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ASTROPHYSICAL PAYLOAD ACCOMMODATION ON THE SPACE STATION

FINAL REPORT NAS8-36124

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Astrophysical Payload
Accommodation On The
Space Station

Final Report
15 October 1985

Prepared For:

George C. Marshall
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FOREWORD

This final report is submitted in accordance with the requirements of Contract NAS8-36124. This deliverable item is provided in direct response to Section H, Article H-3, Item B, "Final Report" of Contract NAS8-36124. The study was performed at the Allied Bendix Guidance System Division's facility in Huntsville, AL. Most of the effort in support of this study was performed by Bernard P. Woods, who prepared all sections of this report. The study was directed from the Guidance Systems Division in Teterboro, NJ. Mr. Joel Levinthal is the engineering manager at this location. The guidance of Mr. Max Nein and Mr. Joe T. Howell of MSFC during the course of this study is gratefully acknowledged.

This study is one of a group of studies dealing with accommodation of astrophysics payloads on the Space Station which were performed under the general direction and guidance of Dr. David Gilman and Mr. Michael Kiya of NASA Headquarters, Code EZ. The other studies in the series were performed by the Jet Propulsion Laboratory, and the GSFC. The subject of these other studies included pointing requirements and technologies, contamination requirements, data requirements, and accommodation of the Solar Optical Telescope on the Space Station. These studies were performed to support the Space Station Definition and Preliminary Design activity which is presently being performed by NASA and associated contractors.

ABSTRACT

Surveys of potential Space Station Astrophysics Payload requirements and existing pointing mount design concepts were performed to identify potential design approaches for accommodating Astrophysics Instruments from Space Station. Most existing instrument pointing systems were designed for operation from the Space Shuttle and it is unlikely that they will sustain their performance requirements when exposed to the Space Station disturbance environment. The study indicates that the technology exists or is becoming available so that precision pointing can be provided from the Space Station manned core. Development of a "disturbance insensitive" pointing mount is the key to providing a generic system for Space Station. It is recommended that the MSFC Suspended Experiment Mount concept be investigated for use as part of a generic pointing mount for Space Station.

Availability of a shirtsleeve module for instrument change out, maintenance and repair is desirable from the user's point of view. Addition of a shirtsleeve module on Space Station would require a major program commitment.

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1.0 INTRODUCTION

The purpose of this report is to present the findings and results of a study entitled "Accommodation of Astrophysical Instruments in the Space Station System" which was performed by the Allied Bendix Aerospace, Bendix Guidance Systems Division (BGSD) for the George C. Marshall Space Flight Center under contract NAS8-36124.

Although the term Astrophysics generally includes a study of the physical nature and properties of the planets and stars, this study concentrated on the pointing requirements of the solar and stellar instruments, which are potential candidates for Space Station, since the earth looking instruments were being adequately covered by other NASA activities.

The study had the following objectives and tasks:

A) To survey and organize astrophysical experiment pointing and control requirements for Space Station and Space Platform missions, especially with respect to anticipated pointing mount requirements;

B) To compare the experiment requirements to anticipate Space Station requirements with respect to pointing mount design.

C) To consolidate the current concepts for pointing mount designs and innovative ideas presently being pursued at NASA and contractor facilities.

D) To identify new pointing mount design approaches for accommodating future experiments especially with respect to isolation mechanisms;

E) To identify potential techniques that will allow a "shirt sleeve" access to the instrument focal plane structures.

1.1 SUMMARY

The Space Station is required to simultaneously accommodate instruments that have Earth, anti-Earth, Solar and Stellar viewing requirements. Satisfaction of this requirement from the Space Station flying in the baseline local vertical orientation will require a pointing system. The Space Station will provide some articulated attachment points for externally mounted pointing instruments. These coarse pointing/tracking gimbals will provide attitude control between ±1 to ±5 degrees and attitude determination to ±.01 degrees. Maximum angular rates would not exceed ±0.02 degrees/sec. In addition, the Space Station will provide knowledge of the position of the instrument mounting surface relative to the navigation base to ±0.002 degrees. Users or customers with instruments that require higher pointing accuracy will be required to provide their own pointing mounts.

The study surveyed the capabilities of eleven (11) past and present pointing concepts and systems. Most of these systems were designed for a Shuttle application which has a more benign disturbance environment than the Space Station environment is expected to be. It is unlikely that existing systems and concepts will sustain their performance when exposed to the Space Station disturbance environment. The study indicates that the technology is either available or rapidly becming available to design a generic pointing system for Space Station. Development of a disturbance insensitive pointing mount is the key to providing a precise pointing mount for Space Station. With respect to pointing mounts, the following conclusions and recommendations are made:

It is recommended that a trade study be performed to determine the cost effectiveness of requiring the individual instruments to provide precise pointing versus the Space Station providing a gerneric precise pointing capability.

An intrtially reacting pointing system is less sensitive to base body disturbances than are gimbal systems. It is recommended that the MSFC Suspended Experiment Mount (SEM) be investigated for potential use as a pointing mount for Space Station.

Both the Spacelab Instrument Pointing System (IPS) and the Talon Gold Gimbal (TGG) should be considered for use on Space Station because of the maturity of these systems and to realize benefits from the investments previously made.

Availability of a module which will permit instrument change-out, maintenance and repair in a shirt-sleeve environment is desirable from a customer's point of view. Inclusion of a shirt-sleeve module would impact most Space Station subsystems. Addition of a facility of this type would require a major program commitment.

It is recommended that a study be performed to determine the cost impact of providing a "shirt-sleeve environment" maintenance and repair module versus the benefits gained by the customer with respect to longer orbital lifetime, increased observing time and less complicated EVA tasks.

2.0 ASTROPHYSICS PAYLOADS

Several sources were used to identify the potential Astrophysics Payloads and their requirements. These are identified in references [1] through [5]. The major source of information regarding potential payloads was the Langley Space Station Mission Data Base, specifically, the Science and Applications Mission Data Base (SAAX). Both the September 1984 and the May 1985 versions were accessed to identify potential payloads and the payload requirements that would impact pointing mount design. Neither version of the data base contains sufficient information to perform detailed pointing mount design; but, the information from the Langley Data Base (both versions) augmented by other sources, such as the MSFC Pinhole Occulter Facility (POF) Phase A Study Report [6], does provide sufficient information to bound the problem. Table 1 identifies potential astrophysics payloads and whether or not they were identified as such in Reference [1] or in the two versions of the Langley Data Base. These payload lists are intended to illustrate typical astrophysical payloads that the Space Station program may have to satisfy. Some of the differences between the payloads identified in the two versions of the Langley Data Base can be explained. The POF (SAAX0009) is included as part of the Advanced Solar Observatory (SAAX0011). The X-Ray Timing Explorer (SAAX0014), the Far UV Spectroscopy Explorer (SAAX0019), and the Solar Corona Diagnostic Mission (SAAX0010) are the payloads for the Leased Platforms (Explorer) 1 through 3 (SAAX0026, SAAX0027, SAAX0028).

The candidate payloads can be classified as: attached payloads, payloads located on co-orbiting free flying platforms, and free flying satellites, which would be serviced by Space Station. Table 2 identifies the performance location for each payload on the composite list.

TABLE 1. POTENTIAL ASTROPHYSICS PAYLOADS				
CODE	PAYLOAD	REF. CONF. DOC.	SEP.'84 LANGLEY BASE	MAY'85 LANGLEY BASE
SAAX0001	SPECTRA OF COSMIC RAY NUCLEI		X	X
SAAX0004	SHUTTLE INFRARED TELESCOPE FACILITY		X	X
SAAX0005	TRANSITION RADIATION & ION CALORIMETER	X	X	
SAAX0006	STARLAB	X	X	X
SAAX0007	HIGH THROUGHPUT MISSION		X	X
SAAX0008	HIGH ENERGY ISOTOPE EXPERIMENT		X	X
SAAX0009	PINHOLE OCCULTER FACILITY	X	X	
SAAX0010	SOLAR CORONA DIAGNOSTIC MISSION		X	
SAAX0011	ADVANCED SOLAR OBSERVATORY		X	X
SAAX0012	HUBBLE SPACE TELESCOPE	X	X	X
SAAX0013	GAMMA RAY OBSERVATORY	X	X	X
SAAX0014	X-RAY TIMING EXPLORER	X	X	
SAAX0016	SMALL SOLAR PHYSICS (NOTE 1)	X	X	X
SAAX0017	ADVANCED X-RAY ASTROPHYSICS FACILITY	X	X	X
SAAX0018	VERY LONG BASELINE INTERFEROMETER		X	
SAAX0019	FAR UV SPECTROSCOPY EXPLORER	X	X	
SAAX0020	LARGE DEPLOYABLE REFLECTOR		X	X
SAAX0021	SUPER CONDUCTING MAGNET			X
SAAX0022	SPACE STATION SPARTAN PLATFORM (NOTE 2)		X	X
SAAX0023	SPACE STATION SPARTAN PLATFORM 2			X
SAAX0024	SPACE STATION SPARTAN PLATFORM 3			X
SAAX0025	SPACE STATION SPARTAN PLATFORM 4			X
SAAX0026	LEASED PLATFORM (EXPLORER) 1	X		X
SAAX0027	LEASED PLATFORM (EXPLORER) 2	X		X
SAAX0028	LEASED PLATFORM (EXPLORER) 3	X		X
SAAX0029	LEASED PLATFORM (EXPLORER) 4	X		X
SAAX0030	SPACE STATION HITCHHIKER 1			X
SAAX0031	SPACE STATION HITCHHIKER 2			X
SAAX0032	SPACE STATION HITCHHIKER 3			X

NOTE 1: IDENTIFIED AS SOLAR MAX MISSION IN SEPT.'84 LANGLEY DATA BASE.

NOTE 2: IDENTIFIED AS SOLAR SEISMOLOGY EXPERIMENT IN SEPT.'84 LANGLEY DATA BASE.

Attached payloads are those which operate attached to the Space Station manned core, but are external to the pressurized modules. Unmanned co-orbiting platforms will accommodate one or more compatible instruments. Co-orbit is used in the sense that the orbital inclination is the same as the Space Station manned core. Free Flyers or Free Flying Satellites refer to major facilities like the Hubble Space Telescope and other satellites that may utilize the Space Station for maintenance and repair. The use of the Orbital Maneuvering Vehicle (OMV) or the Orbital Transfer Vehicle (OTV) is essential for this purpose.

The Space Station Program Elements also include a polar platform. This unmanned platform will accommodate those payloads which require a polar orbit. The payload identified for performance from the polar platforms support a strategy for obtaining integrated earth science measurements from low earth orbit. Polar Platforms and the instrument located on them will be serviced by the orbiter in an orbit that is within the its capabilities.

Appendix 1 provides a summary description of each payload on tables 1 and 2. These data were excerpted from the Langley Data Base.

TABLE 2. ASTROPHYSICS PAYLOADS POTENTIAL LOCATIONS

CODE	PAYLOAD	ATTACHED	CO-ORBIT PLATFORM	FREE FLYER
SAAX0001	SPECTRA OF COSMIC RAY NUCLEI	X		
SAAX0004	SHUTTLE INFRARED TELESCOPE FACILITY		X	X
SAAX0005	TRANSITION RADIATION & ION CALORIMETER	X	X	
SAAX0006	STARLAB	X	X	
SAAX0007	HIGH THROUGHPUT MISSION	X	X	
SAAX0008	HIGH ENERGY ISOTOPE EXPERIMENT	X	X	
SAAX0009	PINHOLE OCCULTER FACILITY	X	X	
SAAX0010	SOLAR CORONA DIAGNOSTIC MISSION	X		
SAAX0011	ADVANCED SOLAR OBSERVATORY	X	X	
SAAX0012	HUBBLE SPACE TELESCOPE			X
SAAX0013	GAMMA RAY OBSERVATORY			X
SAAX0014	X-RAY TIMING EXPLORER			X
SAAX0016	SMALL SOLAR PHYSICS			X
SAAX0017	ADVANCED X-RAY ASTROPHYSICS FACILITY			X
SAAX0018	VERY LONG BASELINE INTERFEROMETER	X	X	X
SAAX0019	FAR UV SPECTROSCOPY EXPLORER			X
SAAX0020	LARGE DEPLOYABLE REFLECTOR			X
SAAX0021	SUPER CONDUCTING MAGNET	X		
SAAX0022	SPACE STATION SPARTAN PLATFORM 1			X
SAAX0023	SPACE STATION SPARTAN PLATFORM 2			X
SAAX0024	SPACE STATION SPARTAN PLATFORM 3			X
SAAX0025	SPACE STATION SPARTAN PLATFORM 4			X
SAAX0026	LEASED PLATFORM (EXPLORER) 1			X
SAAX0027	LEASED PLATFORM (EXPLORER) 2			X
SAAX0028	LEASED PLATFORM (EXPLORER) 3			X
SAAX0029	LEASED PLATFORM (EXPLORER) 4			X
SAAX0030	SPACE STATION HITCHHIKER 1	X		
SAAX0031	SPACE STATION HITCHHIKER 2	X		
SAAX0032	SPACE STATION HITCHHIKER 3	X		

2.1 ASTROPHYSICS PAYLOAD REQUIREMENTS

Table 3 provides a listing of the key payload requirements for payloads which are candidates to be performed attached to Space Station. Table 4 lists the requirement for the payloads which are performed on co-orbiting platforms or are independent free flyers. The listed requirements were derived from the Langley Data base, except in cases where more definitive information was available. Some additional payloads, identified in reference [5], were also included.

Caution should be taken when using summary descriptions like those contained in the Langley Data Base. Often times pertinent characteristics or requirements are not clearly identified in the Summary. For example, consider the Advanced Solar Observatory (ASO), SAAX0011. The ASO consists of: the Solar Optical Telescope (SOT); the Solar Extreme U.V. Telescope Facility (SEUVTF); the Solar Soft X- Ray Telescope Facility (SSXRTF)[7&8]; and the Pinhole Occulter Facility (POF). The SOT, SEUVTF and the SSXRTF form the SOT cluster. Two pointing mounts are required for the ASO; one for the SOT cluster, and the other for the POF. This requirement is not apparent from the entries in the Data Base.

Tables 5 and 6 contain weight statements for the SOT and the POF. These data were excerpted from references [9],[5], and [6]. An examination of the SOT weight statement in table 5 indicates that the SOT Observatory weight is between 4000 kg and 6600 kg. Table 6 shows that the weight of the POF Instruments and mission peculiar equipment would be 1469 kg (3238 lb.). The Langley Data Base reports a total weight for the ASO of 12,500 kg. The combined weight of the POF (1469 kg) and the SOT (4000 kg) is 5469 kg. This would leave 7031 kg for the SEUVTF and the SSXRTF. Requirements for these instruments have not been defined, but a combined weight of 7031 kg for them appears high. The data base does not indicate how the 12,500 kg for the ASO was arrived at. Other examples can be cited. With respect to pointing requirements, the data base shows that the ASO requires a pointing accuracy of 1.0 arc sec. and pointing stability of 0.1 arc sec. It is not clear if these requirements are those of the instrument or those of the pointing system that the instrument is mounted on. In the case of the SOT, image stability is provided for internally by precision control loops within the SOT in addition to that provided by the pointing mount. This achieves a stability of .03 arc sec. The POF pointing requirements, per reference [6], are a pointing accuracy of 30 arc sec. and a stability of 0.1 arc sec.

TABLE 3
ATTACHED ASTROPHYSICS PAYLOAD REQUIREMENTS

PAYLOAD	ALTITUDE (Km)	INC. (Deg)	VIEW	PTN. ACC. (Arc Sec)	STABILITY (Arc Sec)	LENGTH (M)	WIDTH (M)	HEIGHT (M)	MASS (Kg)
SCRN	500	28.5	Anti Earth	36000	36000	3.3	4.8	3.8	3082
TRIC	500	28.5	Anti Earth	36000	36000	3.3	4.8	4.8	5750
HTM	500	28.5	Inertial	180.0	10.0	2.0	2.0	2.0	1000
HEI	500	28.5	Anti Earth	36000	36000	1.0	1.0	0.1	3000
SOT	500	28.5	Solar	1.0	0.1	7.5	4.4 Dia		4000
POF	500	28.5	Solar	30.0	0.1	32.0	3.3	3.3	3600
SCDM	500	28.5	Solar	10.0	2.0	8.0	5.0	5.0	1250
ASO	500	28.5	Solar	1.0	0.1	32.0	5.0	5.0	12500
VLBI	500	28.5	Inertial	30.0	1.0	15.0	15.0	15.0	1354
SUPERMAG	500	28.5	Anti Earth	72000	36000	-	-	-	-

HITCHHIKER - Specific requirements not defined.

TABLE 4
NON-ATTACHED ASTROPHYSICS PAYLOAD REQUIREMENTS

PAYLOAD	ALTITUDE (Km)	INC. (Deg)	VIEW	PTN. ACC. (Arc Sec)	STABILITY (Arc Sec)	LENGTH (M)	WIDTH (M)	HEIGHT (M)	MASS (Kg)
SIRTF	700	28.5	Inertial	0.15	0.10	8.5	4.0	4.0	4000
HST	600	28.5	Inertial	0.20	0.007	15.0	5.0	5.0	11350
STARLAB	500	28.5	Inertial	2.0	0.02	7.0	2.0	2.0	3200
GRO	500	28.5	Inertial	1800	60.0	7.6	4.6	4.6	14000
XRTE	500	28.5	Inertial	360.0	36.0	8.0	5.0	5.0	850
SSP	500	28.5	Solar	10.0	1.0	5.0	3.0	3.0	3000
AXAF	500	28.5	Inertial	30.0	0.2	15.0	5.0	5.0	10267
LDR	700	28.5	Inertial	1.0	0.03	40	35	35	55000
FUSE	GEO	28.5	Inertial	2.0	1.0	8.0	5.0	5.0	1360

SPARTAN PLATFORMS - No specific orbit or pointing requirements.

