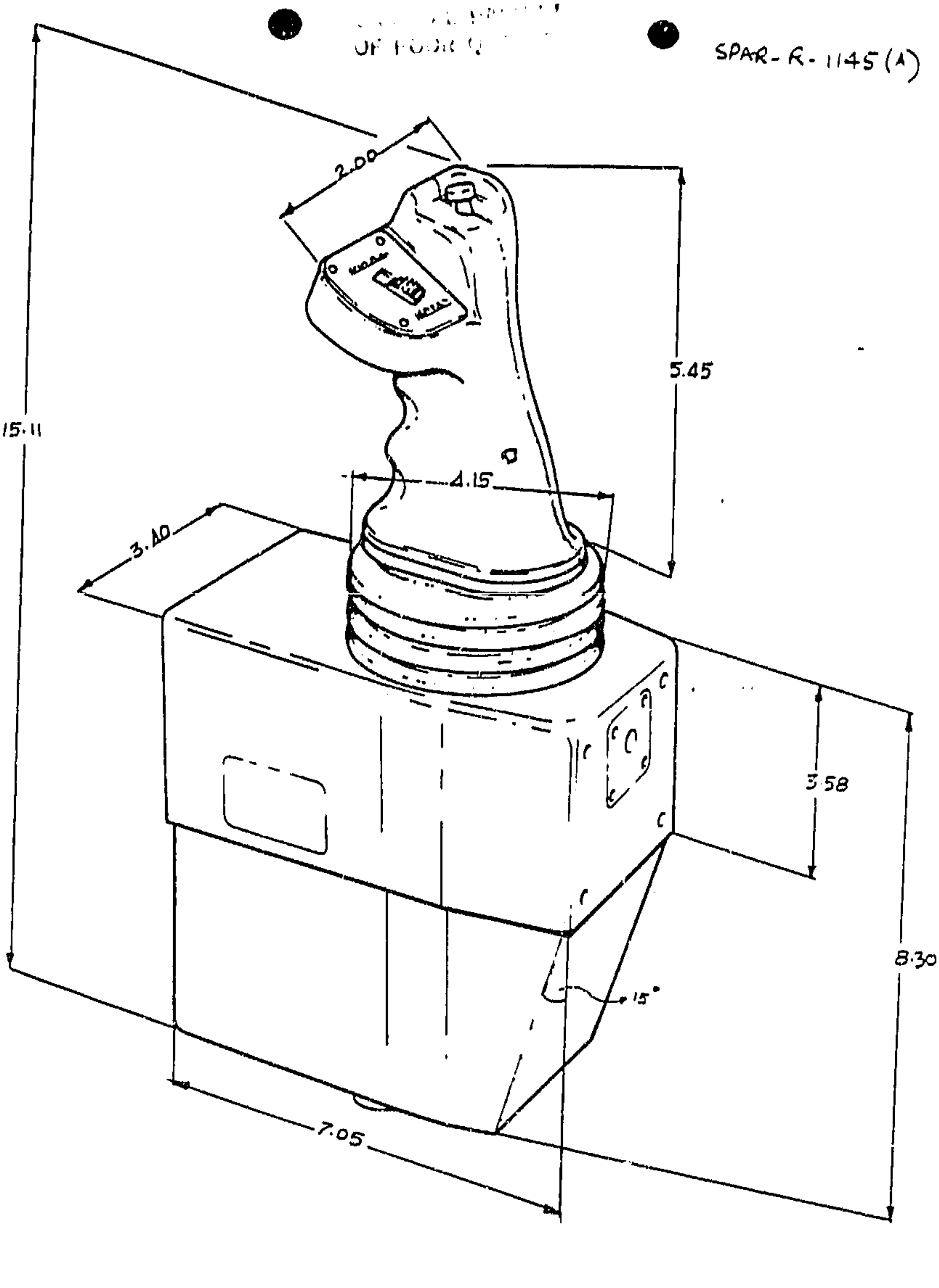

FIG 2.9 ROTATIONAL HAND CONTROLLER (RHC)

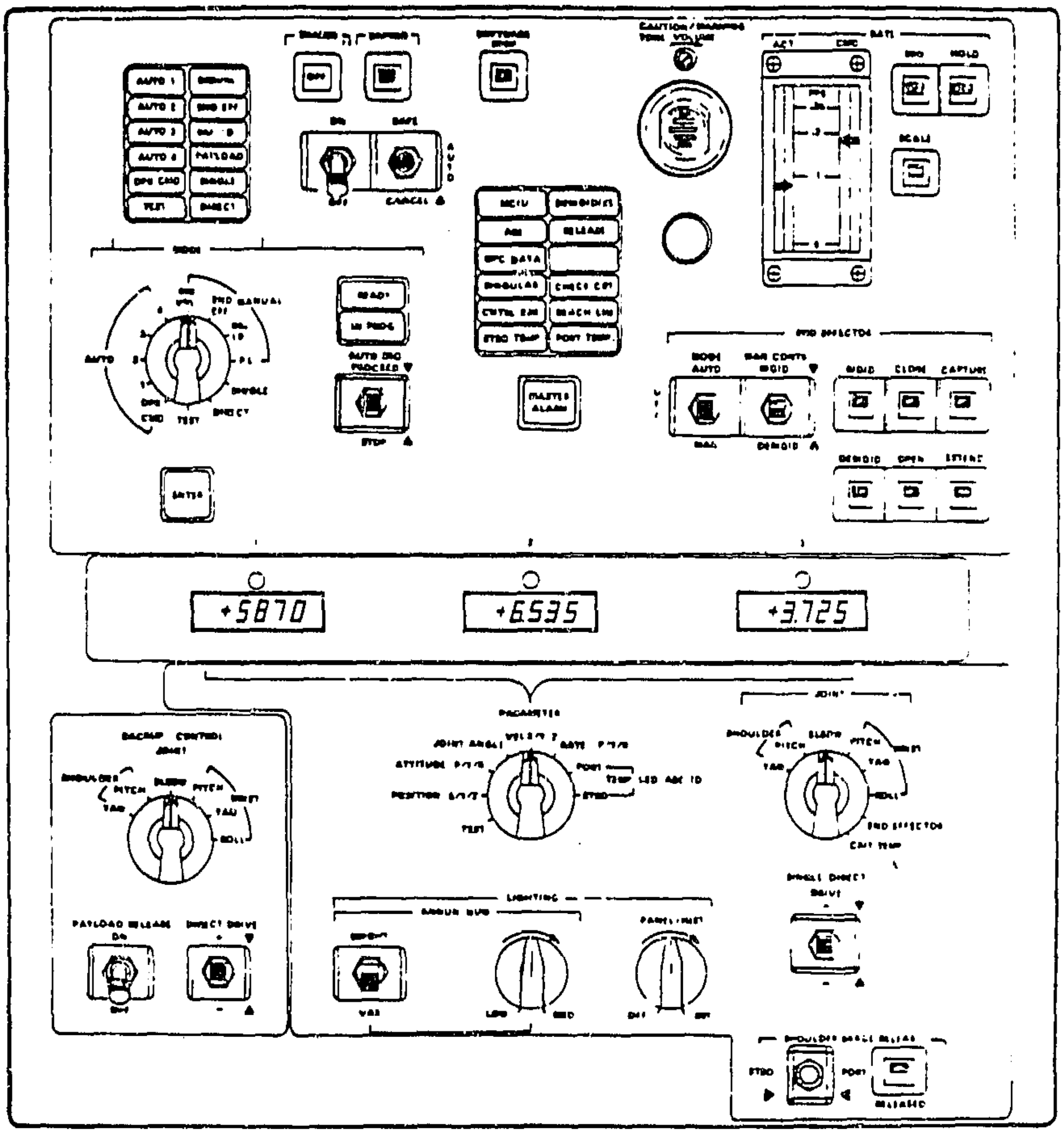

FIG. 2-10 DISPLAY AND CONTROL PANEL

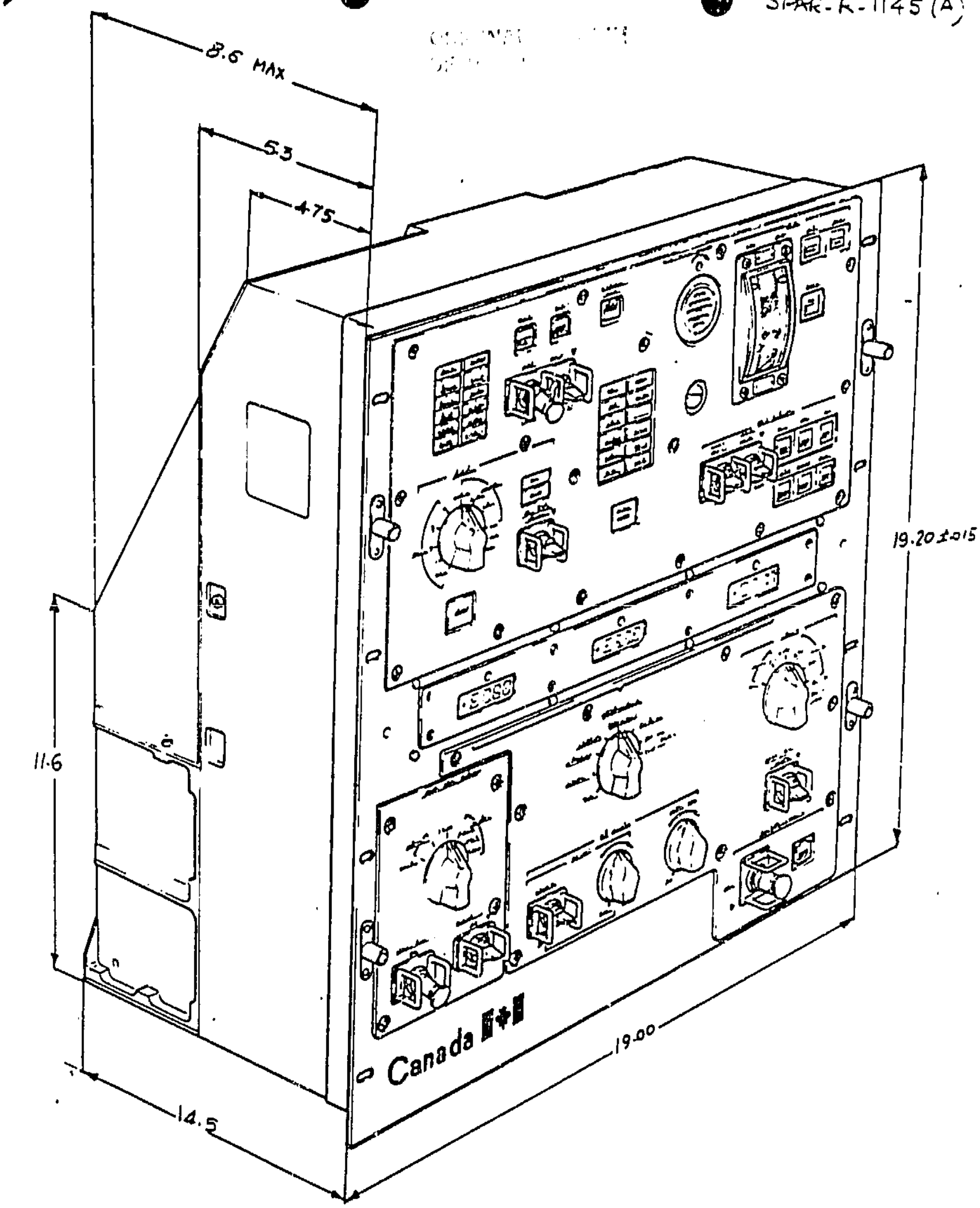

FIG. 2.11 DISPLAY & CONTROLS PANEL

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Table 2-1

SRMS Mass, Power and Size Properties

No.	SRMS Element	Mass - kg (lb)	Power Consumption + 115V 400 Hz (VA)	Power Consumption + 28 VDC (Watts)	Approximate Size
1	Manipulator Arm*	390 (857)		1735	0.4m dia x 16m long
1.1	-Mechanical Arm Assembly	334 (735)		500	0.4m dia x 15m long
1.2	- End Effector	38 (61)		(330)**	0.4m dia x 0.4m long
1.3	- Thermal Protection	15 (33)		975	—
1.4	- Wrist CCTV, Light, Bracket	13 (28)		260	0.55m x 0.45m x 0.2m
2	Displays & Control Subsystem	31 (68)	120	2	
2.1	- Displays & Control Panel	24 (52)			0.5 x 0.5m x 0.4m
2.2	- Handcontrollers (RHC & THC)	7 (16)			0.2m x 0.15m x 0.25m - THC 0.2m x 0.1m x 0.4m - RHC
3	Manipulator Controller Interface Unit (MCIU)	8 (17)		84	0.5m x 0.3m x 0.3m
	Total System	429 (942)	120	1821	

* Elbow CCTV, Pan/Tilt Unit and Switching Unit available as add-on kit @ 15 kg (33 lb.)

** Transient additive to steady state consumption for 10 seconds during E/E rigidization.

\+ Typical Power Consumption for DDT&E Arm - prime channel only

TABLE 2-2

JOINT PARAMETERS

PARAMETER	UNITS	SHOULDER YAW	SHOULDER PITCH	ELBOW PITCH	WRIST PITCH	WRIST YAW	WRIST ROLL
Mechanical Stops	Deg.	±180	-2 +145	-161 +2.4	±121.4	±121.3	±447
Software Stops	Deg.	±177.4	+0.6 +142.4	-157.6 -0.4	±116.4	±116.6	±442.0
Backlash (Output)	Deg.	0.114	0.114	0.114	0.150	0.150	0.150
Gear Ratio	-	1842	1843	1260	738	739	738
Torque Capability (Output)							
Minimum	Ft-lbs.	772	772	528	231	231	231
Maximum	Ft-lbs.	1158	1158	792	347	347	347
Brake Slip Torque	Ft-lbs.	600-1063	600-1063	410-756	195-418	195-418	195-418
Backdrive Threshold	Ft-lbs.	48-155	48-155	34-110	20-65	20-65	20-65
**Joint Rate (Unloaded)	Rad/sec.	0.04	0.04	0.056	0.083	0.083	0.083
**Joint Rate (32K Payload)	Rad/sec.	0.004	0.004	0.0056	0.0083	0.0083	0.0083

** Level C Data

3.0 POTENTIAL IMPROVEMENTS IN SHUTTLE RMS

The present shuttle RMS design has been primarily governed by considerations of payload retrieval and deployment from the orbiter. A number of improvements of the RMS can be envisaged to expand its capabilities in performing tasks related to satellite servicing and module exchange, inspection, space construction, materials handling and transfers, etc. Such tasks are likely to be routinely perfomed on a space station.

The improvements can be broadly divided into the following categories:

(a) improved ability to do precise tasks,
(b) improved operator control aimed at reducing operator effort and/or work load,
(c) increased reach and articulation.

The items that can be considered are discussed below. The needs and priorities of the RMS application would determine which of these items should be considered for a space station.

3.1 Force/Moment Sensing and Feedback

The present RMS has no provision for sensing the force and moments at the end effector/payload interface, although the maximum level of the forces/moments can be adjusted by setting the motor current limits in the joints at the required values. Knowledge of forces and moments at the arm tip may be very helpful to the operator in performing certain tasks. A force/moment sensor can be mounted between the wrist roll joint and the end effector. Feedback can be provided to the operator as a display or as tactile feedback through the hand controllers. As a further step, automatic force/moment limiting can be achieved by providing the feedback to the servo control system.

3.2 Force/Moment Accommodation

The concept of force accommodation is basically a software enhancement once the force/moment sensing and feedback are in place. The operator would enter, or call up, a "force/moment accommodation matrix" describing the task that he wishes to perform. The manipulator would then perform the task using the loading and kinematic constraints resulting from the accommodation matrix. Examples of tasks where such an approach would be useful are:

(a) inserting a peg into a hole,
(b) turning a threaded object,
(c) prying objects apart,
(d) attaching objects without collision or excessive loads.

3.3 Visual Proximity Sensing

The shuttle RMS provides CCTV views from wrist camera (and optional elbow camera) to the operator who can use these views to judge relative distance, orientation and rates, between the arm tip (end effector) and the payload that he/she is trying to capture. A real-time photogrammetric system (RPS) can use the TV view to determine precisely the distance, orientation and rates between the end effector and a payload. Such a system has been developed by the National Research Council of Canada (NRCC) and Leigh Instruments in Canada and has been demonstrated at the Manipulator Development Facility (MDF) at NASA-JSC. The system uses a target pattern on a payload (four dots, for example) and the variation of the target pattern as seen through the TV camera, to compute the position, orientation and rate information, using a microprocessor. This information can be used in open-loop (manual augmented mode) or closed-loop (automatic mode) to position the manipulator (and/or the payload) more precisely than is possible in the present design.

The Leigh/NRCC photogrammetric system could also be incorporated into the SRMS control system to enable the arm to automatically track a free-flying payload after it has been acquired in the wrist CCTV camera's field of view. Once legal capture conditions are achieved, the operator would execute final grappling through manual control. Such a system would be useful in capturing payloads with high relative rates.

A similar system on the orbiter could also be used for berthing payloads or berthing orbiter to a space station.

3.4 Stand-Alone Computer System

The SRMS uses the shuttle GPC as a computing resource and as a repository of SRMS software. A stand-alone computer system for the RMS would eliminate this dependence on the GPC allowing the RMS to be located as a separate system on a space station. Advantage may also be taken of the advances in VLSI technology in designing the new computer system which would be able to provide additional computing resources needed by other features such as force/moment sensing and feedback, force accommodation, photogrammetric sensing, more degrees of freedom, and collision avoidance.

3.5 Collision Avoidance Software

Limited computer resources precluded inclusion of collision avoidance function in the SRMS software to have the capability of predicting potential collisions between the arm, payload, orbiter and its contents. The collision avoidance capability can be considered for applications in Space Stations where routine RMS operations may be automated to a high degree and VLSI technology would make considerable computing resources readily available. The computer memory requirement would depend on the Space Station configuration and the resolution to which its features are described. For the Shuttle orbiter, collision avoidance software was estimated to require about 14K, 32-bit words.

3.6 End-of-Arm Tools

The RMS can be used for performing functions such as activation of latches and mechanisms, attachment and detachment of modules, connection and disconnection of umbilicals, and holding objects to support space construction tasks, if suitable tools are designed for operation at the end of the arm.

An active tool concept, called the Universal Service Tool System for satellite servicing has also been developed by Spar, and is shown in Figure 3-7 and Figure 3-8. The tools could have force/moment sensing and feedback features discussed earlier.

3.7 Voice Activated Controls and Displays

The technology of direct voice communication to and from a computer is becoming increasingly sophisticated. Voice interface systems for caution/warning, data entry, information retrieval and system control are being developed for aircraft cockpits. Such a system can also be considered for RMS to reduce workload for the operator and to enable him/her to exercise more control authority over the RMS without letting go of the hand controllers. A voice activated system could be considered for the following functions:

(a) selection of the operating mde of the RMS,
(b) selection of display parameters (joint, angles, rates, etc.),
(c) some of the end effector operations,
(d) caution and warning, and to "read" the display parameters to the operator.

The hand controllers are not likely to be replaced since they are used to provide continuous and rapidly changing commands. Inputs of these commands verbally would be too slow and probably tiring for the operator.

The additional hardware needed to implement voice interfaces could be incorporated into the display and controls assembly leaving the rest of the system architecture unchanged. Reliability and design studies would have to be done for implementing the system. One approach would be to keep the present functions and switches on the D&C panel and design a voice interface system as an add-on module.

3.8 Control from a Payload Station (MRWS)

In some potential applications of the RMS, a manned remote work station (MRWS) such as the open cherry picker, is envisaged. Such a work station would be an RMS payload. A display and control panel can be provided in the work station to operate the arm. This would be in addition to the main control panel in the RMS crew cabin.

This capability could be provided by interposing a "control station selector" between the MCIU and the main D&C panel. The selector would perform the function of switching communication to and from the MCIU and different control stations.

The capability to operate the direct drive and back-up modes of the arm would be available only at the main D&C panel (the commands for these modes by-pass the GPC and MCIU).

The D&C panel in the MRWS could be similar to the main D&C panel without the following switches and control:

(a) back-up control, payload release and direct drive switches,
(b) the end effector operating switches and controls,
(c) shoulder brace release switches (the shoulder brace may not be needed for the RMS for a space station).

Some additional switches to power-up the D&C panel in MRWS would have to be provided. Similarly, some additional switches to transfer control to the MRWS D&C panel would have to be provided on the main RMS D&C panel.

In addition to the D&C panel, the MRWS would also have a set of hand controllers to "fly" the tip of the RMS.

Since MRWS is a payload, electrical power can be transferred to it from the payload electrical connector at the end effector. A power conditioner at the MRWS would condition and distribute this power to the arm D&C panel and other elements of the MRWS. Communication between the MCIU and the MRWS-D&C panel could either be over the available payload signal lines, or over a separate set of wires/cables to be attached outside the arm structure. Communication would be via two serial data links consisting of two twisted pair wires. Typical EIA RS-232C interface can be used. The additional wires could be incorporated in the present connector with small design changes in the linkage mechanisms to engage the connector.

The above approach may also be adopted in developing a portable D&C panel for the RMS, or to establish control stations at different locations in a space station.

3.9 Increased Reach and Articulation

To position and orient an object in three dimensional physical space, six degrees of freedom are necessary but not always sufficient. The shuttle RMS, with six degrees of freedom, has some singular configurations where one or more degrees of freedom are lost. The software manages singularities when they are encountered. When one or more joints encounter their travel-limits, the SRMS cannot be manoeuvered along the required trajectory.

These problems can be alleviated by providing additional joints. For example, addition of an upper arm roll joint and a lower arm roll joint would reduce the effects of singularities and reduce loss of desired trajectory due to joint travel-limits.

Additional degrees of freedom would, however, complicate the control algorithms and changes in software would be required.

It is also possible to readily develop derivatives of shuttle RMS using the SRMS hardware. An example is a three or four degree of freedom manipulator such as the Handling & Positioning Aid (HPA) being considered by NASA for the Orbiter (Figure 3-10 and Figure 3-11). A longer (or shorter) manipulator, with six degrees of freedom, could be developed by increasing (or decreasing) the lengths of the arm booms. A 100 ft. long manipulator of this type, for instance, would weigh about 1200 lbs. and have a stiffness of about 1.5 lb. per inch minimum at the tip. Reach of the shuttle RMS can also be increased, for tasks such as inspecting or positioning a work station, by using a deployable Astromast at the end of the arm. A stowed Astromast, with a work station and/or inspection equipment attached to it, can be picked up and manoeuvered as a payload. At a desired position, the Astromast can be extended thereby extending the work station and/or the inspection equipment. A concept for such a system has been developed by Spar and was considered by NASA for orbiter inspection tasks such as the inspection of the orbit thermal system tiles (ITRS). This system involved an Astromast with 70 ft. extension, with the total package (Astromast and work station) weight of about 450 lbs. and a stiffness of about 0.5 lb. per inch at the tip of the deployed Astromast. The concept is shown in Figure 3-12.

FIGURE 3-1, FORCE-MOMENT SENSING SYSTEM: BLOCK DIAGRAM

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FIGURE 3-2 FORCE/MOMENT SENSOR CONCEPT

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FIG 3.3: REAL TIME PHOTOGRAMMETRIC SYSTEM (RPS) FOR SHUTTLE RMS

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FIGURE 3-4 A CONCEPT OF A MULTI-PURPOSE PASSIVE TOOL

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FIGURE 3-5 ACTIVE POWER TOOL CONCEPT

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FIGURE 3-5 ACTIVE POWER TOOLS (CONTD.): CONCEPTS

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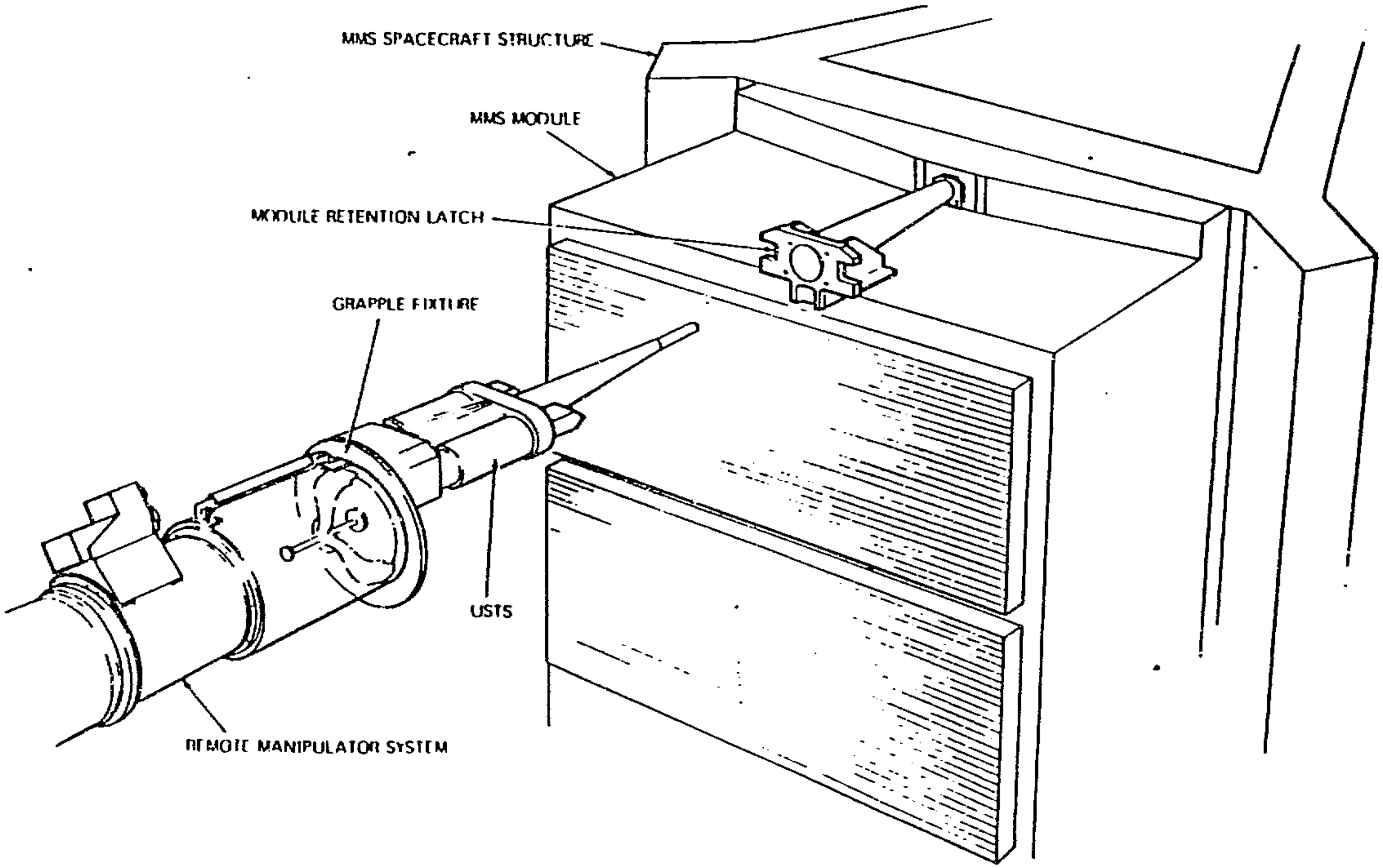
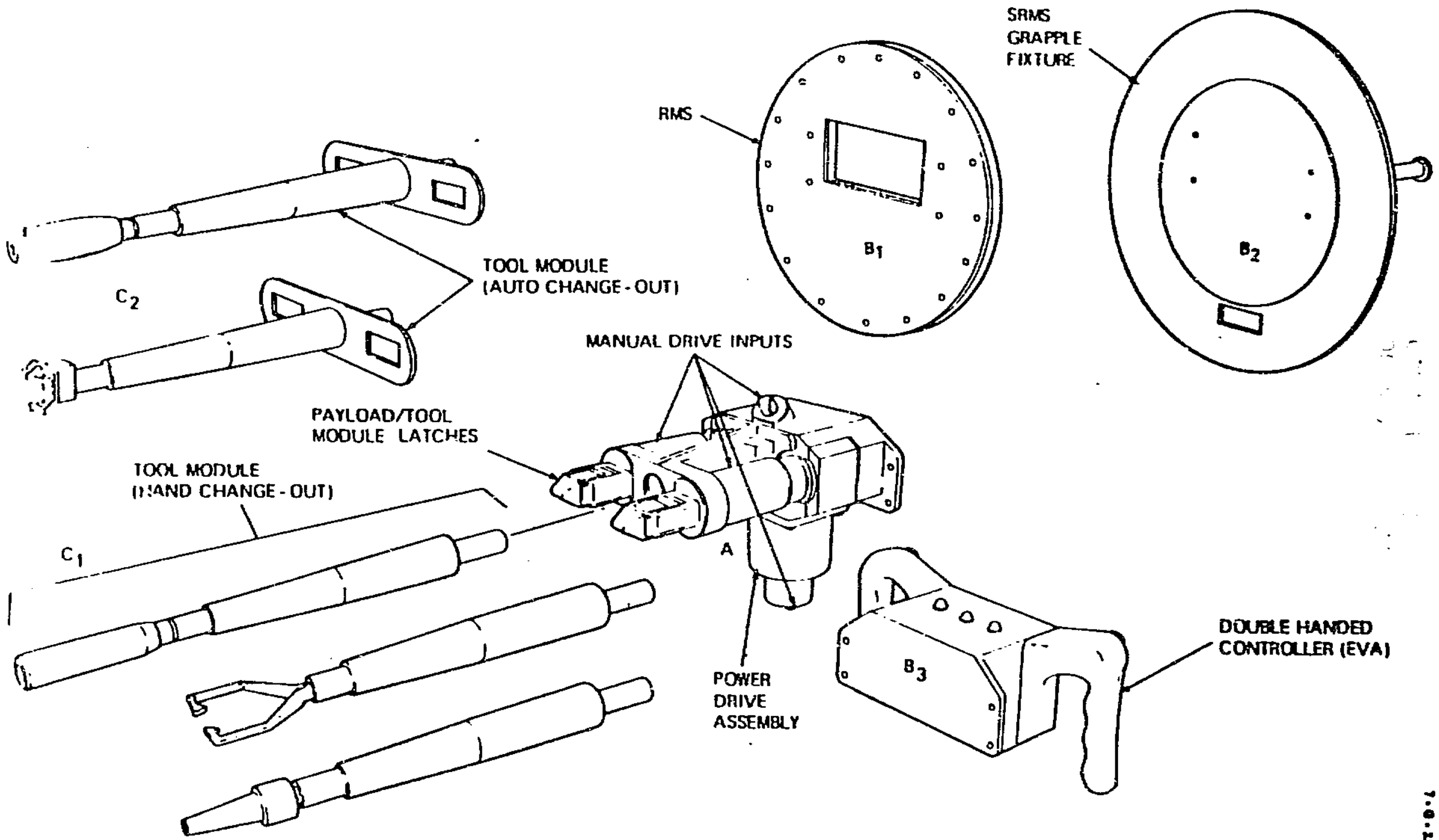
Figure 3-7 UNIVERSAL SERVICE TOOL SYSTEM (USTS) CONCEPT



PATENT APPLIED FOR

Figure 3-8 UNIVERSAL SERVICE TOOL SYSTEM



PATENT APPLIED FOR

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FIG 3-9 DISPLAY & CONTROLS AT THE MRWS (PAYLOAD)

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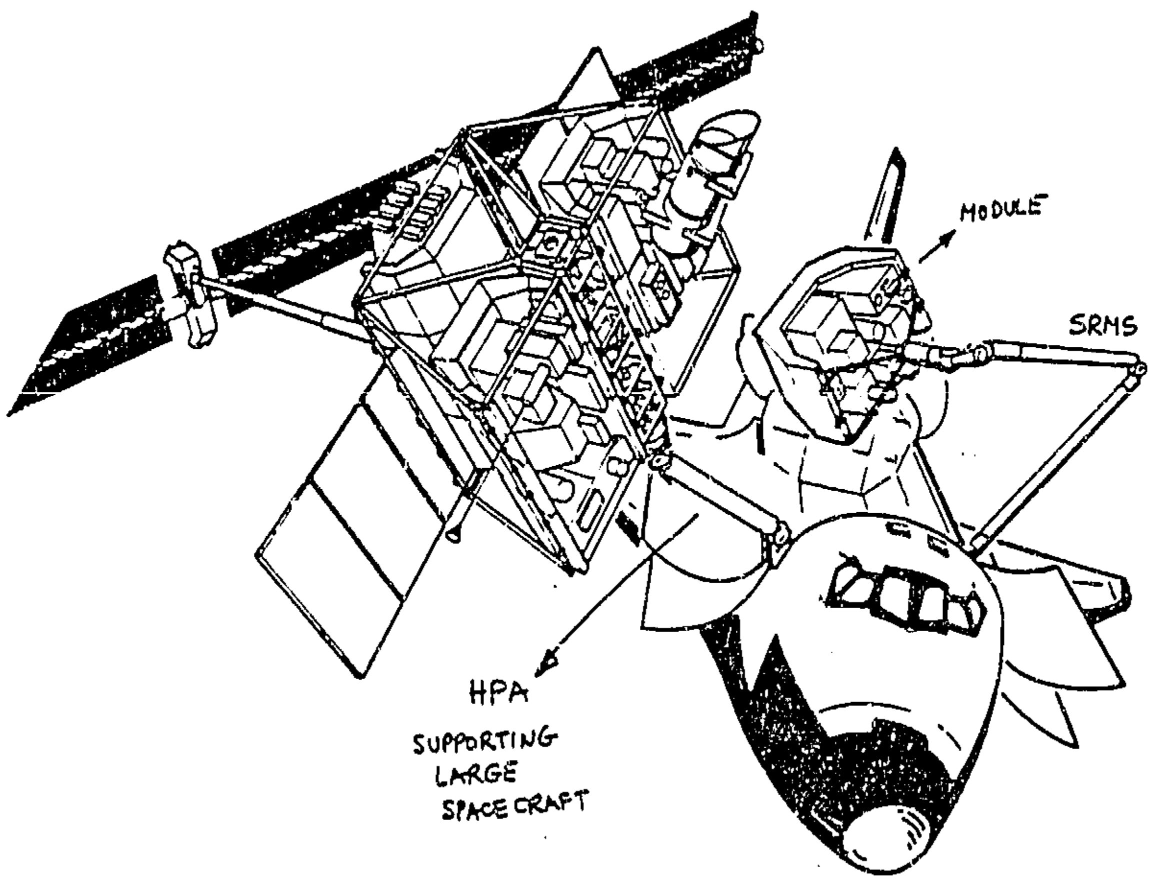

FIGURE 3-10 CONCEPT ILLUSTRATING HPA SUPPORTING SPACECRAFT DURING SERVICING

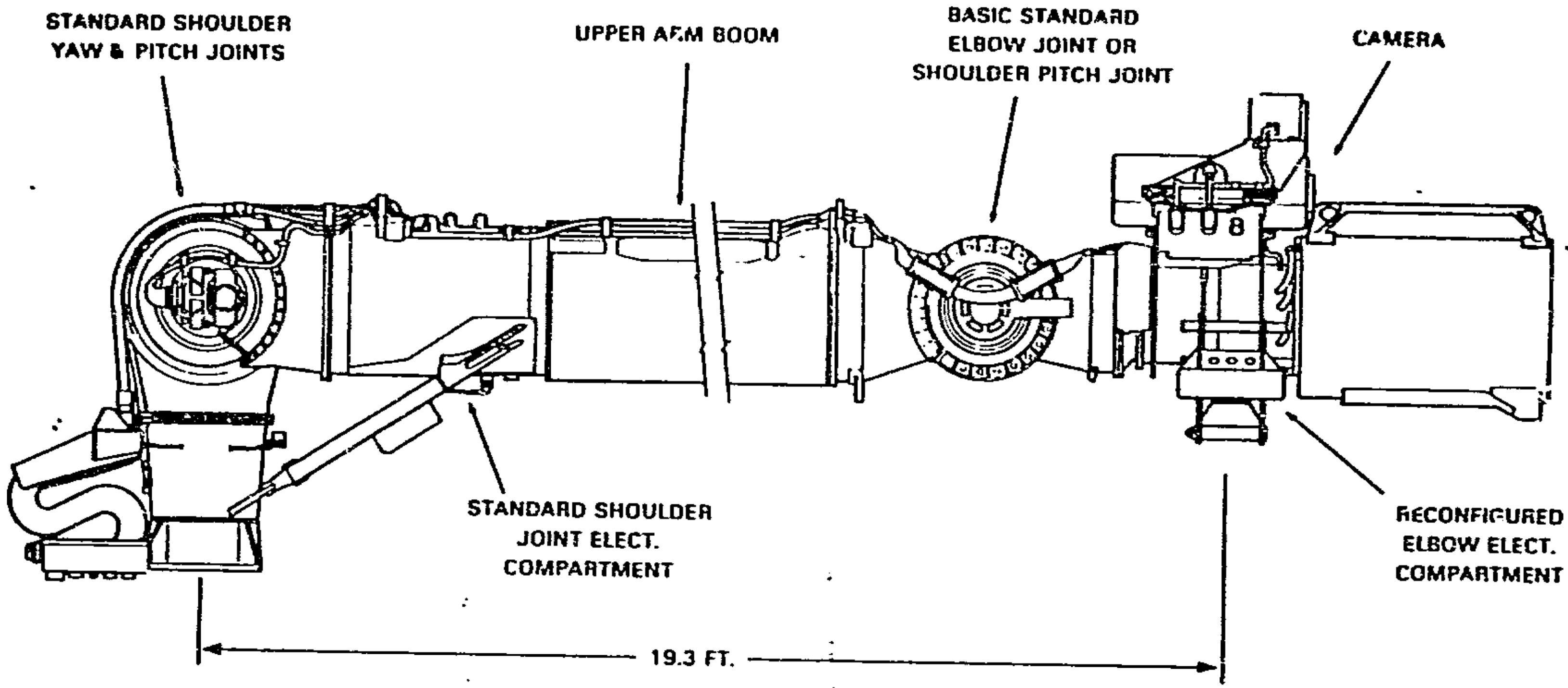

FIGURE 3-11 BASELINE HPA CONFIGURATION

FIG 3-12 ITRS – SYSTEM DESIGN ON-ORBIT CONFIGURATION

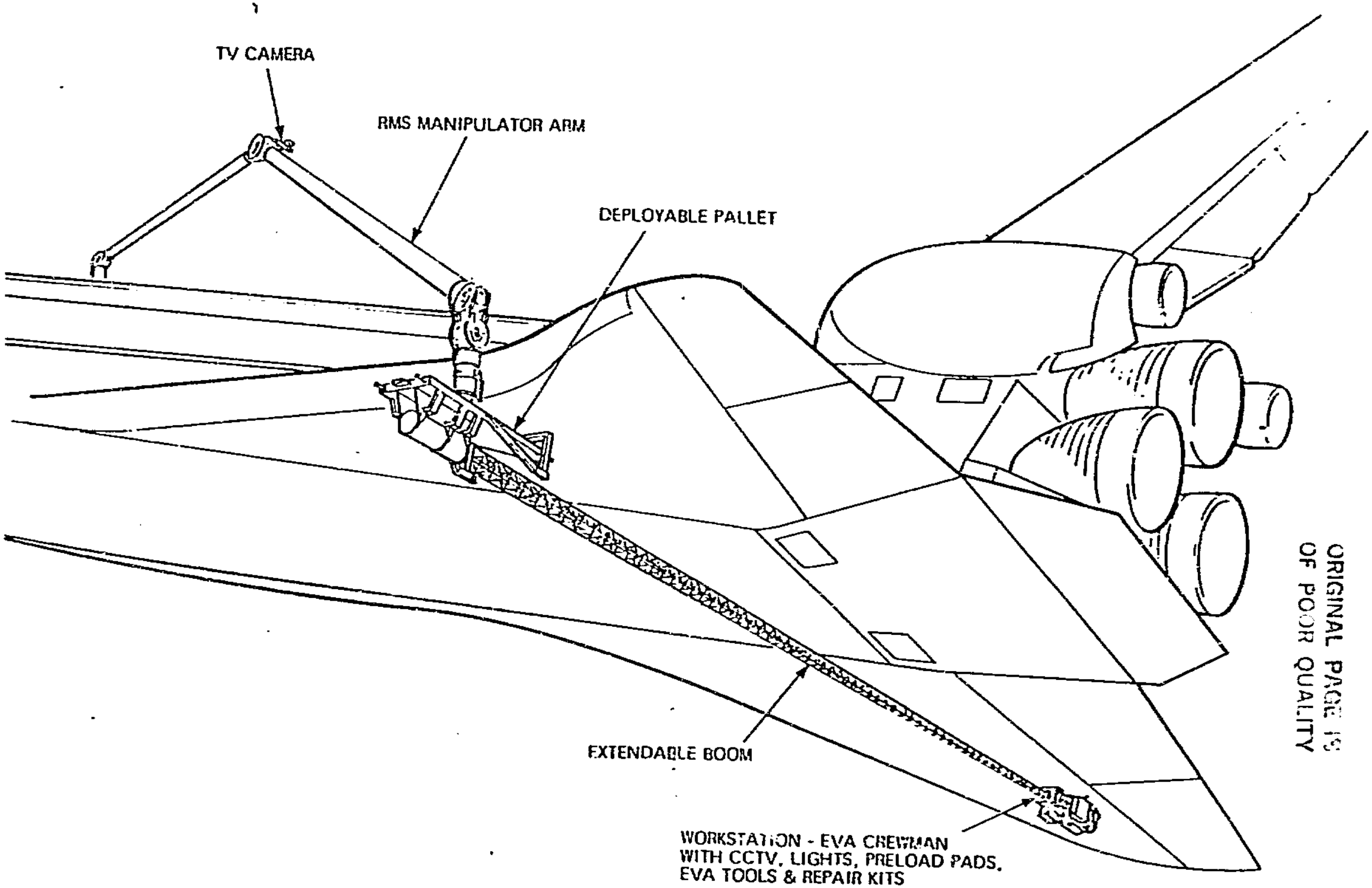

FIG. 3-13: THE SPACE WORM

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4.0 SPACE - MAINTAINABILITY CONSIDERATIONS FOR RMS

The Shuttle Remote Manipulator System hardware is divided into Line Replaceable Units (LRU's) and Shop Replaceable Units (SRU's). The LRU's are hardware assemblies that can be removed from, and replaced on, the orbiter during ground turnaround operations. The SRU's are lower level assemblies that are to be removed from, and replaced on, an LRU in a controlled environment integration area ("shop"). Figure 4-1 identifies the LRU's in the RMS and the SRU's of the mechanical arm assembly.