LEASED PLATFORMS - Pointing requirements will depend on the specifica payload. The platform will provide 0.05 degrees accuracy and stability without fine guidance from the payload. With fine guidance an accuracy of 0.001 degrees is available.

TABLE 5. SOT WEIGHT STATEMENT

(PER SP82-MSFC-2583) ITEM		Mass (Kg)
SCIENCE INSTRUMENTS		
Instruments and contingency/50%		2500
Focal Plane		4100
Soft X-Ray Telescope		
XUV Telescope		
INTEGRATION HARDWARE		
Pointing System		426
Signal Interface Units (2)		20
Power Interface Units (2)		80
Freon Pump Package		63
S/S Cold Plate		14
Experiment Cold Plates (2)		44
Power and Signal Cooling		7
Pedestal		80
CARRIER		
Pallet (or equivalent)		741
Berthing Adapter Assembly		100
	PAYLOAD TOTAL	8175

(PER GSFC SOT ACCOMMODATION STUDY) ITEM		WEIGHT (Kg)
SOT OBSERVATORY		4000
MODIFIED IPS		1400
CMA		1200
SUPPORT EQUIPMENT		2000
	TOTAL	8600

TABLE 6. POF WEIGHT STATEMENT (LBS)
(per MSFC Phase A study)

EXPERIMENTS AND EQUIPMENT			2094
X-Ray Detectors		992	
Ultraviolet Coronagraph		617	
White Light Coronagraph		485	
ESA and NASA Furnished Hardware			8239
Pallet + Igloo		3336	
IPS incl P/L Clamp Assy (PCA)		2760	
OAFD Equip		127	
Fwd Utilities		1166	
Extra F.C. Reactants (H_2 & O_2)		850	
MISSION PECULIAR EQUIPMENT			1144
Expmt Platform STR.		500	
Attach STR for Exp, canister, and sensors		120	
Payload Hold Down Mech. and STR		130	
Integration Wiring		40	
Thermal Insulation (MLI)		12	
Alignment & Pointing Sensors		110	
Must,Canister & Motor (SEPS Type)		120	
Mask		112	
Occulting Plane (Al)	17		
Coded Aperature	24		
Fourier Transform	3		
Lightweight Shield	10		
Attach. Str. to top of Mask	58		
	TOTAL	11,477 LBS	

For purposes of this study, it was assumed that the requirements shown in Table 3 are the payload requirements which need to be accommodated by an instrument pointing system. A generic pointing system would have to accommodate the combined worst case requirements of all the attached payloads. The ASO represents the worst case payload requirements shown on Table 3 with respect to mass, pointing accuracy and stability. This is true even when the requirements for the SOT and the POF are examined separately. As shown previously the mass of the SOT Cluster is estimated to be between 4000 Kg and 6600 Kg. The Transistion Radiation and Ion Calorimeter has a mass of 5750 Kg, which is close to the high estimate for the SOT Cluster. A generic pointing mount capable of satisfying the requirements of the attached payloads would have to point a payload of approximately 7000 Kg with an accuracy of 1 arc sec and provide a stability of 0.1 arc sec. This takes on added significance since the ASO has been recently approved for attachment to the IOC SS in 1993. Figure 1 shows the ASO attached to the SS upper boom.

2.1.1 The Solar Optical Telescope

The core of the ASO is the SOT which will be a SOT-1 observatory with minimun modifications to permit operation on the SS. SOT-1 will be the first of a series of Shuttle missions of 7 to 14 days duration carrying a 1.3 m solar telescope and science instruments as a Spacelab attached payload. The primary goal of the SOT is to study physical processes that occur on the sun using angular resolution and stability that correspond to an atmosphere density scale height of 70 Km or 0.1 arc sec. In addition to the resolution requirement the LOS stability must be maintained to approximately 0.03 arc sec RMS and drift maintained to 0.06 arc sec for at least 60 sec. To meet the science objectives the Orbiter will be placed in a solar inertial attitude and the Spacelab Instrument Pointing Subsystem (IPS) will provide coarse pointing of the SOT-1 to the sun. Fine pointing will be performed within the SOT itself using a control system which moves the primary and tertiary mirrors in tandem.

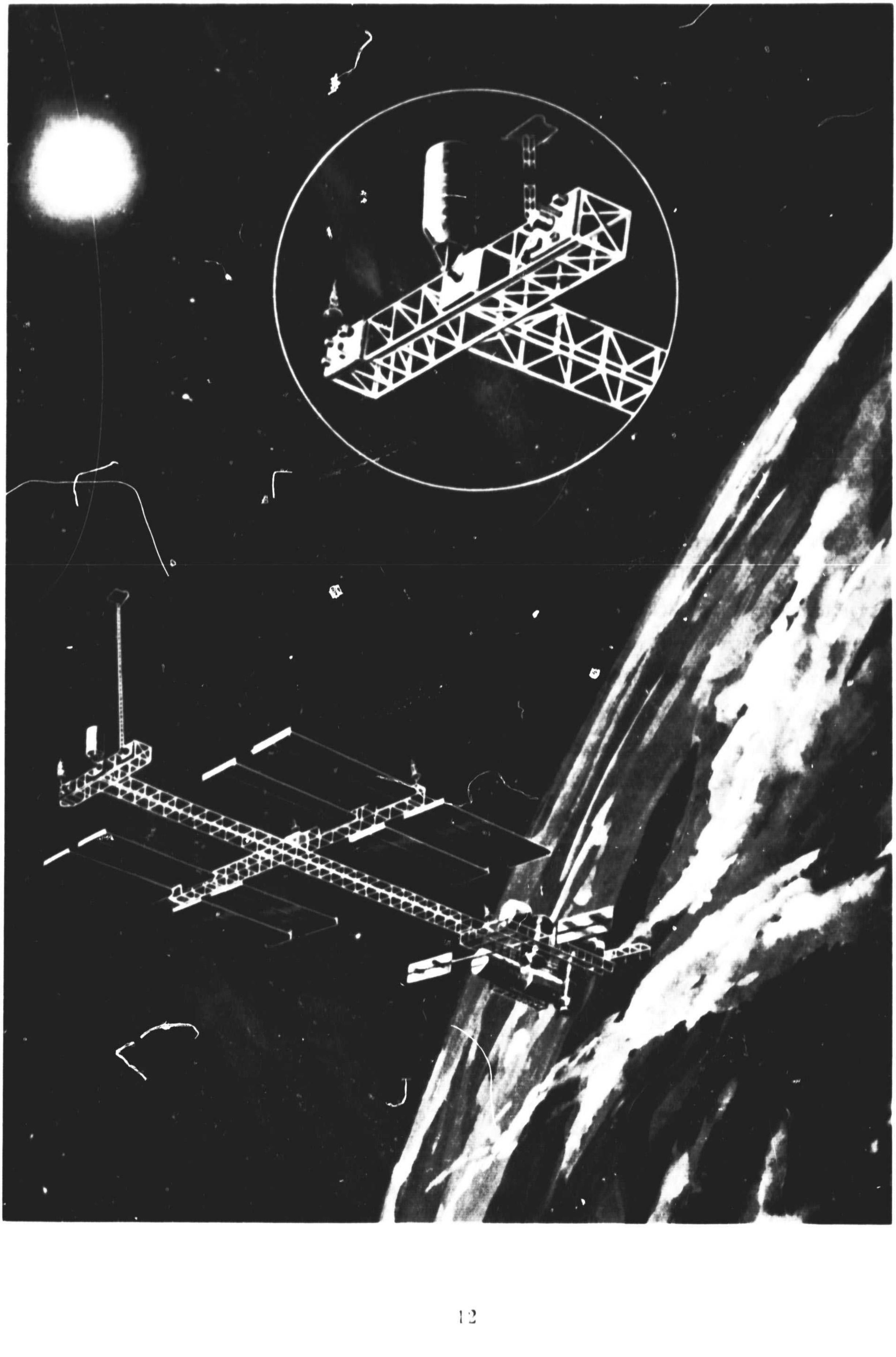

FIGURE 1 ASO ON SPACE STATION UPPER BOOM

Figure 2 illustrates the SOT-1 observatory in terms of overall dimensions and location of components. The facility is approximately 4.4 m in diameter, 7.3 m long and weighs approximately 4000 Kg. The observatory has two major functional components : the Telescope Facility and the Coordinated Instrument Package (CIP). The Telescope Facility is a folded, on-axis, Gregorian system with a 1.3 m diameter primary paraboliod mirror. The primary mirror is mounted in an assembly called the articulated primary mirror (APM) which is supported by six motor driven displacement actuators. These enable the APM to be translated and rotated about the prime focus for offset pointing and rotated about its vertex for LOS stabilization in six degrees of freedom. Additional, higher bandwidth steering of the beam is achieved using the tertiary mirror. This is achieved using three linear actuators which provide control in two degrees of freedom.

The CIP consists of four functional elements:

A dedicated experiment processor
An image processor
An electronic interface unit
A focal plane instrument package

The focal plane instrument package contains the optical and mechanical portions of the SOT instruments. They are a photometric filtergraph, a tuneable filtergraph, and a combined ultraviolet and visible spectrograph.

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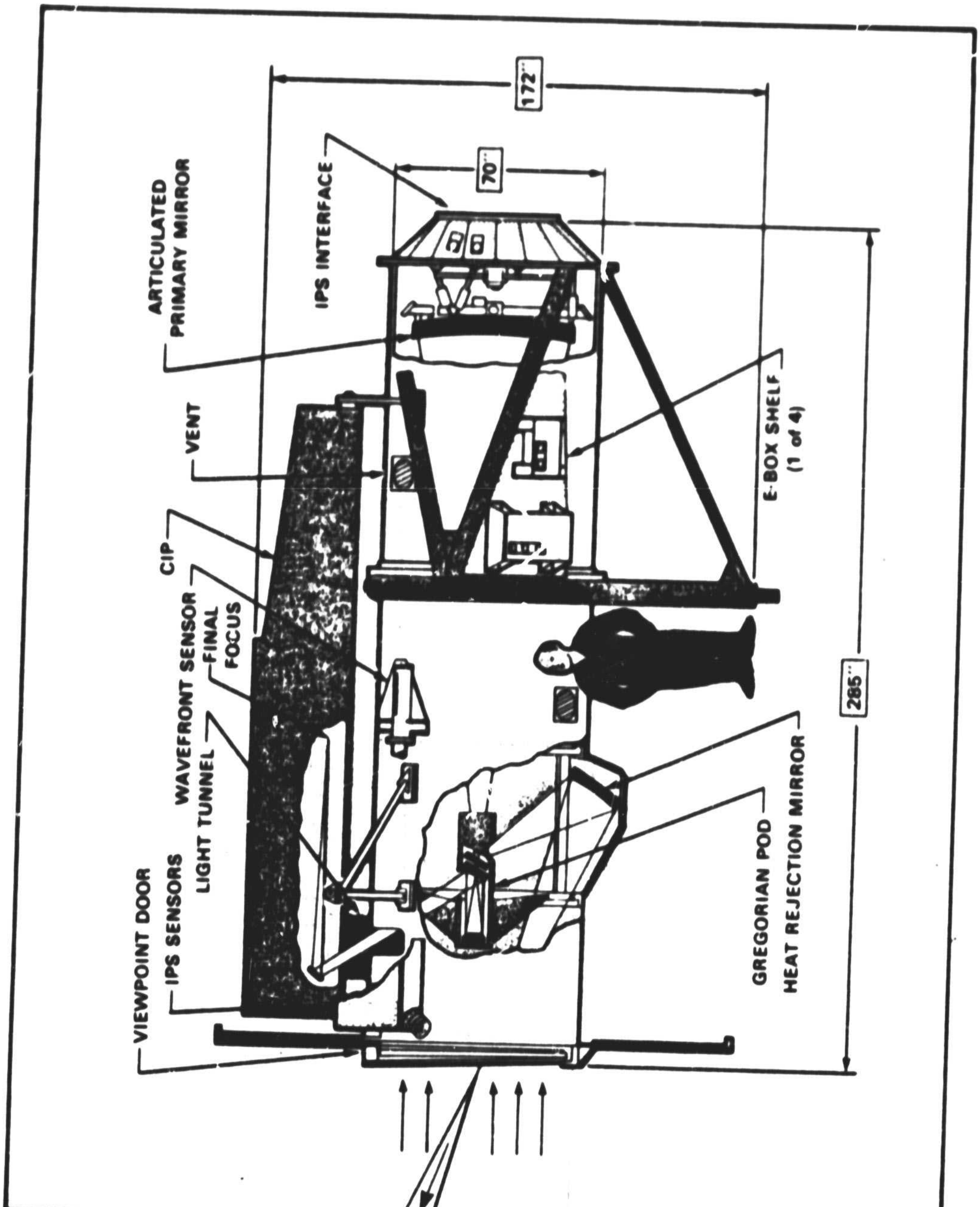

FIGURE 2 SOT–1 GENERAL CONFIGURATION

2.1.2 The Pinhole Occulter Facility

The POF is designed to study the non thermal phenomena that lie at the heart of plasma dynamics in the solar corona and will make possible high energy cosmic X-Ray observations. The facility consists of a self deployed boom which separates an occulter plane from a detector plane. The X- Ray detectors and coronal imaging optics mounted on the detector plane are analogous to the focal plane instrumentation of an ordinary telescope, except that the occulter is used for providing a shadow pattern. The occulter plane is passive and has no electrical interface with the rest of the facility. Figure 3 shows the facility in a partially deployed configuration. MSFC performed a Phase A engineering definition study of the POF [6] operated from the Space Shuttle. The study considered a full instrument complement for the facility which included an X- Ray Coded Aperature Pinhole, an X-Ray Fourier Transform Pinhole, a White Light Coronagraph Imager, and a UV Spectrograph. The facility requires the use of the Spacelab IPS for fine pointing. The primary technical challenge for performing the POF objectives is in the area of pointing and control of the deployable boom. The boom is a continuous longeron Astromast 0.4 m in diameter and deployed length of 32 m. The boom is quite flexible having natural frequencies of 0.0948 Hz and 0.917 Hz (first and second modes) in its deployed condition.

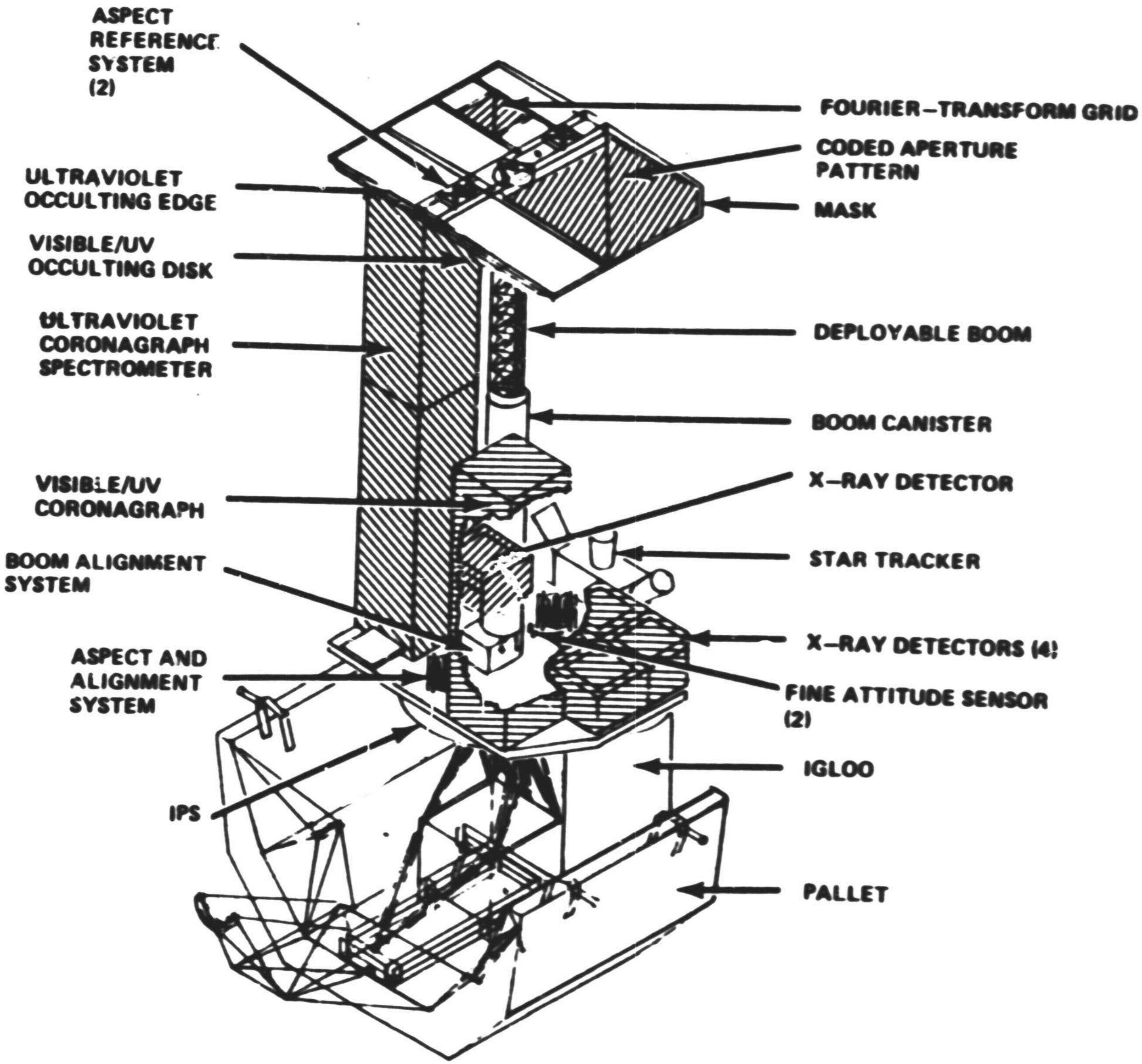

FIGURE 3 PINHOLE OCCULTER FACILITY

3.0 PRESENT POINTING MOUNT CONCEPTS

Information pertaining to past and present pointing mount concepts was gathered from several sources to determine the basic design concepts and system capabilities. The primary source was documentation and publications on the various systems. These sources are identified in references [9] through [41]. Verbal information from telephone contacts and attendance at presentations was also used. Much of the information contained in the following sections describing these systems was summarized or exerpted from references [9] through [41]. Table 7 lists the pointing systems considered, the developer of the system, and the stage of system development. Various stages of development are represented. The Suspended Experiment Mount is in concept development while the Instrument Pointing System flew on Spacelab 2. Three of the systems have flight experience; the Apollo Telescope Mount (ATM),the Modified ATM Star Tracker (MAST), and the Spacelab IPS. The MAST is not a closed loop system but was included to illustrate some of the techniques used to perform experiments on Spacelab. The ATM was included to provided a historical perspective on what was achieved during SKYLAB over a decade ago. The Hubble Space Telescope (HST), although not a gimbal system, represents the present state of the art for precision pointing systems. The basic capabilities of these systems are listed in Table 8. Most of the systems were designed for Space Shuttle applications and the capabilities listed assume the appropriate base body. Examination of Table 8 shows that none of the systems surveyed will accomodate the " worst case " payload requirements as defined in section 4.1, i.e. accommodate a payload mass of 7000 Kg with a pointing accuracy of 1.0 arc sec and a stability of 0.1 arc sec. Appendix 2 provides a general discussion of pointing mount design considerations, architecture, hardware, and terminology.

TABLE 7. POINTING SYSTEMS SURVEYED		
SYSTEM	DEVELOPER	DEVELOPMENT STAGE
SKYLAB ATM	MSFC	FLIGHT HARDWARE
SMALL INSTRUMENT POINTING MOUNT (SIPS)	BALL BROTHERS FOR GSFC	DESIGN
MINIATURE POINTING MOUNT (MPM)	MSFC	DESIGN
ANNULAR SUSPENSION AND POINTING SYSTEM (ASPS)	SPERRY FOR LANGLEY RESEARCH CENTER	GROUND TEST HARDWARE
ADVANCED GIMBAL SYSTEM (AGS)	SPERRY FOR MSFC	GROUND TEST HARDWARE
MODIFIED ATM STAR TRACKER (MAST)	MSFC	FLIGHT HARDWARE
INSTRUMENT POINTING POINTING SYSTEM (IPS)	DORNIER FOR ESA	FLIGHT HARDWARE
GIMBALFLEX	MARTIN MARIETTA	GROUND TEST HARDWARE
SUSPENDED EXPERIMENT MOUNT (SEM)	MSFC	CONCEPTUAL DESIGN
TALON GOLD GIMBAL	LMSC FOR DOD	GROUND TEST HARDWARE
HUBBLE SPACE TELESCOPE	LMSC/BENDIX FOR MSFC	FLIGHT HARDWARE IN ASSEMBLY AND TEST

TABLE 8
POINTING SYSTEM CAPABILITIES

SYSTEM	ATM	SIPS	MPM	ASPS	AGS	MAST	IPS	GIMBAL-FLEX	SEM	TGG	HST
POINTING (Sec)	2.5	1	-10	0.1	0.1	60	2	6	N/A	3	0.01
STABILITY (Sec)	2.5	0.5	2	0.1	2.5	-	1.2	-	1	0.6	0.007
SLEW RATE (Deg/Sec)	X & Y: 1.3 Z: 7	2	-	3	4	3.5	3	3	N/A	3	0.25 (PK)
ACCELERATOR (Deg/Sec^2)	0.28	1.5 to 25	0.1	0.55	0.55	-	-	0.55	N/A	0.12	0.0008
TORQUE N-M	19	10	0.6	34	34	-	23.5	-	N/A	95	0.8
FOV (Deg)	4	180/ 10/90	100	200/ 120	140	160/ 80	130	-	Full Via CMG System	-	Full
SERVO B/W (Hz)	2	1	1	1	1	-	0.5	0.5	1	0.5	1.3
P/L MASS (Kg)	11000	2 @ 500	450	600	60 to 7200	450	7000	2000	Determined by S/L Pallet Capability	2400	11350

3.1 SKYLAB ATM

Skylab was launched on May 14, 1973 and the mission ended on February 8, 1974, with the undocking of the Skylab 4 Command and Service Module. The Skylab Orbital Assembly and its associated control axes are shown in Figure 4. The Skylab Attitude and Pointing Control System (APCS) provided three axis attitude stabilization and maneuvering capability throughout the mission and pointing control of the Skylab experiment package during experimentation periods.

The APCS was comprised of three control systems. The CMG Control Subsystem provided three axis attitude stabilization and maneuvering capability for the vehicle. The Experiment Pointing and Control (EPC) Subsystem was used for maintaining attitude control and fine pointing of the solar instrument package during solar experimentation. While the EPC subsystem was fine pointing the solar instrument package, the CMG and TACS control systems maintained attitude control of the vehicle. The EPC Subsystem also provided the astronaut with the capability of offset pointing the solar instrument package to targets of opportunity on the solar disk or its outer perimeter. The TACS provided additional attitude control capability to the vehicle.

The solar instruments were mounted on a cruciform spar within the ATM canister, which was located within the basic structure of the ATM. The instruments which were mounted on the spar required accurate and stable pointing control.

The requirements were: offset pointing from the sunline of ±24 arc-min., with an accuracy of ±2.5 arc-sec.; a stability of ±2.5 arc-sec. with a jitter of 1 arc-sec/sec for a 15 minute period; and roll about the sunline of ±120 degrees with an accuracy of ±10 arc-min., a stability of ±7.5 arc-min. for a 15-minute period, and a jitter of 3 arc-min for a 1-second period.

To achieve the required pointing accuracy and stability, the instrument canister was decoupled from the main portion of ATM structure as much as possible by the use of frictionless flex pivot bearings. Figure 5 illustrates the ATM fine pointing control mechanism. The canister was attached to the structure by a three-degree-of-freedom mechanism actuator assembly. The outer ring of this assembly was attached to the solar observatory structure by four roller bearings, and was driven by an electric motor to roll the canister. The inner ring was attached to the outer ring by two frictionless flex-pivot

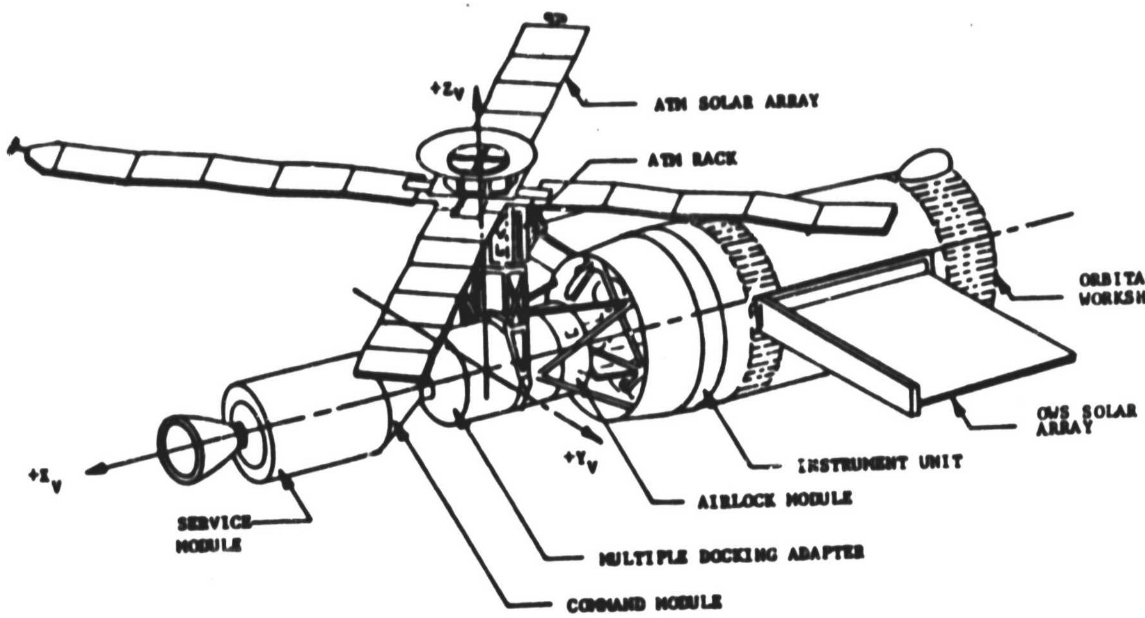

FIGURE 4 SKYLAB ORBITAL ASSEMBLY

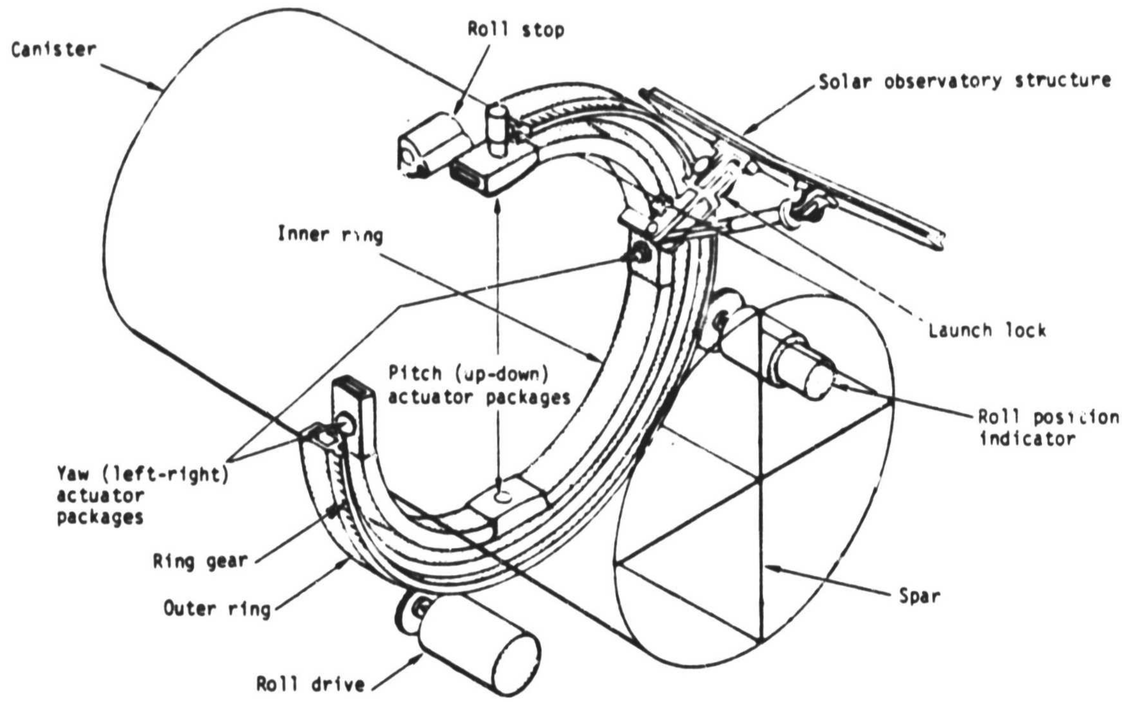

FIGURE 5 ATM FINE POINTING CONTROL MECHANISM

bearings with ±2 degrees freedom and was driven about the up-down axis by redundant actuators. The canister was attached to the inner ring by two frictionless flex-pivot bearings with ±2 degrees freedom and was driven about the left-right axis by redundant actuators. Orbital locks constrained the rings and the canister while the solar observatory was not being used. A roll stop limited the roll to ±120 degrees, to avoid damage to electrical cables. Launch locks protected the assembly from damage during launch. A roll resolver provided experiment roll position information to the digital computer.

The EPC subsystem provided automatic stabilization about the canister's Yaw (left/right) and Pitch (up/down) axis and open loop positioning was provided for the Z (Roll) axis. Figure 6 is a block diagram of the EPC subsystem.

The Fine Sun Sensor (FSS) and Rate Gyro (RG) sensors, mounted on the canister, supplied position and rate feedback to the Experiment Pointing Electronics Assembly (EPEA). The EPEA contained the electronic functions to command the gimbal actuators, closing the control loop. The Z-axis stabilization was provided by the ACS.

Manual positioning about all three axes was provided for offset/roll pointing of the experiment canister. X and Y- Axis offset pointing was achieved by means of rotating optical wedge mechanisms within the FSS. Offset commands could be issued from the Manual Pointing Controller (MPC) or from the ATMDC. The Roll Position Mechanism (RPM) was activated by command switches located on the Control and Display Panel and on the ATM EVA Rotation Control Panel. All offset/roll commands were processed by the EPEA. A line-of-sight pointing capability during roll operations was provided by the ATMDC, utilizing the ATMDC wedge drive capability.

The EPEA also provided an interface between the Star Tracker and the MPC for manual positioning of the Star Tracker gimbals. Other EPCS related functions were canister caging control, experiment alighnment calibration, and experiment and FSS door control.

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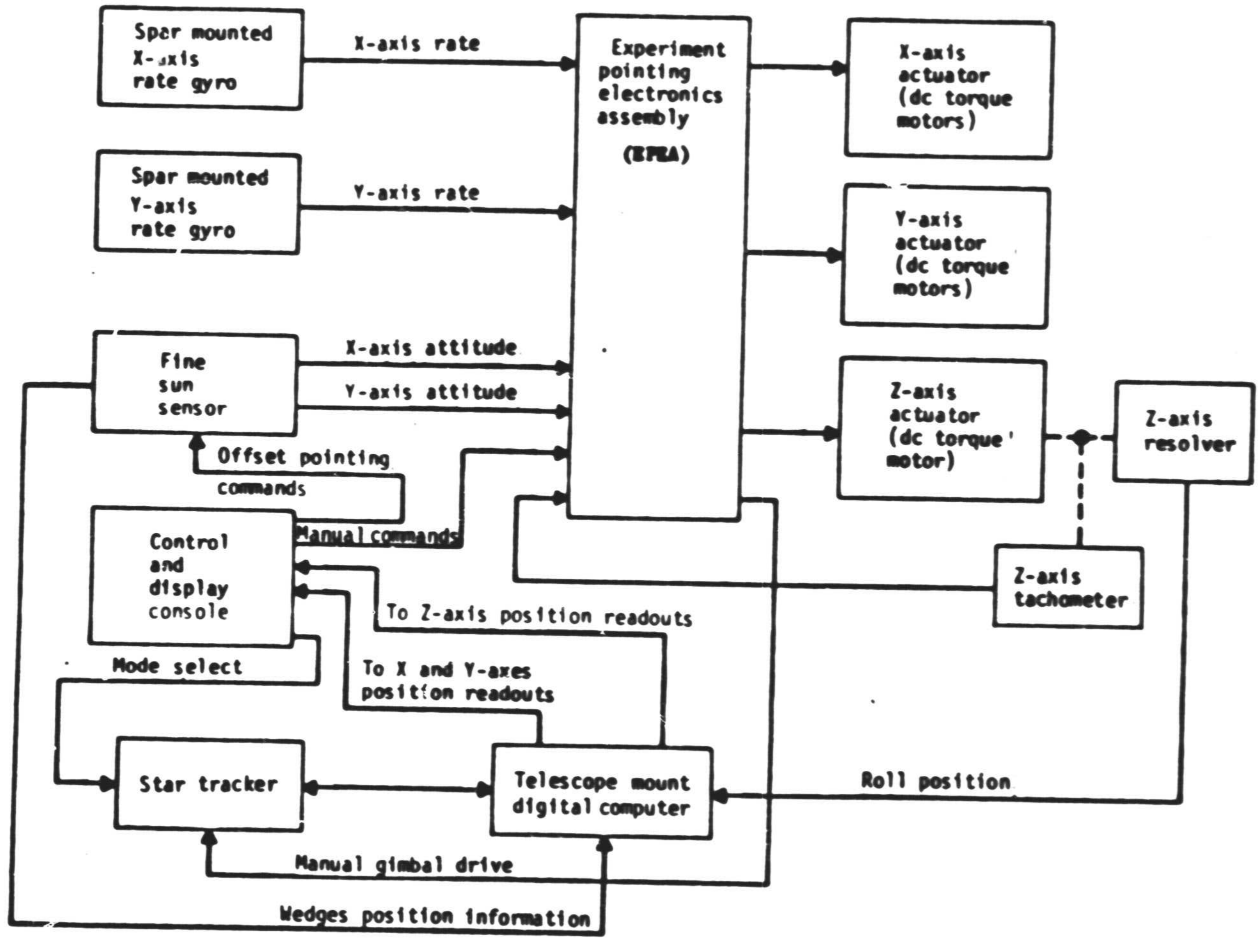

FIGURE 6 EPCS BLOCK DIAGRAM

A number of new concepts were proven by the performance of the Skylab APCS, which are applicable to Space Station. Skylab was the first manned spacecraft to utilize large Control Moment Gyros (CMG) for momentum storage and attitude control, the first to utilize vehicle maneuvers for CMG momentum desaturation, the first to utilize a fully digital control system with in-orbit reprogramming capability and extensive automatic redundancy management, and the first to utilize an attitude reference system based on a four parameter strapdown computation.