Five of the LRU's grouped together to form the manipulator arm, are:

(a) mechanical arm assembly consisting of the six joints and structural elements connecting them,

(b) standard end effector,

(c) arm thermal protection system consisting of multi-layer insulation blankets externally applied to the arm and to the standard end effector,

(d) the wrist CCTV and viewing light subassembly which consists of the following SRU's:

- i) CCTV camera and monochrome lens assembly,
- ii) viewing light and cables,
- iii) RMS to camera cable assembly,
- iv) camera/light mounting bracket.

(e) the elbow CCTV subassembly.

The remaining four LRU's, housed in the aft crew station in the orbiter, are:

(a) Manipulator Controller Interface Unit (MCIU),
(b) Display and Controls (D&C) panel,
(c) Translational Hand Controller (THC),
(d) Rotational Hand Controller (RHC).

The present categorization of hardware could be maintained for an RMS for the Space Station. The manipulator arm LRU's would be handled by EVA. Two or more astronauts may be needed to remove or install LRU's with appropriate handling aids, tethers and tools. Some of the interfaces may have to be redesigned to enable mechanical connections (bolts, clamps, etc.) and electrical connections to be make quickly by an astronaut in a space suit with suitable tools. At the SRMS-orbiter interface (Shoulder to MPM), the mechanical attachment is with 16, 0.25" bolts and washers and two dowel pins, located on the periphery of a rectangular interface 7.2" x 8.86". The electrical connections are through 9 plug-in connectors located over a rectangular interface 7" x 8" (approximate).

The LRU's could be brought into an enclosed, pressurized service bay in the Space Station to act as a work-area where LRU's could be serviced and SRU's replaced. Some design changes may again be required to modify physical interfaces for zero-g assembly and disassembly with appropriate tools and handling aids. Clearly, handling aids, tools and test and integration equipment for zero-g in-orbit servicing would have to be developed.

In-orbit fault detection and isolation procedures would also have to be developed to support the maintenance activities. Some built-in test capability to detect and report failures in the prime operating channel and to isolate faults to the LRU level has been designed into the shuttle RMS. A detailed review of this, along with the development of tests to isolate faults to the SRU level (or lower), would be necessary to support an on-orbit maintenance plan. Trade-off's involving on-orbit servicing versus ground servicing also would have to be carried out.

The shuttle RMS has been designed to provide a minimum operating life of 100 mission cycles. One mission cycle is defined as a launch, five operations of the RMS (from deployed position, through release or retrieval operation of a payload and return to the original deployed position), re-entry into the earth atmosphere, and landing. The operating scenarios on a space station would be different from those on the shuttle orbiter. An "operating life model" of the RMS would have to be developed to track the "remaining life" of components as the RMS operations progress, in order to identify the planned maintenance periods of the system.

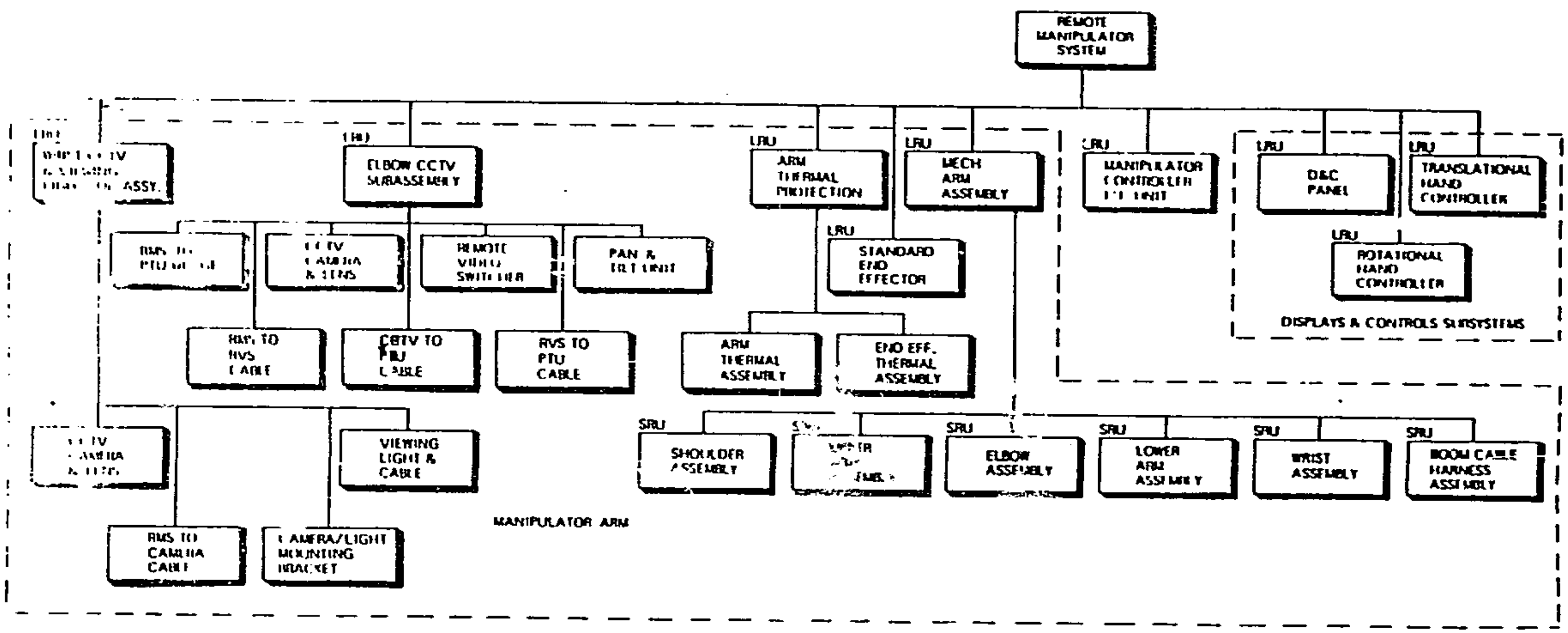

FIG. 4-1 IDENTIFICATION OF LRU's & SRU's WITHIN RMS

5.0 MANIPULATOR DEVELOPMENT ISSUES: SIMULATION AND ANALYSIS

The shuttle RMS is a complex man-machine system involving non-linear dynamics and control, interaction between structural flexibility and control system, and is designed for man-in-the-loop operation in space. The process of design and development involved extensive analysis and simulation activities to:

(a) Support decision making in selecting design options at system, subsystem and component level.

(b) Verify the performance of design configurations.

(c) Develop high-fidelity simulations of the system to be used for determining system behaviour under different conditions, and for training operators of the system.

The analyses and simulations included:

(a) mathematical models of the arm structure and joints to study their stiffness and dynamics,

(b) mathematical models of the joints and their servo control systems,

(c) simulations of a single joint arm,

(d) high fidelity non-real time simulation of the system (ASAD),

(e) real-time simulation facility with man-in-the-loop (SIMFAC).

The shuttle RMS has now become operational and the two simulations of the system, ASAD and SIMFAC, are available at Spar to develop and verify different operational procedures for the system. These simulations are described below.

5.1 ASAD

The non-real-time digital simulation, called ASAD, is a computer program which uses operator control inputs and commands, and produces detailed dynamic outputs of the system including trajectories of the arm, position rate and acceleration of the joints and the arm, and loads in the arm structure and joints. The program is modular in design, the principal modules being the SRMS control algorithms, joint gearbox and servo, arm dynamics (flexibility) and Orbiter Attitude Control System (ACS). Portions of RMS operational scenarios, for example, the tasks of payload capture, berthing and release, can be studied in detail by using hand controller inputs during these tasks and system initial conditions, as obtained from a real-time simulation such as SIMFAC.

ASAD runs on a CDC computer system - CDC 6600 - and operates typically at 30 times real time. It is the most versatile detailed and accurate model of the shuttle RMS system currently available. It is the master simulation validated by flight test results which is used for validation of other simulations of the RMS.

5.2 SIMFAC

The real-time simulation facility, SIMFAC, provides realistic man-in-the-loop operation of the RMS in a flight-like crew station patterned after the orbiter aft flight deck. SIMFAC has the following four major subsystems, as shown in Figure 5-1.

(a) simulation subsystem,
(b) scene generation subsystem,
(c) operator complex,
(d) master control complex.

The simulation subsystem (Figure 5-2) is a self-contained computer complex for manipulator mathematical models, control algorithms, servo software modules and data update to the scene generation subsystem. The simulation subsystem complex has two TI-980B minicomputers, each with an associated floating point processor, used in master-slave configuration. The computers are augmented by an array processor in which the matrix calculations are performed.

Software in the simulation subsystem is divided into three categories:

(a) operating system software,

(b) applications software, including the orbiter model and the RMS model. The orbiter model includes mass properties and rotational dynamics. A number of payload model options have been configured including the maximum envelope payload. The RMS model includes its mass and stiffness distribution, non-linear dynamics equations including the first six modes of arm vibration, servo control systems of the six joints, and control algorithms.

(c) service software, to process all display and control inputs and outputs. It also communicates with the applications software via a data base.

The scene generation subsystem, shown in Figure 5-3, is a three computer complex augmented by an array processor, which receives updated data from the simulation subsystem, and produces the four visual images presented to the operator. Two identical IDIIOM graphics generators are employed, each generating two of the images. One produces direct (window view) images and the other produces CCTV images. The graphics systems are vector type. Up to 2,000 vectors per graphics system can be generated, of which 1,300 are dynamic.

The operator complex is an enclosed flight-like crew station including the man-machine interface, i.e.

(a) display and controls panel,
(b) hand controllers,
(c) CRT monitors and keyboard,
(d) CCTV camera controls,
(e) four TV monitors providing window and CCTV views.

The direct view perspective is adjusted to a specified viewpoint in the operator complex and a lens system effectively places the images at infinity, thereby creating the illusion of depth to the operator.

The Master Control Complex includes work stations for the Test Conductor and Systems Engineer. It provides full interactive control and monitoring of tests and communication with the operator complex. Tasks may be frozen and restarted, or, if desired, re-initialized. The test conductor can monitor engineering data in a CRT display system with a hard copy of the display, if desired. Simulation malfunctions may be inserted and/or cleared by the test conductor.

5.3 Simulations and Analysis for Space Station Manipulators

Simulation and analysis would be needed for development of manipulators for a space station in the following phases:

(a) Requirements Definition

Analysis and simulation could be used to support space station manipulator operations analysis and manipulator tasks. SIMFAC and ASAD, with appropriate changes, could be used to study shuttle RMS type manipulators and their operations for anticipated tasks. Such activities would lead to definition of requirements for space station manipulators.

(b) Design and Development

The analysis and simulation effort needed would depend on the manipulator concept selected to meet the requirements. It may be possible to use ASAD and SIMFAC with small modifications, or new simulations may be developed based on the experience and expertise gained in the shuttle RMS simulations.

(c) Verification and Training

The role of simulations to verify operational scenarios and to train the operators would be similar to those for the shuttle. Some training may be conducted in orbit on the space station.

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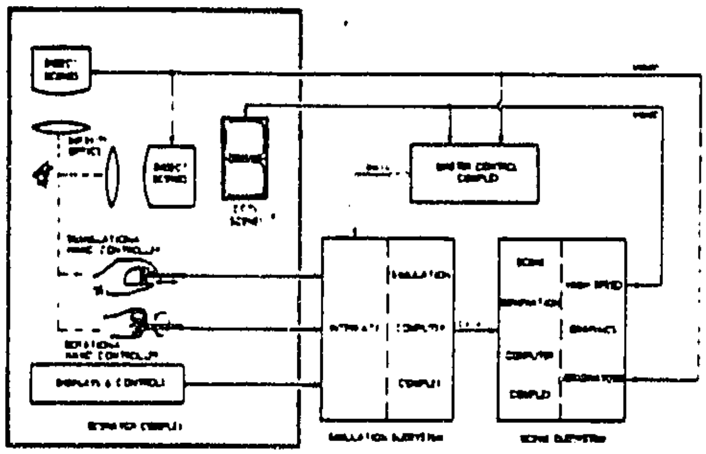

FIG. 5-1 SIMFAC BLOCK DIAGRAM

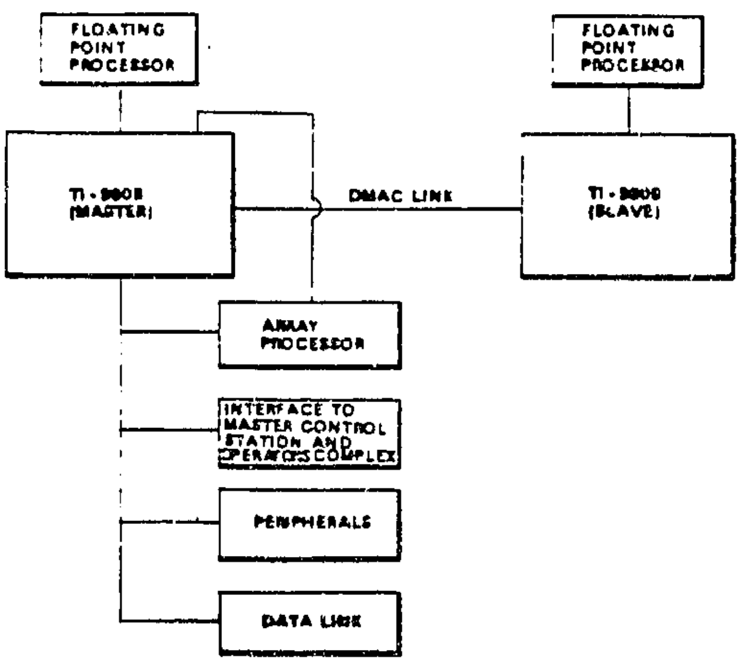

FIG. 5-2 SIMULATION SUBSYSTEM BLOCK DIAGRAM

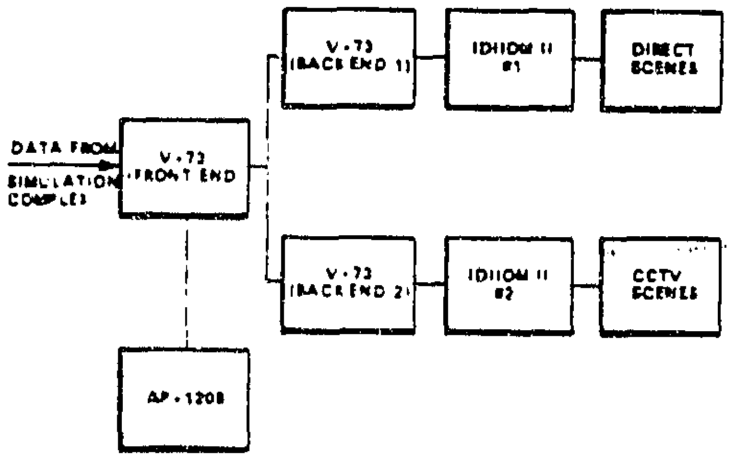

FIG. 5-3 SCENE GENERATION SUBSYSTEM BLOCK DIAGRAM

6.0 RMS TRACK AND BASE ASSEMBLY CONCEPT FOR A SPACE STATION

Boeing Space Station concepts at present have a shuttle RMS type manipulator(s) mounted on a linear track. This increases the operating envelope of the manipulator. Payload handling operations would involve translation of the RMS, carrying a payload, along the tracks.

Such a system would require a track system, a drive system and an interface structure between the RMS and the track/drive assembly. The main functional requirements for the system would be:

(a) Provide a stiff structural system between the RMS base and the space station structure. The stiffness should be at least as high as that provided by the orbiter Manipulator Positioning Mechanism (MPM) for the shuttle RMS. The stiffness requirement may be higher if the manipulator is longer than the SRMS manipulator.

(b) The track system should have no backlash. Backlash would appear as a "deadband" at the arm tip and would degrade the operator's ability to position payloads precisely.

(c) The track system should provide a low stiction/friction ratio system i.e. variations in friction over the speed range of operation, including start from zero speed, should be small. Variations in friction level cause degradation in positioning capability and tend to produce "jittery" motion.

(d) Thermal distortions of the track system should be minimized since they would produce large friction forces, as well as cause positioning errors at the manipulator tip.

(e) The track, base structure and the drive should be designed for repeated applications of mechanical loads. Thus, fatigue, fracture and wear should be considered in the design to provide adequate operational life for the system.

(f) The base assembly should also have adequate provisions for handling the electrical wiring harness, and to manage the length of wire bundles as the system moves along the tracks. A mechanism to feed and retract the wires as the RMS is translated along the track, would be needed.

(g) Some requirements would have to be developed for on-orbit integration of the system, as the space station concepts mature and approaches to on-orbit integration are evaluated.

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FIG 6-1

RMS BASE & TRACK

GENERAL ARRANGEMENT

CONCEPT

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FIG 6-2 RMS BASE, TRACK AND DRIVE ASSEMBLY CONCEPT FOR A SPACE STATION

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7.0 CONCLUSIONS AND RECOMMENDATIONS

Capabilities and potential improvements in the SRMS have been discussed in this report. It appears that requirements for a large manipulator on a space station could be met by either an advanced SRMS (with appropriate end effectors and tools) or a derivative of the SRMS design. Analysis of remote handling operations on a space station should be performed in conjunction with potential manipulator configurations (length, degrees of freedom, tip force capability, etc.) to optimize the remote handling system and its relationship with the space station physical architecture. This analysis activity would benefit from support of real-time simulation (SIMFAC) and non-real-time simulation (ASAD) developed for SRMS and available at Spar. The remote handling system studies should also consider the issue of on-orbit maintenance and repair of manipulators and assess the impact of various approaches and trade-off's for on-orbit servicing on the design of manipulators. The shuttle RMS has not been designed for on-orbit servicing.

APPENDIX A

BOEING STATEMENT OF WORK

SCHEDULE A

STATEMENT OF WORK

INTRODUCTION

This Statement of Work covers technical interchange between Spar and BAC for the Space Station Needs, Attributes, and Architectural Options Study for NASA. The planned study will emphasize mission analysis to establish mission needs, with second emphasis on system architecture, and third on cost and programmatics.

Spar will assist BAC in assessing remote handling aspects of the space station architecture work with a level-of-effort consistent with two man-weeks time available.

TECHNICAL INTERCHANGE TASKS

Task 1 - Manipulation Configuration Concepts - Review BAC's mission operations, handling equipment concepts, and space station architectural options and recommend improved and/or alternative remote handling equipment concepts. These concepts would include configuration sketches and top-level descriptive information, e.g., reach envelop, payload mass rating, control station location, etc.

Task 2 - Control Station Concepts - (a) Report on current RMS performance characteristics and system evolution; (b) discuss issues related to controls and displays located at end of manipulator, i.e., manned remote work station; (c) discuss potential application of voice-actuated control technology.

Task 3 - Manipulator Development Issues - (a) Discuss manipulator development software requirements, e.g., primary issues, development lead time, magnitude of the code, etc., (b) Discuss manipulator simulator hardware and facilities development issues, lead time, etc.

Task 4 - RMS Improvements - Describe RMS improvement potential, e.g., control/display interfaces, degrees of freedom, end effectors, etc.

7.4.8 Instrument Pointing Systems - Dornier System GMBH

DORNIER SYSTEM

Participation in NASA Space Station Study

TITLE: POINTING SYSTEMS
TITEL:

DOCUMENT NO.: TN-SSS-DS-003
DOKUMENT NR:

ISSUE NO.: 1
AUSGABE NR:

ISSUE DATE: 14.02.1983
AUSGABEDATUM:

PREPARED BY: M. Kersten
BEARBEITET: G. Lippner

COMPANY: Dornier System GmbH
FIRMA:

AGREED BY: J. Spintig
GEPRÜFT:

COMPANY: Dornier System GmbH
FIRMA:

CONTRACT NO :

PROJECT MANAGER
PROJEKTMANAGER

LIST OF CONTENTS

1. INTRODUCTION

Dornier looks back on 13 years of experience in the development and manufacture of stabilized gimbal systems, starting with balloon borne telescope pointing systems and maritime antenna stabilization systems. In the space field the following gimbal systems have been developed by Dornier:

- the Instrument Pointing Subsystem (IPS)
- the two axes antenna pointing mechanisms for the German MRSE and MRSE-MAS
- the Position and Hold Mount (PHM) covering phase A, B and demonstration model
- the Antenna Pointing Mechanism (APM).

A detailed system description for IPS, PHM and APM is given in section 2. The Space Station relevant payloads are summarized in section 3. The Space Station accommodation aspects are handled in section 4.

2. POINTING SYSTEM DESCRIPTION

2.1 Instrument Pointing Subsystem

2.1.1 General

Starting in 1976 DORNIER SYSTEM has been developing and manufacturing the Instrument Pointing Subsystem (IPS) under contract of the European Space Agency (ESA). In the years 1979/1980 the IPS has successfully passed an extensive qualification programme.

Due to numerous engineering changes concerning Spacelab and Orbiter interfaces the manufacturing programme for the flight unit had to be interrupted to allow for a redesign of critical components. The whole programme of manufacturing and requalification of the IPS flight unit will be completed by end of 1983.

The first mission of IPS is scheduled to take place in October 1984.

2.1.2 Technical Concept

The Instrument Pointing Subsystem (IPS) provides precision 3-axes 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 2.1.2-1.

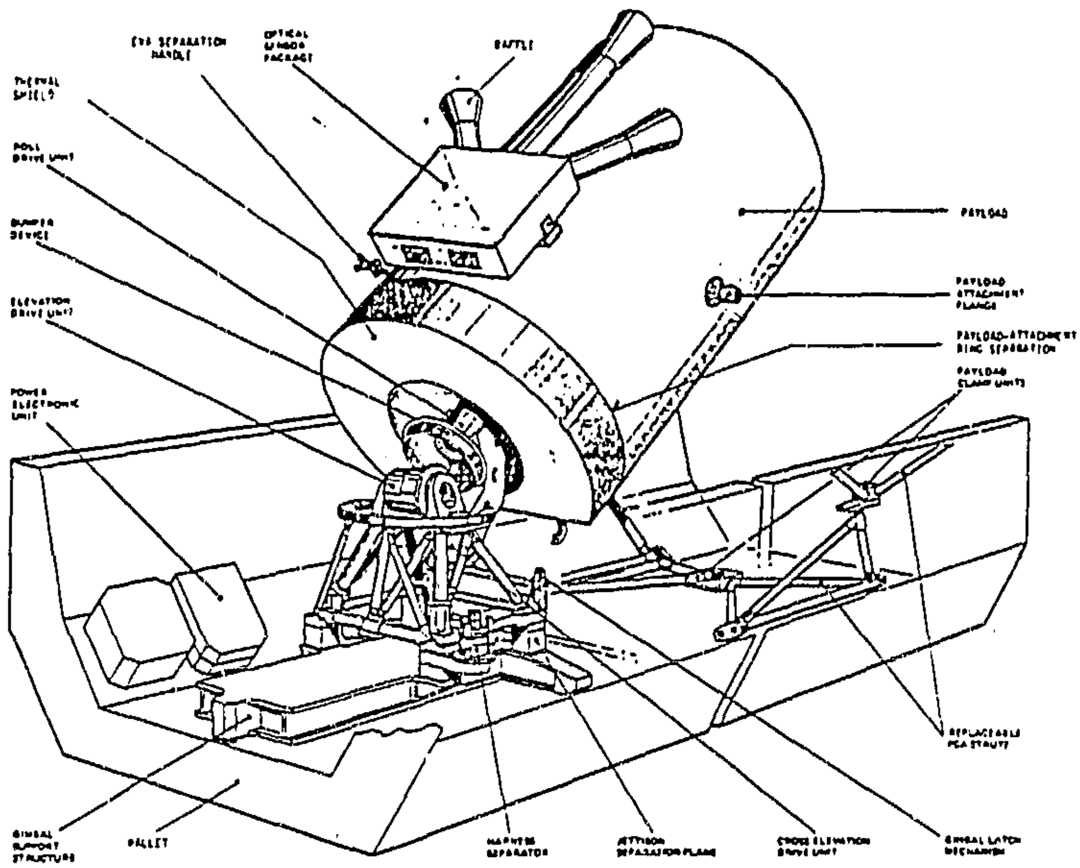

Fig. 2.1.2-1: IPS Configuration on a 2 Pallet Train

The IPS features the following main systems:

- the three axes gimbal system for precision pointing
- the Payload Clamp Assembly (PCA) to support the payload during ascent and descent
- Attitude Measurement Assembly including Optical Sensor Package and Gyro Package
- the Power Electronics Assembly (PEA)
- the Data Electronics Assembly (DEA)

For the wide range of various payloads different IPS configurations can be provided:

- Flexibility is given by potential payload dimensions from 0,5 m to 3 m diameter and up to 2 m distance from Payload CG to IPS attachment plane
- Payloads up to 3000 kg mass can be accommodated
- Single or double pallet train may be used depending on payload mass and dimensions
- The field of view is conical about the Orbiter z-axis ant its half-cone angle can be varied from 30° to 60° by adjustment of the impact ring position
- Any x position of the centre of rotation (COR) can be adapted by moving the gimbal system on the gimbal support structure rails
- The z position of the centre of rotation (COR) can be adjusted in height from 1,3 m to 2 m by means of a mission dependent replaceable column
- The payload clamp assembly can adapt the wide range of payloads by replacement of struts
- Nominal payload characteristics as used for IPS design reference are shown in Table 2.1.2-1
- The IPS electrical and mechanical data and the essential IPS capabilities for experiment accommodation are summarized in Table 2.1.2-2.

	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	1460 kgm^2	20 kgm^2
about LOS axis	1000 kgm^2	25 kgm^2
CG offset referred to P/L interface plane:		
along LOS (Note 3)	1,63 m	0,50 m
perp. to LOS	0,00 m	0,10 m
characteristic structural frequency (Note 1)	7,5 Hz	(Note 2)

Note 1: Lowest bending mode, supported at P/L interface plane

Note 2: Considered not to be critical

Note 3: The LOS is the vector through the COR, perpendicular to the P/L interface plane

Table 2.1.2-1: Characteristics of Nominal 2000 kg and 200 kg Payloads

The optical sensor package includes the capability to have two roll sensors at a skewed angle of either 45 degrees or 12 degrees with respect to the line of sight (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.

Requirement	Value	Comments
IPS Total Mass	1174 kg	For the SL-2 configuration (1.83 m COR height); without mission dependent PCA struts, including integrated SL hardware (RAU/IS)
IPS Power Consumption		
- Inertial Pointing	500 W	Mean Value
- Full Torque Slewing	1400 W	30 Nm Torque About 2 Axes
- Emergency Stowage	200 W	Short Time Peak Value
- Thermal Control	350/1200 W	Cold Case Mean/Peak Power
Payload Accommodation		
- Nominal Mass	2000 kg	
- Max. PCA Loading Mass	3000 kg	With Stiffened Clamping Struts
- Max. P/L Pointing Mass	7000 kg	not covered by actual clamping system
- Diameter	0,5 to 3 m	
- Back to CG Offset	0,5 to 2 m	3 m for 7000 kg P/L
Payload Support Power		
- Main Power	1250 W/22 VDC	8 x 6 AWG 20 independently fused
- Essential Power	100 W	4 AWG 20
Payload Date Lines for		
- Experiment RAU's	3	Data and Power Busses
- High Data Rate	6	6 TSP 125 ohm for 16 Mb/s
- Analog Signals	3	3 TSP 75 ohm for 4,5 MHz
- Control	10	for discrete signals and commands

Table 2.1.2-2: IPS Physical Characteristics and Payload Accommodations

2.1.2.1 Mechanical Concept

The IPS comprises two major mechanical assemblies:

- the three axes Gimbal Structure Assembly
- the Payload Clamp Assembly (PCA)

2.1.2.1.1 Gimbal System

During operation on orbit the payload is attached to the Gimbal System and its three axis attitude and stability control is performed by torquers applied by the three identical drive assemblies. Their axes intersect at one point. Each drive unit employs 4 ball bearings, two brushless DC-torquers, and two single speed/multi speed resolvers for nominal and emergency operations.

The electronic units of the IPS are mounted on the Equipment Platform, except the power electronics unit which is mounted to a cold plate on the Pallet.

During launch and landing the gimbal system and the payload are separated, so that no additional loads or moments will be imposed on the payload by the gimbal structure.

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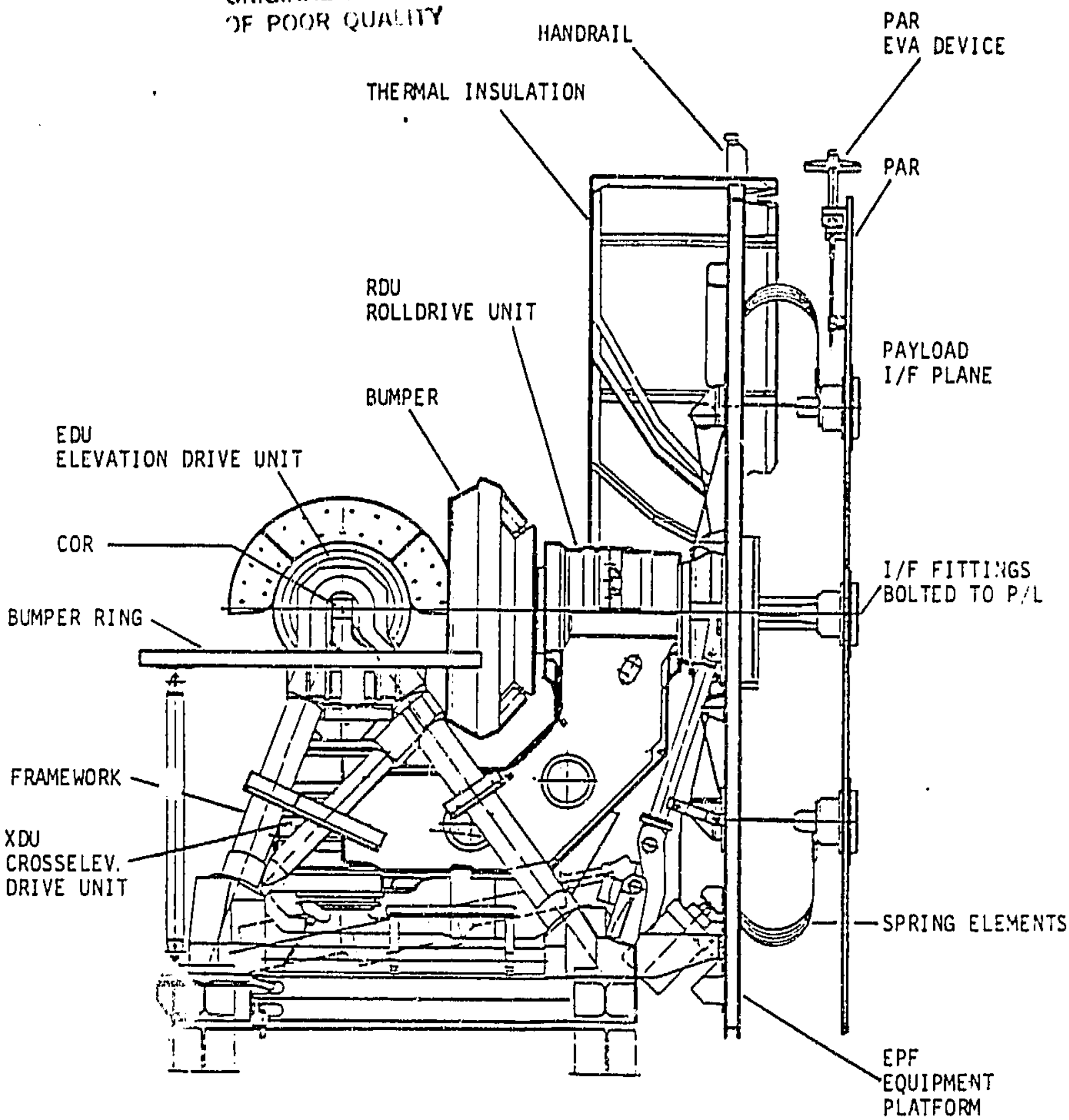

FIG. 2.1.2.1.1-1: GIMBAL STRUCTURE

The gimbal support structure, mounted at 4 hardpoints to the pallet, includes rails with brackets for positioning the gimbal system in x direction. For adjustment in x and y direction the brackets include a screwing device.

A replaceable extension column between the Gimbal Support Structure (GSS) and the jettisoning device at the Gimbal Structure can adapt the height of the COR between 1,3 m to 2 m.

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 and overboard jettison is required.

The payload and the integrated gimbal structure will be installed separately onto the pallet and then be connected to each other via three mounting flanges on the PAR. During the ascent/descent phase the upper gimbal structure is locked in its adjusted location by the Gimbal Latch Mechanism (GLM).

After release of the PCA clamping devices on orbit a mechanism will move the Payload Attachment Ring with the payload towards the EPF, and clamp the Payload onto the Gimbal Structure.

In an emergency case a PAR mounted EVA device is dedicated to separate the P/L, connected to the PAR, from the Gimbal Structure for individual jettison of either part.

2.1.2.1.2 Payload Clamp Assembly

During ascent and descent, the payload is physically separated from the Gimbal Structure to avoid any load path. The payload is supported by the Payload Clamp Assembly (PCA) which distributes the flight loads of the payload into the pallet hardpoints (Figure 2.1.2.1.2-1). The PCA is designed such that the directions of the loads induced in the payload are predominantly tangential.

The Payload Clamp Assembly consists of

- clamping mechanism, i.e. three clamping units (CU) defining a triangle in the $Y_o - Z_o$ plane and an actuator mechanism with replaceable flexible shafts to drive the CU's
- replaceable struts distributing the loads from each clamping unit to four pallet hardpoints. These struts will be tailored to each individual mission configuration and thus determine the size of the triangle mentioned above to accommodate payloads between 0,5 m to 3 m diameter.
- non-replaceable elements distributing the loads from the replaceable struts to the pallet hardpoints.
- for an emergency case an EVA device is provided to enable the removal of the keybolts and so release the payload.

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 3 m 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.

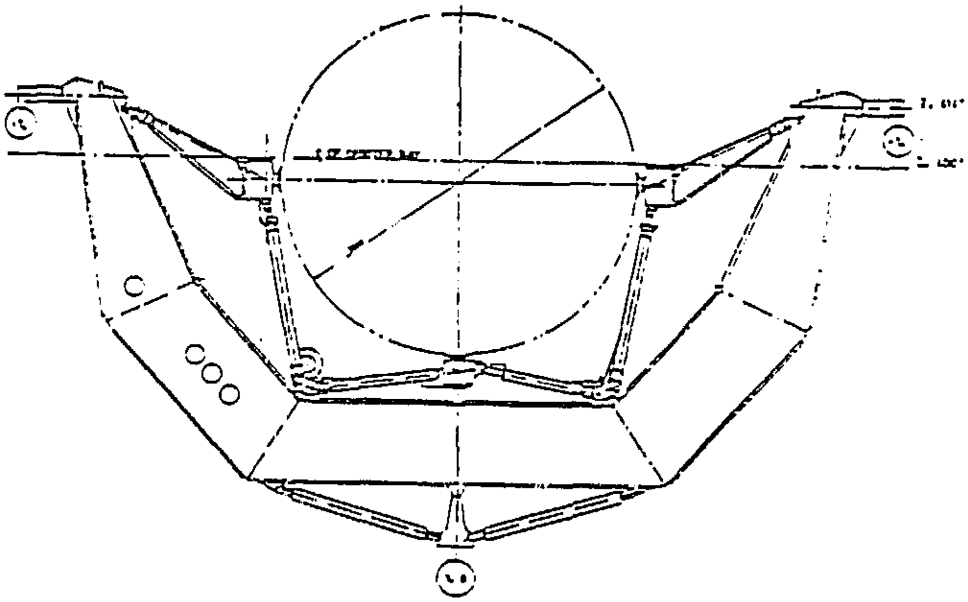

Figure 2.1.2.1.2-1: IPS PAYLOAD CLAMP ASSEMBLY

2.1.2.2 Thermal Concept

During orbital operations the IPS is capable of operating in the

- "cold case", that means in a completely shadowed configuration (worst case) for indefinite time
- "hot case", that means in a continuous full sun illumination (worst case) for a minimum time of 195 minutes when starting with the status after 9 hours cool-down. The time period for cooling off after maximum solar exposure - IPS stowed and units switched off - is 9 hours maximum. For not as extreme environmental conditions, the operational time of IPS is unlimited or can be enlarged. For hot cases TCA puts the following constraints on payload operation:
  - o The total solar and infrared radiation energy which is absorbed by the OSP radiator may not exceed 1.3 kWh during one hot operational phase. This could mean a roll angle restriction.
  - o During non-operation phases, payload and IPS shall remain attached. The maximum time with separated payload (stowed in PCU) shall not exceed 2 hours in sun phase (hot case).