The Skylab APCS successfully met or exceeded all design specifications. In particular, the EPC Subsystem provided pointing accuracy and stability well within the Skylab instrument requirements.

3.2 SMALL INSTRUMENT POINTING SYSTEM (SIPS)

The SIPS was a precision fine pointing system designed by Ball Brothers Research for GSFC in the middle 1970's. The pointing system was designed to accommodate two instruments, each approximately 1 x 1 x 3 meters, from a Spacelab pallet. Individual pointing and control was provided for each instrument. The SIPS configuration is shown in Figure 7. The SIPS consisted of four major sections: a Pallet Interface Structure; a Deploy/Retract Pedestal; a Pointing Section; and Instrument Cannisters.

The Pallet Interface Structure provided load-carrying attachments for mating with the hard points on a standard Spacelab pallet. The deploy/retract pedestal provided the suport structure and mechanisms for raising the pointing section to a position above the shuttle bay, to obtain an unobstructed hemispherical field of view for the instruments. The pointing section supported two individually pointed instruments and provided independent fine pointing for each instrument. The Instrument Canisters were modular, to accommodate instruments of various lengths and provided complete instrument environmental protection.

The design allowed instruments to be integrated and tested in the instrument canisters before being mated to the SIPS gimbals. The SIPS was a center of mass mounted system to decouple shuttle translational motions from payload pointing. The unit, without instruments, weighed approximately 764 Kg. The overall gimbal arrangement is shown in Figure 8. The fine pointing gimbals were normal to the line of sight eliminating the need for coordinate transformations and some independence of spacecraft orientation. Figure 8 identifies the gimbal capability with respect to degrees of freedom. Azimuth is motion about the shuttle "Z" axis. An optional roll gimbal was designed to fit within the SIPS canister gimbal frame for use with instruments requiring roll about the LOS. Depending on instrument size, the SIPS could provide payload angular acceleration from 1.5 to 25 deg/sec^2 and a typical slew rate of 2 deg/sec.

Both the SIPS and the mounted instruments depended on Spacelab for electrical and command and data services. In many ways, the SIPS acted as an "extension" providing the user with easy access to Spacelab systems. Figure 9 is a block diagram of the SIPS Pointing System.

The SIPS was nearly autonomous in that all control loop closures were made through the SIPS control electronics. Sequencing and supervisory control of pointing was accomplished by command inputs from the Spacelab experiment computer by means of the data bus and a Remote Acquisition Unit (RAU).

The payload specialist could monitor and control the operation of the SIPS from the keyboard and CRT that are associated with the experiment computer. Normal operation was largely pre-programmed, but the payload specialist would have full capability to conduct such operations as boresighting, guide star acquisition, and sensor calibrations.

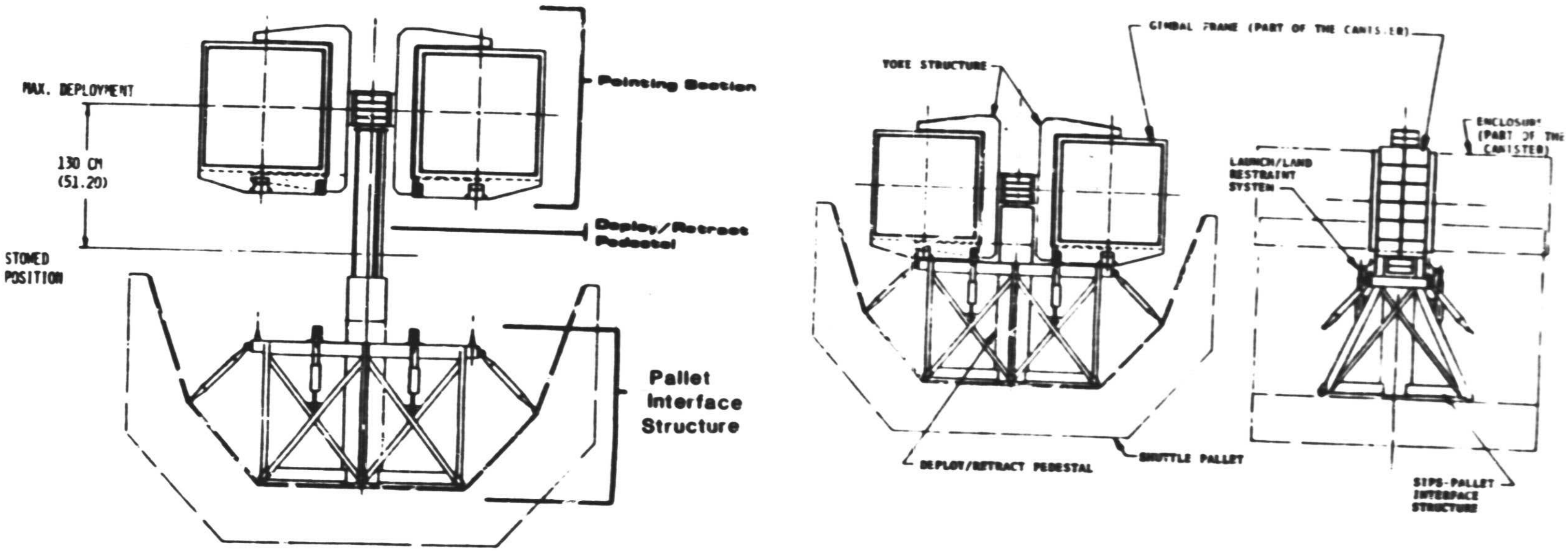

FIGURE 7 SIPS CONFIGURATION

ACQUISITION FREEDOM:

- ± 180° ABOUT AZIMUTH AXIS
- + 120° UP IN ELEVATION
- ± 10° ABOUT RIGHT-LEFT AXIS

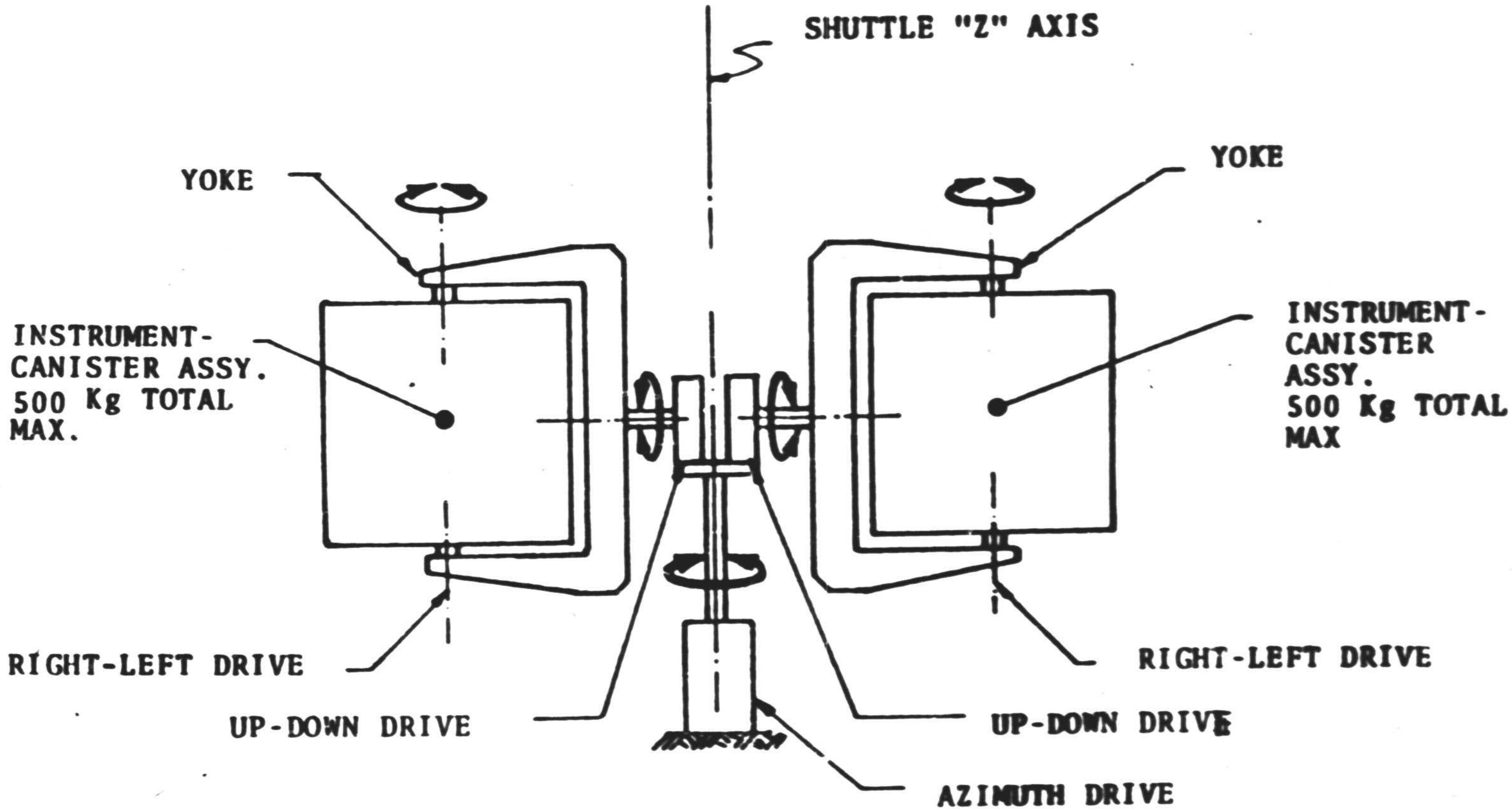

FIGURE 8 SIPS GIMBAL ARRANGEMENT

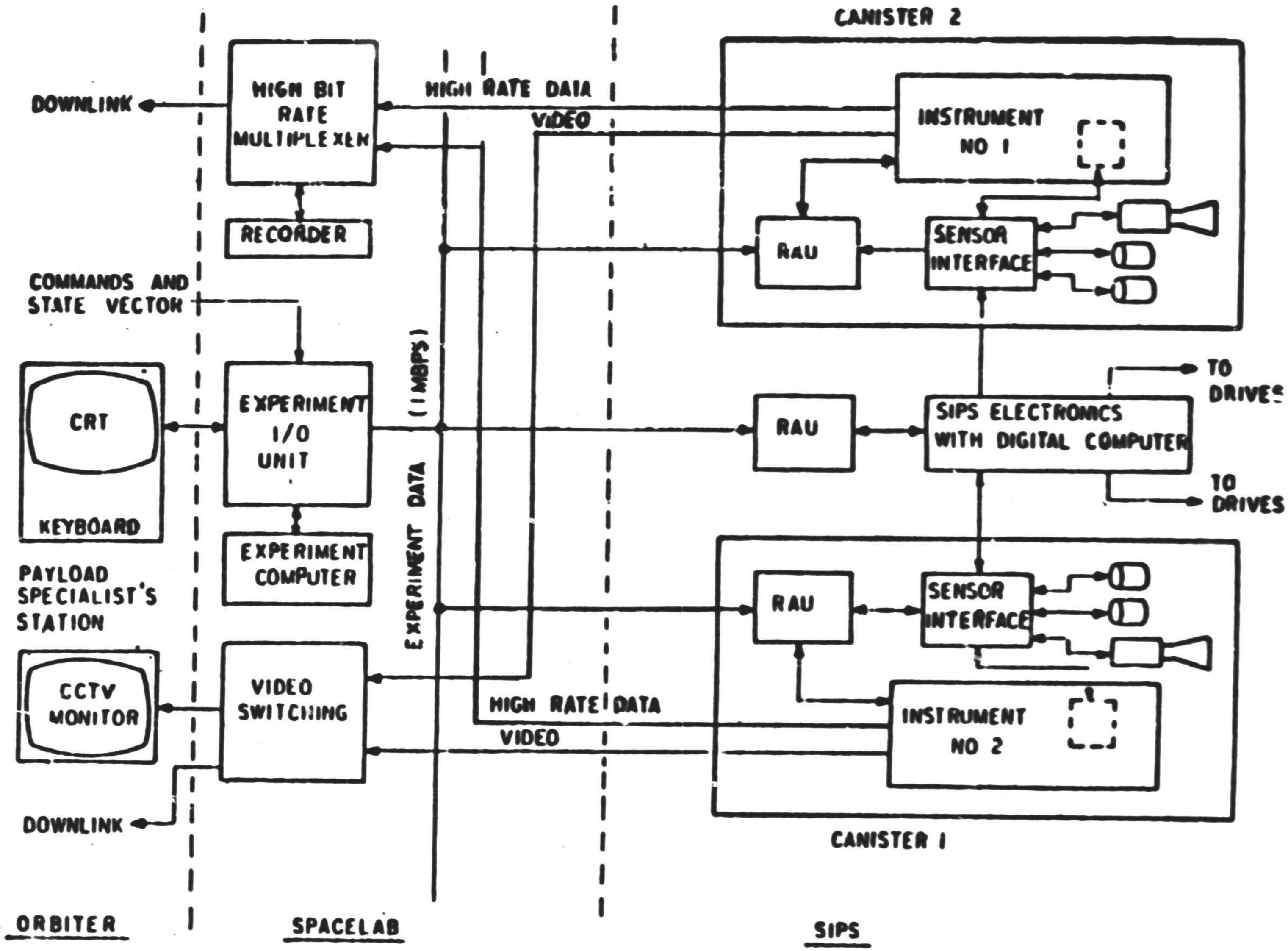

FIGURE 9 SIPS BLOCK DIAGRAM

3.3 MINIATURE POINTING MOUNT (MPM)

The Miniaturized Pointing Mount (MPM) was a modified ATM Star Tracker intended to accommodate relatively small payloads on the spacelab pallet that was studied at MSFC during the middle 1970's.

The configuration of the MPM is shown in Figure 10. The star tracker was modified by removal of the optical telescope assembly; addition of a roll gimbal torquer and resolver; fabrication of a base mount pedestal, isolators, and clamping mechanism; and the addition of an experiment mounting structure. Payloads were end-mounted to the experiment base plate. The gimbal system nomenclature and freedom were as follows: The outer or elevation gimbal ±180 degrees; the middle or cross elevation gimbal ±50 degrees; the inner or roll gimbal ±180 degrees.

This gimbal arrangement allowed a conical field of view of 100 degrees included angle, and when the outer gimbal freedom was along the orbital trajectory, permited horizon to horizon track along the orbital path or fixed point earth tracking. The roll gimbal being the innermost, allowed roll about the payload LOS to be achieved with a one-axis command.

The estimated weight of the MPM system was 110 Kg. The MPM torque capability was 0.6 N-M per axis adequate for solar and stellar fine pointing, but limiting slewing and scanning operations. Slewing and scanning operations are a function of payload inertia and payload center of mass offset from the center of rotation. End mounting the payload increased the payload inertia about the center of rotation, placing additional requirements on the torque motor than would be seen by a c.g. mounted system.

The soft mount isolators on which the MPM was mounted were provided to attenuate the effects of external disturbances to the mount (i.e., those produced by crew motion or Orbiter RCS firings) and to pass only frequencies within the controller bandwidth. Figure 11 is a functional block diagram of the MPM.

The MPM interface with the Spacelab Electrical Power Distribution System (EPDS) at a standard bulkhead connector. The MPM required both 28 VDC and 115 VAC, 400 Hz, 30 for on- orbit operation. The 28 VDC was utilized to power the elements of the MPM, while the 115 VAC was taken across the gimbal for payload use. Crossing the gimbal at

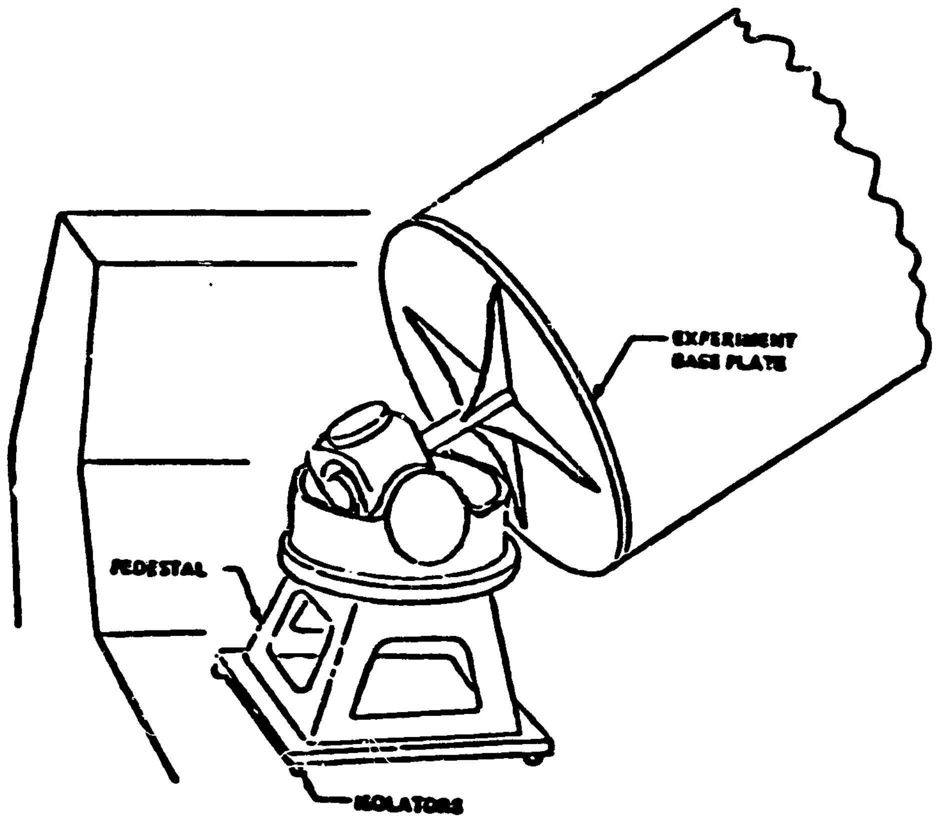

FIGURE 10 MPM CONFIGURATION

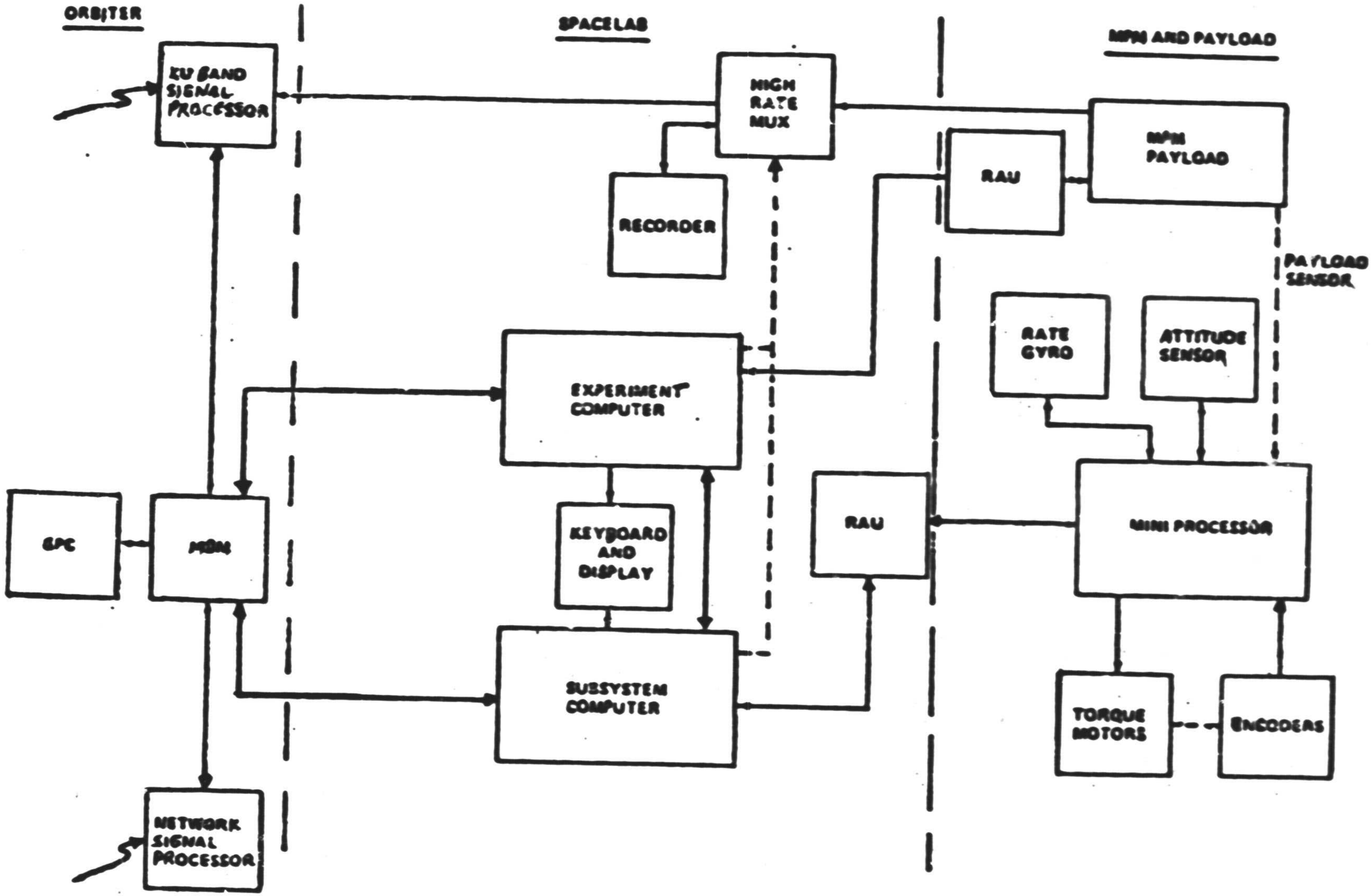

FIGURE 11 MPM FUNCTIONAL BLOCK DIAGRAM

high voltage levels minimized the wire diameter across the gimbal, thus yielding less cable torque. The AC voltage was converted to 28 VDC for those payloads requiring DC excitation. The prime controlling element for the MPM was a self-contained miniprocessor. This computer controled the MPM, utilizing error signals either from the MPM sensors or from a payload supplied source.

The MPM attitude sensors consisted of three rate gyros, a fine sun sensor and a fixed star tracker.

End mounting the payloads on the MPM allowed the experiment canister to be designed independently. The basic interface requirement on the canister was that it mate with the MPM base plate, both mechanically and electrically. Due to the variety of payloads requirements, no "standard" canister design was established as in the case of the SIPS.

3.4 ANNULAR SUSPENSION AND POINTING SYSTEM (ASPS)

The ASPS is a modular payload pointing system which was intended for use on Spacelab. The system was designed by Sperry Flight Systems Division under contract to Langley Research Center. The system, shown in Figure 12, was comprised of: a Vernier Pointing Assemblty, two Coarse Gimbal Assemblies, a Mounting and Jettison Assembly, a Control Electronics Assembly, and assorted connecting hardware.

The complete ASPS was expected to provide payload pointing stability better than 1 arc-sec and, when coupled with a quality sensor, the pointing accuracy should have approached that of the sensor. The unique arrangement of gimbal and vernier subsystem permitted the utilization of ASPS for stellar, solar, and terrestial observations. Actuators in the ASPS were sized to accept payloads up to 600 Kg in mass with center of mass offsets up to 1.5 meters; however, larger instruments could be used if reduced acceleration was acceptable.

Two identical Coarse Gimbal Assemblies (CGAs) were stacked to form an elevation and a lateral gimbal pair. Design of the Gimbal Mounting Bracket and the Gimbal Mounting Structure was such as to provide a mechanically limited travel of ±100 degrees (from vertical) along the lower elevation gimbal axis, and ±60 degrees about the upper lateral gimbal axis. The Vernier Pointing Assembly (VPA) contained the roll axis drive which provided unlimited rotation about the payload longitudinal axis and a vernier rotation of

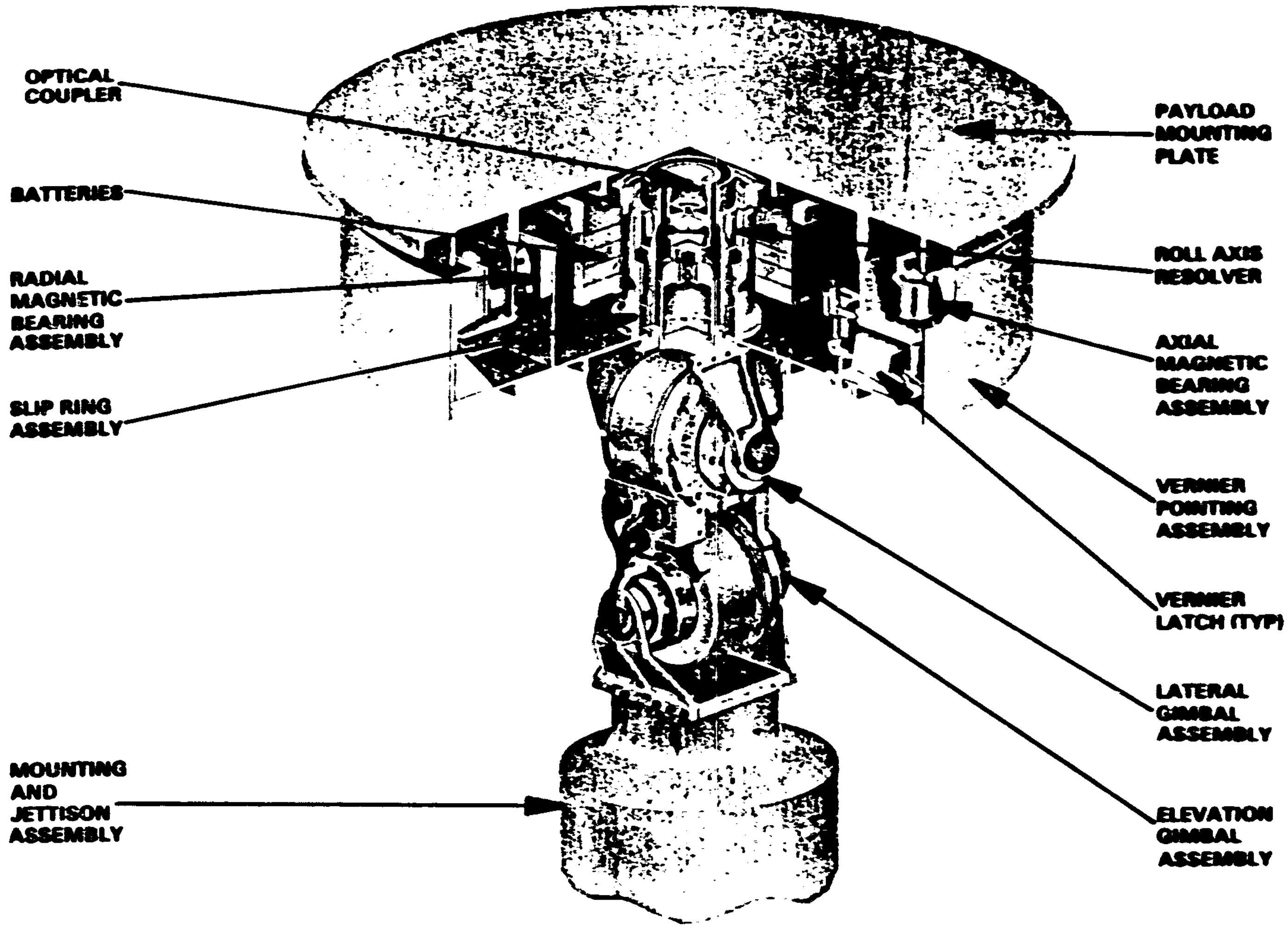

FIGURE 12 ANNULAR SUSPENSION AND POINTING SYSTEM

±.75 degrees about any axis in the plane normal to the payload roll axis. The Payload Mounting Plate formed a removable base plate for mounting and aligning experiments prior to installation on the ASPS. For missions not requiring roll freedom or very high pointing stability, the Payload Mounting Plate could be attached directly to the upper Gimbal Mounting Bracket. ASPS electronics were contained within a separate electronics package, which was mounted to a pallet cold plate near the bottom of the Mounting and Jettison Assembly.

Magnetic Suspension was utilized in the VPA to provide noncontacting isolation and vernier positioning of the payload. Three-axial Magnetic Bearing Assemblies (MBAs) provided axial translation and vernier pointing about the transverse axes. Proximeters associated with each MBA were used to linearize the force/displacement characteristics of each actuator. The proximeter outputs were also combined electronically to compute the axial displacement and the tilt of the payload about the two axes normal to the payload line-of-sight.

The ASPS interfaced with the Spacelab Electrical Power Distribution system for power to the ASPS components and for the payload. Twenty five nickel-cadmium cells were mounted below the rotating top plate to supply 300 watt hours of electrical energy to the payload. The selected cells were NASA Standard and provided a nominal 28 Vdc with a peak power output of 300 watts. A recharge circuit with a one- hour duty cycle was included. Rechanging the batteries required the VPA to be caged, allowing a solenoid operated brush block assembly to connect power required to recharge the batteries.

Figure 13 shows the interfaces of the ASPS with the Orbiter/Spacelab, pallet and payload. The experimenter began control of the system via the Spacelab Digital Display Unit (DDU) keyboard. He provided servo position commands to the ASPS. These commands were routed through the ASPS resident software in the experiment computer, through the CDMS bus to an RAU, and eventually arrived at the Control Electronic Assembly (CEA). The CEA could then command the ASPS to the desired direction, using feedback from the ASPS resolvers and proximeters. An optical coupler, which connected the Spacelab Experiment Data Bus to a second RAU located on the payload plate, allowed the experimenter to communicate with the payload.

Pointing error data generated by experiment sensors could be sent to the CEA through the CDMS for use in the pointing control servos.

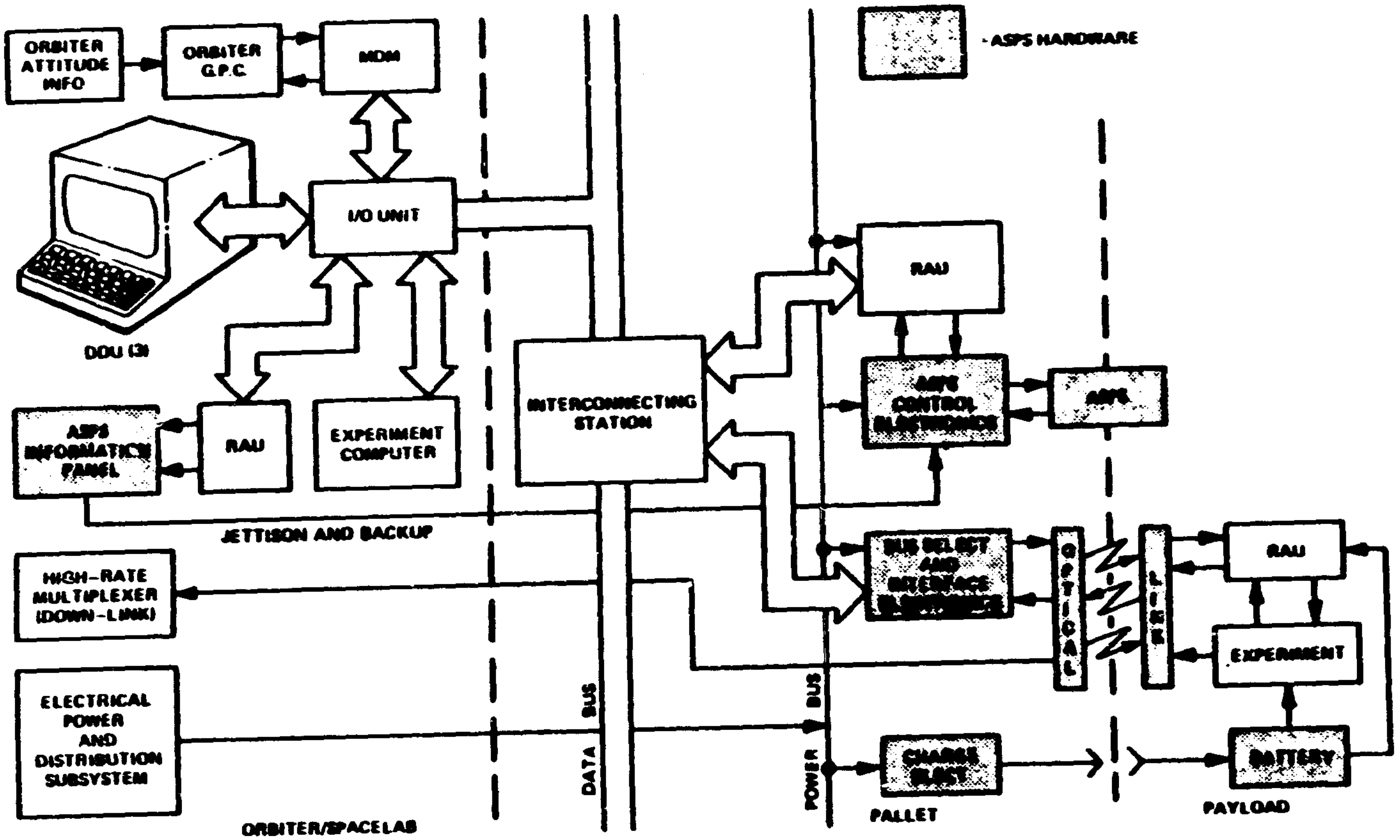

FIGURE 13 ASPS INTERFACES

3.5 THE ADVANCED GIMBAL SYSTEM

The AGS is a three-axis precision pointing system that was designed by Sperry Flight Systems under contract to NASA Marshall Space Flight Center, for use on Spacelab Missions. Its initial design was based on the ASPS with the magnetic system removed. It was comprised of a three-axis gimbal system, which accommodated end mounted payloads through the use of a payload mounted structure (PMS). Total weight of the AGS was approximately 1025 Kg. The AGS could accommodate payloads from 60 to 7200 Kg, which included all hardware mounted above the PMS plane. The dimensions of the payload were restricted only by the allowable movement of each of the three gimbals, by the space in the pallet available to the payload and by other payloads resident in the Orbiter payload bay. Figure 14 illustrates the major hardware elements that made up the AGS. Two identical gimbal assemblies formed an elevation and a lateral gimbal pair. The lower (elevation) gimbal provides an angular range of + 100, - 60 degrees (from vertical), and the upper (lateral) gimbal provides cross-axis positioning of ±60 degrees. The gimbals were mechanically limited by an adjustable stop arrangement to prevent contact with the Shuttle. A roll gimbal permited rotation (±180 degrees) about the payload line-of-sight and was mounted above the lateral gimbal. The gimbals were connected to the payload through a Payload Mounting Structure (PMS). The flat end-mount configuration of the AGS permited overhanging the PMS to accommodate payloads of various sizes and shapes. A Mounting and Jettison Assembly (MJA) was used to connect the gimbals with the underlying mounting structure. During launch and landing, a separation device within the MJA physically disconnected the mounting base from the AGS gimbals, which were independently mounted with the experiment to the support structure.