For other not as extreme orbital conditions than the specified design cases of IPS TCA, the restrictions may be reduced or inapplicable depending upon the results of the mission dependent thermal analysis.

The TCA consists of the following components:

- Multilayer Insulation (common with Spacelab module)
- Radiators covered with second surface mirror foils
- White paint for not insulated external surfaces
- Black paint and tapes for trimming of radiative heat transfer
- Interface fillers for improvement of contact conductance
- Heater mats for heat leak compensation
- Thermistors for heater control and temperature monitoring
- TCA software loaded in CDMS for thermistor signal transformation, heater switching, temperature out-of-limit control and warning.

The 37 thermistors and the 29 heater loops are conditioned by the PEU and the DCU via the S/S-RAU and the IPS-RAU.
The heaters are switched on/off if the temperature is lower/higher than a pre-selected switching limit.

There are three different thermally relevant operation modes:

a) Operational Mode

The operational/in-calibration limits of all IPS units are automatically controlled by heater logic HL-A.

b) Stand-by Mode

The switch-on temperatures of not operated units are automatically controlled by heater logic HL-B (IPS stowed).

c) Non-Operating Mode

IPS is stowed and all units switched off. Every 30 minutes all temperatures have to be checked by HL-B. For temperatures colder than the switch-on limits the stand-by mode must be initiated. If max. operating limits are exceeded, the Orbiter must be turned to a cold attitude.

In order to achieve an optimized cool-down of IPS after a hot mission phase certain switch-off/on sequences of IPS units are to be performed which also provide operational conditions at the end of the cool down phase.

Ascent Constraints

- Pre-launch temperatures of IPS shall be not higher than 30°C when starting into a long duration worst hot ascent
- IPS main power shall be available at least 1 hour after opening of cargo bay doors
- After a worst hot ascent, the Orbiter shall not remain more than 1.5 hours in the worst hot attitude (Z solar inertial, full sun orbit) after cargo bay door opening.

Descent Constraints

The thermal conditions of IPS components which are to be realized by pre-descent orbital conditions before a descent is initiated, shall be evaluated by the mission dependant analysis. However as a minimum the IPS design enables a descent to be made with initial temperatures (when main power is switched off one hour before cargo bay door closure):

a) which are higher than those occuring after being 1.5 hours in the hot case transient (hot descent)

b) which are below those occuring after being 2 hours in the cool down period (cold descent).

An emergency descent can be initiated with steady state cold case and end of hot case transient temperatures without causing a failure which would lead to loss of personnel or damage of Spacelab or Orbiter.

If the temperatures of the pallets and the Orbiter radiators are in the range of -20°C to +50°C at cargo bay door closure, no temperature problems exist for IPS components.

2.1.2.3 Electrical Concept

The IPS electrical concept is determined by the extensive interface to the SL Command and Data Management Subsystem (CDMS) and to the Electrical Power Distribution Subsystem (EPDS). A blockdiagram of the IPS and its interfaces to SL is shown in Figure 2.1.2.3-1.

Clearly indicated are the main IPS electronic systems:

- the Power Electronic Unit (PEU) connected to the EPDS and
- the Data Control Unit (DCU) connected to the CDMS.

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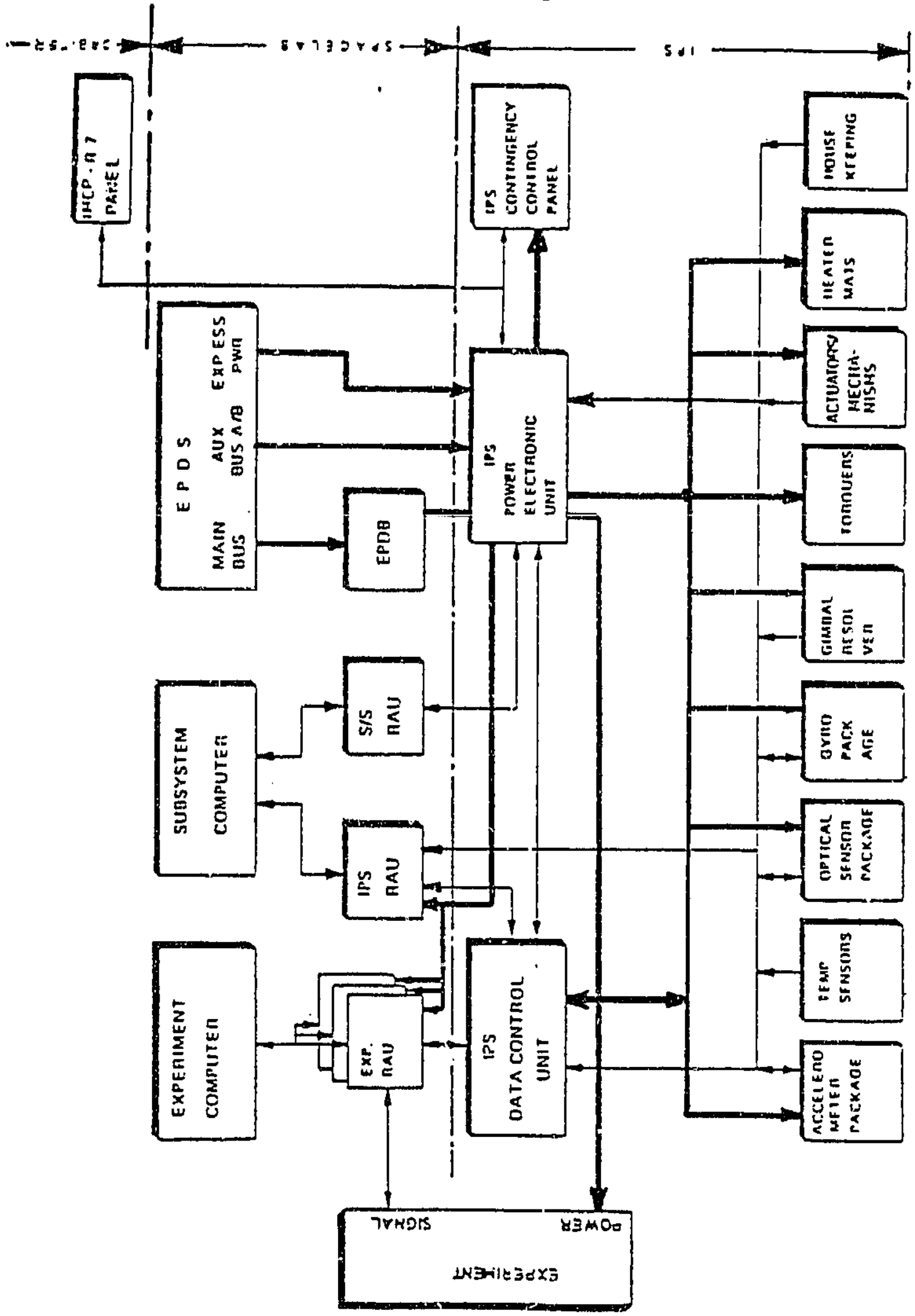

FIG. 2.1.2.3-1: IPS ELECTRICAL BLOCK DIAGRAM

2.1.2.3.1 IPS Power Electronic Unit

The IPS Power Electronic Unit (PEU) receives its primary power from the Spacelab Electrical Power and Distribution Subsystem (EPDS) by the following DC busses:

- Main DC Power from the Spacelab Electrical Power Distribution Box (EPDB) by three independent busses:
  - o DC1 Power Bus for IPS experiments
  - o DC2 Power Bus for IPS and RAU's
  - o DC3 Power Bus for heaters, RAU's and IPS OSP
- Auxiliary Power for contingency operation of IPS (retraction of IPS payload or activation of jettison):
  - o Auxiliary Power Bus A
  - o Auxiliary Power Bus B
- Experiment Essential Power

  from Spacelab Emergency Bos as a power source for experiments redundant to the experiment main power bus DC1.

The PEU provides the distribution and fusing of Spacelab power to the IPS electrical and electronic equipment, to the IPS or payload mounted Spacelab equipment (RAU's) and to the IPS payload.

Functionally the PEU is accomplishing the following tasks (see Electrical Blockdiagram, Fig. 2.1.2.3-1):

Unregulated DC main or essential power will be supplied to IPS experiments, unregulated main power to RAU's, Optical Sensor Package (OSP), Gyro Package (GP), IPS Data Control Unit (DCU).

Regulated DC power will be supplied to Accelerometer Package (ACP) and Jettison.

DC power at limited voltage and current at both polarities will be supplied to the actuator motors.

Controlled power will be supplied to torque motors and heater mats.

In the nominal operation mode the PEU receives control signals from the DCU to operate the torque motors.

For contingency operation of IPS during loss of main power or during nonoperation of CDMS the retraction of the IPS payload will be initiated by the IPS Contingency Control Panel (CCP) which is mounted in the Orbiter Aft Flight Deck. In this case the retraction circuitry for the torque motors is powered by the Spacelab Auxiliary Bus and controlled by the PEU internal "stowage loop".

In case that a safe landing of the Orbiter may be prevented by any failure of IPS, its separation from the orbiter is feasible from a separate section of the CCP, after the jettison function is enabled by a switch on the IMCP-R7 panel in the Orbiter Aft Flight Deck.

During deployment or retraction the PEU, controlled by the CDMS or the IPS CCP is driving the IPS mechanisms PCM, GLM and PGSM.

Furthermore the PEU is supporting the thermal control by heater mat switching initiated from the CDMS or by conditioning of the thermistor signals.

The power data given in Table 2.1.2.3-2 apply for the different power buses available at the S/L interface.

Spacelab Power Bus	IPS Power Consumer	Voltage Range [Volts]	Power Available at S/L Interface Mean [Watt]	Power Available at S/L Interface Peak [Watt]
Main DC Power				
- DC1 Power Bus	IPS Experiments	24,0 - 32 (1)	-	1440 (1)
- DC2 Power Bus	IPS Electronics IPS Heater IPS RAU	23,5 - 32 (2)	- 500	1410 (2) 1400
- DC3 Power Bus	EXP RAU's		350	1200
Experiment Essential Power	IPS Experiment	21,5 - 32	-	100
Auxiliary Power	Jettison & IPS Set 1 Electronics	24,7 - 32	-	(3)

Note:

(1) IPS on pallet 1 through 3

(2) IPS on pallet 4 and 5

(3) 8 AMPS MAX continuous, after Spacelab and payload have been configured to the low power mode

Table 2.1.2.3-2: IPS Power Budget

2.1.2.3.2 IPS Data Control Unit

The IPS interface to the Spacelab CDMS is provided by the IPS Data Control Unit (DCU). The DCU is interfacing with the Spacelab Subsystem Computer (SSC) via the IPS RAU and corresponds with the Spacelab Experiment Computer (EXC) via the payload mounted EXP RAU1. This DCU serial data link provides the data exchange capability between the IPS experiments and IPS. The DCU controls the IPS data and command flow and processes the fast loop portion of the IPS pointing control loop by means of a minicomputer.

Controlled by the CDMS the DCU provides thermal control of IPS.

2.1.2.4 Software Concept

IPS software utilizes two processing units:

- the Spacelab Subsystem Computer (SSC) and
- the DCU "mini-processor"

The overall control of the IPS is exercised from the SL Data Display System (DDS) and subsystem computer via IPS application software except for the emergency stowage and jettison functions. The latter functions are commanded from the IPS contingency control panel (CCP) in the Orbiter Aft Flight Deck (AFD) and require no software.

Stability control of the payload in all 3 axes is processed within the IPS mini-processor "Data Control Unit" (DCU), based on error signals of rate integrating gyros (feedback-control) and an Accelerometer Package (feedforward-control).

The processing of pointing commands is performed in the SSC and the resulting desired attitudes and rates are inputs to the DCU.

Drift and attitude correction by means of real star/sun measurements with the aid of the optical sensor package containing 3 fixed head star/sun trackers is processed in the subsystem computer of the SL CDMS.

Furthermore it is possible to accept either attitude offset commands or replace the function of the boresighted star/sun tracker by an experiment sensor error signal via the SL experiment computer to the IPS DCU. However, the control (crew I/F) of these EXC functions is performed via the SSC.

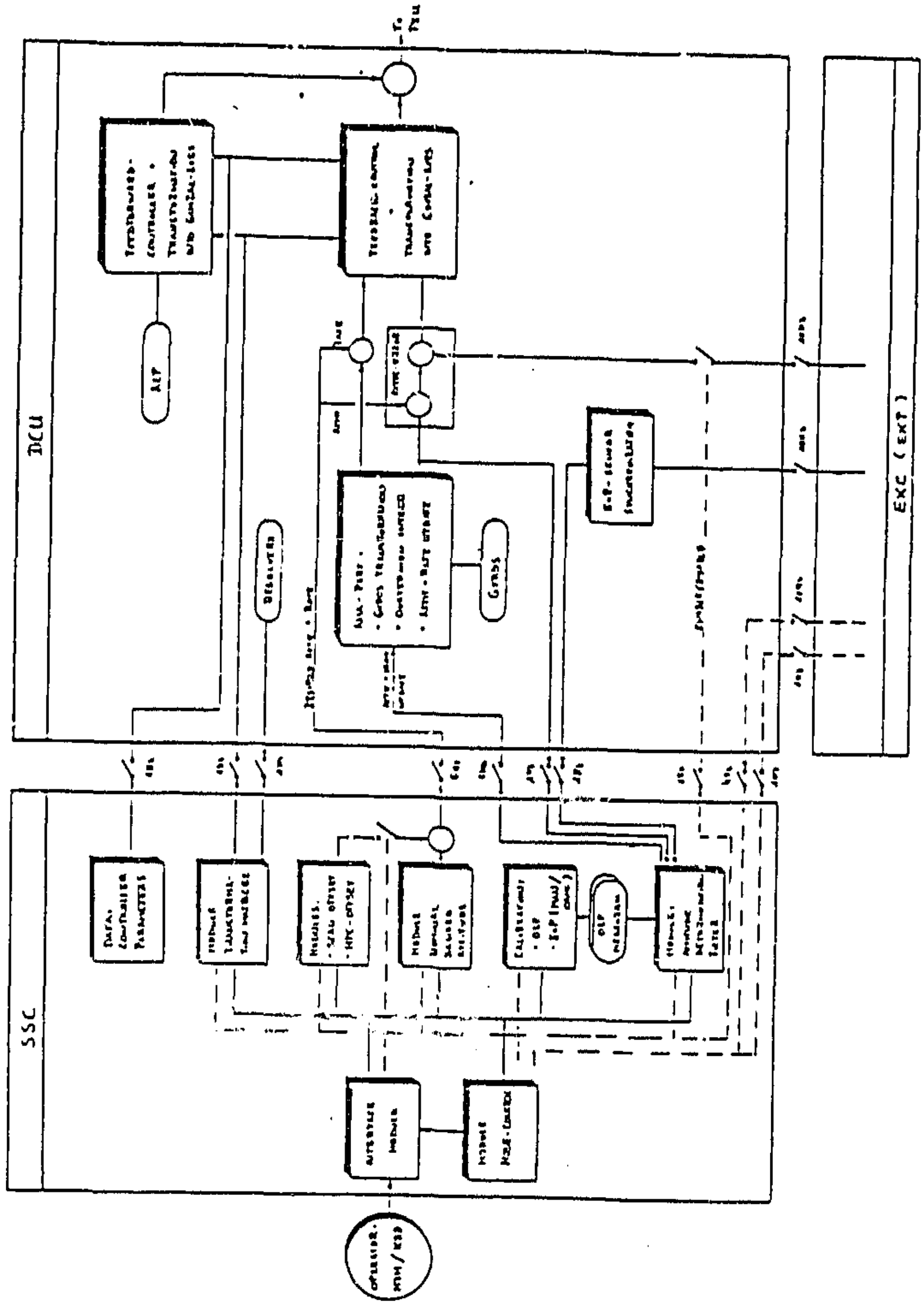

FIG. 2.1.2.4-1: IPS CONTROL LOOP BLOCK DIAGRAM

All these input data are processed in the DCU to give three control signals (one for each gimbal axis) and are sent to the Power Electronic Unit (PEU) which drives the DC torquers in the three gimbal axes.

The partitioning between SSC and DCU software is shown in the blockdiagram of Fig. 2.1.2.4-1.

The IPS related SW can be divided into two major components:

- Fast Loop SW

  residing in the DCU and performing the basic calculation of the inner control loop of the IPS for inertial pointing.

- Slow Loop SW

  residing mainly in the SSC except for the experiment-generated sensor data (replacing OSP BS data) and offset commands, which are input from the Experiment Computer (EXC).

The Slow Loop SW generates all mode option dependent desired and/or corrective attitude data as input to the Fast Loop SW in the DCU.

Fig. 2.1.2.4-2 shows an overview of the IPS SW environment and the data interfaces.

In the scope of this paper the right branch of Fig. 2.1.2.4-2, i.e., the SSC SW-DCU control loop is of main interest.

The EXC slow loop interface with the DCU is controlled via the SSC by enabling the EXC control cmd inputs in the DCU. For the operational functions which employ this interface (EXP CTL and

C-2

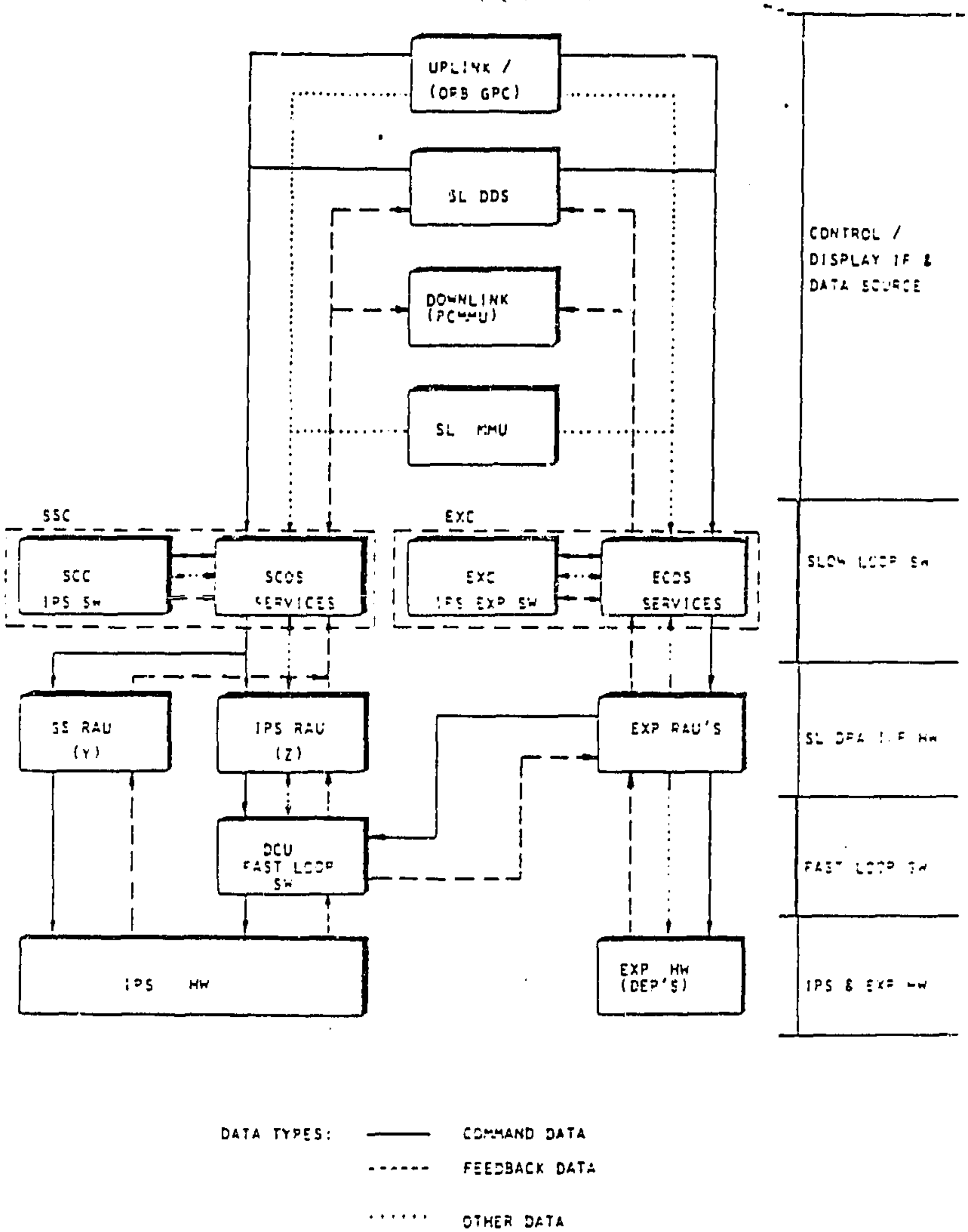

FIG. 2.1.2.4-2: IPS SOFTWARE ENVIRONMENT/DATA FLOW

EXP CMD) it is assumed, that by a TBD experiment/EXC operation the interface is configured correctly and needs only to be enabled/disabled via the SSC slow loop. Output data from the DCU to the EXC are partially enabled by activation of the DCU SW.

DCU Software

The DCU SW is subdivided mainly into two SW packages:

- the DCU Fast Loop SW
- the DCU Test SW.

The Fast Loop and Test SW run within the DCU data processor loaded and started by IPS application SW. Only one program is in the DCU memory at one time.

DCU Fast Loop SW

The main task of the DCU is to perform the calculation of the inner control loop of the IPS, done by the fast loop program. This program provides the capability of the IPS to keep the inertial attitude based on fast sensor information delivered from gyros (100 Hz) and accelerometers (50 Hz). Additionally it acquires attitude and drift signals based on 1 Hz optical sensor information processed within the SSC by the IPS Application SW. The DCU then updates every 40 ms its output to the PEU, which drives the gimbal torquers. The data between the DCU and the CDMS (i.e. SSC and EXC) are interfaced via the serial RAU data link.

DCU Test SW

The DCU Test SW executes a selftest of the DCU data processor.

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SSC IPS Application Software (IPS SW)

The IPS flight SW consists of one functional configuration (FC).

Within the flight FC, the IPS SW comprises different memory configurations (MCs):

- 1 MC for ACT/DEACT mode,
- 8 MCs for stellar mode,
- 5 MCs for solar mode,
- 1 MC for earth mode.

The structure of all IPS flight MCs is quite similar. Each MC for flight application contains:

- the always core resident task for 10 Hz communication (PCOMM-task) between SSC and DCU
- the always core resident task for temperature control (TCA), MMU-access and GIMBAL HOLD (PTGGI-task),
- one MODE-task controlling different operational options (PMXXX-task),
- one KBD/ITEM-task supporting the operators/IPS SW interface (PINXX-task),
- one OSP-dialogue task (POXX-task),

- one AMA-task (PAXXX) which supports the attitude determination (Star Identification, Attitude Determination Filter)

- one SCHEDULE task (PSCXX-task) required to allocate the Interrupt Levels and to start all tasks of a MC.

To allow for required operations, each task comprises up to 6 internal programs, each supporting a special function.

Thus each MC supports a dedicated operational function identifiable from the respective acronym as follows:

MC Name	Operational Function/Mode
ACDEAC	Activation/DEActivation mode
SLEWST	SLEWing in STellar mode
IDINST	Star IDentification/INitial in STellar mode
OSPCST	OSP Calibration in STellar mode
IDOPST	Star IDentification/OPerational in STellar mode
OHSCST	Optical Hold plus SCan in STellar mode
OHMPST	Optical Hold plus MPc in STellar mode
SCMPST	SCan plus MPc in STellar mode
OHOAST	Optical Hold plus OAms in STellar mode
SLEWSO	SLEWing in SOlar mode
IDINSO	star sun IDentification/INitial in SOlar mode
OHSCSO	Optical Hold plus SCan in SOlar mode
OHMPSO	Optical Hold plus MPc in SOlar mode
SCMPSO	SCan plus MPc in SOlar mode
SCMPEA	SCan plus MPc in EArth mode

Display Processing Concept

All data necessary to operate IPS during flight are provided on fixed format displays (FFDs).

The IPS FFDs basically utilize the SCOS display services.

The respective FFD is only operative as long as the associated MC is loaded.

Command Processing Concept

IPS utilizes the SCOS capability for keyboard inputs and related SCOS command processing to issue individual commands for IPS operations.

By uplink commands to the IPS via the Orbiter (MDM uplink), the ground has the same functional capability to control IPS that the crew has in using the keyboard.

The IPS SW is designed such that "parallel" operator inputs are accepted by SW. Every task which has to receive commands for IPS applications SW sets a request in the first run after having been started. Therefore, the IPS application SW accepts any command delivered by SCOS from the KBD (Item-, FK entry), the ground via MDM or the EXC via the DCU for execution at any point in time on the basis of a 1 Hz repetition cycle with the assumption that no 2 command inputs from the same source occur less than 1 second apart.

2.1.2.5 Operations Concept

Operational Mode Diagram

The IPS operational mode diagram in Fig. 2.1.2.5-1 gives a schematical overview, how the various IPS operations are functionally processed.

Two generic processing blocks

- Desired Attitude Processing
- AMA Processing (Attitude Measurement Assembly)

provide the inputs to the IPS dynamics which allow to perform the required pointing modes with the required accuracy.

The desired attitude processing allows to select, acquire and hold desired attitude (including offsets), which are defined in stellar, solar or earth coordinate systems or relative to the Orbiter.

Without AMA processing, the desired attitude acquisition and -hold is based on IPS gyro data only (with static, stored drift updates).

The AMA processing serves to increase the pointing accuracy by experiment dependent static calibration and/or dynamic attitude updates based on attitude data from the OSP or from experiment sensors (including OAMS).

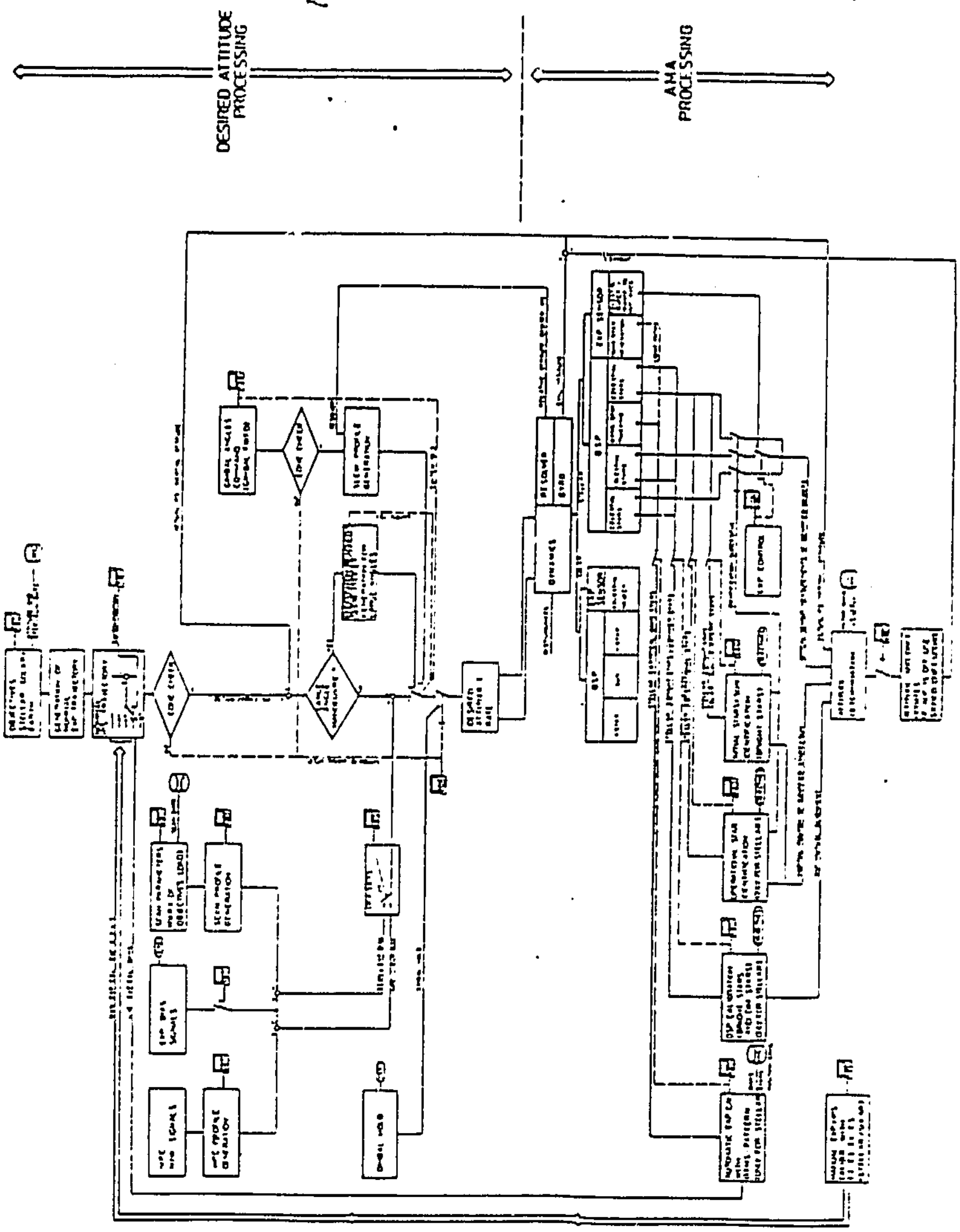

FIG. 2.1.2.5-1: OPERATIONAL MODE DIAGRAM

Groundrules for IPS On-Orbit Operations

- IPS will be inactive during the ascent/descent
  - o No normal or contingency IPS operations will be performed during these phases

- The IPS on-orbit activities are performed in the on-orbit period between end of SL activation and start of SL deactivation.

- During this phase IPS is in one of the three states:
  - o inactive
  - o operated in a normal mode
  - o operated in a contingency mode

  All flight operations to achieve and maintain the three states will be covered by related IPS sequences.

- All normal IPS operations are performed from an SL DDS, employing ITEM-commands and feedbacks on dedicated IPS FFDs in the SSC. Unlike the basic Spacelab, which is primarily operated via the Orbiter GPC CRT and Keyboard, the GPC is not involved in IPS operations.

- Although the ground (MCC/POCC) has the same command capability as the crew has on board, the ground control of IPS will be procedurally restricted to objective loads and modifications as well as start and stop of experiment dedicated IPS operations (scan, exp. command and exp. control) for solar and stellar missions. Activation/deactivation as well as earth pointing will always be performed by the crew.

IPS Operational Functions

Definition:

An operational function is an IPS operation or operational mode, which - success oriented - is always performed or established and maintained as an operational entity.

In general, an operational function is initiated and terminated procedurally by the IPS operator.

The operational functions are grouped into the five categories:

- Set-up
- Attitude Control
- Acquisition
- Offset Pointing
- Experiment Support

Table 2.1.2.5-1 identifies all IPS operational functions, the assignment to the functional categories and their applicability in the basic modes.

F1: Activation

During Activation the Orbiter- and SL subsystem resources are provided, IPS equipment is activated, the IPS and its payload are attached, unclamped and slewed into the upright position.

F2: Deactivation

During deactivation the IPS is slewed into the stowed position, IPS and its payload are separated and clamped and all IPS equipment is deactivated.

Operational Functions Overview			Basic IPS Mode				Funct. #
Category	Operational Function	Short Name	ACT/ DEACT	Stellar	Solar	Earth	
Setup	Activation	Activation	X				F1
	Deactivation	Deactivation	X				F2
	IPS Standby	IPS STBY	X				F18
	DCU Test & Dump	DCUT	X				F19
Attitude Control	Objective Load and/or Modification	OBJ LOAD/MOD		X	X	X	F5
	Attitude Command	ATT CMD		X	X	X	F6
	Gimbal Angle Command	GMBL ANG CMD	X	X	X	X	F4
	Gimbal Hold (on Orb. State Vector)	GMBL HOLD	X	X	X	X	F3
Acqui-sition	Initial Stellar Identification	ST INIT ID		X			F7
	OSP Calibration	ST OSP CAL		X			F20
	Initial Solar Identification	SO INIT ID			X		F8
	Operational Star Identification	ST OPNL ID		X			F9
	Attitude Hold (Gyros Only)	ATT HOLD		X	X	X	F12
	Optical Hold (Gyros + OSP)	OPT HOLD		X	X		F13
Offset Pointing	Scan	SCAN		X	X	X	F15
	Manual Pointing Control	MPC		X	X	X	F16
Experiment Support	Experiment Bias Command	EXP CMD		X	X	X	F17
	Experiment Control	EXP CTL		X	X		F14
	Manual Experiment Calibration	MAN EXP CAL		X	X		F10
	Experiment Calibration With OAMS	EXP CAL OAMS		X			F11

TABLE 2.1.2.5-1: IPS OPERATIONAL FUNCTIONS

F3: Gimbal Hold (GMBL Hold)

GMBL Hold commands IPS to hold the current gimbal angles relative to the Orbiter. This is accomplished by use of the Orbiter state vector (SV), superimposing the rotation of the Orbiter on top of the IPS inertial hold.

GMBL Hold is primarily a contingency option, available throughout any IPS operation when IPS is unstowed. During normal IPS operations it is employed automatically during mode-transition phases to exclude uncontrolled motions of IPS.

F4: Gimbal Angle Command (GMBL ANG CMD)

GMBL ANG CMD allows to command IPS gimbal angles relative to the Orbiter. It can be performed in any basic mode, it is always performed as large angle manoeuvre, i.e., the slew SW must be loaded, upon GMBL ANG CMD IPS acquires the commanded gimbal angles and holds these based on resolver data. In Stellar, Solar and Earth mode a plausibility check is performed which sets IPS into GMBL Hold if the cone limits would be violated. In the ACDEAC mode this check is omitted to allow stowage/erection.

F5: Objective Load and/or Modification (OBJ LOAD/MOD)

OBJ LOAD/MOD is an essential precondition for any IPS Stellar, Solar or Earth pointing. Objective data including optional scan data are loaded from MMU, modified or restored from SSC core as desired objective.

The performance is functionally similar for all three basic pointing modes.

F6: Attitude Command (ATT CMD)

ATT CMD acquires a desired objective which is defined by a preceeding OBJ LOAD/MOD operation in the Stellar, Solar, Earth (Local Vertical) or Earth (Landmark) coordinate system. During execution of ATT CMD, IPS will be moved until the current attitude equals the desired.

F7: Initial Stellar Identification (ST INIT ID)

After launch or long periods of attitude hold on Gyros only, ST INIT ID has to be performed to run the strapdown attitude determination system which realignes the inertial IPS attitude. INIT ID identifies unique bright stars only.

F8: Initial Solar Identification (SO INIT ID)

SO INIT ID identifies a solar target (sun + bright start). After the identification process the strapdown attitude determination is performed which realignes the IPS inertial attitude.

F9: Operational Star Identification (ST OPNL ID)

The operational star identification is performed in stellar mode only using a set of operational stars loaded as part of an objective load.

During the operational star identification process no attitude updates are performed i.e. IPS is in attitude hold on gyros only.

F10: Manual Experiment Calibration (MAN EXP CAL)

MAN EXP CAL allows to determine the actual alignment between the EXP LOS and the IPS LOS. Employing attitude corrections until an EXP-provided output indicates the optimum attitude, the predetermined alignment matrix (or unity matrix if not predetermined) is updated with the optimum values and will statically be used when IPS is pointing in support of the respective experiment.

F11: Experiment Calibration with OAMS (EXP CAL OAMS)

IPS SW determines and updates the alignment of one experiment versus the boresighted sensor during stellar mode operation using the experiment provided On-orbit Alignment Measurement System (OAMS). In this option the boresighted sensor continuously tracks five artifical OAMS stars. The difference between the expected (ground determined) and the actual location of the artificial stars in the BS tracker field of view is used to update the specific experiment alignment matrix dynamically.

F12: Attitude Hold (ATT HOLD)

In ATT HOLD IPS points fixed relatively to the basic coordinate system as listed below:

- in stellar mode: inertial mission true of date,
- in solar mode: solar ecliptic reference system,
- in earth mode (Local Vertical): local vertical reference system
- in earth mode (Landmark): geodetic reference system

ATT HOLD is based on Gyro data only. If provided, the last update of the drift estimation will be used statically for drift compensation.

F13: Optical Hold (Gyros plus OSP) (OPT HOLD)

High pointing accuracy and stability control are not only based on gyros only, but additionally on optical sensor package (OSP) updates which provide for a gyro drift compensation by means of an absolute attitude reference signal. Therefore, the optical hold is an attitude hold under gyro control with optical sensor update to the strapdown attitude determination system, invoked automatically or by operator command after acquisition and identification of a celestial target, OPT HOLD is provided only in stellar or solar mode.

F14: Experiment Control (EXP CTL)

IPS accepts two axis control commands derived from the experiment sensor measurements in stellar or solar mode. The experiment provided sensor replaces the measurements of the OSP boresighted sensor. Experiment data are transmitted to IPS via the Experiment Computer interface. Under experiment control IPS uses for the strapdown-filter a prelaunch determined set of filter constants, defined for each experiment.

To achieve the full IPS bias-pointing-performance capability, the experiment sensor must track the same object (star/sun) as the IPS boresighted sensor.

F15: Scan

In all three basic pointing modes IPS is capable to superimpose a scan on top of attitude - or optical hold (plus optional EXP CMD).

When scan is started, IPS scans a rectangular field according to scan parameters defined in the desired objective data.

The operator has the option to start, stop, interrupt/resume the scan or define the current position as new scan center.

F16: Manual Pointing Control (MPC)

In all three basic pointing modes IPS is capable to superimpose MPC manoeuvres on top of attitude or optical hold (plus optional EXP CMD). MPC allows the crew to manually command yaw, pitch and roll signals. Rate commands generated by the handcontroller are superimposed on rate commands from other enabled options. Inertial attitude commands are derived from the total rate command.