The MJA also contains a pyrotechnically actuated jettison system which provided for jettisoning of the experiment/AGS gimbals in the event of a multiple failure in orbit. An accelerometer unit mounted on the AGS structure or the pallet sensed Orbiter motion for use in a feed forward decoupling control law.

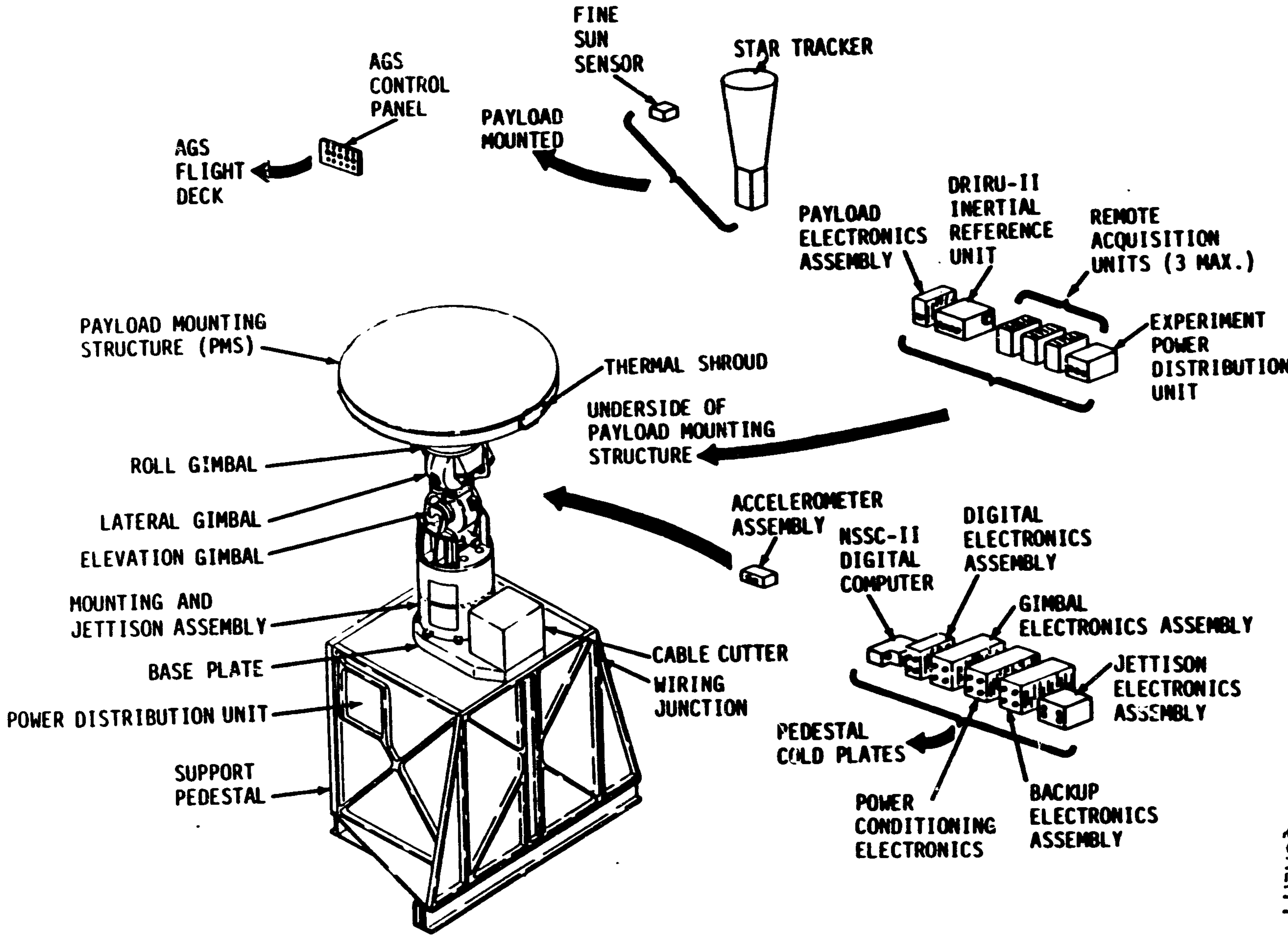

FIGURE 14 AGS ELEMENTS

The AGS angular error sensors (NASA standard fine sun sensor and star tracker) were mounted on top of the payload mounting structure with the experiments. The DRIRU-II rate integrating gyro package was mounted inside the thermal shroud on the bottom of the payload mounting structure, along with the Payload Electronics Assembly (PEA), and up to three (RAUs).

Figure 15 identifies the AGS control and data interfaces. Interfaces with the attitude sensors were provided by the PEA which is connected to the Digital Electronics Assembly (DEA) through a serial data link. The PEA is a general purpose microprocessor-controlled interface unit, which provided a Remote Interface Unit (RIU) compatible serial interface with the angular error sensors, and a special purpose interface with the DRIRU-II. Experiment control and data was handled by the Experiment RAUs, which were connected to the Spacelab Command and Data Management System (CDMS). Additional lines were also provided to the experiments for power, control, and Spacelab High Rate Multiplexer (HRM) input data. A Power Distribution Unit (PDU) supplied primary power through the AGS gimbals to an Experiment Power Distribution Unit (EPDU) mounted within the thermal shroud under the PMS. The EPDU could be "patched" to supply different power feeds on a mission-dependent basis.

Other AGS electronics were a Gimbal Electronics Assembly (GEA), Power Conditioning Electronics (PCE), Backup Electronics Assembly (BEA), Jettison Electronics Assembly (JEA), Digital Electronics Assembly (DEA), and NASA Standard Spacecraft Computer (NSSC-II), all of which were mounted to Spacelab coldplates attached to the AGS Support Pedestal. The GEA contained analog electronics for modulation and demodulation of the gimbal angle resolvers and for power drive of the two-phase, brushless dc torquers used to position the gimbals. The GEA also contains electronics which operated the payload clamping system, separation mechanism, and retractable gimbal stop. A passive Wiring Junction Unit (WJU) mounted adjacent to the coldplate mounted electronics and provided for flexibility in mission- dependent harnessing.

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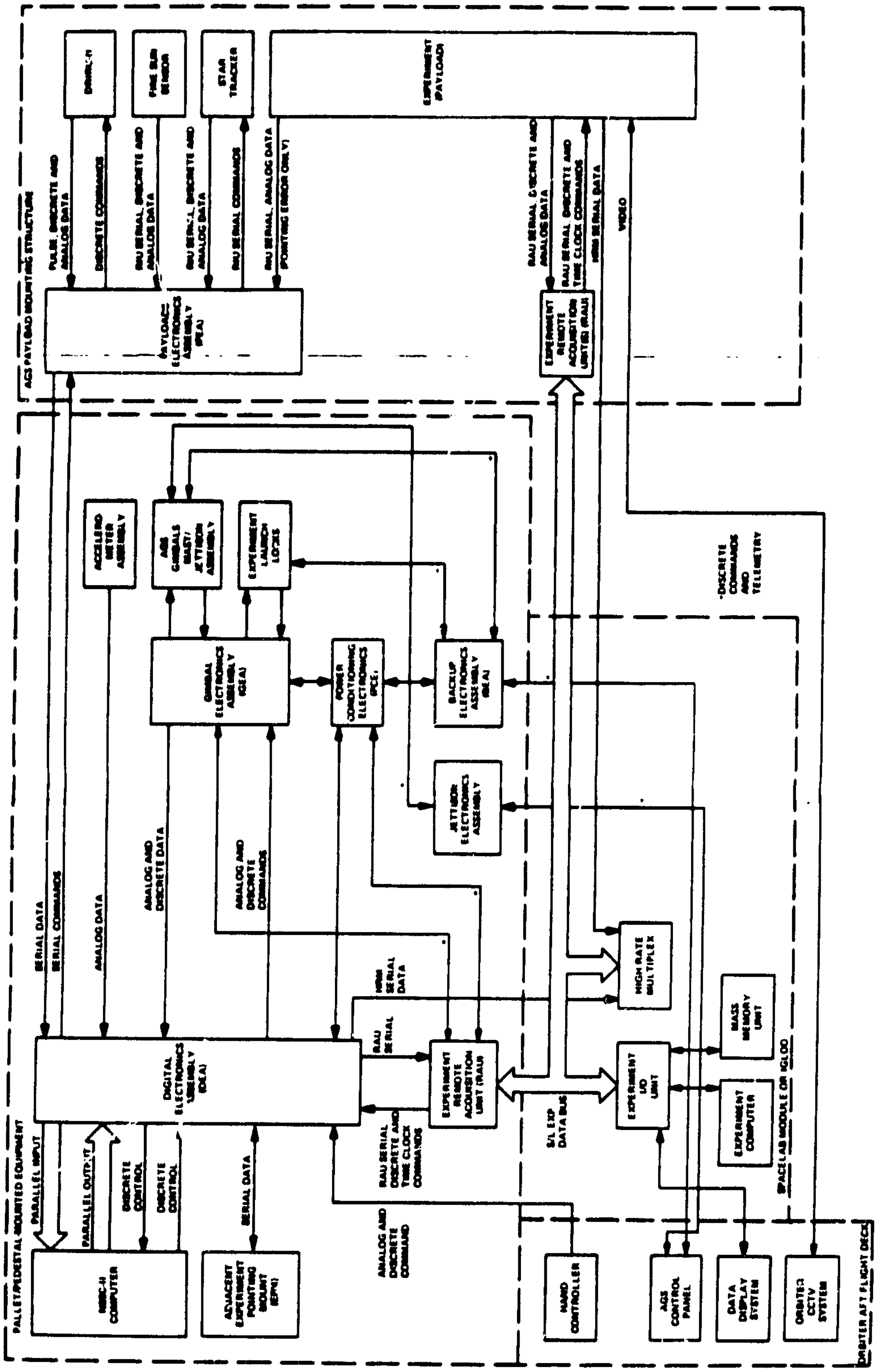

FIGURE 15 AGS CONTROL AND DATA INTERFACES

Power switching for all of the AGS electronics (sensors, GEA, BEA, PEA, DEA, and NSSC-II) and for the AGS heaters was performed in the Power Conditioning Electronics (PCE). The PCE also provided EMC filtering of the primary power utilized by the GEA and BEA.

3.6 MODIFIED ATM STAR TRACKER (MAST)

The MAST is a two-degree-of-angular-freedom, manually controlled, gimballed system that was used on STS-9 (Spacelab 1) as part of the "Atmospheric Emissions Photometric Experiment" (AEPI). The pointing mount (a modified ATM Star Tracker) and its associated electronics was designed by MSFC to accommodate the AEPI instrument and could be adapted to support instruments with similar requirements and mass properties. A general description of the MAST includes the description of several of the AEPI components and functions since these were required by the pointing system for power, command and control interfaces.

The AEPI experiment was designed to (1) observe both the natural and induced aurora, (2) observe air glow and other atmospheric phenomena, and (3) observe and measure contamination around the Shuttle/Spacelab. The major hardware components of the AEPI included Spacelab pallet mounted equipment and rack mounted equipment within the Spacelab pressurized module.

The Pallet mounted equipment, shown in Figure 16, was comprised of:

- MAST
- MAST Control Electronics Package
- A Load Isolator
- Linear Actuator
- Detector Assembly
- AEPI Support Pedestal
- Cradle/Locks Assembly
- Lens Covers

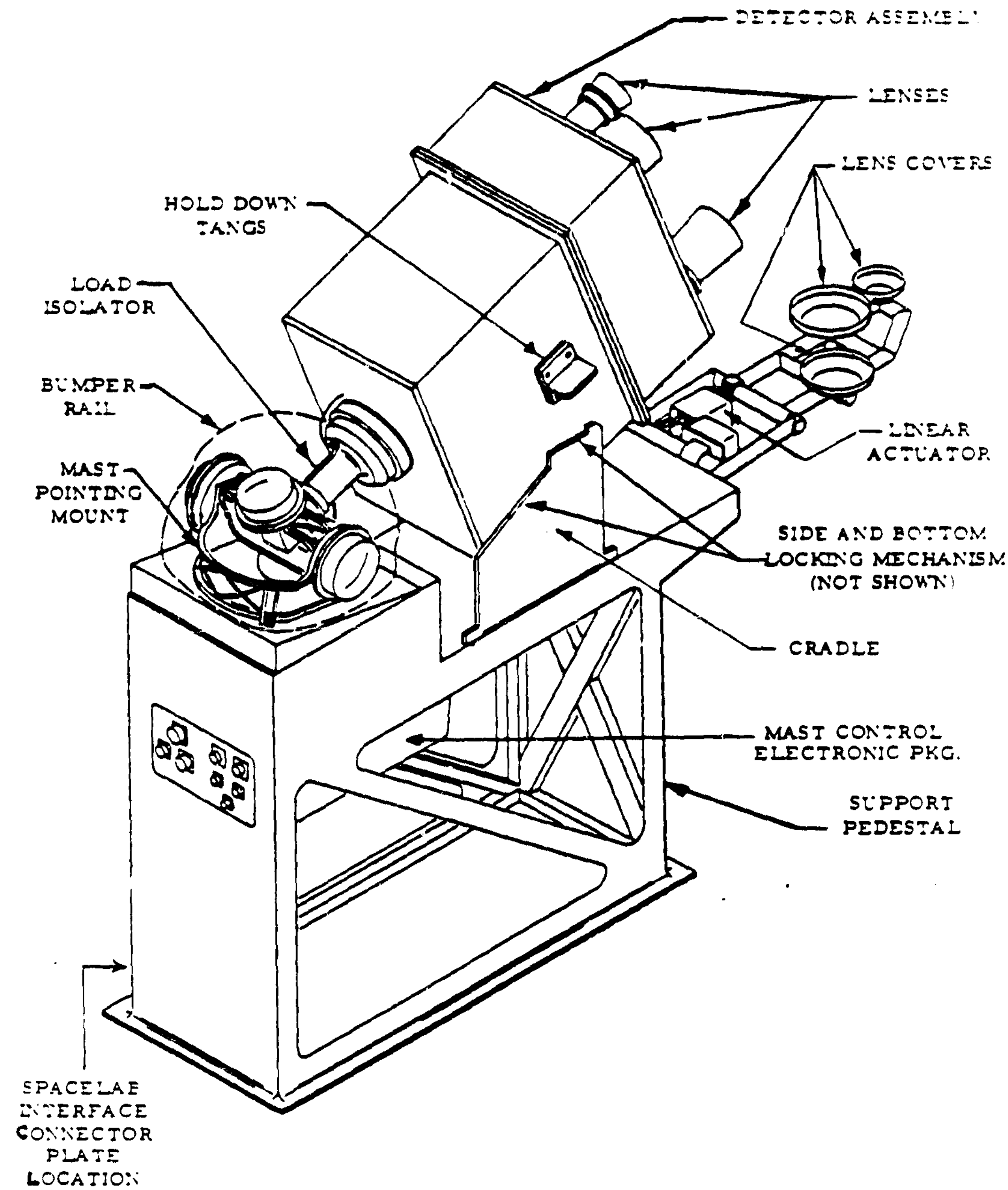

FIGURE 16 AEPI PALLET MOUNTED EQUIPMENT

The experiment detector assembly is affixed to the MAST pointing gimbal system. The lens covers (which also contained the experiment calibration source) are attached to the pedestal. A cradle provides for mounting of the locking mechanisms near the cg of the detector assembly. The MAST electronics box is located beneath the pedestal.

The rack mounted equipment, shown in Figure 17, located in the Spacelab pressurized module include:

- a Dedicated Experiment Processor (DEP)
- a Video Data Encoder (VDE)
- a Mount Manual Control (MMC) Panel

The MAST is a two-degree-of-angular-freedom gimbal system. Each gimbal has two pivots. Each pivot contains a duplex bearing pair. One pivot contains a multi-speed resolver, which permits accurate control of the angular position of the gimbal with respect to the MAST mounting plane. The other pivot contains a d.c. torquer, a d.c. tachometer- generator, and a system of flexible wires to carry electrical signals and power to the MAST components and the load isolator. The torquer is driven by servo-electronics to provide pointing control torques. The tachometer permits compensation for control stability, and is also the rate reference. Each pivot also contains a thermostat, a heater, and a thermistor for thermal control of the pivot environment.

The MAST Control Electronics Package is used for pointing control and for power distribution for the experiment. Relays in the box switch power to heaters, the detector assembly, and the pointing controls. They also control the shutter, load isolator and launch lock linear actuators. The analog servo system for the MAST gimbals are part of the box, as are the digital interfaces with the computer (DEP).

The load isolator is a device which mechanically connects the detector assembly package to the gimbal system. It contains a screw mechanism driven by a d.c. torquer. Driving the torquer in the extend direction decouples the detector assembly package from the gimbal system which is required to prevent damage to the gimbal system during launch.

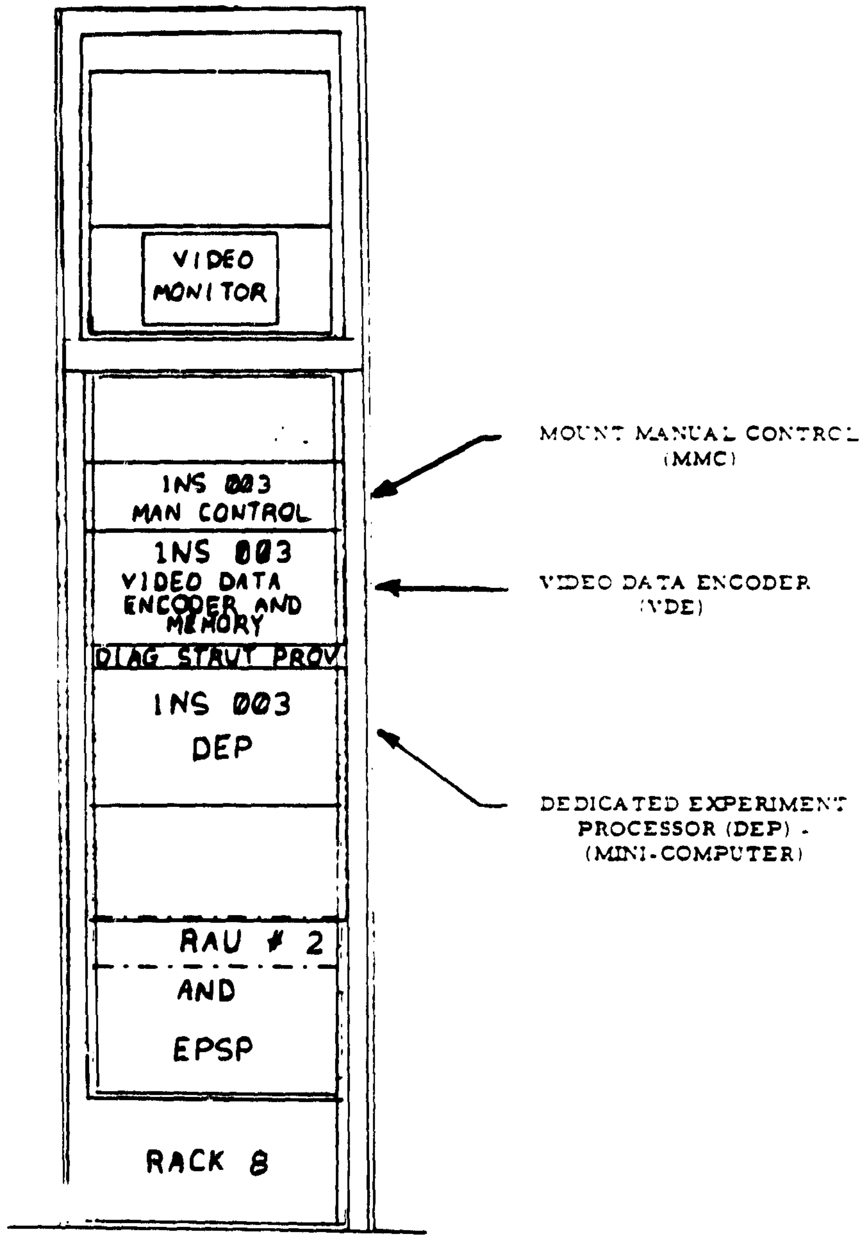

FIGURE 17 AEPI RACK MOUNTED EQUIPMENT

Driving the torquer in the retract direction mechanically couples the detector assembly package to the gimbal system which is required for pointing in orbit. The load isolator is launched in the extend (decoupled) position. Once in orbit, it is retracted (coupled), and remains this way until the end of the mission.

The detector assembly contains two cameras and three lens systems and associated control electronics. One of the cameras is a low light level television camera (LLLTV) and can operate through either of two lens systems, 5 or 20 degrees field of view. The other camera is a 10 x 10 diode array which operates through a separate 5 degree field of view.

The linear actuator is an electric motor provided to open or close the TV lens covers and to actuate the lock mechanism (close or open).

The lens cover assembly is mounted on the open end of the TV lenses. The covers in this assembly cover the input aperture of the TV lenses during periods when the camera is not in operation and when the cameras are being calibrated. The cover assembly is released to the open or closed position by a linear actuator within the assembly.

The detector assembly is held in position during boost phase and re-entry by launch locks. The locks are contained in a structural "U"-shaped cradle that engages the detector assembly at the two sides and bottom surface. The lock mechanisms are actuated by three linear actuators.

The support pedestal is constructed of fiberglass material. Fiberglass structure is used because of its low thermal conductivity. The modified ATM star tracker, the MAST control electronics package, the detector assembly experiment, the stowage device (cradle), lens covers, and the "launch locks" locking mechanism are all mounted on the pedestal. Thermal insulation blankets cover all external surfaces of the pedestal and equipment with the exception of the star tracker gimbals and cradle/lock assembly, which will be covered to the maximum extent possible.

The DEP is an off-the-shelf minicomputer which is used to compute pointing commands and to cycle the detector assembly for its specific functional objective.

The VDE is the control panel for the detector assembly. It provides power to the detector assembly and interfaces the camera and video display. It facilitates operator interaction with the experiment when under computer control. In addition, it also provides an emergency lock switch, which automatically stows and secures the detector assembly.

The MMC panel has the basic tasks of controlling experiment mechanisms involving the pointing system, shutter system, locks system, and manual stowing of the experiment system should compute controlled stowing fail. The mount manual control system receives primary power (28 Vdc) from the VDE and essential power (28 Vdc) from the Spacelab. The MMC provides manual control of the launch locks, load isolator, lens covers, calibration source, and servo-electronics. In the event of a major system failure, the MMC provides switch capability to the servo-electronics from the primary power mode to the secondary system (essential power). The MMC also provides for manual operation of the gimbal system to stow the experiment and activate the launch locks.

3.7 THE INSTRUMENT POINTING SYSTEM

The Instrument Pointing System (IPS) was developed at Dornier Systems under contract from the European Space Agency (ESA) as a multipurpose, precision pointing system for Spacelab missions. Figure 18 contains an assembly view and an exploded view of the IPS. The IPS is a three-axes gimbal pointing system, which points an inertially stabilized mounting plate on which an end-mounted payload is attached.

The total operational range of the gimbals is ±180 degrees about the roll axis and 30 degrees to 60 degrees, 1/2 cone angle, about the two lateral axes. The range of the lateral gimbals is limited by adjustable mechanical stops. For safing any uncontrolled motion of the gimbal structure during an orbit operation, the Gimbal Support Assembly has a bumper device which is adjustable to different angles before the flight and prevents any damage to the Orbiter, Spacelab or Payload. The payload may be a single or a clustered scientific array of instrument payloads having varied mass and dimensions that observe inertially fixed or moving targets.

The IPS design permits pointing of payloads with mass properties up to 7,000 kg, moments of inertia about their C.G. up to 30,000 kg-m^2 about the lateral axes and 12,000 kg-m^2 about the roll axis and C.G. offsets from the payload interface plane up to 3 m. Payload masses above 3,000 Kg will require modifications to the Payload Clamp Assembly (PCA), which clamps and structurally holds the payload during launch, ascent, reentry and landing phase of the flight. Payload dimensions are restricted by the width and available length of the Spacelab pallet and diameter restrictions associated with the PCA. Wiring for payload power, for three RAUs and for direct data transmission to Spacelab/orbiter, is provided across the gimbal at the payload mounting plate.

Attitude control of the payload is based on the inertial reference of a three-axis gyro package mounted on the payload mounting plate that is updated by star trackers or sun sensors mounted on the payload. A Digital Control Unit (DCU) also mounted on the payload mounting plate, governs all slew maneuvers, payload stabilization and tracking operations by issuing commands to the three-axis gimbal drive assembly. A three-axis accelerometer package, mounted to the IPS support structure, measures accelerations caused by external disturbances. Outputs from the accelerometers are fed forward to the primary feedback control loop to allow for compensation of external disturbances. The pointing performance requirement for the IPS are listed in Table 9. The disturbance

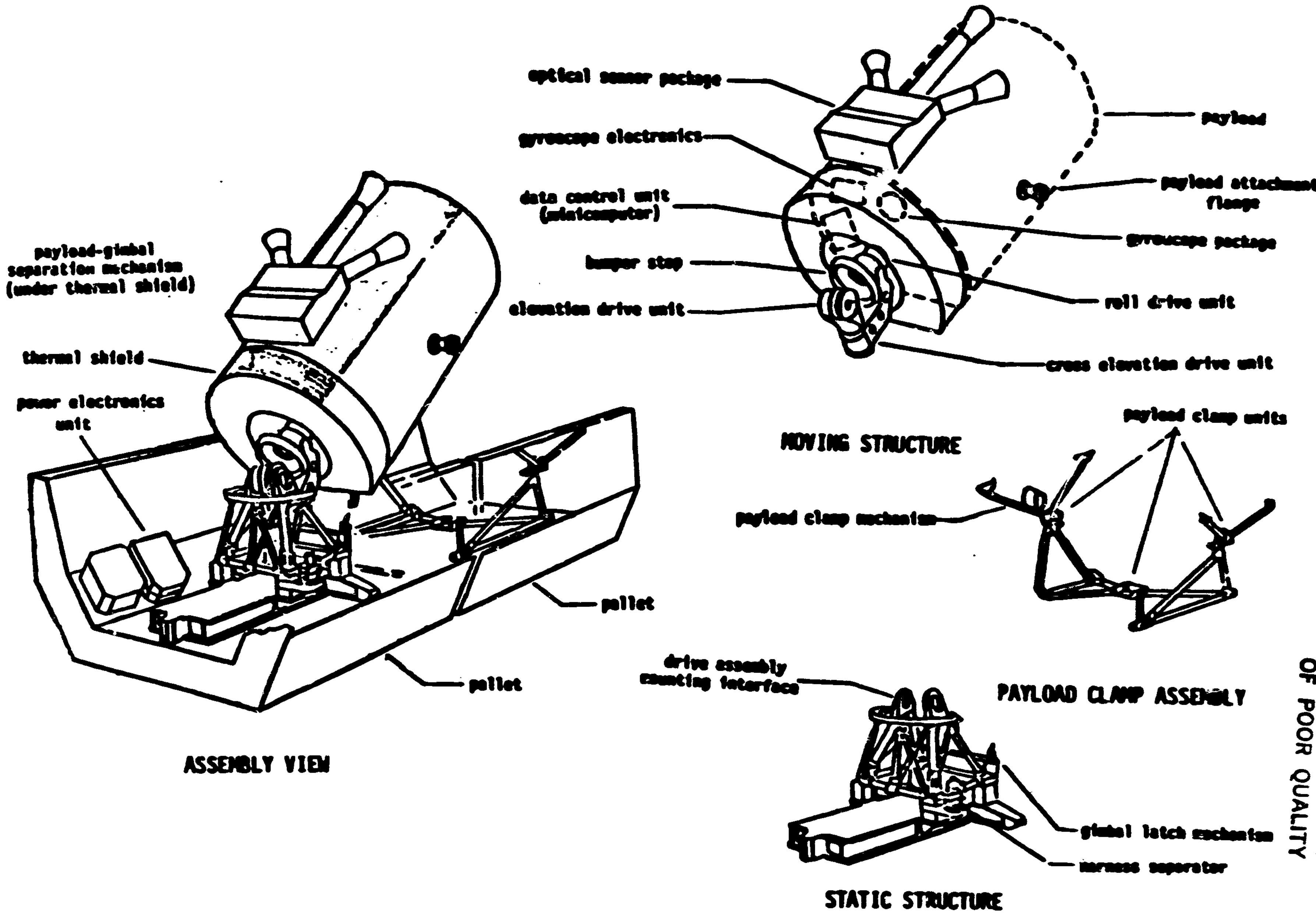

FIGURE 18 IPS ASSEMBLY

TABLE 9. POINTING REQUIREMENTS OF THE IPS		
PERFORMANCE REQUIREMENTS	IN THE 2 AXES PERPENDICULAR TO THE EXPERIMENT LINE-OF-SIGHT	ABOUT THE ROLL AXIS
Pointing accuracy of the experiment line-of-sight	2 arc sec	20 arc sec
Quiescent stability error	1.2 arc sec	3 arc sec
Man-motion disturbance	4 arc sec	15 arc sec
Stability rate	60 arc sec/sec	130 arc sec/sec

response requirements listed apply for a 2,000 Kg payload with the IPS center of rotation (COR) located on a "Z" axis through the Orbiter center of gravity (c.g.). For a 200 Kg payload, the disturbance error is required to be less than 1.7 times that of the 2,000 Kg payload.

The IPS has seven major assemblies, which are:

- Gimbal Structure Assembly
- Attitude Measurement Assembly
- Power Electronics Unit
- Data Electronics Assembly
- Thermal Control Assembly
- Payload Clamp Assembly
- Harnesses

The total mass of the IPS, excluding the PCA, is approximately 1,176 Kg.

The Gimbal Structure Assembly incorporates all static and moving mechanical structures, drive units, stow/unstow mechanisms, and all special devices mounted onto it.

The Attitude Measurement Assembly consists of a three-axis gyro package and an optical sensor package that provide inertial and optical references for IPS attitude control.

The Power Electronics Unit distributes power to all other IPS elements and to the Payload. It also provides contingency control functions to enable manually controlled contingency retraction or jettisoning of the system.

The Data Electronics Unit contains the circuitry which processes the signals required to maintain the IPS attitude control. It also contains the three-axis accelerometer package, which enables feed forward control of disturbances introduced by external sources.

The Thermal Control Assembly protects critical components from the extreme temperature variations of space.

The Payload Clamp Assembly (PCA) is a mechanical structure designed to secure the payload whenever it is separated from the IPS. It clamps and structurally holds the IPS payload and payload attachment ring during launch, ascent, reentry and landing phases of a flight. The PCA holds the payload in a statically determinate manner by three attach flanges lying in a plane through the payload center of mass.

The IPS wiring harness provides all the necessary interconnection of the electronic units and assemblies. It is subdivided into Pallet Harness, the Gimbal Support Assembly Harness and the Payload Interface Harness.

The IPS electrical configuration envolves extensive interfaces with the Spacelab Command and Data Management System (CDMS), and the Electrical Power Distribution System (EPDS). A functional Block diagram of IPS electrical and data interfaces is snown in Figure 19 .

Pointing, tracking, and stabilization control of the IPS is maintained by a digital control loop operated within the IPS DCU and the CDMS subsystem computer. Data flow within the control loop is seperated into two computation loops: fast loop computations operating at 25 Hz between the DCU and IPS attitude components, and slow loop computations operating at 1 Hz between the DCU and subsystem computer.