The crew can change the maximum MPC rate for yaw and pitch which also defines the max. roll rate via a premission defined scale factor, furthermore, the crew can select medium and low rates.

F17: Experiment Bias Command (EXP CMD)

In all three basic pointing modes IPS is capable to superimpose three axes experiment bias commands on top of attitude or optical hold (plus optional scan or MPC). These experiment data are transmitted to the DCU via the experiment computer interface. Experiment off-set commands are only accepted by the IPS SW when this option has been selected by a separate keyboard command from the SSC.

F20: Stellar OSP Calibration (ST OSP CAL)

By tracking in each tracker of the OSP two identified stars (1 bright and 1 dimmer star) simultaneously, the OSP calibration is performed. This compensates the alignment errors of the OSP skewed versus the boresighted sensor.

2.1.2.6 Safety Concept

2.1.2.6.1 General

Safety of human life has the highest priority during all operational phases of IPS. In particular, special emphasis is given to crew safety during ascent/descent and orbital operations. Therefore catastrophic events must be excluded under all circumstances to prevent:

- loss of personnel and/or
- loss of Orbiter or Spacelab.

From this the following top catastrophic hazards exist a priori for IPS:

- hitting the Orbiter or Spacelab
- inability to configure for safe return.

In consequence the IPS design has payed special attention to the safety critical functions:

- prevent hitting the Orbiter or Spacelab
- configure for safe return.

Before the occurance of an event, the hazard (i.e. potential of occurance of a hazardous situation) is latent and exists regardless of the introduction of means to reduce the probability of the event. The level of an hazard cannot be changed, the hazard can only be controlled by introducing appropriate means (safeguards).

The top catastrophic hazards and the corresponding safety critical functions for IPS are shown in the following table. To control the top catastrophic hazards, means must be available to preclude the occurance of the hazardous events. Such safeguards are shown in the table. Causes of the top catastrophic hazards are identified and the relevant controls are defined.

top catastrophic hazard	safety critical function	safeguards
A. hitting the Orbiter/SL	prevent hitting Orbiter/SL	- bumper device
B. inability to configure for safe return	configure for safe return	- stowage via CDMS - stowage via CCP - jettison - EVA
C. premature jettison	prevent premature jettison	- appropriate inhibits o RPC switch o ARM function o EXECUTE function

Control of Hazard A

For hazard A the introduction of the passive, mechanical bumper device provides adequate control for the occurance of the hazard as long as the PL is attached to IPS and provided the bumper is properly designed. In case of inadvertent operation of PGSM during pointing mode or in case of inadvertent operation of Payload Clamp Mechanism (PCM) or Gimbal Latch Mechanism (GLM) during ascent/descent the bumper device is no longer a safeguard against collision between IPS/PL and Orbiter/SL.

Control of Hazard B

In the case of hazard B a combination of various active means with a reliability less than 1 are used to control the hazard.

Besides the reliability a further argument for the introduction of several safeguards is the fact that not every safeguard is valid for all operational phases.

The standard operation to configure IPS/PL for safe return is stowage by using the Gimbal Latch Mechanism, Payload Gimbal Separation Mechanism and Payload Clamp Mechanism. For safety reasons there are 3 options to stow (Fig. 2.1.2.6.1-1):

- normal stowage (SET1 via CDMS)
- back-up stowage (SET2 via CDMS)
- contingency stowage (SET1 via CCP).

Control of Hazard C

The jettison device is introduced as a safeguard for hazard B. It generates itself the top catastrophic hazard C, named "premature jettison". To preclude the hazardous event certain design features (inhibits) are introduced, which are:

- RPC 2 switch
- ARM function
- EXECUTE funtion

Items introduced in order to close a top catastrophic hazard are per definition safety critical items. The identification of critical items will be based also on these top catastrophic events (hazards), i.e. items being involved in (or part of) functions which may result in top catastrophic events shall be considered primarily as a critical item and will be examined for its criticality.

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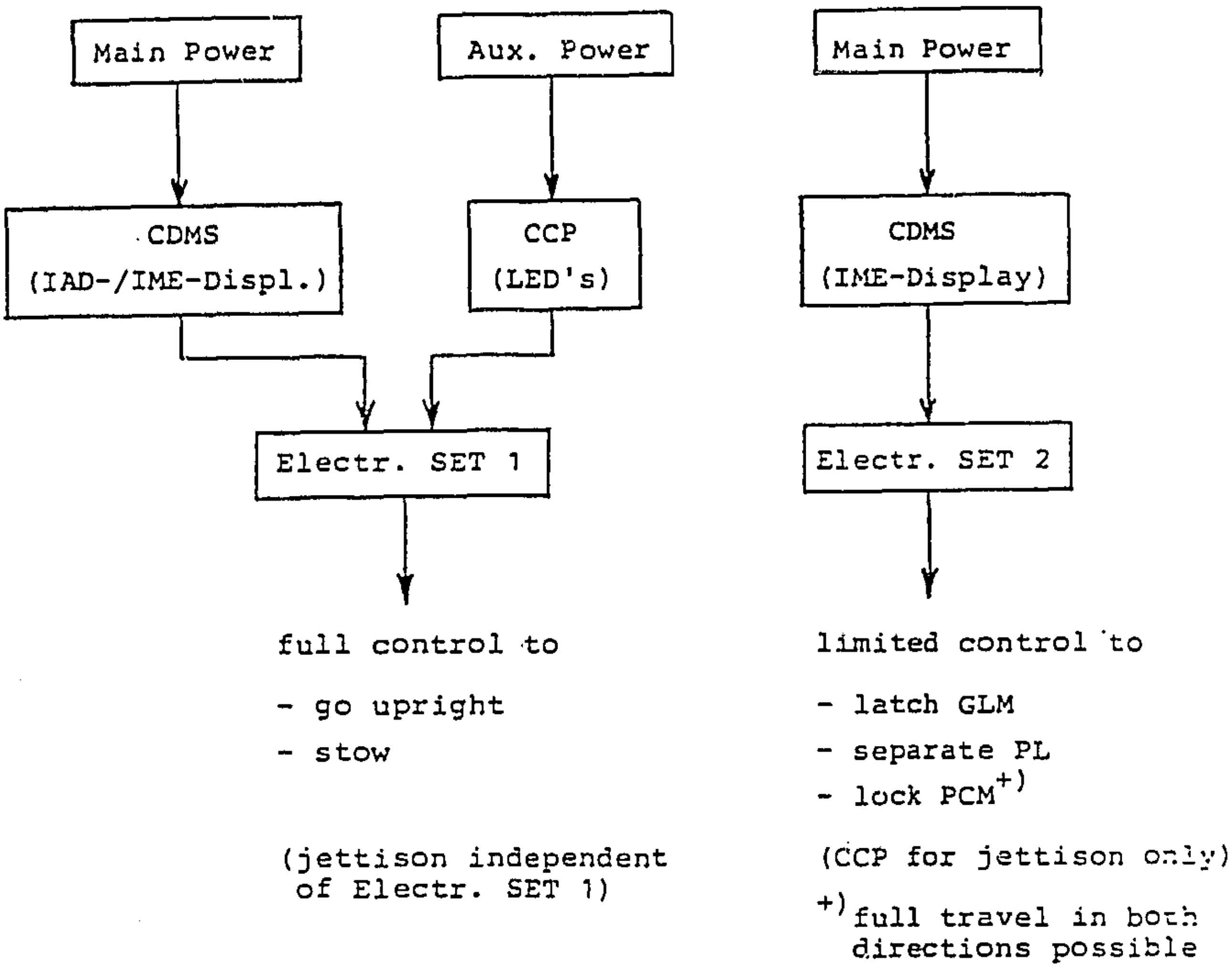

Fig. 2.1.2.6.1-1: OPTIONS FOR IPS ACTIVATION/DE-ACTIVATION

2.1.2.6.2 Safeguards for Top Catastrophic Hazards

For the redesigned IPS there is a significant safety improvement with regard to the top catastrophic hazard A: "hitting the Orbiter/SL" because of the

- addition of a bumper device which
  - o represents a passive mechanical stop for motions of IPS and its payload as long as the PL is attached to the IPS and the cargo bay doors are open
  - o is more reliable than the concept of the former design for active electronical limitation of range and rate (a failure of a LBP then resulted in an increased probability of collision with Orbiter/SPACELAB)
- elimination of two complex safety critical systems
  - o hardware range/rate limitation electronics
  - o LBP (brakes)
- elimination of items from the critical items list. By introduction of the bumper all items taking part in manoeuvres required to support the mission objective (i.e. manoeuvres) only, are no longer safety critical items. This does however not apply to those items which are required for configuration for safe return.

  It must be recognized that the bumper device itself is a safety critical item.

The safeguards for the top catastrophic hazard B: "<u>inability to configure for safe return</u>" are in the order of preference:

- back-up stowage (SET2) via CDMS
- stowage via CCP (SET1)
- jettison
- EVA

In the following these safeguards for the redesigned IPS are discussed:

- The principal features of the revised stowage concept which constitutes the primary function for hazard B are listed below:
  - o introduction of gimbal latch mechanism (GLM)
  - o elimination of load by-passes and formlocks within drive units
  - o rate limitations no longer necessary for safety reasons
  - o GLM less complex (more reliable) than brake design
  - o GLM redundant for the locking function (SET2)
  - o PGSM redundant for 'separate' (SET2)
  - o IPS during stowage in pointing mode (normal stowage) until GLM is locked
  - o redundant switches indicating locked position of EPF and GLM.

- The main features of stowage via CCP are:
  - o zero positioning axis by axis with removal of the control torque after zero position is reached in the relevant axis
  - o no locking of axes in zero position for roll and X-EL since effects introduced by disturbances (man motion, VCS firing and mass unbalanced about axes) are negligible compared to friction of bearings and CFT
  - o CCP stowage represents a back-up for normal stowage and stowage using SET2 via CDMS.

  Apart from those features no further modifications resulting from safety considerations have been introduced within the new design for contingency stowage.

- The main features of the jettison concept are:
  - o separation plane is located below the GLM.
  - o a third inhibit is introduced (RPC 2 switch on R7 panel).

  Since the LBP brakes are eliminated in the new design, a failure of the torquers may cause IPS to swing (supported by deflections at the bumper ring) for max. time of 20 min. until it stops. This constitutes a time constraint for the use of the jettison capability.

  No further modification have been introduced resulting from safety considerations into the jettison design.

- For EVA some modifications resulting from safety considerations compared to the baseline are introduced:
  - o deletion of the Drive Unit's EVA handles
  - o redesign of the EVA features of the PCA
  - o introduction of EVA Payload retention.

The control of the top catastrophic hazard "premature jettison" is provided by verification of JSC 08060 B requirements (Appendix I to Safety SR-IS-0002) and additionally by

- introduction of 3 inhibits
  (RPC 2 switch, ARM function, EXECUTE function).

2.1.3 Interfaces

2.1.3.1 Mechanical Interface

2.1.3.1.1 Interface with Orbiter

2.1.3.1.1.1 IPS Equipment located in the Orbiter Payload Bay

There is no direct mechanical interface between the Orbiter and IPS equipment located in the Orbiter Payload Bay. However, the performance of the IPS is influenced by the behaviour of the Orbiter. The Orbiter behavioural characteristics and dynamic model used to investigate the IPS performance and structural integrity are those defined in section 2.1.4.

2.1.3.1.1.2 IPS Equipment located in the Orbiter Aft Flight Deck

The only piece of IPS equipment located in the Orbiter Aft Flight Deck is the Contingency Control Panel.

2.1.3.1.1.3 Interfaces with Orbiter Tools

The interface between IPS and the Orbiter Tools for the manual opening of the jettison bolts is detailed in 20-ICD-IPS.

2.1.3.1.2 Interfaces with the Spacelab Pallet

IPS is attached to the pallet at the hardpoints and sill fittings. The IPS payload clamp assembly attachments are dependent on the payload characteristics. The IPS gimbal support structure (GSS) is attached to the pallet hardpoints, numbers 10, 12, 14 and 18.

2.1.3.2 Thermal Interface

2.1.3.2.1 Interfaces with the Orbiter

2.1.3.2.1.1 IPS Equipment located in the Orbiter Payload Bay

2.1.3.2.1.1.1 Thermal Design Configurations and Models

The Orbiter TMM used for the Spacelab Mission 2 thermal analyses is detailed in ES3-76-7.

Specular solar energy reflection from the forward Orbiter radiators is addressed in NASA TM-78270.

2.1.3.2.1.1.2 Mission Thermal Environment

The temperatures of the Orbiter and IPS elements used in and derived from the IPS baseline thermal analyses are shown in Table 2.1.3.2.1.1.2-1. These temperatures are those used in and derived from the IPS design cases defined in IF-IS-0001 and envelope the temperatures to be expected during the Spacelab Mission 2. The temperatures of the Orbiter and IPS elements resulting from the Spacelab Mission 2 thermal analyses are detailed in NASA-Ref-TBD3.

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(4) NODE No	NODE DESCRIPTION / TEMPERATURE (°C)	END OF ASCENT (1)		ON-ORBIT		DESCENT (3)	
		HOT	COLD	(2) HOT	COLD	HOT	COLD
1210	Gimbal Support Structure	44	- 10	99	-122	99	-26
1220	Gimbal Support Structure	44	- 4	76	-117	78	-21
1230	Gimbal Support Structure	44	- 4	76	-118	78	-21
1240	Gimbal Support Structure	45	- 4	75	-116	76	-21
1250	Gimbal Support Structure	45	- 4	76	-117	77	-21
2201	Framework/Gimbal Support Struture Connection	46	- 1	53	-114	54	-14
2202	Framework/Gimbal Support Structure Connection	45	- 1	59	-115	60	-15
2203	Framework/Gimbal Support Structure Connection	46	- 2	60	-114	61	-16
2204	Framework/Gimbal Support Structure Connection	45	- 1	60	-114	62	-16
2520	Framework/Elevation Drive Unit MLI	52	- 36	87	- 73	87	-18
2521	Framework/Elevation Drive Unit MLI	50	- 22	70	- 41	70	- 8
4511	XDU Cable Feedthrough MLI	45	- 10	66	-107	66	-24
4219	Yoke Radiator	44	0	53	- 9	55	10
4515	Yoke +X Sidewall MLI	47	- 27	90	-106	90	-32
6231	Equipment Platform SSM	57	- 47	43	- 60	64	-11
6232	Equipment Platform SSM	52	- 31	40	- 58	62	-16
6233	Equipment Platform SSM	45	- 12	36	- 62	59	-23
6236	Equipment Platform SSM	44	- 12	27	- 60	51	-24
6237	Equipment Platform SSM	52	- 31	31	- 59	55	-16
6238	Equipment Platform SSM	57	- 46	42	- 57	64	-11
6251	Payload Attachment Ring	49	- 35	47	-100	26	-17
6321	RAU/ICS Baseplate	46	1	29	- 3	49	- 1
6501	Equipment Platform MLI	54	- 55	109	-134	109	-32
6502	Equipment Platform MLI	54	- 55	109	-135	109	-33
6503	Equipment Platform MLI	53	- 49	110	-114	110	-29
6504	Equipment Platform MLI	53	- 49	110	-115	110	-29
6505	Equipment Platform MLI	47	- 18	108	-103	108	-26
6511	Equipment Platform MLI	56	- 69	47	-163	47	-34
6512	Equipment Platform MLI	42	- 20	49	-132	48	-45
9101	Payload	50	-120	70	-120	70	-20
9111	Pallet below outer longeron	40	- 20	93	-143	93	-62
9112	Pallet above outer longeron	40	- 20	93	-147	93	-62
9113	Sill	40	-148	85	-146	85	0
9121	Module Aft End Cone Top	60	-113	37	-164	37	-24
9122	Module Aft End Cone Bottom	48	- 74	61	-135	61	-30
9125	Module Top	65	-167	-54	-170	-53	-19
6320	PAU	46	1	30	- 4	50	- 1
6330	ICS	45	2	33	- 13	53	- 1
9131	Orbiter Aft Bulkhead Top	59	- 91	33	-152	34	-26
9132	Orbiter Aft Bulkhead Top	51	- 47	67	-158	67	-40
9141	Orbiter Radiators	66	- 20	15	- 20	15	-20
9142	Orbiter Wings	66	-194	55	-194		

Notes: 1) Defined as PBD open, SL Services available
2) Defined as end of 3.25 hrs operating in sun mode
3) Defined as end of descent
4) Node numbers from IPS thermal analyses

TABLE 2.1.3.2.1.1.2-1: ORBITER/SPACELAB/IPS INTERFACE TEMPERATURES

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2.1.3.2.1.2 IPS Equipment located in the Orbiter Aft Flight Deck

The only item of IPS equipment mounted in the Orbiter Aft Flight Deck is the Contingency Control Panel (CCP). The CCP will be surface cooled by the cabin gas.

The dissipation of the CCP is 3 watts maximum. The temperature of the structural mounting interface and the mean radiant environment temperature is 49°C maximum.

2.1.3.2.2 Interfaces with the Spacelab

2.1.3.2.2.1 Thermal Design Configurations and Models

The Spacelab TMM used for the Spacelab Mission 2 thermal analyses is that shown in NASA-Ref-TBD5.

2.1.3.2.2.2 Structural Attachment Thermal Interfaces

The detail design of the mechanical connections between IPS and Spacelab is described in 20-ICD-IPS.

2.1.3.2.2.3 Mission Thermal Environment

The temperatures of the Spacelab and IPS elements used are derived from the IPS baseline thermal analyses. These temperatures are those used in and derived from the IPS design cases defined in IF-IS-0001 and envelope the temperatures to be expected during the Spacelab Mission 2. The temperatures of the Spacelab elements and the corresponding temperatures of the IPS elements resulting from the Spacelab Mission 2 thermal analyses are detailed in NASA-Ref-TBD7. In particular the thermal environment for the Spacelab Remote Acquisition Unit and Interconnect stations mounted on the IPS is shown in Table 2.1.3.2.2.3-1.

	Mean Radiant Environmental Temperature	Mean Enrivonmental Emissivity
Hot Case	+ 60°C	0.9
Cold Case	- 40°C	0.9

Table 2.1.3.2.2.3-1: RAU and IS Thermal Enrivonment

2.1.3.3 Electrical Interface

The electrical interface between IPS and Spacelab is descripted in Fig.'s 2.1.3.3-1 to -4.

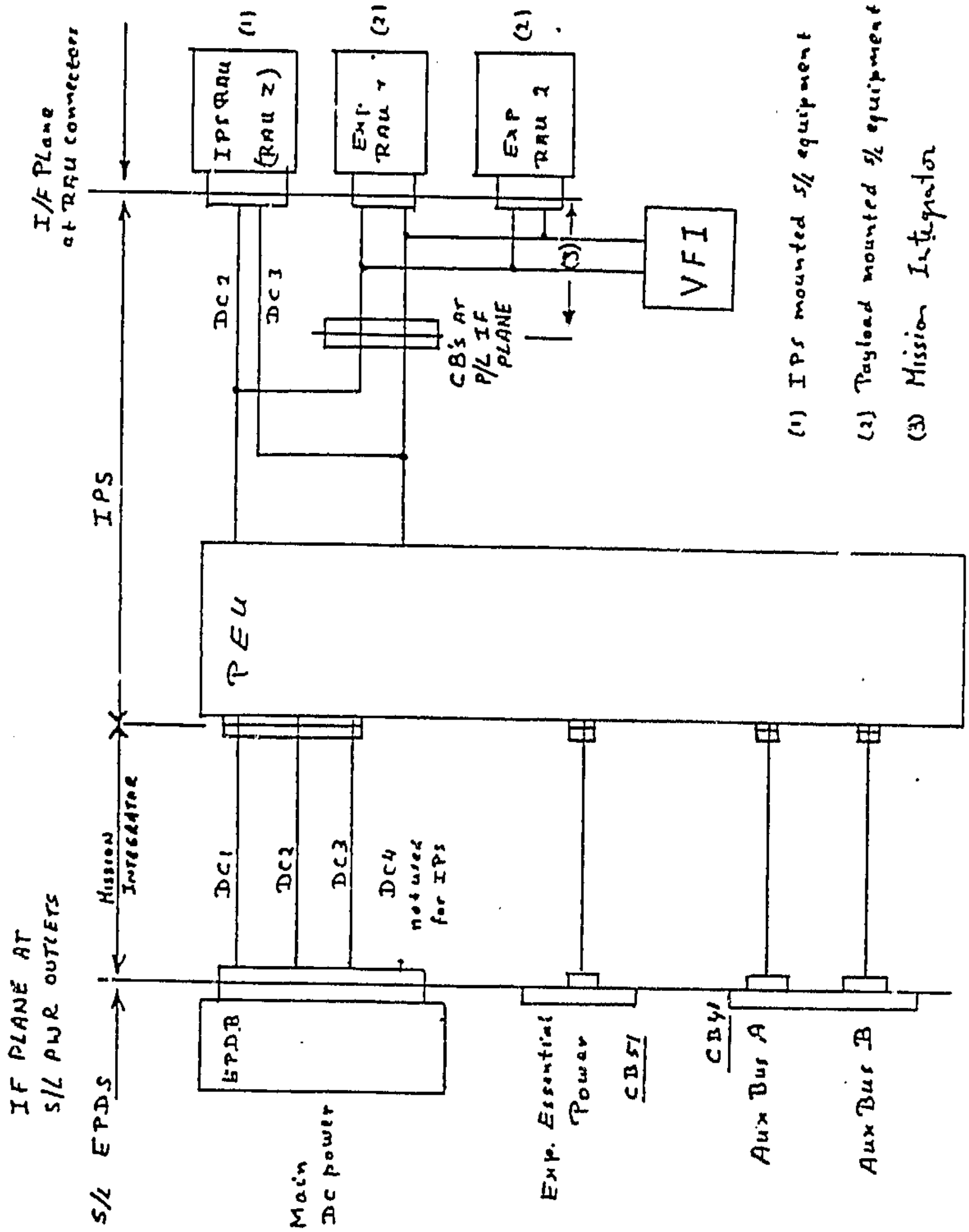

Fig. 2.1.3.3-1 IPS-SPACELAB POWER INTERFACES

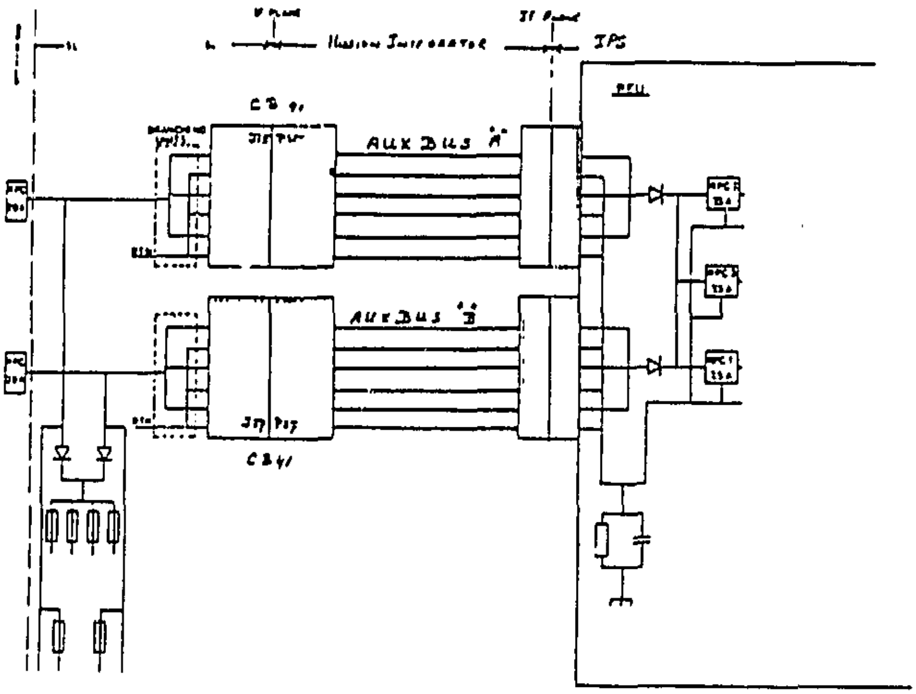

FIG. 2.1.3.3-2: AUXILIARY POWER INTERFACE

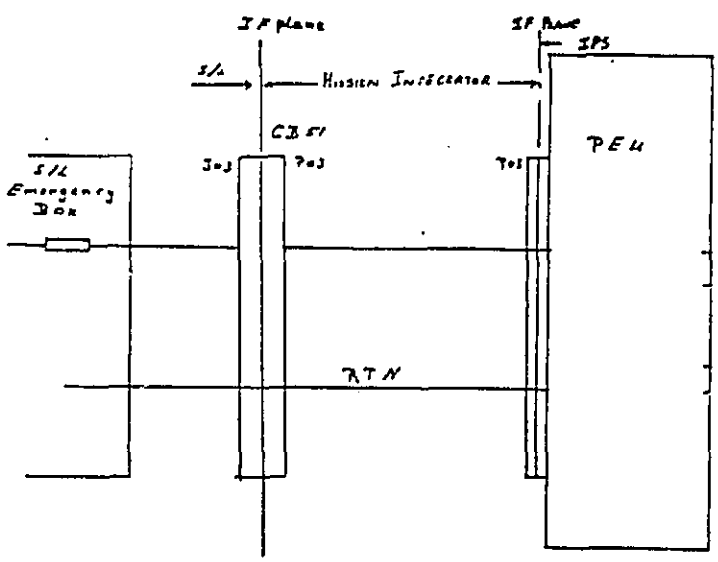

FIG. 2.1.3.3-3: EXPERIMENT ESSENTIAL POWER INTERFACE

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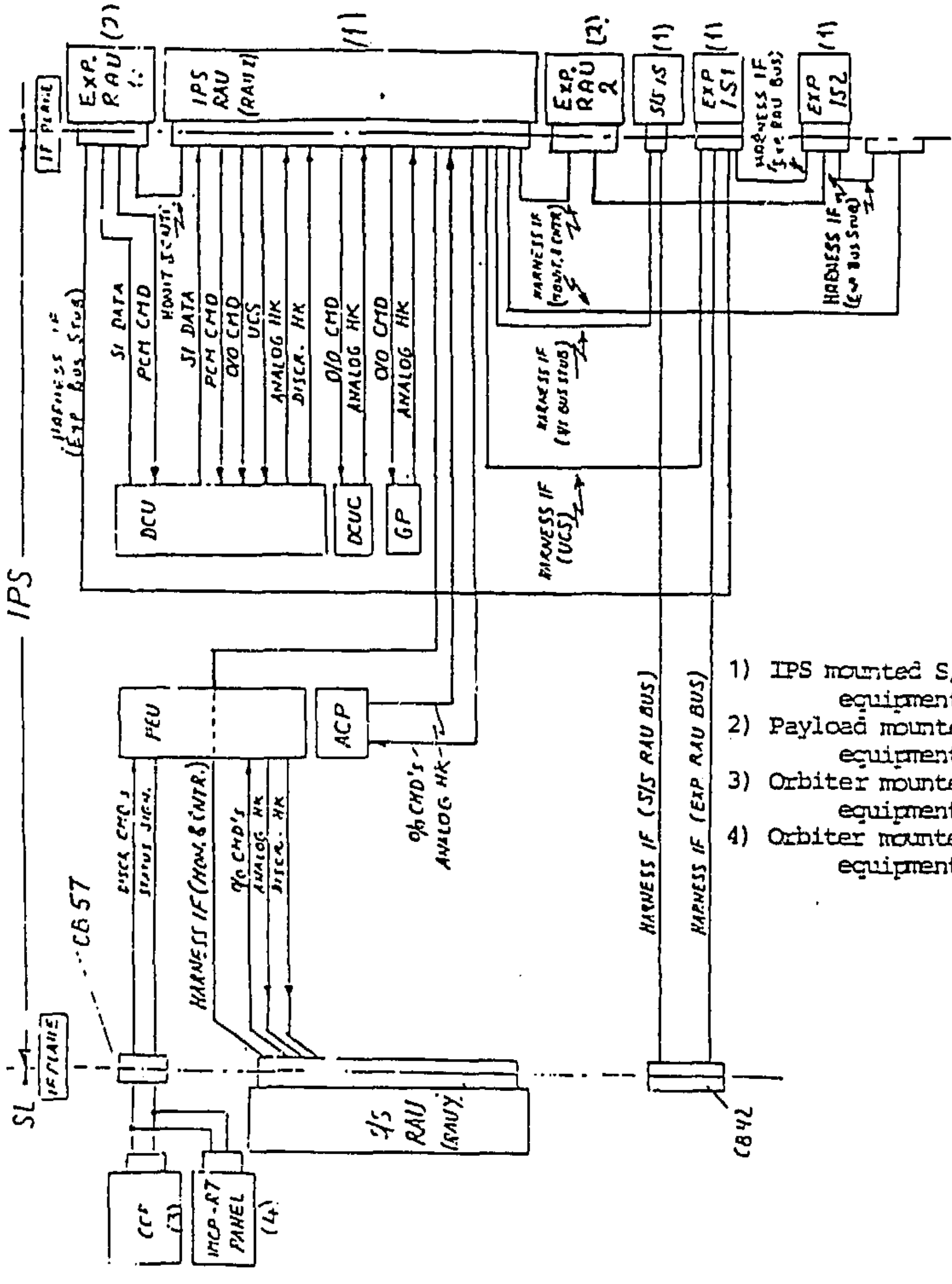

1) IPS mounted S/L equipment
2) Payload mounted S/L equipment
3) Orbiter mounted IPS equipment
4) Orbiter mounted S/L equipment

FIG. 2.1.3.3-4: IPS-SPACELAB SIGNAL INTERFACES

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2.1.3.4 Software Interface

2.1.3.4.1 Interface with the Orbiter/STS

2.1.3.4.1.1 Uplink

2.1.3.4.1.1.1 Orbiter Data

The GN&C data provided by the Orbiter via SCOS to IPS will be in the format defined in Appendix A §A.1.1.2.28 and A.1.1.2.29 of ICD-02-05301; the data are updated at the rate described in § 3.4.1.3. of ICD-02-05301.

2.1.3.4.1.1.2 IPS Uplink Commands

By uplinking commands to the IPS via the Orbiter (MDM-link) the ground will have the same functional capability to control the IPS that the crew has using the keyboard.

IPS uplink commands can be subdivided into commands which are executed directly by the Subsystem Computer Operating System (SCOS) without IPS SW intervention, and commands executed by IPS SW.

Commands which are directly executable by SCOS are beyond the scope of this document.

The number of uplink commands which are to be executed by the IPS SW will be one for every ITEM in each IPS FFD, with the exception of IEL and IME FFD's, plus one command for each of the Function Keys (FK's) dedicated to the control of the IPS (however, some of the commands will never be used via uplink).

IEL and IME FFD's only contain items belonging to the category of commands directly executable by SCOS.

2.1.3.4.1.1.3 Formats of IPS SW Data Files on MMU

The following data files are used by IPS SW:

1) Objectives (File OBJECT)

2) Filter Gains (File AGAINS)

3) OSP Data (File LORDFL)

4) Mission Dependent Parameters (File COMPAR)

5) DCU Fast Loop SW (File DCUFAS)

6) DCU Self Test SW (File DCUTES)

7) Monitor Parameters (File MONLIM)

8) TCA Parameters (File TCALIM)

Data files used by IPS SW will be stored on MMU as SCOS "User Files". The general format of an SCOS User File is described in MA-MA-0075, para 5.4.2.3.

2.1.3.4.1.2 Downlink

2.1.3.4.1.2.1 IPS Data in TMB

IPS data downlinked as part of the Telemetry Buffer (TMB) are grouped into downlink frequency subparts. Within each frequency sub-part, data are grouped into blocks of consecutive words and for each word the ID#'s of the items allocated to that word are listed.

The data are located in the TMB in the order given below:

ANALOG adjacent ID numbers are located in one 16 bit word

WORD 1	WORD 2	WORD 3	WORD 4
WORD 5	WORD 6	WORD 7	WORD 8

DISCRETE 2 rows of ID numbers are contained in one 16 bit word

bit 1	bit 2	bit 3	bit 4	bit 5	bit 6	bit 7	bit 8
bit 9	bit 10	bit 11	bit 12	bit 13	bit 14	bit 15	bit 16

For fields not filled with ID numbers the following applies:

(blank)	:	bytes or bits not contained in TMB
SL	:	bytes or bits already acquired by SL, respective words not accounted for in IPS TMB (see IF-IS-0001, para 4.2.1.3)
x	:	bytes or bits not available for IPS, but respective words accounted for in IPS TMB
--	:	bytes or bits available as IPS spares, accounted for in IPS TMB
IN	:	invalid bytes or bits due to 20 Hz acquisition, accounted for in IPS TMB

2.1.3.4.2 Interfaces with Spacelab

2.1.3.4.2.1 Interfaces with Spacelab SW

The Spacelab generated SW packages listed in SR-IS-0001, § 3.1.5.2, in the versions specified below in Table 2.1.3.4.2.1-1, are utilized for the production or for the on line support of IPS SW ("Version" defines the revision status).

SW Package	Version
- Host Macro Assembler (XMAS)	4.0
- Host Linkage Editor (XEDL)	4.0
- HAL/S-CII Compiler, HALLINK	6.01
- HAL/S-IBM Compiler	16.46
- Subsystem Computer Operating System (SCOS)	8.6
- System Generator (SYSGEN)	3.6
- FFD Skeleton Generator (SKLGEN)	3.6
- Memory Configuration Generator (MCTGEN)	3.6
- Flight Tape Generator (FLTGEN)	3.6
- Data Base Generation and Maintenance (DBGM)	3.11

Table 2.1.3.4.2.1-1: Spacelab SW Packages Versions

2.1.3.4.2.2 CDMS SW Resources Utilization

2.1.3.4.2.2.1 SSC SW Sizing

SSC core size utilization (budget and actual) is as follows:

SW Item		Words
a)	SSC SW without IPS	40000
b)	IPS SW (Cat. 1 and 2)	22500
c)	Second Fixed Format Display Buffer	1500
	TOTAL	64000
	available	65500

Item a) is defined as the SSC SW System as optimized to a SL-2 configuration.

Item b) is as specified in IF-IS-0001, § 5.2.1.

Item c), actually operations dependent, is required when it is expected to operate two DDS's concurrently with the SSC.

2.1.3.4.2.2.2 SSC SW Timing

SSC CPU load and access budget is as specified in IF-IS-0001, para 5.2.2.

Due to the interrupts allocation specified in IF-IS-0001, para 5.2.3 it is expected that it will not be possible to update two concurrent FFD's at the nominal rate of 1 Hz. The consequence is that the FFD update frequency must be set at SCOS generation time to a lower value which is consistent with the overall SSC SW load, in order to permit tasks running at priorities lower than the FFD-updating task to access the CPU and in order to avoid the generation of repetive SOE's 8A01 (Overrun of the FFD updating task).

A nominal FFD update frequency for an IPS mission is established to be 0.75 Hz. However, as it is considered operationally acceptable that FFD updates could be slowed down to a minimum of 0.33 Hz, the CPU margin obtainable by reducing the "nominal IPS FFD update frequency" to this value is the current CPU reserve. Reduction to a minimum of 0.5 Hz is controlled by ESA, for IPS SSC SW and operation contingencies. Lower update frequency values need joint approvals from ESA and the organization responsible for the SL SW maintenance.

2.1.4 IPS Performance

The IPS performance has been analyzed using a finite element based simulation model incorporating the following substructures:

Orbiter from NASA/MDTSCO

Pallet from ESA/ERNO

IPS from DORNIER

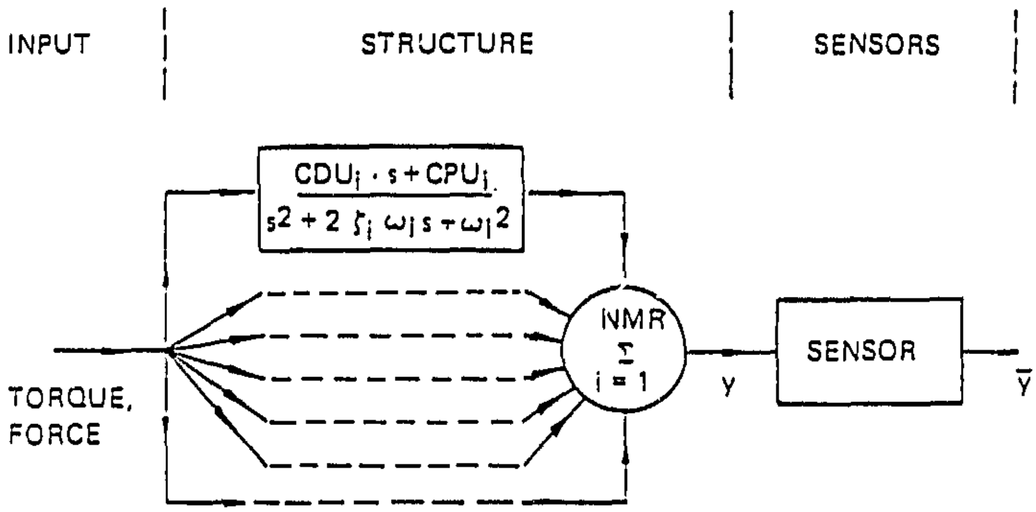
Payload from DORNIER

The overall structure is modelled to be free-free.

The following data apply to the Orbiter finite element model (payload doors open):

- Mass $m = 81074$ kg
- MOI $J_{xx} = 1.2 \cdot 10^6$ kgm²
  $J_{yy} = 8.8 \cdot 10^6$ kgm²
  $J_{zz} = 9.1 \cdot 10^6$ kgm²
  $J_{xz} = 0.3 \cdot 10^6$ kgm²

The basis of the overall structural model is a system representation using parallel second order oscillators between control and disturbance inputs and sensor outputs as in figure 2.1.4-1.



NMR	:	number of retained modes
y	:	displacement at sensor station
$\overline{y}$	:	measured displacement at sensor station
CPU_i	:	proportional coupling coefficient (for i'th mode)
CDU_i	:	derivative coupling coefficient (for i'th mode)
ζ_i	:	damping coefficient (for i'th mode)
ω_i	:	modal frequency (for i'th mode)

Figure 2.1.4-1: Parallel Oscillator Representation

For every IPS look angle and for every IPS payload an individual structural model is used.

External disturbance forces applied to this model are:

- VRCS thruster firing and
- Man motion.

Location of Orbiter Disturbance Sources
(Co-ordinates relative to the spacelab co-ordinate system in millimeters):

- Man Motion X = 8806 Y = 0 Z = 1198
- Vernier Thrusters

	X	Y	Z
1	3456	-1516	-1267
2	3456	1516	-1267
3	34968	-3807	1499
4	34968	3067	1499
5	34968	-2997	1408
6	34968	2997	1408

Magnitude of Orbiter Disturbances

- The man motion to be considered by the IPS design shall be as shown in Figure 2.1.4-2. This motion shall be applied in each axis individually.

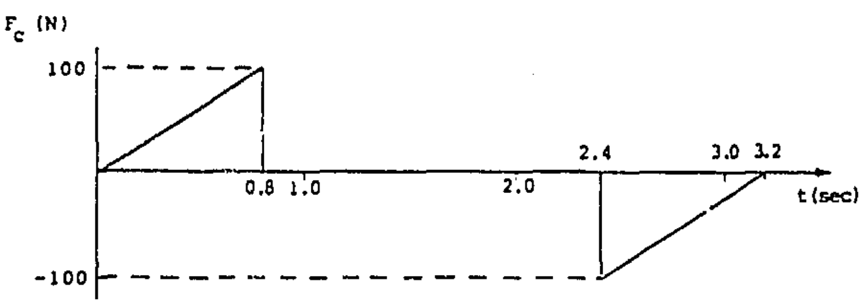


Figure 2.1.4-2: Man Motion Disturbance

- The IPS design takes into account the vernier thruster thrust levels and six firing combinations shown in table 2.1.4-1 for an 80 msec duration. The IPS design shall also take into account an Orbiter limit cycle motion of ± 0.1 degree.

Thruster No.	X	Y	Z	Remarks
1	-3.56	75.62	- 78.29	
2	-3.56	- 75.62	- 78.29	
3	0	106.8	- 2.67	
4	0	-106.8	- 2.67	
5	0	0	-106.8	
6	0	0	-106.8	
Thruster Combinations	1,3,5 2,4,6	1,2 5,6	1,4 2,3	+ ve rotation - ve rotation

Table 2.1.4-1: Thrust vectors (N) and firing Combinations

With these orbiter dynamics and FEM's for 200, 2000 and 7000 kg payloads the following performance values have been simulated (Table 2.1.4-2):

Payload Mass	Disturbance		Sim. Result (arcsec)
200 kg	Man Motion		
		Lat	3.2
		Roll	10.0
	Thruster Firing		
		Lat	5.6
		Roll	10.0
2000 kg	Man Motion		
		Lat	3.9
		Roll	4.0
	Thruster Firing		
		Lat	5.3
		Roll	5.1
	Quiescent Stab.		
		Lat	0.8
		Roll	3.0
7000 kg	Man Motion		
		Lat	6.0
		Roll	-

Table 2.1.4-2: IPS Performance

2.2 Position and Hold Mount (PHM)

2.2.1 General

The PHM is a small pointing facility for experiments weighting up to 200 kg and calling for low or medium stability. The pointing stability is based on the orbiter (± 0.1 deg.).

After performing a feasibility study (Phase A), which included a two axes demonstration model, Dornier System has completed the definition of the PHM (Phase B) in October 1982.

2.2.2 Technical Concept

The Position and Hold Mount (PHM) is a two axis pointing facility for smaller Spacelab payloads or payload clusters of up to 200 kg mass (see Fig. 2.2.2-1, 2.2.2-2). Its elevation-over-azimuth two axes gimbal assembly provides up to 360 degrees of freedom range in azimuth and up to 180 degrees of freedom range in elevation. The medium pointing stability based on the stability of the Orbiter is ± 0.1 degrees.

As a Spacelab subsystem the PHM relies on Spacelab/Orbiter support in the areas of data management (CDMS), power supply, thermal control services for electronics boxes (cold plates) and attitude information (IMU). It was one of the driving design goals to ease the use of these Spacelab/Orbiter services for the potential PHM user and to make interfaces as simple, reliable, and modular as possible to cover a broad application spectrum for the PHM payloads.

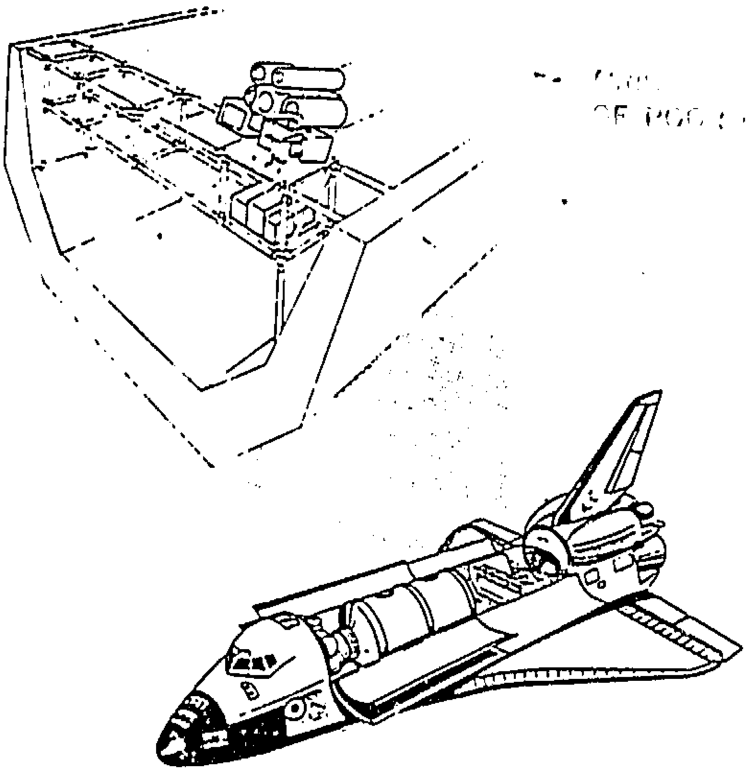


FIG. 2.2.2-1: PHM WITH TYPICAL PAYLOAD CLUSTER

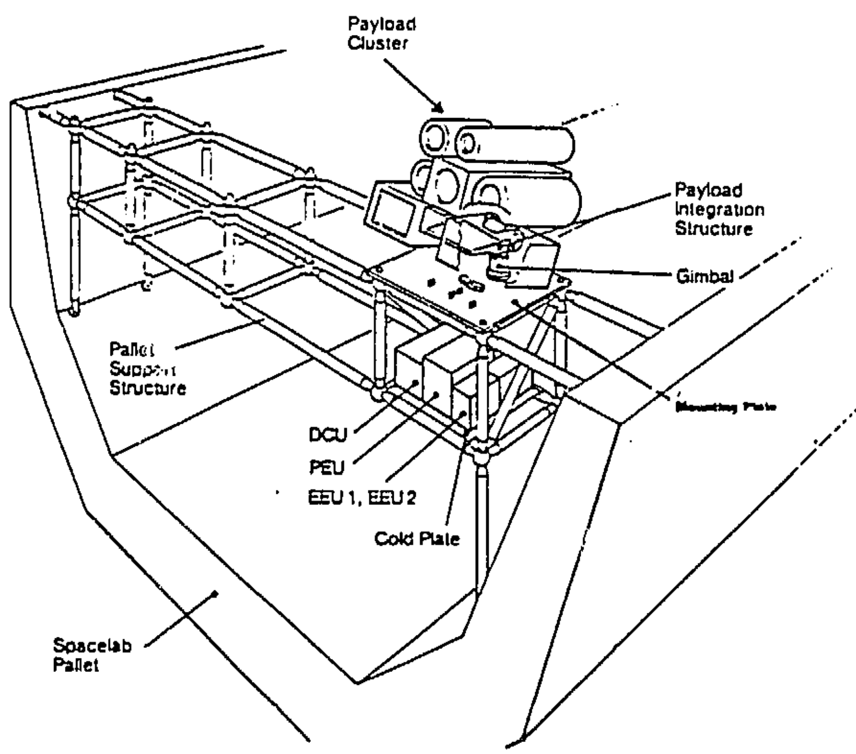


FIG. 2.2.2-2: PHM WITH TYPICAL PAYLOAD INTEGRATED ON A SPACELAB PALLET SUPPORT STRUCTURE

The PHM Spacelab subsystem consists of the following major assemblies:

- The Gimbal Assembly (GA) with the Drive Mechanism Subassembly (DMSA) for rotating the payload, the Clamping Mechanism Subassembly (CMSA) for securing the payload during launch and landing and the Structural Elements Subassembly (SESA) for overall stiffness.

- The Electronics Assembly (EA) with its Power Electronics Unit (PEU) for power supply and the Data Control Unit (DCU) for data and operation management. The Emergency Electronics Units (EEUs) are special electronic units for contingency back up operating modes.

- The Ground Support Equipment (GSE), partitioned in the Electrical Ground Support Equipment (EGSE) and the Mechanical Ground Support Equipment (MGSE), which both supply the necessary checkout and verification tools.

Special attention throughout the whole PHM design was payed to the safety concept and its mechanical and electrical implementation. The safety elements have to guarantee the integrity of the PHM and its payload during any mode of operation, especially during launch of the Shuttle, in orbit operations and during descent and landing of the Orbiter. Safety is among others achieved by a completely redundant retraction and stowing capability for the payload.

This safety aspect was one of the design drivers for the overall PHM technical concept.

Due to its modular design the operating capabilities and performance of the basic PHM design as described in this document can be extended relatively easy by adding attitude sensors (e.g. sun sensor), the necessary sensor couplers and software in the DCU.

This aspect was not covered in this phase B study, but experience with the PHM demonstration model (PHM phase A-study) indicated with a Dornier supplied sun-sensors, that accuracy improvement by a factor of 5 is easily achieveable.

As the PHM is conceived to cover a broad range of applications, some mission dependent hardware has to be taylored according to the specific payload requirements and has to be supplied by the user. These are the following elements:

- a Payload Integration Structure (PIS) as linking element between payload and PHM
- a Mounting Plate (MP) as linking element between PHM and the Spacelab Pallet support structure
- a mission tailored harness for linking PHM and payload elements to the electronics
- a thermal protection for the PHM elements and for the payload according to the thermal requirement of the particular mission
- some structural elements to optimize the load paths from Spacelab to the payload.

2.2.2.1 Mechanical Concept

The major design drivers leading to the mechanical design concept are the requirements:

- for accommodation of payloads of various shapes and sizes,
- for a gimbal freedom permitting pointing ranges of any sector within a total of a hemisphere,
- for the ability to secure payload and gimbals for safe return under all circumstances, and
- for the flexibility of mounting the PHM at various places and in any direction on the Spacelab pallet by means of mission dependent support structures.

Thus, the Gimbal Assembly (see Fig. 2.2.2.1-1) fulfils the mechanical payload accommodation and safety requirements, given in short form:

- payload mass: 200 kg (including Payload Integration Structure)
- payload moment of inertia: $50 \leq J \leq 500$ kg/m^2 (incl. PIS)
- payload centre of gravity relative to PHM coordinate system for

  side mounted payloads $Y_{max} \leq 625$ mm

  end mounted payloads $X_{max} \leq 1125$ mm
- payload service
  - o securing of payload during ascent and descent.

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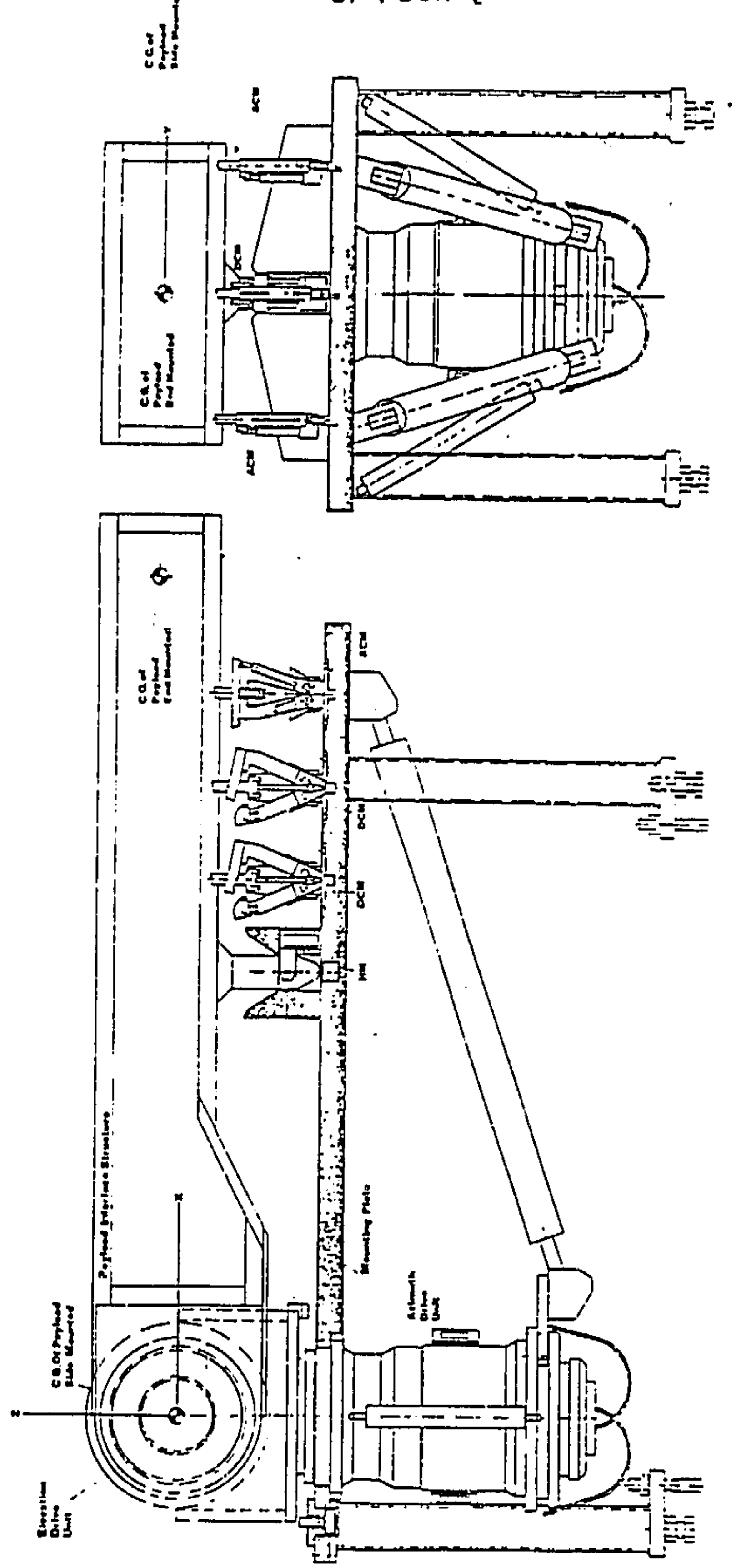


FIG. 2.2.2.1-1: PIM GIMBAL ASSEMBLY

- position keeping at zero power by means of brakes (Orbiter vernier thrusters only)
- safety requirements
  - o structural integrity to be guaranteed by gimbal assembly and clamping mechanism
  - o payload not to penetrate the dynamic envelope of the Orbiter and not to damage surrounding equipment or experiments.

The Gimbal Assembly (see Fig. 2.2.2.1-2) covers the following subassemblies:

- The Drive Mechanism Subassembly (DMSA) has the task to rotate the payload in the desired direction or to perform the commanded motions like scanning or tracking. Main elements within the DMSA are the Drive Units (DU) (see Fig. 2.2.2.1-3) for the azimuth (ADU) and elevation (EDU) axes. They contain the DC-motors for torque generation, the resolvers for relative angle measurement, the brakes to hold the payload in any direction, the position indicators as a backup to enable emergency retraction and the bearings to take the load. The Yoke links ADU and EDU together and carries auxiliary items like connector brackets etc.

- The Structural Elements Subassembly (SESA) consists of the user supplied Payload Integration Structure and Mounting Plate. Furthermore the struts as linking element between the ADU and the MP supply the necessary stiffness for the Gimbal Assembly. The adjustable mechanical endstops may limit the range of the payload and serve as ultimate safety device in case of malfunction.

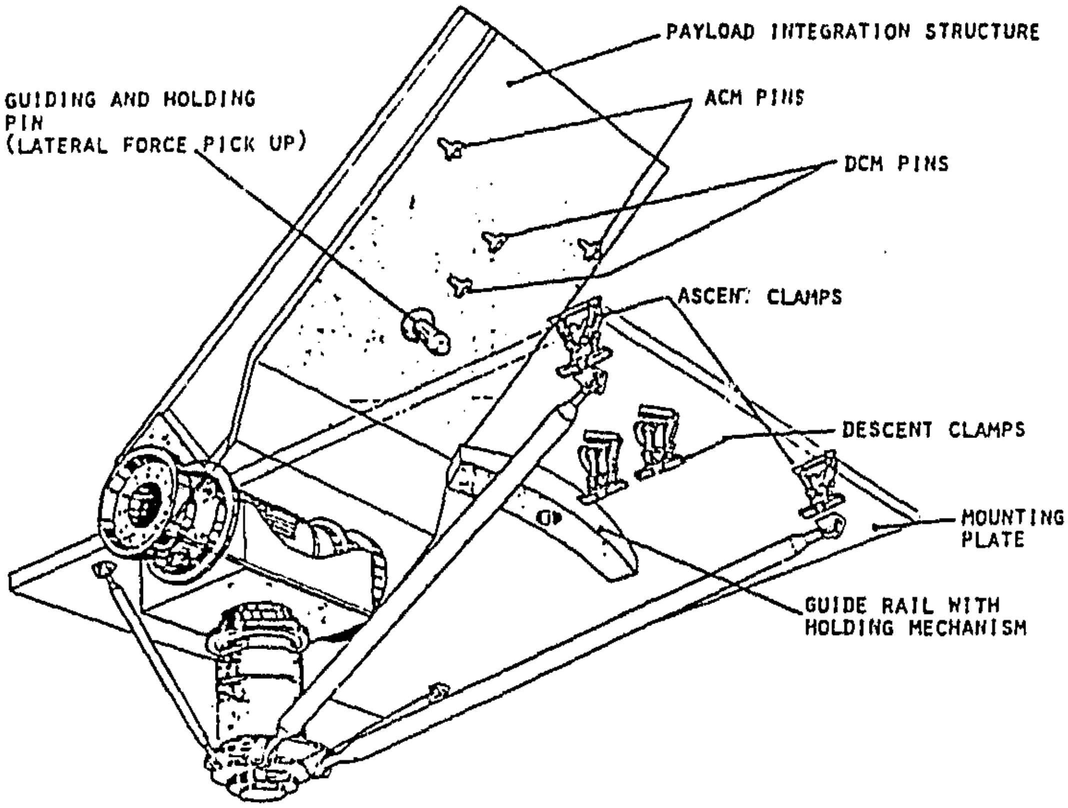
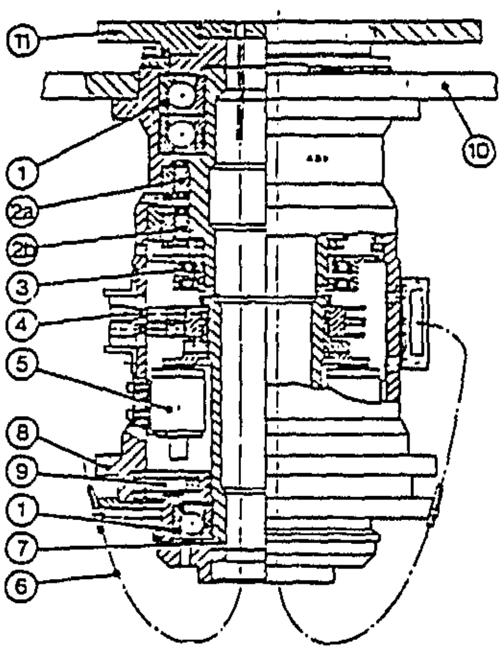


FIG. 2.2.2.1-2: PHM GIMBAL SUBSYSTEM AND CLAMPING MECHANISM



The Components of the Drive Units are:

1 Ball Bearings
2a Motor for nominal operations
2b Motor for emergency operations
3 Resolver
4 Position Indicators
5 Solenoid Brakes
6 Flexible Wire Harness
7 Shaft
8 Housing
9 Membrane
10 Mounting Plate } Mechanical
11 Yoke } Interface

Housing and shaft are one piece components for maximum stiffness.

FIG. 2.2.2.1-3 PHM DRIVE UNIT COMPONENTS

- The Clamping Mechanism Subassembly (CMSA) design features different elements for different tasks. The Ascent Clamping Mechanism (ACM) takes the loads imposed on the PHM/payload ensemble during launch of the Space Shuttle. The Descent Clamping Mechanism (DCM) is responsible for securing of PHM/payload during Orbiter descent and landing. Main actuating element in both units are memory metals which, subjected to heat, change their physical dimensions, thus actuating the clamp.

  The Holding Mechanism serves as a mechanical guide for the zero-clamping-position and as intermediate stowage clamp during Orbiter RCS main thruster firing.

2.2.2.2 Thermal Concept

The thermal concept of the PHM requires to cover the Spacelab so called "hot case" and "cold case" operational conditions.

As it is not possible, to cover all possible user and payload requirements each payload + PHM thermal concept must be individually tailored.

The PHM reserves nevertheless within its PEU some power, dedicated for thermal control, as well as the software is flexible enough to incorporate even complex thermal control switching functions.

It is assumed that whatever thermal concept will be chosen for the PHM and its payload, it is as much as possible of the "passive thermal control" concept.

2.2.2.3 Electrical Concept

The Electronics Assembly of the PHM is designed for operation via the Spacelab CDMS in terms of data acquisition, data transmission, operations service, etc. and for supply by the Spacelab power system in terms of main- and essential power.

The electronics concept is based on the following interfaces, shown also in the interface blockdiagram (fig. 2.2.2.3-1):

- PHM-SSC link for data exchange with the CDMS by an IPS-type of RAU (which is an EXP-RAU)
- PHM-EXC link for data exchange with the payload by an EXP-RAU
- PHM main power bus supply by Spacelab EPDB for normal operations
- PHM emergency retraction and stowage commanded and controlled from Orbiter AFD panel R-7, via bracket 57.

Three different electronics units will control/perform the PHM operations/functions as follows:

- the Data Control Unit (DCU) will handle all data traffic and software duties imposed on the PHM by operations requirements
- the Power Electronics Unit (PEU) will supply all necessary power and switching functions for the PHM, except the emergency functions
- the Emergency Electronics Units (EEU), which is present in redundant form as EEU-1 and EEU-2, will serve as power supply, command receiver, and control signal generator in case of emergency operation.

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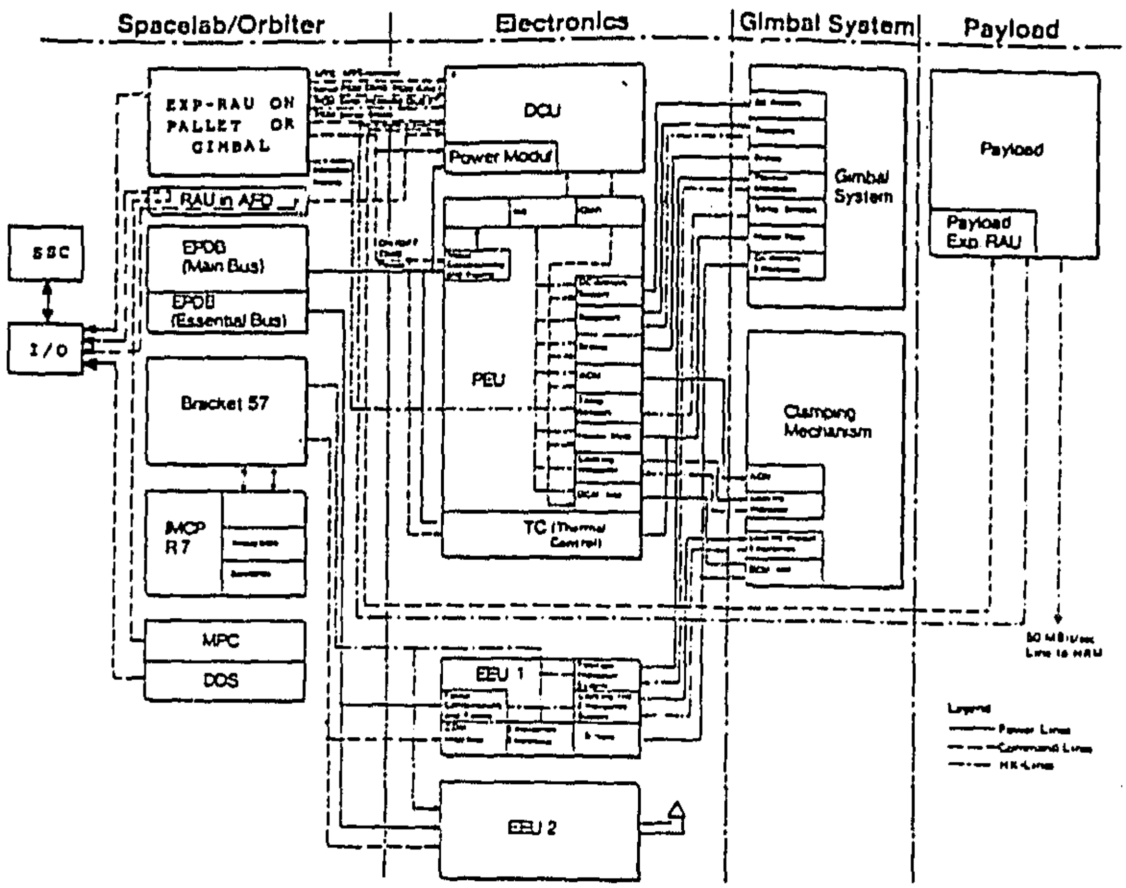

FIG. 2.2.2.3-1: ELECTRONICS ASSEMBLY

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The DCU, as most complex unit of the Electronics Assembly, is described in the following in more detail:

The DCU handles and synchronizes the data traffic between the PHM and the Spacelab CDMS (Subsystem Computer, Experiment Computer) via RAU.

The dataprocessor 430-R controls the traffic on the MUDAS dataway and is able to perform logic and arithmetic operations. This capability is used for calculation of the attitude measurement and control algorithms, and for operating the PHM according to the PHM modes and their routines. The programs of the data-processor are stored in the memories, which are of PROM and RAM type.

The DCU is built to the Dornier-MUDAS space standard. It consists of functional modules, which are connected to the MUDAS dataway. The modules are controlled by the data-processor, which besides the control function is able to perform logic and arithmetic operations.

2.2.2.4 Software Concept

The PHM software concept is governed by the rule to cover as many operational and software tasks in the PHM dedicated processor as possible, to make the PHM a real self standing, autonomous pointing facility to ease the usage and to facilitate the testability.

Nevertheless it is clear, that the offered basic Spacelab Subsystem Computer (SSC), Data Display System (DDS) services are used.

That leads to a distribution of the PHM software in the PHM DCU, covering all operational sequencing and pointing control functions and in the SSC for usage of basic CDMS software services. Additionally the S/W concept is so flexible to incorporate easily additional sensors to increase the pointing accuracy (for example: sun sensor).

2.2.2.4.1 DCU Software

The DCU software covers the following topics:

- Controller S/W
- Timing Control and Organization S/W
- Housekeeping S/W
- PHM-routines and subroutines S/W
- Transformation, and calibration S/W

2.2.2.4.2 CDMS Software

The main task of the CDMS software for PHM is to transfer data, as shown in Fig. 2.2.2.4.2-1.

Inputs are acquired from

- MDM: commands and orbiter attitude information
- Keyboard: commands
- RAU: values to monitor or to display and hand controller information

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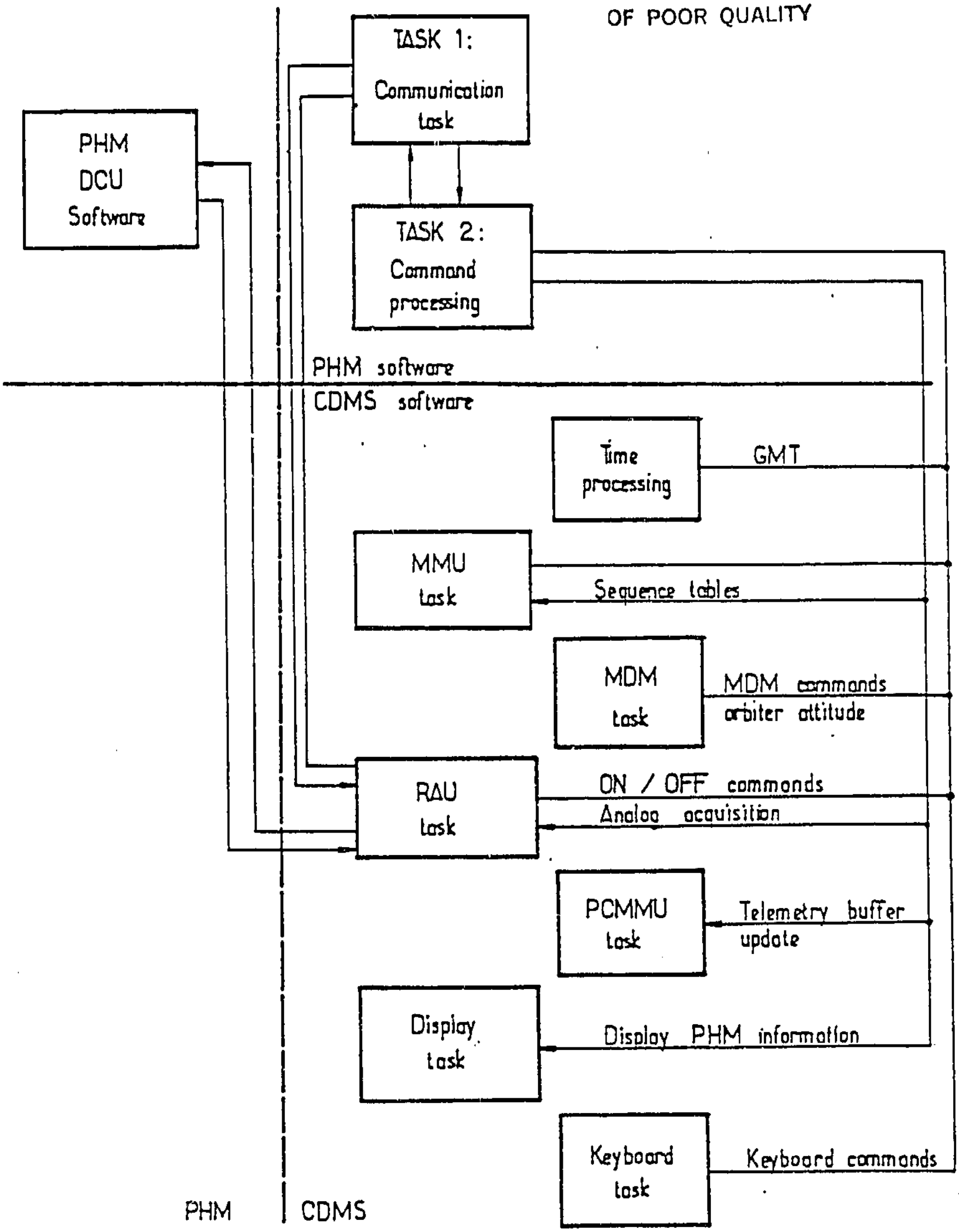

FIG. 2.2.2.4.2-1: CDMS SOFTWARE BLOCK DIAGRAM

- MMU: command sequences
- MTU: time information

Outputs are routed to

- Display: measurement points, monitoring status error messages, etc.
- RAU: ON/OFF commands and serial output
- MMU: update command sequence and user files
- PCMMU: Error messages, mesaurements point values, CDMS hardware status, etc.

SCCS is in charge to access the different peripherals but some application software has to be provided to initialize the different transfers.

The proposed solution for PHM CDMS software is based on a two tasks structure

- The first task will have to handle dialog with the PHM via the RAU serial channels
- The other task will have to handle all other transfers required by the PHM utilization.

For both tasks the application language will be HAL/S.

2.2.2.5 Operations Concept

The operations concept of the PHM (see Fig. 2.2.2.5-1) is based on one side on the CDMS-subsystem computer (SSC) and on the other side on the PHM-DCU processor (MUDAS).

These two processors control together the operations of the PHM.

Four major PHM modes cover the operating requirements and give individual payload operating flexibility.

- Individual command mode (ICM)
- Manual Pointing Controller mode (MPC)
- Timeline Program Mode (TPM)
- Power Down Mode (PDM).

Included in all modes is an "alert trigger", which indicates PHM malfunctions, to alert the Spacelab crew (FFD-SSC message line).

The Individual Command Mode reflects the necessity, mainly for safety reasons, to have a step-by-step command possibility, initiated and controlled by CDMS-keyboard. Within the ICM the following submodes are possible:

- the ICM-end item-mode (ICM-EIM)
- the ICM-PHM-routine-mode (ICM-PRM).

The ICM-EIM is characterized, that by CDMS command, a list of PHM-end items may be commanded and controlled.

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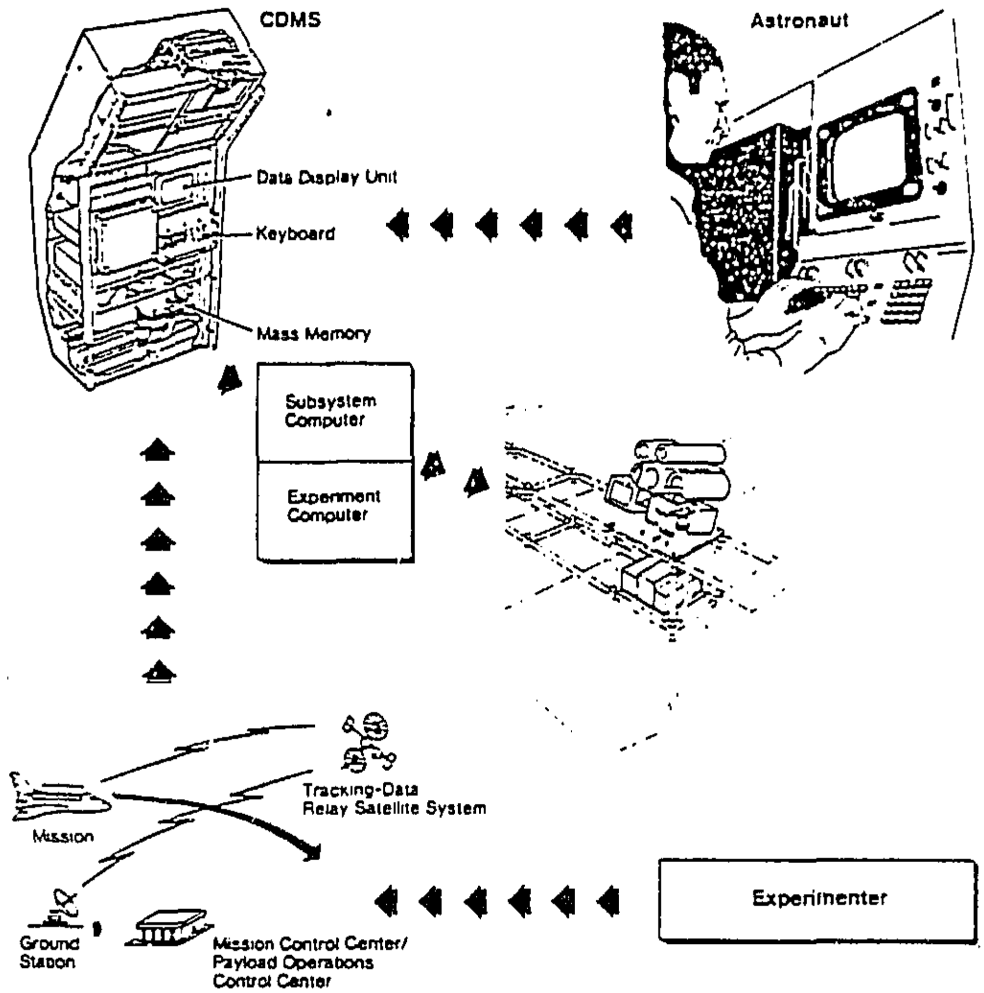

FIG. 2.2.2.5-1: COMMAND LINES FOR PHM OPERATIONS

End items are:

- DC-motors
- clamps
- brakes
- retraction bolt
- resolvers
- thermal elements

The ICM-PRM is characterized, that by CDMS command, individual PHM-routines may be initiated by a GO-command and stopped arbitrarily by a STOP-command, not dependent on GMT.

The Timeline Program Mode is an automatic, time dependent mode which is preprogrammed by the appropriate PHM-routines and their characterizing parameters.

The TPM may be operated from:

- the CDMS keyboard by the on-board crew
- the MMU by initiation of the on-board crew
- the POCC to make corrective actions or change the contents of the MMU.

The sequencing element operating the PHM in the TPM is a "Sequence Table" where the various PHM routines are expressed by their associated parameters. The "Sequence Table" (S.T.) shall be initiated by a GMT start-time and ended by a GMT end time. The S.T., may be constituted of several Sequence Table blocks. One block shall comprise one specific parameter set.

To have enough flexibility within the preprogrammed routines, a "Sequence Table Change Procedure" permits fast onboard modifications by the Spacelab crew.

The following PHM routines are defined:

- Activation and Initialization Routine (AAIR)
- Control Parameter Loading Routine (CPLR)
- Relocking Routing (RELR)
- Slewing And Holding Routine (SAHR)
- Positioning And Holding Routine (PAHR)
- Tracking Routine (TRAR)
- Scanning Routine (SCAR)
- Bias Controller Routine (BICR)
- Deactivation Routine (DEAR)

The Manual Pointing Controller Mode shall enable the on-board crew of manual pointing and slewing the PHM. The MPC-commands will be treated as bias commands. The operating post of this mode is a Keyboard/joystic device in the AFD to produce the bias commands. After having selected an adequate PHM-routine, motion-characterizing parameters of this routine will be regarded as dummies and overwritten by the bias commands which are:

- Azimuth angular rate
- Elevation angular rate.

The Power Down Mode is introduced to keep the PHM thermally controlled without switching on the PHM processor. The PHM is thus a purely CDMS operated mode concerning the commands as well as the feedback data.

For temperature measurements and control, the identical items are used as for the normal PHM thermal control.

The power will be drawn from a special power supply within the PEU.

2.2.2.6 Safety Concept

The PHM safety concept is based on the following design requirements:

- the PHM and its payload shall be non-jettisonable devices
- the PHM and its payload shall require no EVA - operating or a back up to come to a safe configuration
- the payload shall be fixed to the PHM during launch, in-orbit-operations, and descent/landing.

The safety concept thus is most concerned with the structural integrity of the PHM and its payload. To guarantee this, the PHM/payload combination requires through all high stress phases, a maximum strength and stiffness of the mechanical configuration which is only met, when proper clamping is achieved.

Thus, the safety concept is mainly a problem of guaranteed clamping of the PHM through ascent and descent of the Space Shuttle/Orbiter. As clamping can be visually verified before launch, the main concern is with the clamping prior to descent. This is created in the following:

If normal flight mode retraction and clamping has failed (even in case of CDMS failure) or if normal flight mode is no longer possible, an emergency retraction and clamping mode will be performed which relies on fully redundant hardware components, separate command, monitoring, and power supply lines.

Emergency clamping thus is performed manually within the following scheme:

- a redundant set of electrical and electro-mechanical items, will rotate the PHM axes to null position, independently of CDMS and DCU
- the emergency power electronics (there are two for additional redundancy), will power the above mentioned items. Power will be drawn from essential power bus
- the redundant clamping mechanism and its actuating elements will secure PHM/payload
- redundant end switched will indicate safe locking
- initiation, control, and completion of the emergency retration will be done via Orbiter Panel R-7, completely independent from CDMS command lines.

The announcement to the crew, that PHM requires an emergency retraction, will be performed by SCOS services to DDS, based on PHM housekeeping data.

The PHM mechanical safety devices are shown in figure 2.2.2.6-1.

The emergency retraction mode will be checked out for proper function after launch prior to start of the PHM's scientific mission.

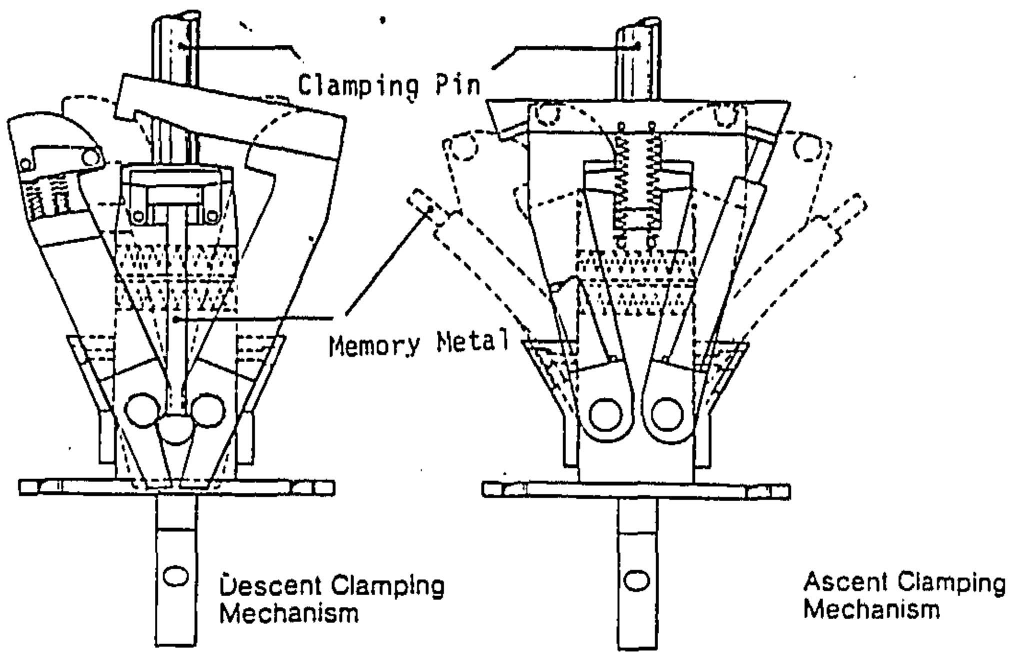

Fig. 2.2.2.6-1: PHM Safety Devices

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2.2.3 Interfaces

The PHM will be integrated into Spacelab. This section describes the relevant Spacelab and Orbiter interfaces of PHM.

2.2.3.1 Mechanical Interface to Spacelab

- PHM/Payload Dynamic Envelope	not to exceed the Orbiter dynamic envelope and not to hit any surrounding Spacelab payload
- Total mass of PHM	m_{PHM} = 95 kg
o Mass of Gimbal Assembly	m_{GS} = 75 kg (incl. clamps)
o Mass of Electronic Assembly	m_{ES} = 20 kg
- Total mass of PHM and Payload	m_{TOT} = 295 kg (max.)
- Mounting interface for	
o Gimbal	- Azimuth Drive Unit flange, mountable to the user supplied Mounting Plate - Struts - Clamping Mechanism flange to be mounted to the Mounting Plate - Clamping Pins, to be mounted to the user supplied Payload Integration Structure - Endstops, according to PHM/Payload Dynamic Envelope
- Electronic Assembly	Common baseplate to be mounted to a Spacelab cold plate
- Mounting Plate	To be mounted to Mission dependent Spacelab Pallet Support Structure

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- Mounting direction	any direction relative to the Spacelab coordinate system
- Coupled analysis	coupled analysis necessary for the combined Payload/PHM/Support Structure

2.2.3.2 Thermal Interface to Spacelab

As stated in 2.2.2.2 "Thermal Concept" there is at the moment no specific thermal interface to the Spacelab. But one can easily forsee, that the PHM would require thermal services from the Spacelab in the field of

- space and heat rejection capability for the power and data electronics by means of using a total or a portion of a Spacelab coldplate.

- space and heat rejection/injection capabilities for the Drive Mechanism Subassembly to comply with the "passive thermal control" concept. Space is needed in this case for a heat pipe radiator surface mounted to the cold plate.

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2.2.3.3 Electrical Interface

To Spacelab:

- Data Control Unit

  Connected with one Connector to Spacelab Main Power Bus
  Connected to an EXP-RAU

- Power Electronics Unit

  connected with one connector to Spacelab Main Power Bus

- Emergency Electronic Unit

  EEU1 and EEU2 connected individually to Spacelab Essential Power Bus

- PHM Power Requirements

  180 W mean value, 2 axes
  280 W peak value, 2 axes
  150 W emergency, 1 axis

To Orbiter:

- AFD R7 Panel

  hardwired lines to PHM Emergency Electronic Unit via Bracket 57

2.2.3.4 Software Interface

- Payload to

  Experiment Computer Operating System (ECOS-S/W)

- PHM to

  Subsystem Computer Operating System (SCOS-S/W)

2.3 Antenna Pointing Mechanism

2.3.1 General

The Antenna Pointing Mechanism (APM) is the coupling/discoupling member between, for example, a heavy satellite and its spot beam antenna reflector, as shown in Fig. 2.3.1-1. It is specially designed for precise pointing within a limited pointing range. It incorporates within its cardanic suspension direct drive motors and precision angle pick-offs, controlled by its specially tailored electronics.

The development status is as follows:

Mechanism model built:

- Vibration Model
- Engineering Model
- Qualification Model
- Antenna Deployment and Pointing Mechanism (Fig. 2.3.1-1)

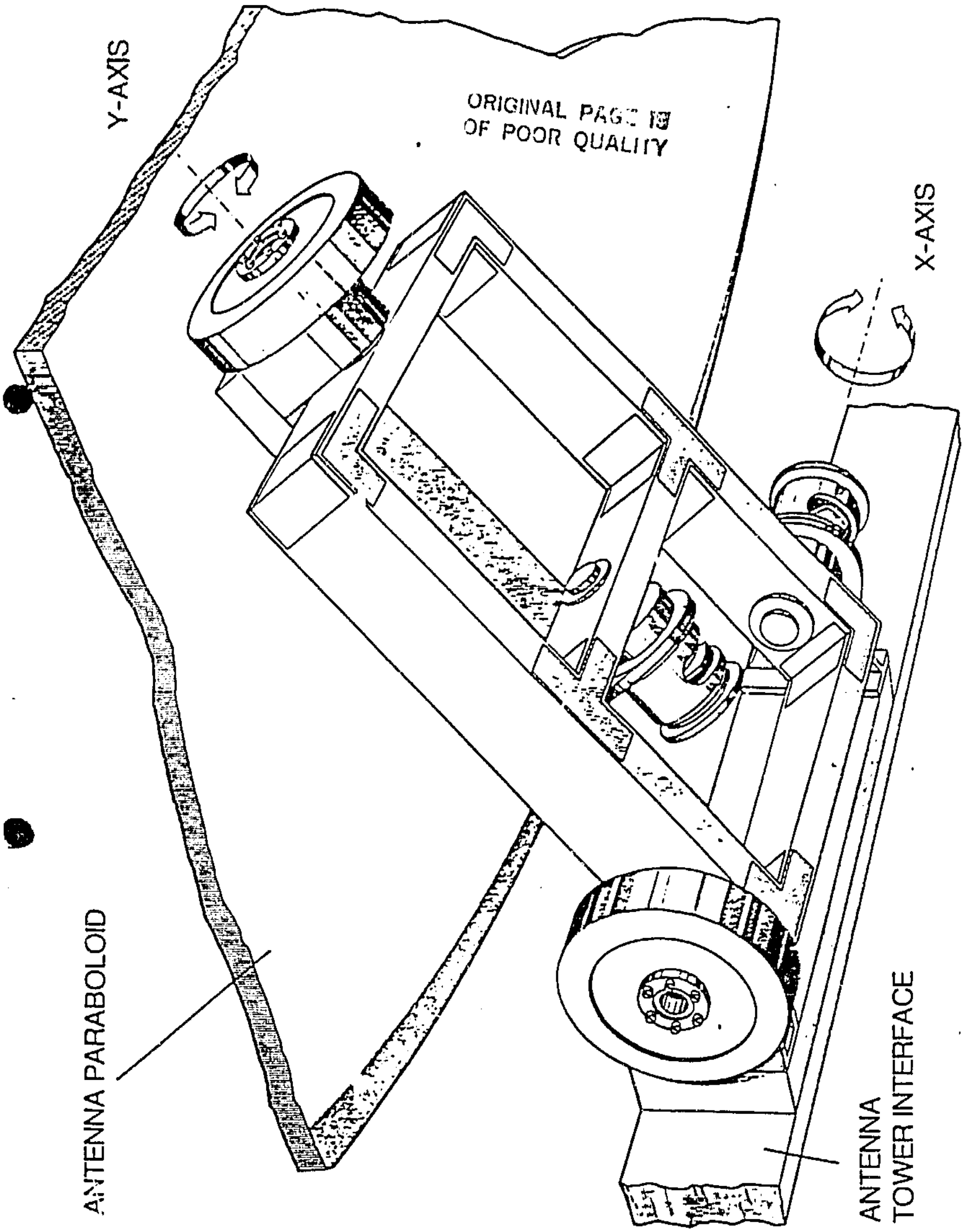

FIG. 2.3.1-1: ANTENNA POINTING MECHANISM (APM)

The Qualification Model survived successfully sine vibration inputs up to 20 g and interfacial temperatures from -70°C to +65°C during qualification tests. An accelera· life test programme at ESTL/England was passed successfully.

For the APM-Electronic a Breadbroad Model was followed by a Prototype Model which was manufactured to flight standard and has passed successfully electric performance tests.

2.3.2 Technical Concept

The technical concept aims at performing the following two pointing tasks:

1. precision pointing of an antenna as required by the WARC regulations for direct satellite broadcasting with a pointing accuracy of 0.01 degrees for the mechanism

2. antenna beam shift from one country to another (repointing)

The APM development at Dornier led to the following measured performance data:

Main Performance Data

- Antenna Inertias up to 18 kgm^2
- Deployment Range up to 180 degree
- Fine Pointing Range +/- 1,5 degree
- Pointing Accuracy +/- 0,01 degree
- Dynamic Performance above 2 Hz

- Speed above 0,3 degree/sec.
- Mass 3,5 kg
- Thermostable CFP (Carbon Fibre Plastic) structure
- Temperature Range -160°C up to +120°C
- Vibration Level up to +/- 50 g

2.3.2.1 Mechanical Concept

The APM is of modular design. Drives, angle pick-offs, bearings are the same on both axes. The frames of CFP material can be adapted to the specific interface requirements of both antenna and spacecraft body.

Basically the framework corresponds to a cardanic suspension. Its axes arrangement can be built centrally or shifted apart to allow deployment actions for the antenna from the launch to an operational condition.

Each axis has a direct driving stepper motor and a resolver as angle pick-off. The bearings are dry lubricated and therefore free of maintenance and free of backlash as well.

If required the APM can be equipped with a clamping device for direct load transfer during launch. A relocking device may be implemented to fix the antenna in a central position upon command.

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2.3.2.2 Thermal Concept

The mechanism's heat input is low enough not to require an active thermal control.

The used CFP material has good thermal stability.

2.3.2.3 Electrical Concept

APM Electronic

It includes all the necessary electronic equipment of the Antenna Pointing Mechanism. The APM-Electronic controls the drives, feeds the angle pick-offs which are precise resolvers, processes their output signals into digital format, accepts commands and generates status signals. The input and output signals of the APM-Electronic are adaptable according to various satellite interface requirements. Each of the two pointing axes is connected to a completely redundant set of electronic circuits.

Technical Data of the APM-Electronic

- Power Input	15 W
- Mass	3.9 kg
- Overall Dimensions	363 mm long 166 mm wide 145 mm high

2.3.2.4 Software Concept

Not applicable

2.3.2.5 Operations Concept

The APM can be operated in the following modes:

- open loop

  specific commands related to effective antenna beam direction

- closed loop

  control loop using the angle pick-offs as position measuring devices

- closed loop with RF sensor

  control loop using an RF sensor as position measuring device located apart from the mechanism and sensing the effective antenna beam direction.

  Introduction of individual bias settings resulting from on-orbit calibration of the antenna performance poses no problem.

The APM is able to perform continuous antenna pointing in order to counteract residual satellite nutations over a life time of ten years.

Its dynamic capability corresponds to a bandwidth above 2 Hz.

The APM's stiffness allows for complete antenna pointing performance testing and calibration on ground with gravitation influence.

2.3.2.6 Safety Concept

The mechanism incorporates complete redundancy of motors and angle pick-offs. The electronic is completely redundant as well.

The relocking capability adds further reliability, which means, that in case of any failure within the drive branches the mechanism can be relocked to its central position by means of its telecommanded Relocking Device.

The life tests have proved the APM's reliable performence under qualification conditions. Special bearing tests confirmed their life endurance.

2.3.3 Interfaces

Mechanical I/F

Both the antenna frame and the ground plate can be adapted to special versions of reflector dish and satellite structure.

Electrical I/F

Mechanism and APM Electronics are to be interconnected by cable lines. Adaptation of the Electronics to the S/C power bus poses no problem.

Thermal I/F

Thermal shielding of the mechanism should be foreseen according to the specific implementation conditions.

Vibrational Dynamic I/F

Depending on the S/C structure and antenna characteristics the APM's stiffness allows for loads up to 70 g.

Operational Dynamic I/F

The APM provides a high dynamic bandwidth and high inherent damping. Therefore the matching of the dynamic behaviour of the APM to specific satellite requirements poses no problems.

3. REQUIREMENTS ANALYSIS FOR SPACE STATION POINTING SYSTEMS

3.1 General

In this section the mission models are analyzed with respect to pointing system aspects. The following relevant mission areas have been identified

- Astrophysics
- Earth and Planetary Remote Sensing
- Environmental Observations
- Communications
- Life Sciences and
- Material Processing

The suitability of the Space Station for pointed experiments can be discussed with the following parameters

- altitude limitation
- orbit limitation
- mission duration
- attitude pointing error/stability
- data management & transfer
- power
- heat rejection
- cleanliness

The micro-g requirement is in the area of Life Sciences and Material Processing the dominant requirement. This results in special requirements to the altitude control/vibration isolation of the whole Space Station. No demand for pointing systems has been identified in these areas, thus they are no more considered in the following sections.

3.2 Astrophysics

The major objectives of astrophysics are

- investigate properties of extragalactic space, the milky way galaxy, and the solar system
- investigations with respect to cosmic evolution.

All wavelengths are used, like e.g.

- visible (cameras)
- IR-astronomy
- x-ray astronomy
- RF-astronomy

The driving requirements in the region of astrophysics are

- sensitivity (aperture, size, mass)
- pointing accuracies
- contamination limits
- thermal control (e.g. cryo systems etc.)

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Typical payloads in the field of astrophysics are summarized in table 3.2-1 and table 3.2-2.

The manned Space Station can provide the following support for astrophysics experiments:

- manned operation
- manned maintenance & refueling of consumables
- contamination control
- perform special calibration procedures etc.

The location of an experiment at the Space Station will significantly enhance the overall utility (costs, operational mission efficiency).

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	MASS (KG)	ALTITUDE (KM)	INCLINATION (DEG)	POWER (KW)	HEAT REJECTION (KW)	FIELD OF VIEW (DEG)	POINTING (ARCMIN)	STABILITY ARCSEC/ TIME	DATA RATE (MBPS)
SOT	8,200	400	57	6.8	0.9	0.025	0.017/90	0.1/15	50
SIRTF		100		1.3		0.125		2/20	
STARLAB		100		2.2		0.8		10/30	
SCRN		100		0.8		70		N/A	
SOLAR SOFT X-RAY TELESCOPE	1,300	430	57	0.2	0.2			0.1	
STO	16,60		57				8-12		
PINHOLE X-RAY CAMERA	10,00		97						
X-RAY OBSERVATORY	3,600	400	28.5	0.9	0.9		1.0		
HRS	1,800	400	< 45	0.5	0.5	10	6/90	36/0.02	0.03
XTE	1,000	400	28.5	0.6	0.6				
AXAF	10 TO 12,000	500	28.5	2.0	2.0		30	1.0	
LAMAR	9,500	400	28	3.4	0.4	1	3/67	10/0.02	0.1
VLBI	1,400	400	57	0.9	0.9	0.1	2.5/45	150/60	12
ASO	12,500	400	57	4.1		0.025	0.17/90	0.1/15	42

Mass (kg)
Altitude (km)
Inclination (deg)
Power (kW)
Heat Rejection (kW)
Field of View (deg)
Pointing (arcmin)
Stability sec/Time
Data (mbps)

Table 3.2-1: Typical astrophysical experiments I

Experiment Title	Pointing Accuracy	Major Dim. cm.	Mass kg	Power	Data
EUV Stellar Spectrometer	1' 30"/hr	60x40x20 30x20x20	20	20	
Large Proportional Counter	0.5 0.5	140x100x100	336	312	11M
Super Wide Angle Camera	0.3° 0.17°	500 x 130	30	100	
Heavy Ion Package (FSLP) ES 24, Beaujean	-	500 x 7	20	1	
Very Wide Field Camera (FSLP) ES 22, Courtès	5° 0.1°	100x100x50	90	50 150 pk	
High Resolution X-ray Background	0.5°	65x50x100	115	47 128 pk	3K
X-ray Sky Surveyor	0.5° 0.01°/sec	40x40x40	100	50	75K
UTEX	3' 3-10"/sec	1300 x 30 20x20	269 1	125 275 pk (300 thermal)	200K
Gas Scintilation Proportional Counter (FSLP) ES 23 ESA Andresen	1° 1°	30x30x55 18x22x10	20 2.8	12	42K
Wide Field X-ray Surveyor	1° 0.01°	240x110x110	100	35	100K
CIRBS	15' 15'	770 x 90 Pointing-sys.	90 30	58 100 pk	3K

Table 3.2-2: Typical astrophysical experiments II

Experiment Title	Pointing Accuracy	Major Dim. cm.	Mass kg	Power	Data
Large Area Detector Array	3' 0.5'/min				
Wide Field Camera for X-ray Bursts					
GIRL + Expt. package	10" 0.3"/sec	116.5 x 375	665	160	15K

Table 3.2-2 cont'd: Typical astrophysical experiments II

However, trade off studies will be required for the location if the contamination requirements will be very stringent. The payload may be preferred to be part of the space station, an unmanned space platform or a dedicated free-flyer. This trade off must include also manual versus automated operations aspects.

3.3 Earth and Planetary Remote Sensing

A set of typical payloads in the field of remote sensing is assembled in table 3.3-1.

The major objectives of the earth and planetary remote sensing are:

- Exploration of the solar system, incl. planets
- Earth dynamics, crustal motion, potential fields
- Resources Study
  - o crops
  - o minerals
  - o petroleum etc.
  - o ocean

The characteristics are

- Planetary landings (not relevant from pointing system view)
- Remote sensing
- Development of instruments for future missions.

The last two items are of interest for pointing system aspects. They have the following design driving requirements:

Experiment Title	Viewing Direction	Pointing Accuracy	Major Dim. [cm]	Mass [kg]	Power	Data
GIRL + Focal Plane Instr.	Limb	2' 1'/30 sec	116.[illegible]0 x 375	605 (50)	340 w	16 Kbps
Ebert Fastie - GIRL + Focal Plane						
Optical Meso/Thermospheric Experiment	Limb	0.5^{o} 0.01^{o}/sec	20 x 60 x 15 15 x 60 x 15 35 x 60 x 15	19	80 w	8 Kbps
Grille Spectrometer (FSLP)	Limb	Exp. provided	180 x 60 x 80 (16.5 x 45 x 54)	80 (15)	300 w	50 Kbps
Lyman (FSLP)	Limb Nadir	S/L	71 x 16 x 25	8	35 w	200 bps
Microwave atmospheric Sounder	Limb	$\pm 2.5^{o}$ 0.01^{o}/sec	30 x 30 x 50 NO 30 x 30 x 30 MRSE (10 x 30 x 10) AFD	11 (1)	40 w	3.4 Kps
Temp. + Wind in the Meso- + Thermosphere (FSLP)	Limb//vv	0.1^{o} 0.05^{o}/sec	52 x 59 x 81	37	52 w 70 wpk	2 Kbps
Waves in the OH Layer (FSLP)	Limb	S/L	19ø x 52.6	10.4	9 w 42 w	
Tropo/Stratospheric Wind	Limb	can provide own scan	100 x 100 x 60 (50 x 38 x 18)	25 (5)	8 w 12 w	10 Kbps
Na D-Line	Solar	1 - 3' 4"	183 x 24 x 25	12.5	30 w	ccTV

Table 3.3-1: Atmospheric and Solar Experiments

Experiment Title	Viewing Direction	Pointing Accuracy	Major Dim. [cm]	Mass [kg]	Power	Data
Solar Obla'eness	Solar	1 - 5' 1 - 2"/sec	30ø x 100	20 (5)	25 w	9.6 Kbps
SCALP	Solar	2' 2° 20" 0.5°	30ø x 70	30	60 w	1.4 Kbps
Solar Spectrum	Solar	1.25° 10'/15 min.	44 x 32 x 61	23	85 w 115 wpk	476 bps
Solar Constant 1 (FSLP)	Solar	±2° 0.5°/hr	16 x 16 x 44	5	3 w	
Solar Constant 2	Solar	±2.5° ±15	25ø x 0.5 22ø x 0.3 (23 x 48 x 35)	16 (7.5)	30 w 80 wpk	150 bps
Pall t mounted metric camera	Nadir	-	TBD	387	TBD	TBD
Scanning Radiometer	Nadir	S/L	25 x 14 x 30 (45 x 10 x 30)	15 (<10)	20 w	7 Kbps
Microwave Pressure Sounder	Nadir	S/L	50 x 50 x 20 30 x 60 x 20 50 x 25 x 25	31	400 w	100 bps
PICPAB (FSLP)	Earth	S/L	30 x 65.5 x 52.5 40 x 30 x 40 (44.7 x 30 x 40)	15+30 (25)	30 w	1 Mbps
MOMS	Nadir	0.1° 0.01°/sec	100 x 80 x 80 (single rack)	100 (65)	100 w	100 bps

Table 3.3-1 cont'd.: Atmospheric and Solar Experiments

Experiment Title	Viewing Direction	Pointing Accuracy	Major Dim. [cm]	Mass [kg]	Power	Data
Microwave remote sensing exp. (FSLP)	Earth	S/L	Antenna 200 x 125 x 180	152	1310 w 914 w 428 w	32 Mbp 377 Kbp
SAR	Earth $45^{o}/19^{o}$	S/L	Antenna 920 x 110 x 10 + Band Ant. 50 x 20 x 20 20 x 20 x 50	150 (2)	2400 w 520 w 50 w	57 Mbp
Low energy electron distribution function (FSLP)	Any	S/L	50 x 37 x 37	20 (1)	20 w	333 Kbp
Magnetic field vector (FSLP)	Any	S/L	11 x 15 x 45 14 x 14 x 16 Boom	5 10	7 w	256 bps

Table 3.3-1 cont'd.: Atmospheric and Solar Experiments

- Orbit
- Instruments pointing (optics, RF, etc.)
- data rates (RF-sensors)
- electrical power (LIDAR, Radars)
- RF-generation and susceptibility

For earth resource operational missions, where global coverage is required, highly inclined orbits (up to 90 deg) are required.

The payloads will comprise small to large microwave antennas and/or passive or active (large LIDAR) optical systems. Pointing requirements are today in the region of 1/10 of a degree, future large antennas may reduce this down to about 1/100 of a degree.

3.4 Environmental Observations

The major objectives of the environmental observations are

- atmospheric and ocean observation to further understanding of
  - solar terrestrial interactions
  - effects of man on environment
  - effects of natural phenomena on environment
- Contribute to the development of global environmental models.

The major design drivers are

- global coverage (highly inclined orbit)
- broad spectral coverage (multisensor measurements)
- Cross Track Scanning, Viewing (multidirectional measurements)
- High data rates (up to 120 MBPS)
- very large antennas

The instruments may be single antennas or grouped on platform/pallets/bridges. The instruments comprise passive remote sensors, active stimulation by lasers, plasma wave injection facilities, electron beams and powerful radars. Typical characteristics for some payloads are shown in table 3.4-1. Early missions at low inclination may include missions for man supported equipment development missions.

The major design driving requirements are

- Orientation & pointing
- Data rates
- Power
- Orbit range.

	MASS (KG)	ALTITUDE (KM)	INCLINATION (DEG)	POWER (KW)	HEAT REJECTION (KW)	FIELD OF VIEW (DEG)	POINTING (ARC MIN)	STABILITY (ARC SEC/ TIME)	DATA RATE (MBPS)
OCEAN	10,000	400	57–90	10	10	1	720	720	120
LARS	1,200	780	> 60	1.7					50
UARS	2,400	400	56, 70	1.3	0.8	VARIOUS			0.02
SPACE PLASMA PHYSICS	3,200	3–400	57–90	2.7	1.8	45	60	60	7.5
ZERO-G CLOUD PHYSICS	500	ANY	ANY	1.4		N/A	N/A	N/A	0.5
METEOROLOGY	1,200	400	57	1.2	0.74		6	6	0.01
ICE AND CLIMATE EXPERIMENT	3,500	275	87	2.3					1.4 TO 17.8

TABLE 3.4-1: TYPICAL ENVIRONMENTAL OBSERVATION PAYLOADS

3.5 Communications

The space station mission of interest with respect to communications and pointing systems will be the technology development for advanced communications technology. The space stations large size, high prime power supply, availability of man to observe, and the recovery of the hardware makes it ideal to employ it as an in situ laboratory.

The major areas of interest are

- large deployable antennas
- Laser communications
- Space borne Interferometer
- millimeter wave propagation.

Larger antennas, Laser communications, interferometer etc. require all higher pointing performance than delivered by the Space Station itself.

3.6 Requirements Summary

3.6.1 Attitude Pointing/Stability/Mass

Astrophysics Bold IC

Three major groups can be identified

1. Pointing accuracy — 1 arcmin down to arcsec high stability requirements

   Most payloads are in the range from 200 to 8000 kg but some are heavier)

2. Pointing accuracy — in the range of 0.1 arcsec

   Most payloads are in the range from 10 to 100 kg and a few up to 270 kg)

3. Pointing accuracy — greater 1 deg

   Payload mass — 10 - 150 kg

Earth Remote Sensing/Environmental Observations Bold IC

A broad spectrum of sensors is considered

- RF
- optical

with pointing requirements of 0.05 to 0.3 deg

Payload mass e.g. LIDAR		up to 3500 kg
	RF Optical	up to 150 - 200 kg
	(one payload up to 10000 kg, but only 720 arcmin pointing accuracy)	

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Communications } Sold IC

Large antenna pointing	<< 0.1°
Major pointing constraints	low payload eigenfrequencies

3.6.2 Altitude/Orbit Limitations

- Altitude

  Nearly all missions can be satisfied by the 400 to 600 km circular orbit, only some earth viewing mission prefer altitudes up to 1000 km

- Inclination

  The inclination requirements can be summarized in three groups

  - o Astrophysics and low "g" prefer 28.5 deg inclination
  - o Earth viewing missions which can be satisfied by 57 deg
  - o Earth viewing missions requiring global coverage (i = 90 deg)

3.6.3 Mission Duration

The potential benefit of the Space Station lies in the capability of supporting scientific research by man's presence of more than 7 to 30 days.

Most missions (e.g. Astrophysics, Earth observation) require mission duration in the region of months and years.

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The useful life of the payloads can be increased by maintenance, replenishment of consumables and by the update or changeout of new technology equipment (e.g. smarter sensors etc.) thus increasing the utility of the observatories through longer on-orbit life.

3.6.4 Data Management & Transfer

High data rates will be required for

- astrophysics payloads (< 50 MBPS)
- environmental Instruments (< 120 MBPS) (e.g. RF equipments)

3.6.5 Power

The missions, related to the pointing system, with highest requirement for electrical power will be the LIDAR and RF missions with up to 25 kW.

Peak astrophysics requirements are 6.8 and 3.4 kW/payload. A variety of payloads of less than 1 kW power demand exists.

3.6.6 Heat Rejection

The requirements for heat rejection will ly in the same magnitude of the power demand. Special effort is required for RF and optical amplifiers active cooling.

3.6.7 Cleanliness

Contamination control for the sensor systems is a stringent requirement (e.g. IR-astronomy, x-ray astronomy). This may cause problems with a manned space station (e.g. local atmosphere cabin leakage or other sources), an accommodation of the affected payload on a space platform may be preferred.

3.6.8 Man Operated Functions

- Manned operation & resource provisoning of station-attached telescopes
- Assembly & checkout techniques
- In rare cases, even, it is conceivable that the investigator could actually be sent to the Space Station to perform his experiment
- Man conducted development of station mounted sensors, analytical & automated techniques
- Manned development of station-attached advanced systems (communications)

3.7 Space Station Constraints Summary

- Preferred space station orbit
  - o Altitude 400 - 500 km
  - o Inclination 28.5

- Space Station Characteristics
  - o Moments of Inertia >> Shuttle MOI
  - o Space Station 1. Eigenfrequency about 1 Hz (Shuttle/Pallet first eigenfrequency about 4 Hz)
  - o Local angular deflections at first eigenmode = large with respect to high pointing requirements
  - o Local disturbances due to actuators
    man motion
    RVD events

- Space Station attitude reference data can be delivered to pointing subsystems.

4. POINTING SYSTEM ACCOMMODATION ON THE SPACE STATION

4.1 Accommodation Analysis

The pointing stability requirements versus the mass of potential European Spacelab Experiments are summarized in Fig. 4.1-1. The mass/accuracy ranges of IPS and the PHM are indicated, a wide range of experiments can be covered by these two systems.

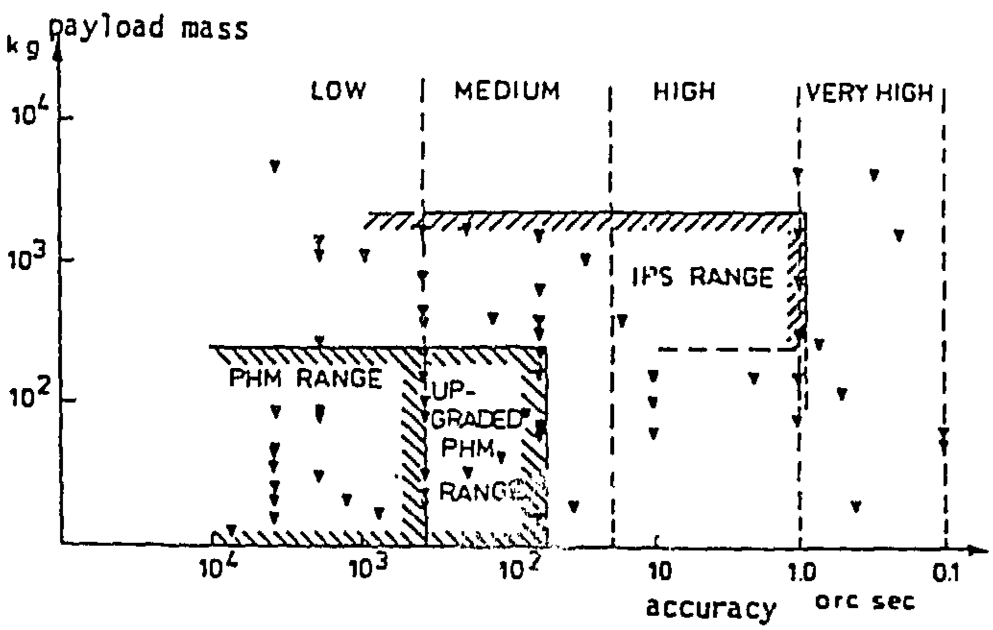

Fig. 4.1-1: IPS and PHM stability and payload range

4.1.1 IPS

Dynamics

- Orbiter
  - Orbiter Limit cycle: ± 0.1 deg/± 0.01 deg/sec
  - Lowest Orbiter/Pallet Eigenfrequency: < Hz
  - Disturbances: Man motion, Thruster firing
- Space Station
  - Limit cycle: TBD
  - Lowest Space St :ion Eigenfrequency: ca. 1 Hz (expected)
  - Lowest eigenfrequency of Solar Array System: << 0.1 Hz
  - Disturbances:
    - Man motion
    - Distributed actuators (e.g. thrusters)
    - RVD activities
    - Moved parts (RMS etc.)

C - 3

Rigid body angular accelerations of the Space Station due to disturbances and attitude control are expected to be lower in amplitude and frequency.

The disturbances are depending on

a) rigid body rates & angular accelerations

b) distance of IPS mounting location from Space Station C.O.M.

c) local translatorial accelerations and angular deflections due to Space Station flexibility.

to a)

Rigid body rates and angular accelerations are expected to be much lower than for the STS due to the Space Station high moments of inertia.

to b)

IPS performance simulations have been executed with a distance of about 1.6 m from C.O.M. For the Space Station a distance up to 10 - 15 m seems to be more realistic. Great attention has to be payed to the fact that resultant disturbances (lower rates, angular accelerations but much greater distances) are in compliance with the IPS-torquer capabilities (30 Nm).

to c)

Space Station flexibility

The first space station eigenmode of about 1 Hz requires at a first glance a lowering of the IPS bandwidth to less than 0.5 Hz. This is valid if the assumption can be made that the angular deflections due to the space station first eigenmode are negligable. An IPS performance similar to the IPS/Orbiter configuration may be achieved, a quick analysis has shown that the IPS can handle a larger but slower disturbance better, than it can handle a smaller but faster disturbance.

If the local angular deflections of the first eigenmode have to be compensated by the IPS, the controller bandwidth has to be increased to 2 to 4 Hz, lying than within the Space Station structural frequencies. So the lower structural frequencies have to be notched in the controller.

The controller structure will be different to the existing one. Modifications which can improve the situation are e.g. decoupling (e.g. magnetic bearing) and control by inertial systems (Reaction wheels, CMG).

Much more investigations have to be performed for stability assessments. The Space Station FEM has to be used for detailed analysis. Interaction is also expected with the payload model. An adaptive controller is recommended due to space station and payload changing characteristics. The feedforward loop (accelerometers) is recommended not to be used.

Operations

Task sharing is performed between CDMS and DCU. For the Space Station an additional processor is recommended, it has the following advantages

- required for adaptive controller
- increase autonomy
- increase flexibility

Safety

Reduced safety requirements are expected, because no reentry is planned (no cargo bay door closing constraints).

Power and Data

- the payload support power can be upgraded according to future payload requirements.
- the payload data lines are according to RAU, CDMS, STS capabilities.

Payload mass

- the IPS has been designed for payloads from 200 to 7000 kg, this seems to fit also with most of the space station candidate payloads.

Improvements for IPS

- better Gyros (noise, drift)
- separate Sun-Sensor
- wide FOV Acquisition Sensor
- on-board alignment calibration

  between IPS and space station inertial measurement unit reduces initial IPS AMA attitude error which relieves from the wide FOV acquisition sensor after first acquisition after launch

- additional control-loop based not on gyros, but on gimbal-resolvers for pointing relative to Space Station (e.g. during Space Station rotations or during IPS stowage/deployment or parking, back-up mode for loss of gyros) → additional processor or RAM extension

- improvement of command-capability from the Experiment Computer (e.g. automatic sequencing)

- Improvement of bright star-triplet acquisition procedure (→ SW) for bright stars search

- sun-sensor as fast attitude-sensor for fast loop control and not for attitude determination filter (ADF)
  - → ADF works only for roll-attitude and not for LOS in solar pointing
  - → different AMA-concepts for

$$\left.\begin{array}{l}\text{stellar}\\\text{solar}\\\text{earth}\end{array}\right\}\text{pointing}$$

- new/additional scan profiles
  - o raster-point scan (stop-and-go)
  - o sin/cos scan
  - o etc.

- earth sensors in control-loop (landmark, horizon-sensors)

4.1.2 PHM

In general, the PHM is for hemispherical coverage for

- low to medium 2 axes pointing and stability requirements for
- small to medium sized payloads,

requiring from the Space Station in its non-autonomous operation mode the

- state vector of the Space Station to calculate a quasi-inertial attitude for inertial pointing or earth tracking.

Possible PHM users in the field of Astrophysics are smaller experiments running in parallel with advanced large astrophysical payloads who want to maintain independence and flexibility from those experiments.

Possible PHM users in the field of Environmental Observation are all kinds of antenna- or telescope-based experiments fitting the PHM capabilities.

The PHM can be upgraded without problems by use of dedicated sensors (Gyros, Optical Sensors). With the demonstration model a pointing accuracy of 0.5 arcmin was achieved with a Dornier off-the-shelf sun sensor.

No accommodation problems exist with payload power and data requirements.

The PHM controller bandwidth is nominally between 3 to 4 Hz. No interaction (as for the IPS) is expected between PHM and Space Station.

4.1.3 APM

Typical application would be in the fields of:

- the Space Stations own infrastructure such as TM/TC antennas for up-downlink purposes,
- antenna pointing for experiments with small, light weight antennas, and
- surveillance operations by supporting a (video) camera.

The APM can accommodate payloads with the performance and interface data as given in sect. 2.3.

4.2 Identification of Design Improvements

4.2.1 IPS

- Improvement of performance
  - o Adaptive/self optimizing control
  - o Modified controller/actuator concept
- Updated distributed microprocessor system
  - o more flexibility, more autonomy, intelligence distribution
- Technology improvements
  - o Sensor improvements, smart sensors (CCD/CID sensors etc.)
  - o decoupling from Space Station or carrier e.g. magnetic bearings

- o cryo or fluidic connections to the payload

- Improvements with respect to maintenance/operations.

4.2.2 PHM

- Accommodation of payload dedicated sensors (inertial, sun, earth reference)
- Development of standardized interfaces (mech. and data)
- Use of dedicated processor
- Increase slew rates

4.2.3 APM

- Accommodation of larger antennas
- more powerful motors to increase slew rates

4.3 Pointing Systems Accommodations Summary

Most of the considered payloads of section 3. can be accommodated by IPS, PHM and APM. Additional investigations primarily have to be done with respect to:

IPS

- IPS/Space Station dynamics
  - o definition of disturbances
  - o set up a coupled Space Station/IPS finite element model
  - o analyse modified controller concepts
  - o perform simulations
- IPS Processor Accommodation
  - o S/W Requirements
  - o Task sharing between DCU, new processor and CDMS
- Analysis of future sensor developments

PHM

- Accommodation of payload dedicated sensors
- Development of standardized interfaces

APM

- Analyse accommodation of larger antennas

All three systems seem to be very well suited to be used as standard equipment for future Space Station missions.

7.4.9 External Radiators - Dornier System GMBH

DORNIER SYSTEM

Participation in
NASA
Space Station Study

TITLE: EXTERNAL RADIATOR CONCEPT FOR THE CORE MODULE OF THE SPACE STATION
TITEL: BOEING DESIGN

DOCUMENT NO.: TN-SSS-DS-002
DOKUMENT NR:

ISSUE NO.: 1
AUSGABE NR:

ISSUE DATE: 24.03.1983
AUSGABEDATUM:

PREPARED BY: Dr. Kreeb
BEARBEITET:

COMPANY: Dornier System GmbH
FIRMA:

AGREED BY: H. Preiß
GEPRÜFT:

COMPANY: Dornier System GmbH
FIRMA:

CONTRACT NO :

PROJECT MANAGER
PROJEKTMANAGER

Technical Note

External radiator concept for the core module of the Space Station

- Boeing design -

1. Introduction

The radiator concept, described in this technical note, is based on the requirements of Telex 9424 from 83-02-02 from Boeing and is part of the cooperation between Boeing and Dornier System within the 'Space Station Study'. The requirements mentioned are derived from the overall core module concept of the space station designed by Boeing. The radiator needs to be assembled in space, the individual radiator modules are stowed in up to 4 packages which are 1,75 m long by 0,5 m square. The modules will be attached to a central freon 21 loop system.

Main requirements :

- o Q_{max} = 25 KW
- o T_{Rad} = 323 K ÷ 280 K
- o Packages dimensions : 4 x 1,75 x 0,5 x 0,5 m
- o No sun-shielding possible

The overall design consists of these radiator modules which possess the following main design features :

- o Heat dissipation on the modules by means of heat pipes.
- o Module seize 1,7m x 0,5m.
- o 2 Heat pipes per module.
- o Radiation to both sides.
- o The heat pipes may be replaced by VCHP's (gas-stabilized heat pipes) for temperature control reasons.
- o Heat pipe and radiator sheet material is an aluminium alloy.

Problem areas :

- Heat pipe performance.
- Panel thickness and possibility to stow.
- High connecting area to central loop.
- Attachment mechanism of the panel modules.
- Low weight.
- Low cost and low development risk.
- Lifetime (10 years).

The design is based on the state of the art technique so that no development risk and minimized manufacturing and design costs exist.

2. Module Design

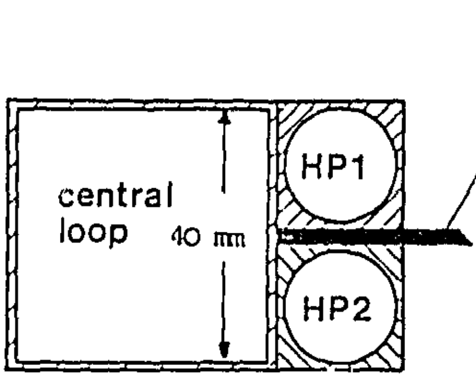

Fig. 2

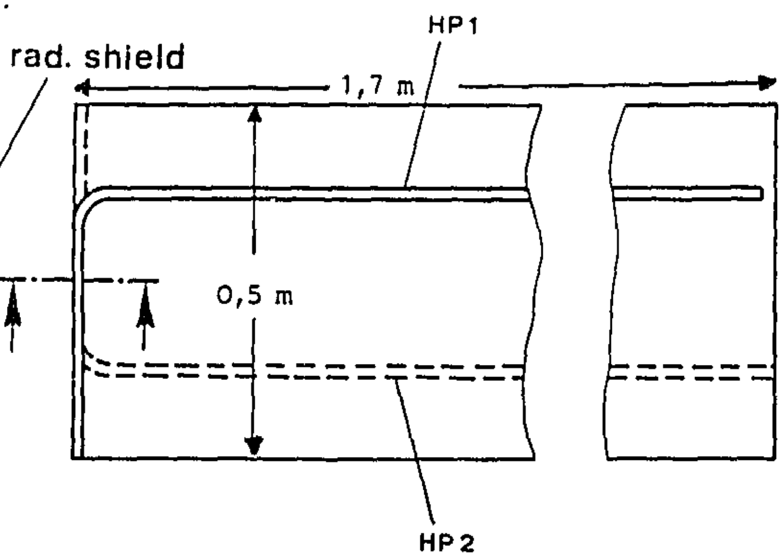

Fig. 1

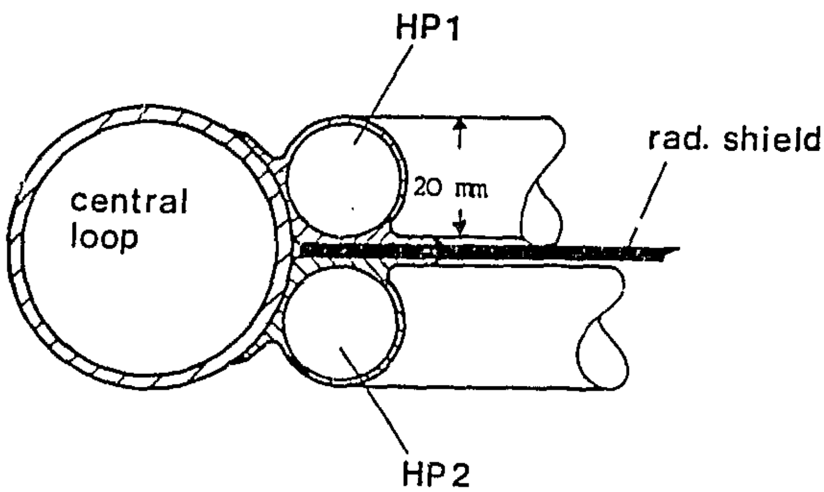

Fig. 3

Fig. 1 shows the radiator module and the heat pipe routing.

Figs. 2 & 3 show two possibilities for the heat pipe contact to the central loop.

The heat pipes must possess a high contact area to the central freon- loop in order to minimize the temperature drop and the radial heat flux density. A long coupling area has therefore been designed. For weight reasons the radiator consists of a single aluminium plate, a honeycomb construction will not be necessary because of the stiffness of the heat pipe profiles which possess integrated fins (see Figs. 4 & 5 - the profiles already available at Dornier). The heat pipes can be welded (on the fin) or bonded with an adhesive to the aluminium radiator plate.

3. Lay-Out Calculations

Radiated heat : $Q_{max} = 450 \text{ W/m}^2$

$Q_{min} = 150 \text{ W/m}^2$

Assumption for Q_{min} : 50% of the surface area (one side) in sunlight, $T_{Rad} = 280$ K; Rad. efficiency included.

Assumption for Q_{max} : no sun; $T_{Rad} = 323$ K; Rad. efficiency included.

Maximum heat radiation per panel :

$Q_{max} = 450 \text{ W/m}^2 \cdot 1{,}7\text{m} \cdot 0{,}5\text{m} \cdot 2 \text{ (sides)} = \underline{765 \text{ Watt}}$

$Q_{min} = 150 \text{ W/m}^2 \cdot 1{,}7\text{m} \cdot 0{,}5\text{m} \cdot 2 \text{ (sides)} = \underline{255 \text{ Watt}}$

Maximum heat load on one heat pipe : <u>385 W</u>

Minimum heat load on one heat pipe : <u>130 W</u>

Desired heat pipe performance : <u>400 Wm</u>

Material combination : Aluminium/ammonia of the heat pipes.

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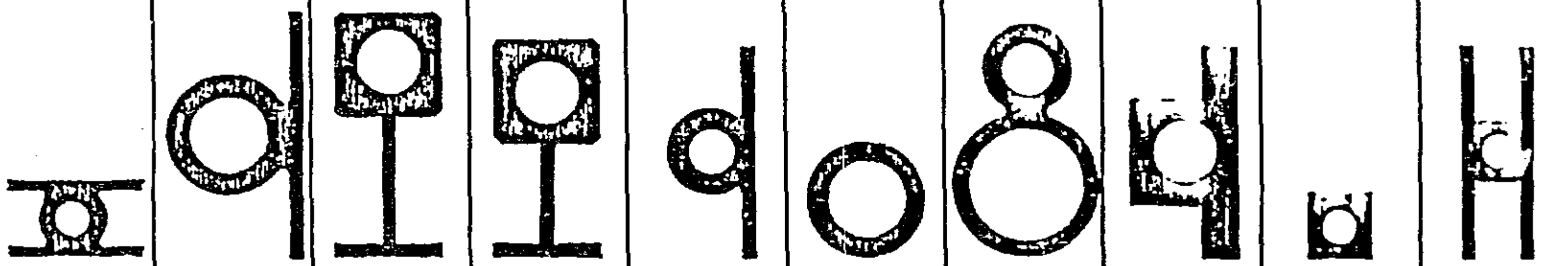
	WR 1a	WR 1b	WR 1c	WR 2	WR 3	WR 4	WR 5	WR 6	WR 7	WR 8
Rillentiefe A [mm]	0,7	0,9	1,2	0,9	0,9	0,9	0,9	0,8	0,8	1,0
Rillenbreite B [mm]	0,5	0,7	0,9	0,7	0,7	0,7	0,7	0,5	0,5	0,5
Rillenanzahl NR	24	18	12	30	30	18	30	20	20	20
Innendurchmesser [mm]	8,6	8,2	7,6	13,2	16,2 x 4,2	8,2	16,2 x 4,2	5,4	5,4	5,0
Außendurchmesser [mm]	12,0	12,0	12,0	18,0	20,0 x 8,0	12,0	20,0 x 8,0	12,0 x 12,0	12,0	9,0
Findicke [mm]	–	–	–	–	–	1,5	2,0	–	1,5	1,5
Finbreite [mm]	–	–	–	–	–	30,0	40,0	–	30,0	30,0
Profilmasse [kg/m]	0,122	0,129	0,140	0,266	0,200	0,235	0,309	0,300	0,333	0,200
Minimale Bauhöhe [mm]	12,0	12,0	12,0	18,0	8,0	12,5	8,0	12,0	12,0	9,5

Fig. 4 : Dornier System Axial Grooved Heat Pipe Profiles



	WR 9	WR 10	WR 11	WR 12	WR 13	WR 14	WR 15	WR 16	WR 17	WR 18
Grooves depth A [mm]	1,0	1,4	1,4	1,4	1,5	1,5	1,5	1,4	1,0	1,0
Grooves width B [mm]	0,5	0,8	0,6	0,6	0,7	1,0	0,7	0,6	0,5	0,5
No of Grooves NR	20	32	32	32	22	24	22	32	20	20
Inner diameter [mm]	5,0	11,2	9,2	9,2	7,0	10,6	7,0	9,2	5,0	5,0
Outer diameter [mm]	9,0	17,0	15,5×15	15,5×15	12,0	16,0	12,0	15,0×30,0	9,0×9,5	9,0
Fin thickness [mm]	1,5	2,0	2,0	2,0	1,5	–	–	5,0	–	1,8
Fin width [mm]	30,0	35,0	15,0	15,0	30,0	–	–	30,0	–	30,0
Mass [kg/m]	0,335	0,335	0,604	0,572	0,250	0,210	0,465	0,5	0,150	0,34
Min height [mm]	10,0	17,5	35,0	31,0	12,5	16,0	32,2	16,0	9,0	9,5

Fig. 5 : Dornier System Axial Grooved Heat Pipe Profiles (cont.)

The already existing (and space-qualified) profile DS-WR 10 possesses a performance with ammonia as heat carrier of about 300 Wm (open axial grooves). Therefore a heat pipe with a slightly higher performance must be developed or the heat pipe routing according to Figs. 1 and 2 must be provided with 2 heat pipes accordingly. This point will not be a basic problem area.

Radial heat flux :

Contact area length : 40 cm

"   " width : 2 cm

$\dot{Q}_{Radial\ max} = \frac{385}{80}\ W = \underline{4,8\ W/cm^2}$

A heat flux density of 6 W/cm^2 has already been qualified for heat pipes in the L-SAT programme. The max. temperature drop is about 5 K.

Weight :
- o Weight of one heat pipe with integrated fin (according to DS-WR 10) : = 0,350 Kg/m
- o Weight of 2 heat pipes on one Module : 1,5 Kg
- o Weight of the radiator panel : 3,5 Kg (surface : 1,7 m x 0,5 m)
- o Connection parts : 0,250 Kg
- o Total weight of one module : <u>5,250 Kg</u>

4. Overall arrangement of the radiator

The radiation of one module without sun is about 600 Watt (average value) and about 425 Watt with full sun on one panel side (average value). Therefore a heat dissipation of 25 KW can be reached with a number of modules between 42 and 59. The arrangement can be done according to Fig. 6. The attachment of 60 modules (30 double modules) leads to an attachment length of 15 m if both sides of the loop are provided with these radiator modules (Fig. 6). The total length of the central loop in the radiator area will be about 20 m.

5. Radiator packages

Before assembly the radiator modules have to be stowed in 4 packages with 1,75 x 0,5 x 0,5 m each. Because each module possesses an outer shape of 1,75 x 0,5 m, the modules may have a thickness of less than 3,3 cm (60 modules). A solution is sketched in Fig. 7. Here we reach about 20 modules per package or 80 modules in 4 packages.

6. Redundancy aspects

Each module possesses a certain redundancy because a failure of one heat pipe does not mean a failure of the entire module but a certain temperature drop of the module and therefore a reduced amount of radiation heat.

A failure of one module does not influence in any way another module. Nevertheless, some spare modules may be connected to the central loop for redundancy reasons.

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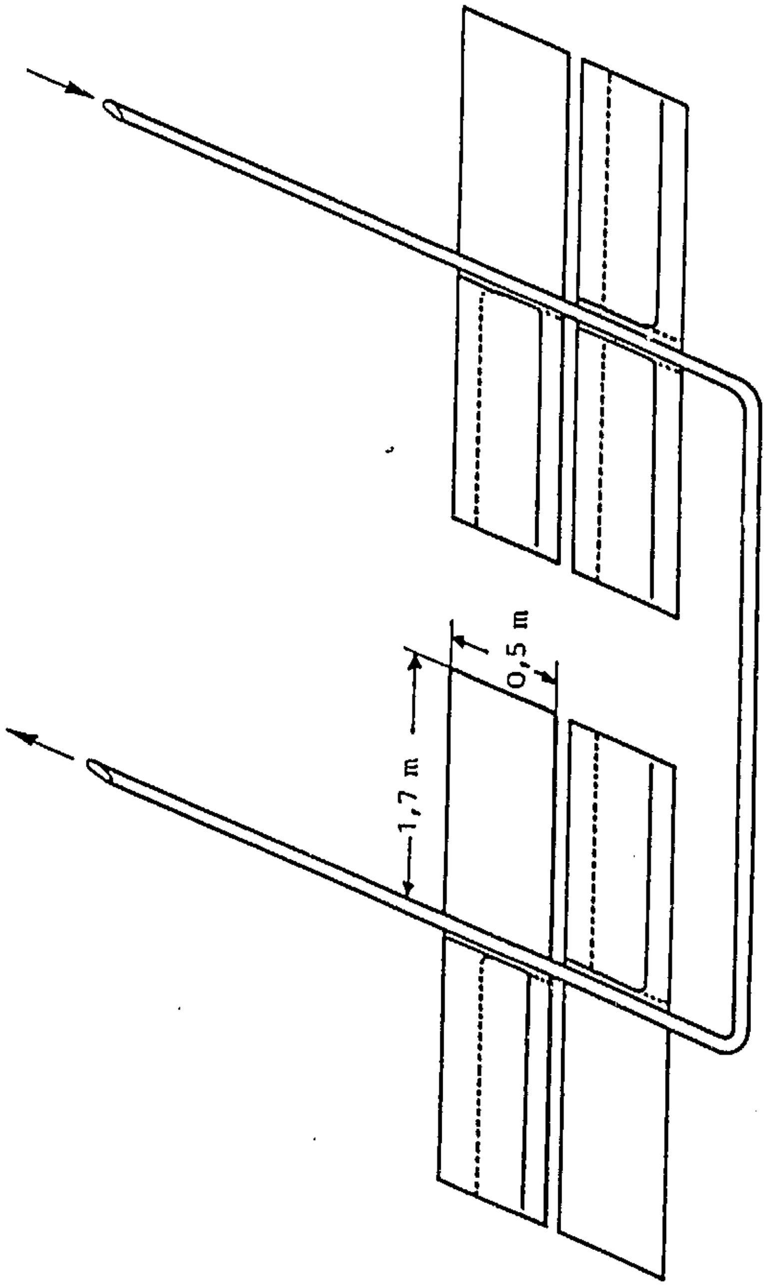

Fig. 6: Principal Modular Heat Pipe Radiator Configuration

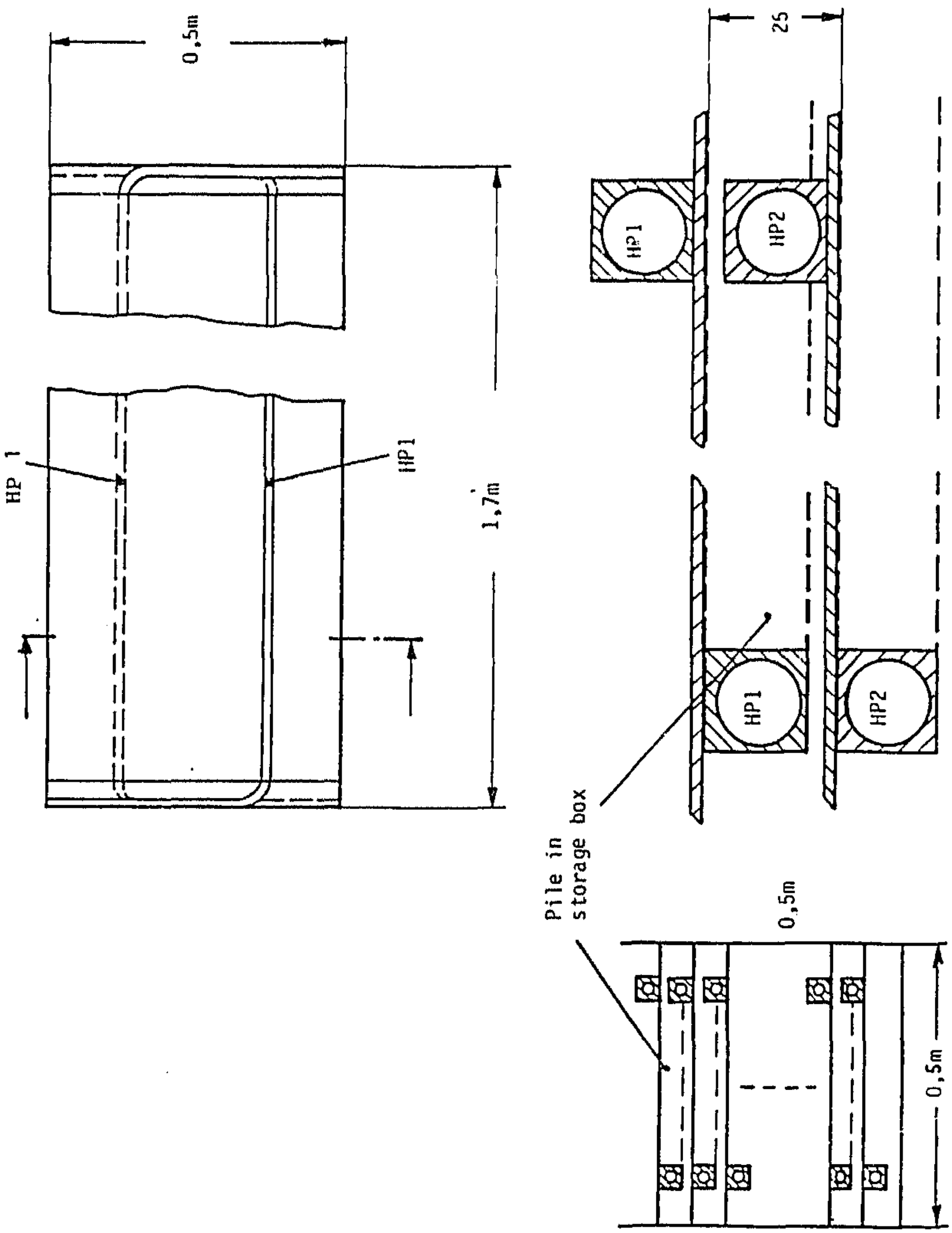

Fig. 7: Module Configuration and Packaging Concept

Without any great effort in designing and manufacturing, such a radiator module may be provided with gas-stabilized heat pipes (VCHP) so that these modules serve not only for radiation but also for the temperature stabilization without any active electrical system such as a heater or controller.

The most critical part is the central cooling loop system itself (no redundancy) and possibly the attachment of the individual modules.

7. Overall Configuration

A conceptual layout of a modular heat pipe radiator installed on the Space Station solar array boom was envisaged (Fig. 8). The central loop system is attached to the boom and the individual radiator panels are mounted to the central loop by means of e.g. a manipulator system. In case of damage to a panel it can easily be disassembled and replaced.

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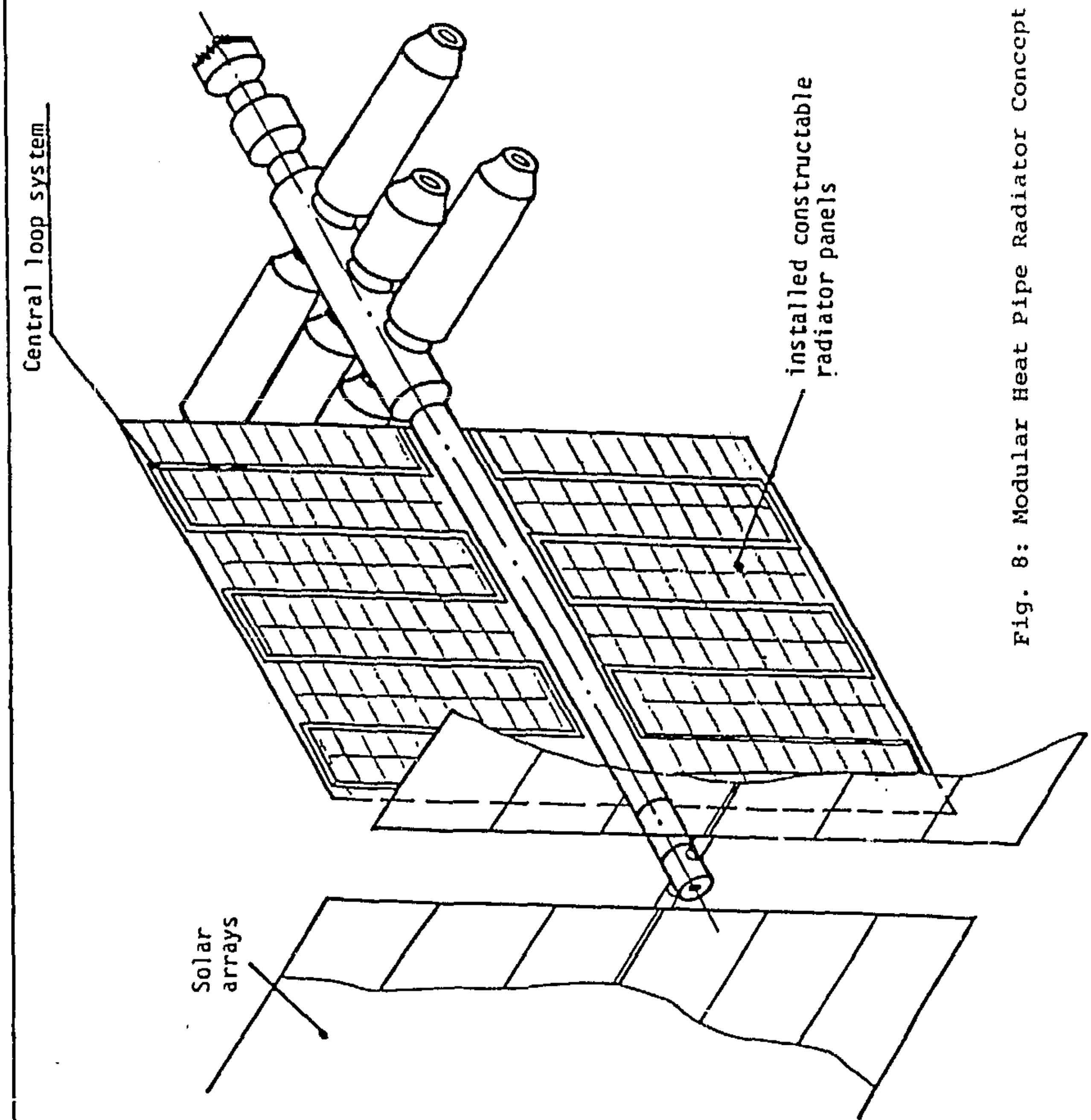

Fig. 8: Modular Heat Pipe Radiator Concept

8. Mecnanical Attachment of Modular Radiator Panels

Several possible attachment mechanisms have been studied and a preliminary concept (Fig. 9) was selected.

Following major design features were considered:

- Mounting by means of manipulator,
- Good contact between thermal saddle and central loop,
- Attachment of panels to both sides of central loop,
- Capability to remove a panel if required, and
- Maintaining envisaged envelope of foreseen packaging concept.

In the selected concept (Fig. 9) the radiator is moved to the central loop 1 by the manipulator, and then slided longitudinally 2 along the central loop to insert the attachment plates into the housings mounted on the radiator panel.

The shapes of the central loop plates and the radiator panel housings (Fig. 10) have been selected such as to ensure a good contact between the thermal saddle and the central loop pipe by means of a positive pressure between the two, when the radiator panel is installed. Compatibility of the materials used for easy insertion and especially for removal after continuous contact must be foreseen.

A locking device to prevent longitudinal movement of the panels has been forseen (Fig. 11). The locking device is actuated by the manipulator arm when holding the panel for installation or removal.

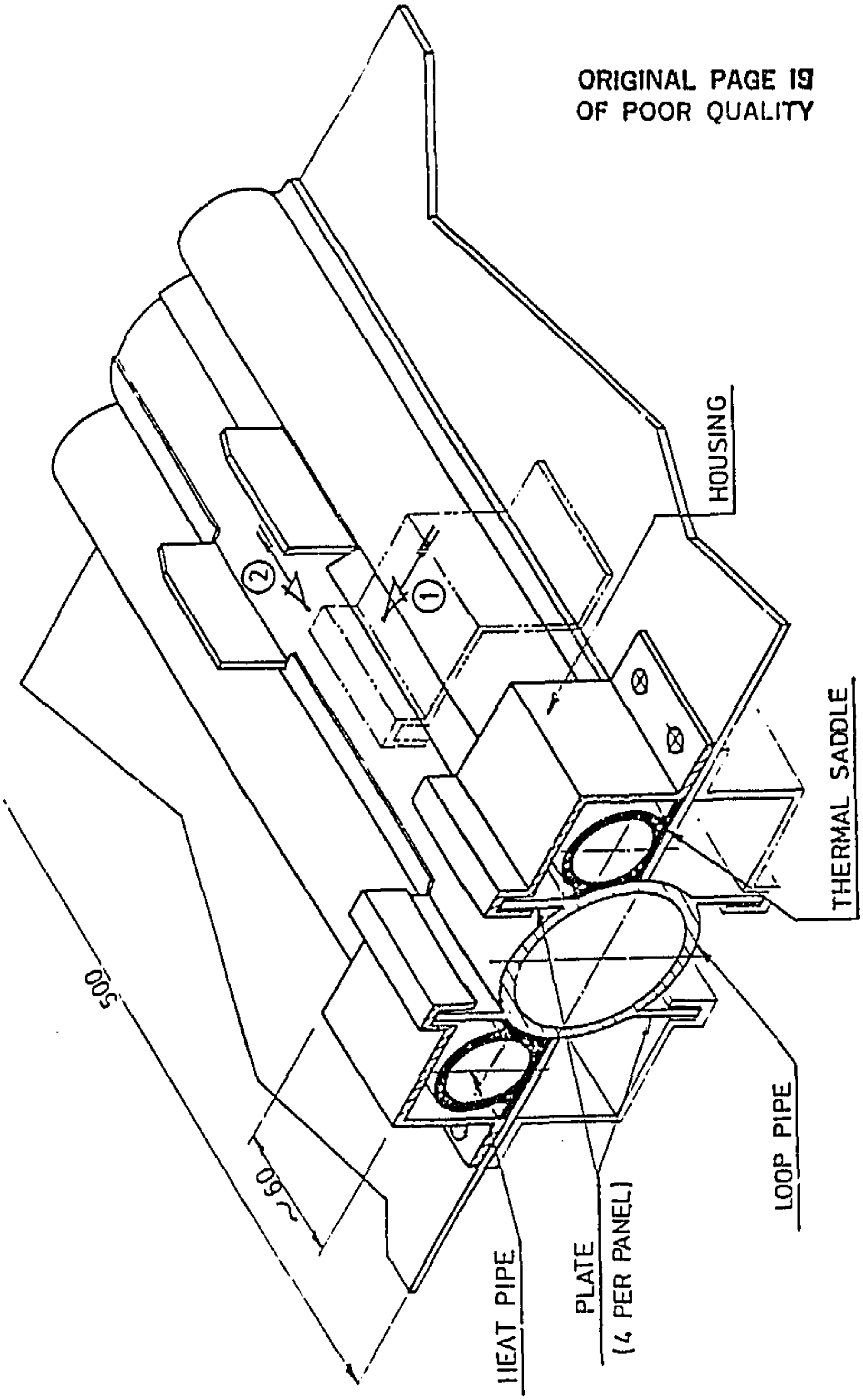

Fig. 9: Modular Radiator Panel Attachment Concept

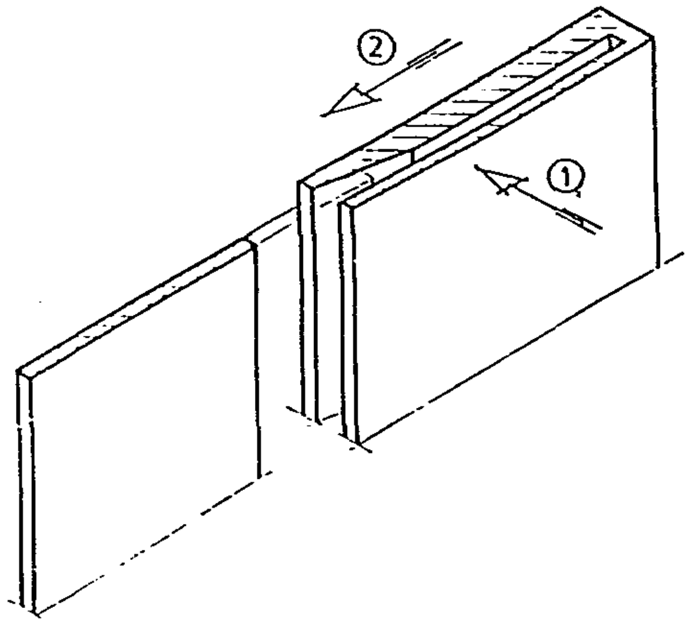

Fig. 10: Plate and Housing of Attachment Mechanism

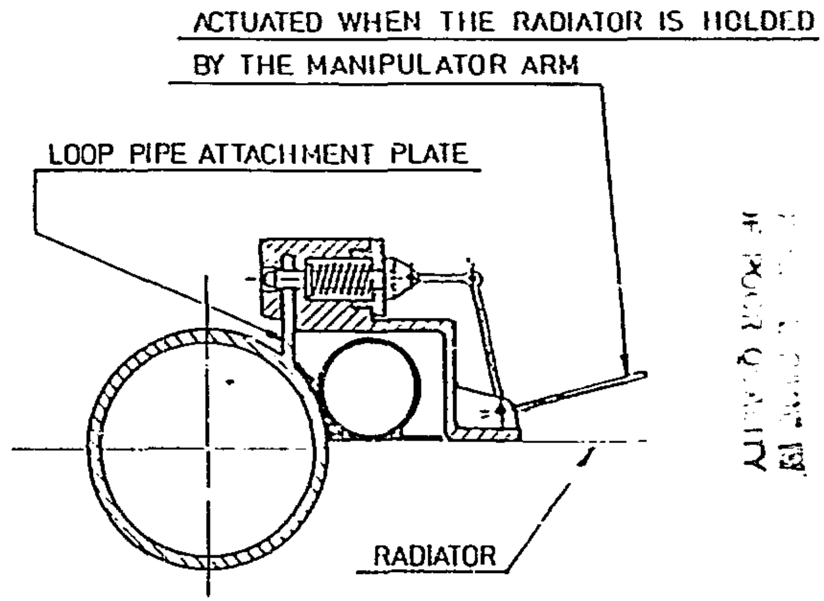

Fig. 11: Locking Device

7.4.10 Interface Requirements and Standardization

5.2 Interface Requirements/Standardization

The principal interfaces for the four basic classes of User and Transportation Vehicles or Facilities as summarized in Figure 5.2-1 were examined.

The objective was directed toward determining the services and configuration driver influences upon the Space Station system.

Three principal influences were defined.

(1) The requirement for a standard docking/berthing interface.
(2) OTV hangar facility (unpressurized).
(3) Propellant storage and transfer.

5.2.1 Module to Module

The nature of a Modular Space Station concept dictates a requirement for standardization of at least the module to module interfaces and the berthing and docking interfaces. This allows the highest operations utility since any module may be placed either permanently or temporarily at any port location. Table 5.2-1 summarizes the basic interfaces requirement areas for Space Station Transportation and User Interfaces. Programmatically, greater flexibility is gained in the growth configurations by being able to change the planned growth sequence while not being constrained to a particular sequence and distinct module positions. This standard interface is not easy to achieve and represents a complex mechanical, electrical and fluid interface. During the SOC study, Rockwell examined the orbiter to Space Station interface and the module to module interface to basically determine if the 40 inch hatch allows enough peripheral area to contain the mechanical and electrical connections. This is shown in Figure 5.2-2.

In the Needs, Attributes and Architecture Study, Hamilton Standard analyzed a Space Station buildup sequence to determine the equipment and requirement growth as the space station is assembled. Figure 5.2-3 shows the interconnect arrangement at the end of a six STS flight buildup operation. This analysis indicates that from an ECLSS point of

TABLE 5.2-1 USER & ON-ORBIT TRANSPORTATION INTERFACES

OTV/SPACE STATION INTERFACE

- HANGAR
- ELECTRICAL POWER
- ACTIVE THERMAL
- SERVICING & CHECKOUT
- EVA/IVA CREW SUPPORT
- MAINTENANCE
- COMMUNICATIONS & TRACKING
- GUIDANCE & NAVIGATION

TELEOPERATOR MANEUVERING SYSTEM (TMS)

- STRUCTURAL INTERFACE
- ELECTRICAL POWER
- SERVICING & CHECKOUT
- EVA/IVA CREW SUPPORT
- MAINTENANCE
- COMMUNICATIONS & TRACKING
- GUIDANCE & NAVIGATION

PLATFORMS & FREE FLYERS

- STRUCTURAL INTERFACE
- ELECTRICAL POWER
- ACTIVE THERMAL
- EVA/IVA CREW SUPPORT
- MAINTENANCE
- COMMUNICATIONS & TRACKING
- GUIDANCE & NAVIGATION

ATTACHED MISSION FACILITY (PRESSURIZED)

- STRUCTURAL INTERFACE
- ELECTRICAL POWER
- ACTIVE THERMAL CONTROL
- ECLSS
- IVA CREW SUPPORT
- ORIENTATION

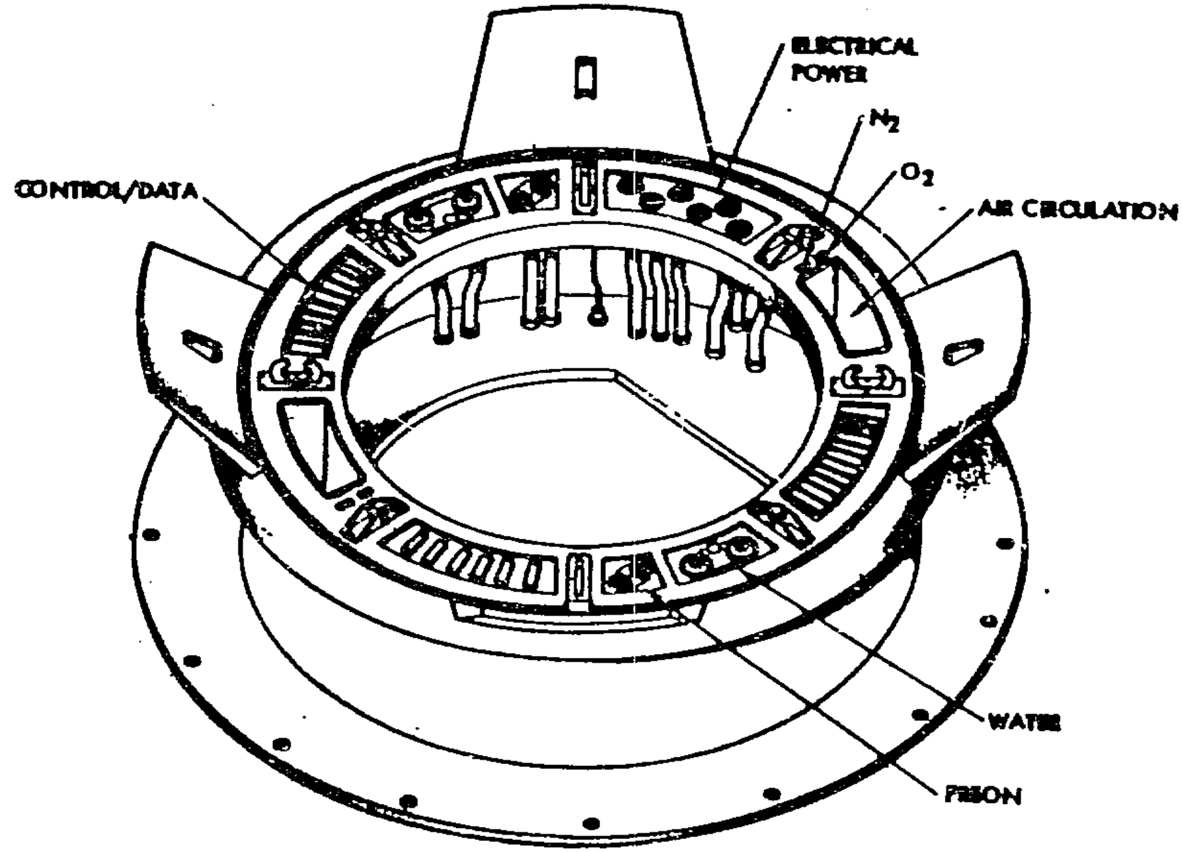

Figure 5.2-2A. Spacc Station Module to Module Interface

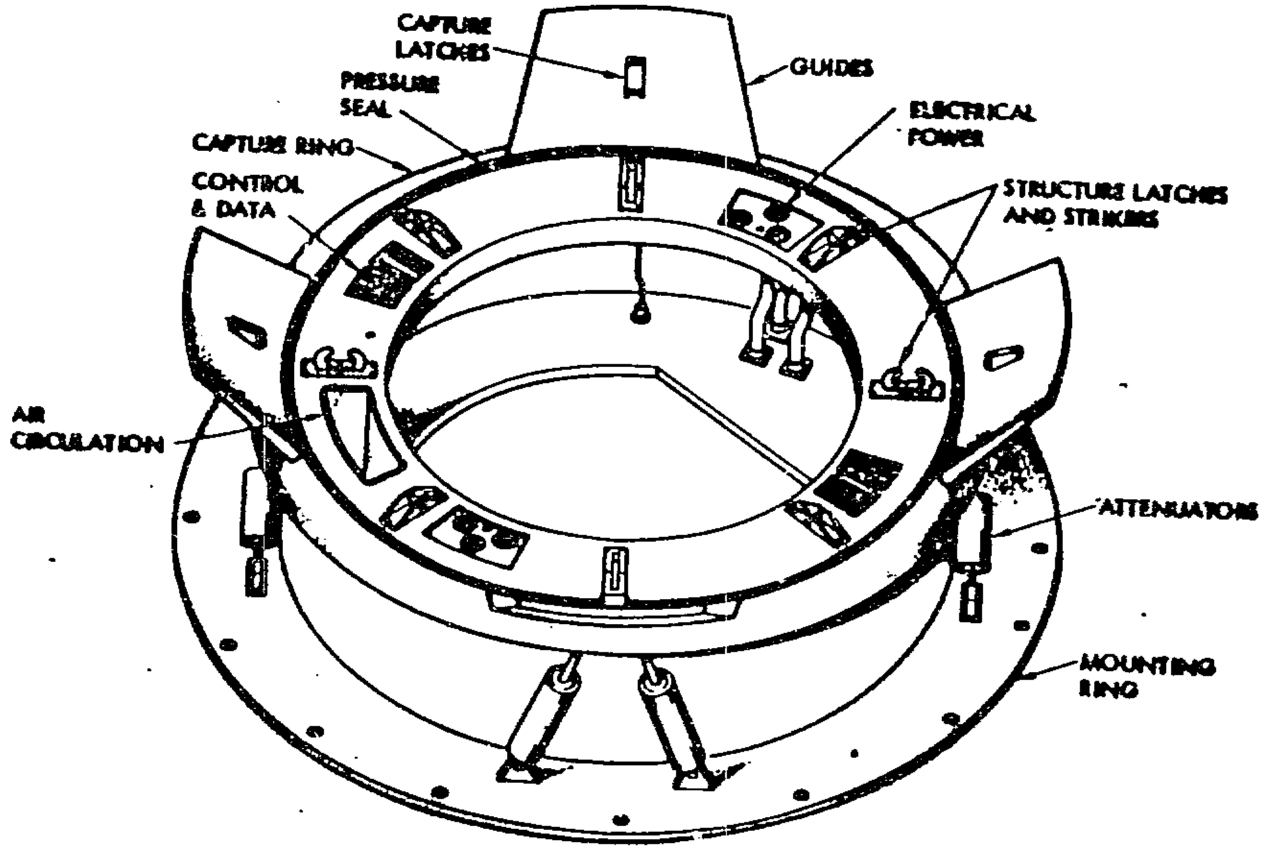

Figure 5.2-2B. Orbiter to Space Station Docking Interface

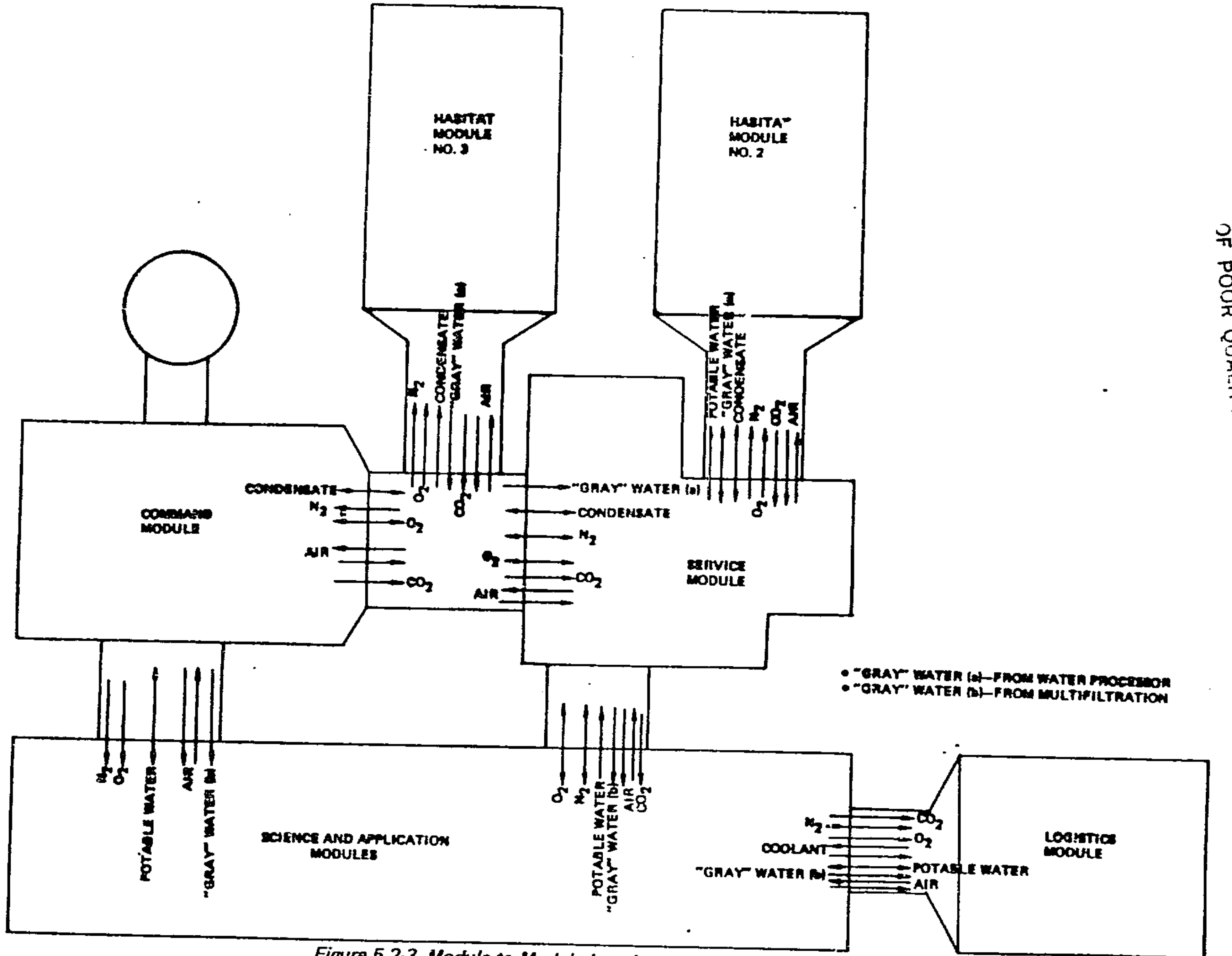

Figure 5.2-3 Module to Module Interfaces After Launch No. 5

view, a standard interface concept is viable and should be pursued further. A greater depth of study across all subsystem areas is required to define and size the interface.

The next phase of the Space Station study should stress the definition of the design of this standard interface port. Involved is not just the mechanical/structural layout but the maintenance aspects for the fluid and electrical disconnects and data ports. The ability to replace and leak test the hazardous fluid disconnects while confining the fluid lines and disconnects to be isolated outside the life support volume is a desirable requirement. This task is complicated further by the requirement for compatibility with the shuttle docking interface and its attenuation features. Some work was done in early 1970's Space Station studies and in the SOC study. Some of that work may have application.

The requirements for this key standard interface is as follows:

(1) It shall be compatible with the Space Station shuttle docking interface.

(2) The standard interface shall have reserved locations for each standard component.

(3) If necessary, it would be permissible to retain a capability to rerig an interface within reserved locations for components.

(4) It is desirable to operate the hazardous fluid lines outside the life support volume with capability for maintainable and test by an IVA crewmember.

With this concept, it would not be necessary that all interfaces carry the 100% hardware standard interface. It would be required that the interface be easily converted to a standard interface. With this modified requirement, it would relieve considerable system cost pressure while retaining the standard interface flexibility.

This concept variation would not be free because it would involve storing several berthing port/docking port hardware sets on orbit ready for reconfiguration as well as tools and the crew training for performing these tasks.

As an aid to the designers, a mockup should be initiated to aid the design and be updated at the end of the study to illustrate the selected design and its utility for maintenance and rigging.

5.2.2 Operating Interfaces

The Space Station system operating interfaces are divided into two classes, (1) Transportation and Servicing Systems and (2) User Elements as defined in Figure 5.2-5.

5.2.2.1 Transportation and Servicing Systems

The STS System is the logistics support of the Space Station System. In this role it delivers the logistic module on a regular basis to resupply the basic needs of the station and crew, it delivers construction equipment and additional modules in support of the on-orbit system growth. Additionally, it delivers user payload elements for installation on or servicing by the space station. In this role it prefers several docking locations on the configuration to allow maximum use of the Orbiters manipulator in berthing its cargo on the Space Station or parking it externally. In general, equipment transported to be installed inside the pressurized space station areas will be transported in the logistics module. Berthing and docking interfaces and their proximity to one another is important to the up and down transportation system.

The shuttle docking ports interface locations are dictated by approach path clearance, plume impingement limitations, and configuration orientation and control considerations. Additional operations considerations are involved in the delivery and return of station equipment, modular elements, construction equipment, and logistics module resupply. The ability to offload and load the orbiter cargo bay efficiently depends on the operating relationship between the orbiter manipulator and the Space Station manipulator in order to efficiently and safely berth or locate the cargo elements on the Space Station and effectively reload the orbiter with down cargo elements.

The growth Space Station progresses into operations supporting Ground Launched and Space Based OTV's. These operations introduce to the system the requirement to base large quantities of propellants in orbit, transport propellants to orbit and transfer of propellants from the orbiter to the on orbit storage tank and transfer of proellant to fuel the OTV's in the course of GEO OTV operations. The interfaces become predominately external to the Space Station pressurized volume. There are propellant storage tanks,

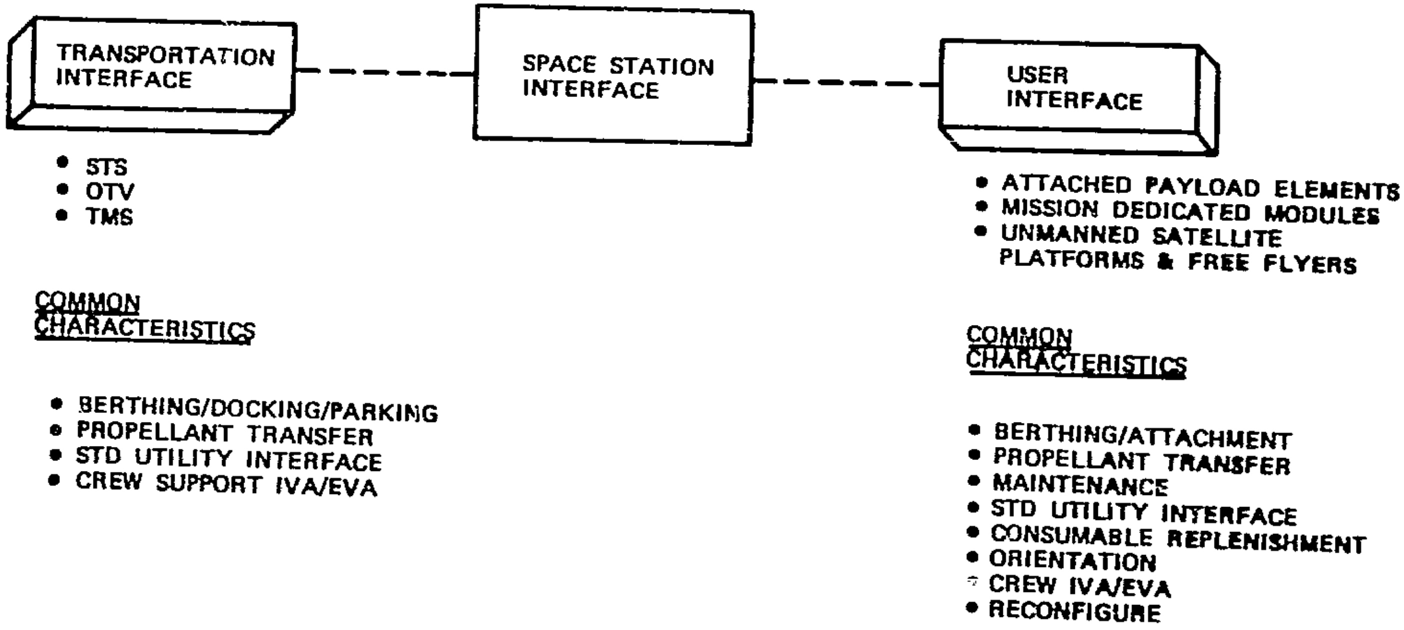

Figure 5.2-5. Space Station Interface Characteristics

hangars for OTV assembly, checkout and protection and areas for storage of equipment and OTV payloads.

The Space Station subsystem support interface for the on-event transportation and servicing systems is not over demanding and involves support from electrical power, thermal, navigation & tracking, and communications as shown in Table 5.2-1.

5.2.2.2 User Elements

The user elements are defined as all of those useful payloads that require Space Station as an experiment platform, a servicing station, or a repair station and reservice station. These user elements can be installed in the Space Station in available laboratory or equipment space pressurized or unpressured or be delivered by the shuttle orbiter as a pressurized module installed on a berthing port.

The third class of user elements is unmanned platforms or free-flyers delivered by the shuttle orbiter and parked or berthed temporarily while servicing and checkout operations are performed under control of the Space Station crew.

These three classes of user elements require radically different interface support from the Space Station.

The internally installed user elements require structural mounting, electrical power, data management and an occasional space station reorientation.

This second element class, the mission dedicated module requires a standard berthing port for a semi-permanent location and becomes an integral part of the Space Station since it is pressurized and requires considerable IVA crew support. Subsystem interface support is supplied in the form of Electrical power, ECLSS, communications, data management and active thermal control. In performance of its mission pointing and pointing stability is usually required supplied by Space Station short term or long term orientation.

The third element, Satellite Platforms and Free-Flyers utilize the Space Station principally as a staging area. Upon delivery to orbit by the STS, a free-flyer is parked or berthed on the Space Station, serviced with propellants, checked out and launched where its operations are monitored by the station periodically or continuously as required. The free-flyer uses the Space Station as its logistics base and periodically requires revisits to the Space Station for servicing and replenishment of consumables and replacement of instruments and equipment in an IVA/EVA maintenance mode.

The Space Station interface is involved in subsystem support of electrical, power, tracking, propellant transfer, active thermal, and navigation and communication. Crew support is heavily involved in IVA checkout and servicing tasks and EVA servicing and maintenance tasks. The Space Station structural and mechanical interface becomes one of supplying berthing ports and hangar work locations suitable for maintenance, transfer of propellants and checkout and monitoring of subsystems by the crew.

5.2.3 Orbital Transfer Vehicle Servicing

The Growth Space Station will serve as a space servicing base for ground based and space base orbital transfer vehicles and their payloads principally supporing GEO Satellite Operations from LEO. These vehicles require the following interface and services.

- Propellant Storage
- Assembly/Storage Hangar
- Electrical Power
- Active Thermal
- Payload C/O & Servicing/Storage
- IVA/EVA Crew Support
- Vehicle Assemble, Checkout and Servicing

The center of this basing concept is the unpressurized hangar and propellant storage and transfer capability. The Boeing concept approach deviates from that proposed by the Space Operations Study being performed by MDTSCO in these areas.

A separate launch and fueling platform is not recommended because of its high cost implications. The pressurized hangar has not been adopted because of its complexity, potential failure modes and associated high costs.

5.2.4 Conclusions

There is an early need to baseline interfaces in two principal areas:

(1) Berthing/Docking Ports

(2) Hargar Servicing/Maintenance/Checkout

5.2.4.1 Standardized Berthing/Docking Port

Emphasis should be placed on defining the physical requirements for a standardized berthing/docking port. The objective would be to conceptually design a standard berthing/docking port and derive the requirements for the following subsystems at this interface.

Structures/Mechanical
Docking/Berthing
Electrical Power
Thermal Control
ECLSS
Communications
Data Management

The conceptual designs should test the feasibility of developing a standardized fully maintainable interface that can meet the safety criteria involved.

5.2.4.2 Hangar Servicing/Maintenance/Checkout Interface

The Hangar Servicing/Maintenance/Checkout interface is a predominant factor in the arrangement architecture of the growth Space Station. It is important to the initial station configuration to know the growth restraints. It is recommended that this area of operation with OTV's, Free Flyers, and Platforms be explored in enough detail to define the basic requirements and to estimate a realistic traffic model to derive the number of maintenance/service stations required. The objective would be to develop a greater understanding of the Space Stations architecture sensitivity to this important growth interface area.

APPENDIX I

SUMMARY OF STUDY TASKS AND FINAL REPORT TOPICAL CROSS REFERENCE

SUMMARY OF STUDY TASKS

The study accomplished 3 major objectives:

1. Identified, collected, and analyzed science, applications, commercial, national security, technology development and space operations missions that require or benefit by the availability of a permanently manned space station. The space station attributes and characteristics that will be necessary to satisfy these requirements were identified.
2. Identified alternative space station architectural concepts that would satisfy the user mission requirements.
3. Performed programmatic analyses to define cost and schedule implications of the various architectural options.

Figure A-1 shows the summary task flow that was used to accomplish these objectives.

In Tasks 1.1 thru 1.5, missions were identified, screened, and their needs and benefits analyzed. Mission investigators were assigned to each of the mission classes (science and applications, commercial, technology development, space operations, and national security). In general, these investigators (and their supporting subcontractors) contacted potential users and analyzed available data to characterize potential mission needs. They worked in conjunction with designers and operations analysts to characterize the potential payloads and operational interfaces. In Task 1.6, the missions were allocated to orbits, and were assigned to platforms, free-flyers, or space stations, as appropriate. During Task 1.7, the various missions were integrated into time-phased mission models. The time-phasing took into account available budgetary constraints, prioritization, time sequencing constraints, and transportation availability. A computer program was used to process the integrated time-phased mission model to derive a year-by-year shuttle manifest schedule. The computer program was also used for Task 1.8 to derive the integrated time-phased space station accommodation requirements, i.e., power and thermal demands, berthing requirements, and crew skills. These mission analyses have been reported in Volume 2 of the final report.

Also included in Volume 2 are the results from Task 1.10. In this task, some of the primary commerical opportunities were examined to define the economics of the use of a space station and to define the benefits of doing business on a space station relative to doing it using the shuttle.

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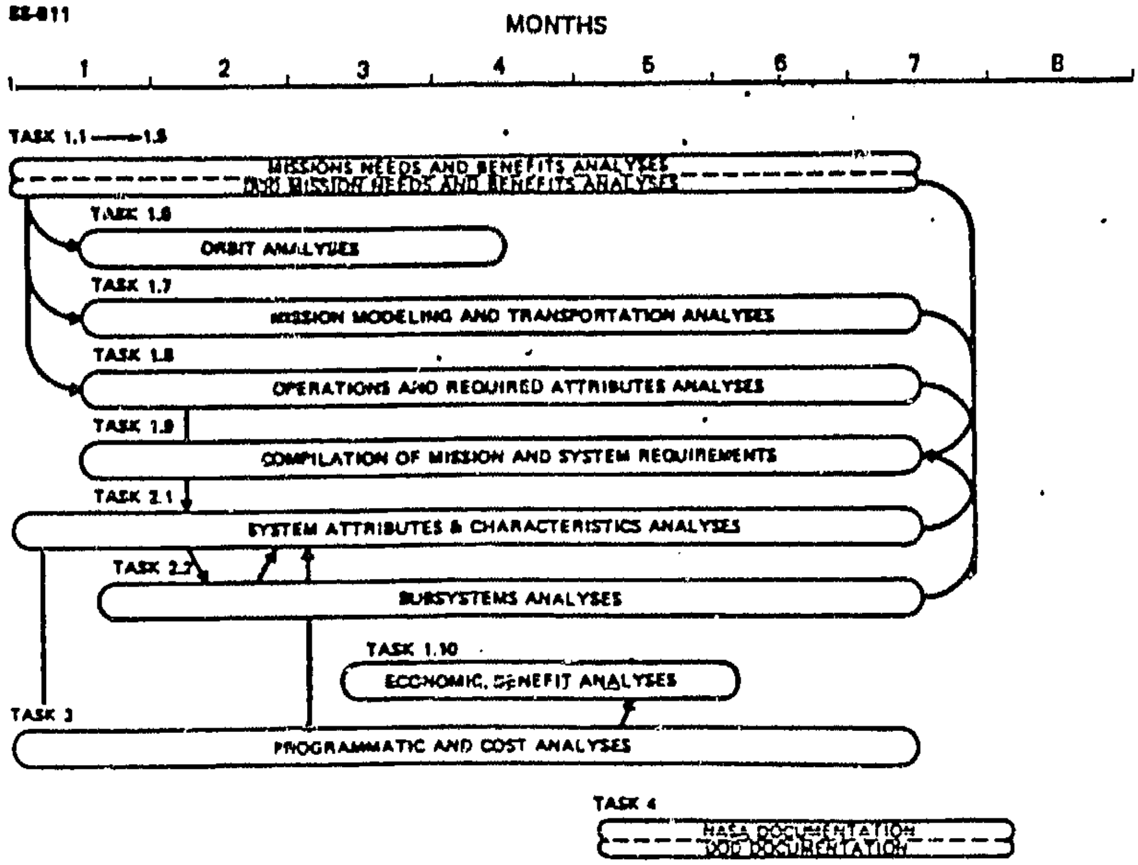

Figure A-1. Summary Diagram Outlines Major Task Traffic

In Task 1.9, mission requirements and space station design requirements were identified. An aggregate of these requirements are reported in Volume 3.

Volume 4 of the final report contains the results from Tasks 2.1, 2.2 and 3. Specifically in Task 2.1, a methodology for defining realistic architectural options was established. This methodology was applied using the requirements defined in the previous tasks. From this, we have created 3 architectural options and have shown some reference space station configuration concepts for each architectural option. Task 2.2 was performed to obtain analysis and trades of some of the principle subsystems, i.e., data management, environmental control and life support, and habitability. Task 3 provides the analyses of programmatics and cost options associated with the concepts derived during the study.

A cross reference guide to enable locating study topics within the volumes and volume sections of the final report is presented in Table A-1.

TABLE A-1

Final Report Topical Cross Reference Guide

Topic	Vol. 1 Exec Summ	Vol. 2 Mission Anal	Vol. 3 Rqm'ts	Vol. 4 Archit	Vol. 5 DoD	Vol. 6 Final Brief	Vol. 7-1 Sci/App Data Book	Vol. 7-2 Commer Data Book	Vol. 7-3 Tech Demo Data Book	Vol. 7-4 Archit Data Book	Vol. 7-5 Mission Data Book
Commercial Missions											
o Communication Satellites	o	3.2.1				o		o			
o Reconfigurable											
o Multibeam											
o Materials Proc.	o	3.2.2		I-1.3.2.3, 1.2.2.1		o		o			
o Semiconductors											
o Biological											
o Glass Fibers											
o Earth Observation		3.2.3									
Industrial Services		3.2.4						o			
o Crew Selection & Training											
o In-Space OPS											
Technology Demo's	o	3.3				o			o		
Space Operation	o	3.4				o					
o Construction											
o Flight Support											
o Servicing											

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Science & Applications Missions											
o Space Environment Missions	o	3.1.2				o	o				
o Astrophysics Missions	o	3.1.3				o	o				
o Earth Environment Missions	o	3.1.4				o	o				
o Life Sciences Missions	o	3.1.5				o	o				
o Materials Science Missions	o	3.1.6					o				
Scenarios of Operational Capabilities o Mission Constrained o Station Constrained o No Space Station	o	4.0, 5.0				o					

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Mission Requirements Summary		5.0									o
o Low Inclination Space Station	o	5.2,5.3	3.2.1	I-1.2.2.4		o					o
o High Inclination Space Station	o	5.2,5.3		I-1.2.2.4		o					o
o Platform only	o	5.4				o					o
o Manifesting o Shuttle o OTV o TMS	o	5.2, 5.3, 5.4				o					o
o Crew Size	o	5.2,5.3 5.4	3.2.1			o					o
o Crew Skills		5.2.5.3 3.1.2.5, 3.1.3.5, 3.1.4.5, 3.1.5.5, 3.2.1.5, 3.2.2.6, 3.2.3 3.3		II-2.2.3							o

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Mission Requirements Summary (Continued)											
o Accommodations Reqm'ts	o	2.2	3.2.1			o					o
o Power		5.2,5.3 5.4	1-1.2.1.2, 1.2.2.4 1.2.3.3 1.2.3.4								
o Internal Vol											
o Berthing Ports											
Benefits		6.0									
o Semiconductor Manufacturing	o	6.2				o					o
o Glass Fiber Manufacturing	o	6.3				o					o
o Communications Satellite Assembly	o	6.4				o					o
o Biological Materials Manufacturing	o	6.5				o					o

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Mission Analysis											
o Manifesting Analysis Software	o	2.2				o					o
o Accommodations & Crew Activity Analysis Software	o	2.2				o					o
o Crew Skills											
o Crew Size											
o Berthing Ports											
o Electrical power											
o Internal volume											
Design Requirements											
o Mission Accommodation Reqm'ts		5.0	3.2								
o Interfaces				II-10.0						o	
o Berthing/Docking Port				I-1.3.2.1							
o Hangar		3.3		I-1.3.2.2							

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Architectural Options											
o Architecture Development Methodology	o			I-1.1		o				o	
o Space Station Architectural Options	o			I-1.2		o				o	
Build-up and Growth	o	5.0		I-1.2.3.4, 1.3.1.3, 1.3.2.3, 1.3.3.3							
Data Management											
o Architecture				II-3.2						o	
o In-Flt Checkout				II-3.3						o	
o Space-Ground Integration				II-3.4						o	
o Ground Lab				II-3.5						o	
o Software Devel.				II-3.6						o	
o Hardware Stds				II-3.7						o	
o Software Stds				II-3.8						o	
o Verif/Valid.				II-3.9						o	

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Logistics/Resupply											
o Logistics Module				II-7.1, 7.3,7.4							
o Resupply Reqm'ts				II-7.2							
Environmental Control and Life Support Subsystem				II-5.0						o	
o ECLS Evolution				II-5.2.1, 5.3.2						o	
o Safe Haven Logistics Module				II-5.2.1						o	
o Air Revitalization System				II-5.0,5.3.2						o	
o Water Revitalization System				II-5.0,5.3.2						o	
o Performance and Loads Specification										o	
o Overboard Venting				II-5.2.1,5.2.2						o	
o Architecture				II-5.2.1						o	
o Water Recovery System				II-5.0,5.3.2						o	
o CO_2 Concentration				II-5.0,5.3.2						o	
o Regenerative-Fuel-Cell-Based ECLS				II-5.0,5.2.1, 5.3.2						o	
o Recommendations				II-5.0, 5.3.2						o	
EVA/EMU				II-5.0, 5.2.2						o	

TABLE A-1

Final Report Topical Cross Reference Guide

Topic	Vol. 1 Exec Summ	Vol. 2 Mission Anal	Vol. 3 Rqm'ts	Vol. 4 Archit	Vol. 5 DoD	Vol. 6 Final Brief	Vol. 7-1 Sci/App Data Book	Vol. 7-2 Commer Data Book	Vol. 7-3 Tech Demo Data Book	Vol. 7-4 Archit Data Book	Vol. 7-5 Mission Data Book
Communications & Tracking Subsystem			3.2.2.1.11	II-4.0						o	
Manipulator System				II-6.0						o	
Pointing Systems				II-8.0						o	
Thermal Management				II-9.0						o	
Crew				II-2.0							
o Tasks				II-2.2						o	
o Skills		5.2.5.3 3.1.2.5, 3.1.3.5, 3.1.4.5, 3.1.5.5, 3.2.1.5 3.2.2.6, 3.2.3 3.3		II-2.2.3							
o Capabilities				II-2.2.2						o	
o Role Relationships				II-2.3.2						o	
o Accommodations			3.2.2.1.11	II-2.4						o	

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Final Report Topical Cross Reference Guide

Topic	Vol. 1 Exec Summ	Vol. 2 Mission Anal	Vol. 3 Rqm'ts	Vol. 4 Archit	Vol. 5 DoD	Vol. 6 Final Brief	Vol. 7-1 Sci/App Data Book	Vol. 7-2 Commer Data Book	Vol. 7-3 Tech Demo Data Book	Vol. 7-4 Archit Data Book	Vol. 7-5 Mission Data Book
Crew (Continued)											
o Habitability	o		3.2.2.1.11	II-2.0,2.4						o	
o IVA Work Stations				II-2.5.2						o	
o EVA Work Stations				II-2.5.3						o	
o Maintenance				II-5.2.2 II-2.5.4						o	
o Stowage			3.2.2.1.11							o	
o Windows			3.2.2.1.11	II-2.4.1						o	
o Hygiene			3.2.2.1.11	II-2.4.2.4						o	
o Scheduling			3.2.2.1.11	II-2.3.1						o	

APPENDIX 2

KEY TEAM MEMBERS

KEY TEAM MEMBERS

Subject	Boeing Team	Subcontractor Team	
Study Manager	Gordon Woodcock	ADL:	Dr. Peter Glaser
		Battelle:	Kenneth E. Hughes
		ECON:	John Skratt
		ERIM:	Albert Sellman
		Hamilton Standard:	Harlan Brose
		Intermetrics:	John Hanaway
		Life Systems:	Franz Shubert
		MRA:	Col. Richard Randolph (Ret.)
		NBS:	Dr. B. J. Bluth
		RCA:	Dr. Herbert Gurk
		SAI:	Dr. Hugh R. Anderson
Technology Manager	Dr. Richard L. Olson		
Mission Analysis			
Science & Applications	Dr. Harold Liemohn David Tingey (Earth Obs.)	SAI:	Dr. Hugh R. Anderson (Environmental Science)
	Dr. Derek Mahaffey (Mission Integration)		Dr. Peter Hendricks (Meterology/ Oceanography)
	Melvin W. Oleson (Life Sciences)		Dr. Gil Stegen
	Dr. Robert Spiger (Plasma physics, astro-physics, solar physics)		Dr. John Wilson (Life Sciences)
			Dr. Robert Loveless (Integration)
			Dr. Robin Muench
			Dr. Stuart Gorney (Life Sciences)
			Ms. Monica Dussman (Life Sciences)
		ERIM:	Albert Sellman (Earth Obs.)
			Dr. Irvin Sattinger (Earth Obs.)
Commercial	Dr. Harvey Willenberg	RCA:	Dr. Herbert Gurk
			Thaddeus (Ted) Hawkes
		ADL:	Dr. Peter Glaser
		Battelle:	Dr. Kenneth E. Hughes
		MRA:	Col. Richard Randolph (Ret.)
			Robert Pace

KEY TEAM MEMBERS (Cont'd)

Subject	Boeing Team	Subcontractor Team	
Mission Analysis (Cont'd)			
Technology Demonstrations	George Reid Dr. Alan G. Osgood David S. Parkman Steve Robinson Richard Gates Tim Vinopal		
National Defense	Robert S.Y. Yoseph	ERIM:	Mirko Najman
Space Operations	Keith H. Miller		
Architecture and Subsystems			
Architecture & Configurations	John J. Olson Brand Griffin Tim Vinopal David S. Parkman Steve Robinson		
Communications		RCA:	Donald McGiffney
Crew Systems	Keith H. Miller George Reid Dr. Alan G. Osgood	NBS:	Dr. B. J. Bluth
Data Management and Software	Les Holgerson	Intermetrics:	John Hanaway
ECLSS	Keith H. Miller	Ham Std: Life Systems:	Harlan Brose Ross Cushman Al Boehm Ken King Todd Lewis Dr. R. A. Winveen Franz Schubert Dr. Dennis B. Heppner
Operations Analysis	Keith H. Miller George Reid Dr. Alan G. Osgood		
Orbit Analysis	Dani Eder		

KEY TEAM MEMBERS (Cont'd)

Subject	Boeing Team	Subcontractor Team	
Architecture and Subsystems (Cont'd)			
Orbit/Survivability Analysis	Stephen W. Paris Merri Anne Stowe		
C^3I	H. Paul Janes		
Radiation Effects	Dr. William C. Bowman		
Requirements Analysis	Lowell Wiley		
Programmatics & Cost			
Cost Analysis	Ken verGowe	ECON:	Ed Dupnick
Programmatics	Gordon Woodcock		

APPENDIX 3

ACRONYMS AND ABBREVIATIONS

LIST OF ACRONYMS AND ABBREVIATIONS

AAP	Airlock Adapter Plate
AC	Alternating Current
ADM	Adaptive Delta Modulation
AM	Airlock Module
APC	Adaptive Predictive Coders
APSM	Automated Power Systems Management
ACS	Attitude Control System
ARS	Air Revitalization System
ASE	Airborn Support Equipment
BIT	Built in Test
BITE	Built in Test Equipment
CAMS	Continuous Atmosphere Monitoring System
C&D	Controls and Displays
C&W	Caution and Warning
CCA	Communications Carrier Assembly
CCC	Contaminant Control Cartridge
CCTV	Closed Circuit Television
CEI	Critical End Item
CER	Cost Estimating Relationship
CF	Construction Facility
CMG	Control Moment Gyro
CMD	Command
CMDS	Commands
CO_2	Carbon Dioxide
CPU	Computer Processor Units
CRT	Cathode Ray Tube
dB	Decibels
DC	Direct Current
DCM	Display and Control Module
DDT&E	Design, Development, Test, and Evaluation
DOD, DoD	Department of Defense
DT	Docking Tunnel
DM	Docking Module
DMS	Data Management System
DSCS	Defense Satellite Communications System
ECLSS	Environmental Control/Life Support System
EDC	Electrochemical Depolarized CO_2 Concentrator
EEH	EMU Electrical Harness
EIRP	Effective Isotropic Radiated Power
EMI	Electromagnetic Interference
EMU	Extravehicular Mobility Unit
EPS	Electrical Power System
ET	External Tank
EVA	Extravehicular Activity
EVC	EVA Communications System
EVVA	EVA Visor Assembly
FM	Flow Meter
FMEA	Failure Mode and Effects Analysis
ftc	Foot candles
FSF	Flight Support Facility
FSS	Fluid Storage System
GaAs	Gallium Acsenide

LIST OF ACRONYMS AND ABBREVIATIONS (Continued)

GN&C	Guidance, Navigation and Control
GEO	Geosynchronous Earth Orbit
GHZ	Gigahertz
GPC	General Payload Computer
GPS	Global Positioning System
GSE	Ground Support Equipment
GSTDN	Ground Satellite Tracking and Data Network
GFE	Government Furnished Equipment
GTV	Ground Test Vehicle
HLL	High Level Language
HLLV	Heavy Lift Launch Vehicle
HM	Habitat Module
HMF	Health Maintenance Facility
HPA	Handling and Positioning Aide
HUT	Hard Upper Torso
Hz	Hertz (cycles per second)
ICD	Interface Control Document
IDB	Insert Drink Bag
IOC	Initial Operating Capability
IR	Infrared
IVA	Intravehicular Activity
JSC	Johnson Space Center
KBPS	Kilo Bits Per Second
KM, Km	Kilometers
KSC	Kennedy Space Center
lbm	Pounds Mass
LCD	Liquid Crystal Display
LCVG	Liquid Cooling and Ventilation Garment
LED	Light Emitting Diode
LEO	Low Earth Orbit
LiOH	Lithium Hydroxide
LM	Logistics Module
LPC	Linear Predictive Coders
LRU	Lowest Replaceable Unit
LSS	Life Support System
LTA	Lower Torso Assembly
LV	Launch Vehicle
lx	Lumens
MBA	Multibeam Antenna
mbps	Megabits per second
MHz	Megahertz
MMU	Manned Maneuvering Unit
MM-Wave	Millimeter wave
MOTV	Manned Orbit Transfer Vehicle
MRWS	Manned Remote Work Station
MSFN	Manned Space Flight Network
N/A	Not Applicable
NBS	National Bureau of Standards
NSA	National Security Agency
N	Newton
NiCd	Nickel Cadmium
NiH_2	Nickle Hydrogen

LIST OF ACRONYMS AND ABBREVIATIONS (Continued)

Nm,nm	Nautical miles
N/m^2	Newtons per meter squared
OBS	Operational Bioinstrumentation System
OCS	Onboard Checkout System
OCP	Open Cherrypicker
OMS	Orbital Manuevering System
OTV	Orbital Transfer Vehicle
PCM	Pulse Code Modulation
PCM	Parametric Cost Model
PEP	Power Extension Package
PIDA	Payload Installation and Deployment Apparatus
P/L	Payload
PLSS	Portable Life Support System
PM	Power Module
POM	Proximity Operations Module
ppm	Parts per Million
PRS	Personnel Rescue System
PSID	Pounds per Square Inch Differential
RCS	Reaction Control System
REM	Roentgen Equivalent Man
RF	Radio Frequency
RFI	Radio Frequency Interference
RMS	Remote Manipulator System
RPM	Revolutions Per Minute
RPS	Real-time Photogrammetric System
SAF	Systems Assembly Facility
SAWD	Solid Amine Water Desorbed
SPGaAs	Space Produced Gallium Arsenide
scfm	Standard Cubic Feet per Minute
SCS	Stability and Control System
SCU	Service and Cooling Umbilical
SDV	Shuttle - Derived Vehicle
SDHLV	Shuttle - Derived Heavy Lift Vehicle
SEPS	Solar Electric Propulsion System
SF	Storage Facility
SM	Service Module
SOC	Space Operations Center
SOP	Secondary Oxygen Pack
SRB	Solid Rocket Booster
SRMS	Shuttle Remote Manipulative System
SRU	Shop Replacable Units
SSA	Space Suite Assembly
SSME	Space Shuttle Main Engine
STS	Space Transportation System
SSP	Space Station Prototype
STAR	Shuttle Turnaround Analysis Report
STDN	Spaceflight Tracking and Data Network
STE	Standard Test Equipment
TBD	To Be Determined
TDRSS	Tracing and Data Relay Satellite System
TFU	Theoretical First Unit
TGA	Trace Gas Analyzer

LIST OF ACRONYMS AND ABBREVIATIONS (Continued)

TIMES	Thermoelectric Integrated Membrane Evaporation System
TLM	Telemetry
TM	Telemetry
TMS	Teleoperator Maneuvering System
TT	Turntable/Tilttable
TV	Television
UCD	Urine Collection Device
VCD	Vapor Compression Distillation
VDC	Volts Direct Current
VLSI	Very Large Sacle Integrated Circuits
VSS	Versatile Servicing Stage
WBS	Work Breakdown Structure
WMS	Waste Management System