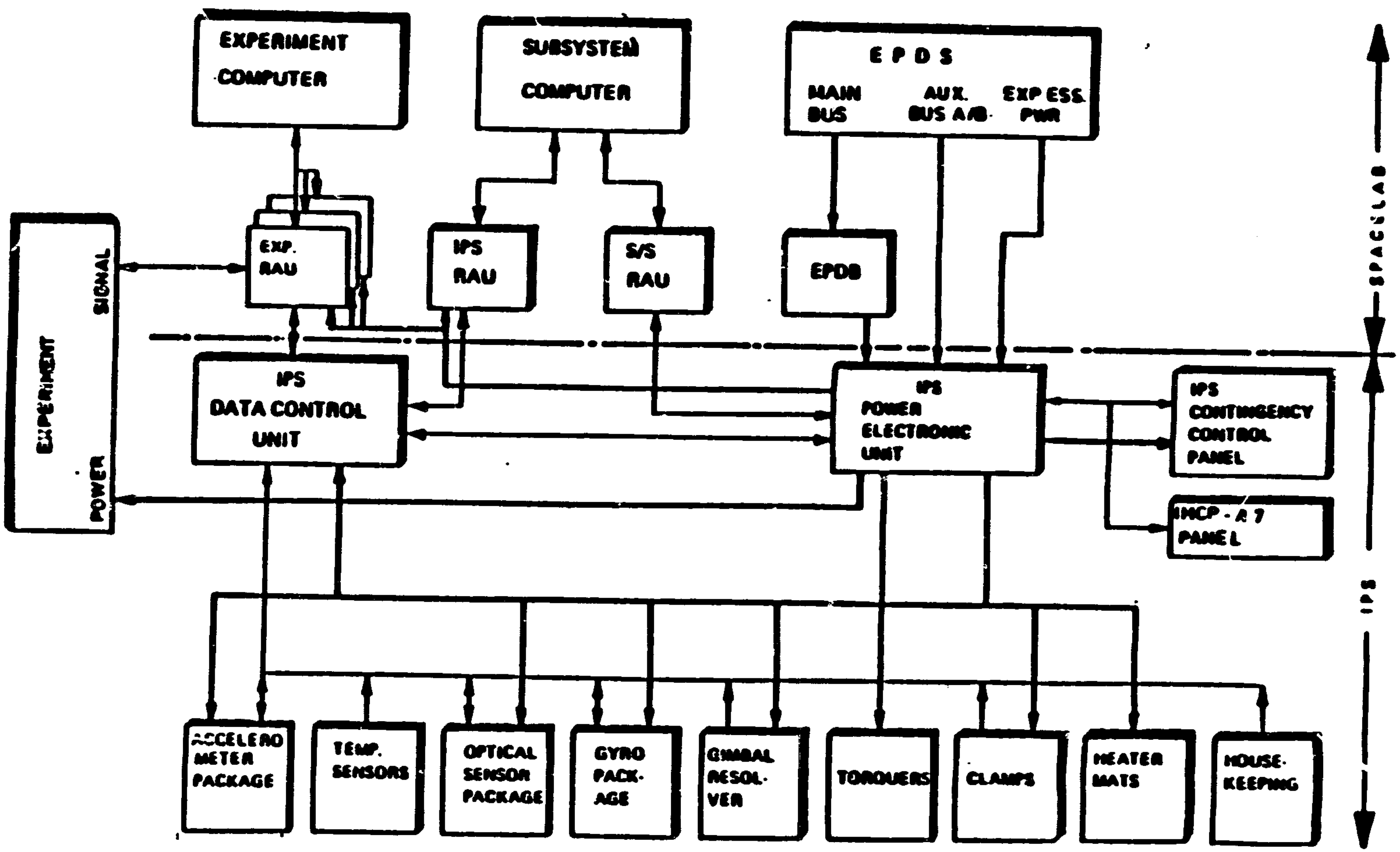

FIGURE 19 IPS ELECTRICAL AND DATA INTERFACES

The IPS also contain all software necessary for monitoring and exercising ground and mission operations. A modular software structure is used to facilitate modifications and to minimize the demand on the Spacelab CDMS.

The IPS Attitude Control loop is shown in Figure 20 .

The fast loop computations are based on the inertial reference furnished by a three-axis rate-integrating gyro package mounted on the equipment platform that sends 100 Hz signals to the DCU. The DCU averages the data over every four samples, integrates by using quaternion notation, and compares it with the desired attitude for generation of the attitude errors to be used in the subsequent control arithmetics. When the inertial reference is disturbed by Orbiter and/or IPS axis deviations, the DCU issues control signals to the PEU to apply control torque to the drive unit torquers. In this manner, any attitude discrepancy is compensated for by the IPS automatically moving to a corrected attitude.

In addition, a three-axis accelerometer package mounted onto the supporting framework measures accelerations caused by Orbiter perturbations and sends a feed-forward control signal to the DCU. The DCU adds this signal to the fast loop computations to compensate for the accelerations.

To compensate for mechanically-induced gyro drifting, slow loop computations are used to update the inertial reference of the fast loop by taking optical measurements supplied by the optical sensor package (OSP). The OSP consists of three fixed head star trackers mounted on a precision base. For stellar missions, a boresight and two skewed trackers provide IPS line-of-sight and roll attitude references respectively. For solar missions, the boresight tracker is reconfigured as a solar sensor with the two skewed trackers maintained for roll references. The OSP signals are sampled once a second by the subsystem computer and processed by a Kalman filter to generate the attitude and rate updates. The OSP is mounted on the payload to assure optimum viewing and to minimize misalignments between the IPS line of sight and the experiment sensor line of sight.

Besides using the OSP for pointing references, attitude offset commands generated by the experiment computer or experiment sensor signals can be used to provide additional pointing references.

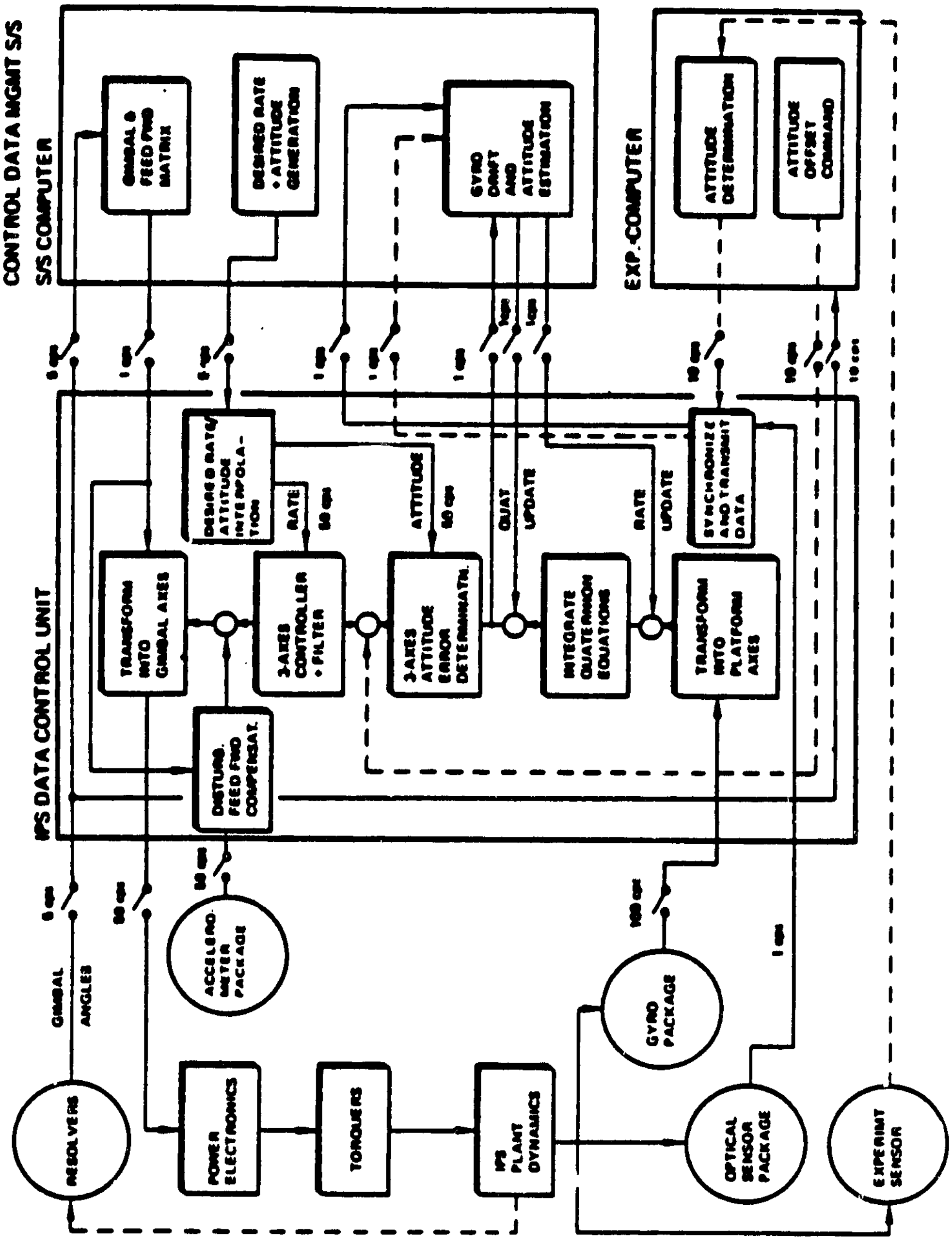

FIGURE 20 IPS ATTITUDE CONTROL LOOP BLOCK DIAGRAM

3.8 GIMBALFLEX

Gimbalflex is a fine pointing system, which has been under development by the Martin Marietta Aerospace Co. The system is meant to serve as a vernier stage on top of another gimbal system such as IPS or AGS, as shown in Figure 21 . The system combines a linear isolation system with rotation axis control to achieve a fine pointing system with vibration isolation. Figure 22 illustrates the basic mechanical configuration. Vibration isolation is made possible by the use of flex hinges and an intermediate gimbal. As shown in Figure 22, the outer linkage set allows motion along the Y-axis between the payload and the gimbal. The inner linkage allows motion along the X-axis between the gimbal and the base body or base pointing system. This provides complete isolation in the X and Y planes. Angular flexures between the linkages and the gimbals allow payload pointing about the Z-axis. Direct drive brushless torquer motors are used to apply the pointing and stabilization torques. This linear isolation system provides a mechanically stiff path back to the base body along all rotational axes. Linear isolation frequencies must be high compared to the angular control frequencies. An active control system can be used so that the isolation frequencies can be adaptive in flight. The isolation system will not remain orthogonal when a large angle pointing is required. Therefore, this system is limited to angles below 15 degrees of rotation.

Based on an early presentation to MSFC by MMC [33], the system could accommodate a payload mass up to 500 Kg and achieve a pointing stability of .01 sec. The system would provide linear motion of ±1 inch along two-axes and ±10 degrees of angular rotation about two axes. The gimbalflex system would weigh about 50 lbs.

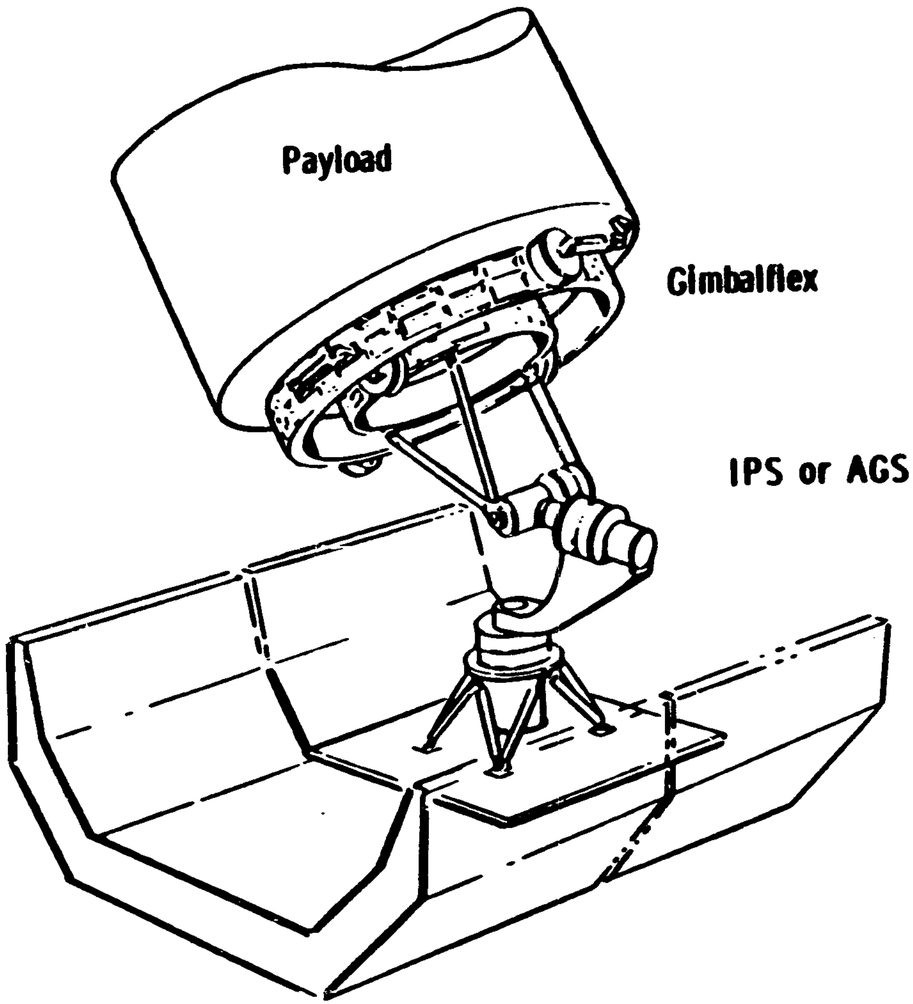

FIGURE 21 GIMBALFLEX - A VERNIER STAGE

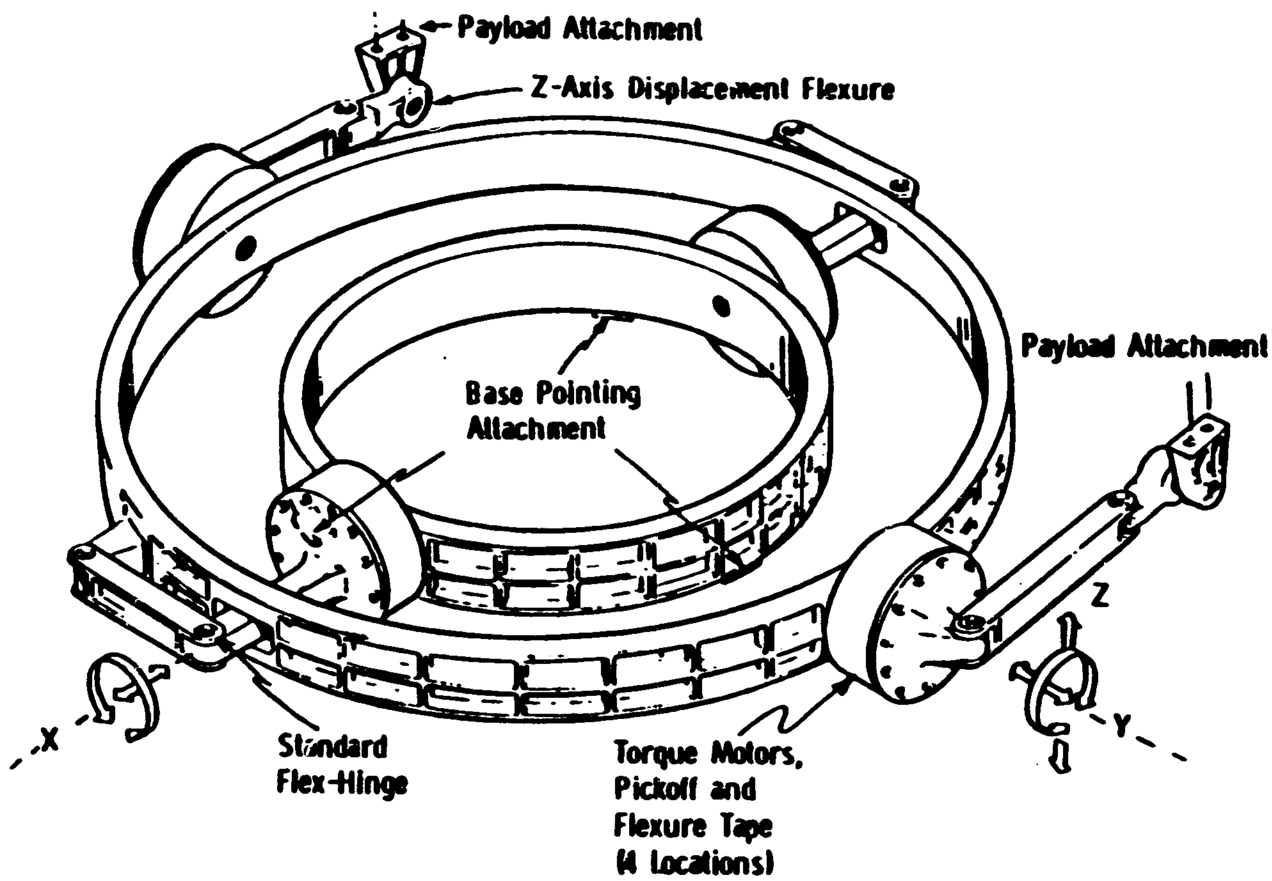

FIGURE 22 GIMBALFLEX MECHANICAL CONCEPT

3.9 SUSPENDED EXPERIMENT MOUNT (SEM)

The Suspended Experiment Mount (SEM) is a payload isolation and stabilization system concept that has been under study at MSFC for several years. As shown in Figure 23, the CMG stabilized structure would mount in the Orbiter payload bay and could be constructed from Skylab spare parts and other existing hardware. The system consists of a spacelab pallet (or pallet train), which is isolated with respect to the shuttle by means of passive spring dampers or other non- rigid suspension devices. These devices would isolate the payload from high frequency disturbances and the CMG system would provide active control of low frequency disturbances and stabilization of the LOS. Three (possibly four) Skylab type double gimbal CMGs mounted on the pallet would provide pointing stabilization to 1 arc second for pallet mounted payloads.

Besides isolating the pallet from vibrations, the isolation system allows coarse attitude control of the Orbiter via the CMG's mounted on the pallet. Thus, the Orbiter Vernier Control Cystem (VCS) is not required to operate. Although especially suitable for instruments with the same viewing requirement, instruments with multiple viewing requirements can be accommodated by the use of simple position and hold pointing mechanisms. The capability of these pointing mechanisms would be determined by the payload requirements. In addition to viewing type instruments, this concept also has the capability to provide a disturbance free environment for other payloads that are sensitive to relatively small accelerations.

The suspension system performs two functions. It isolates the payload from high frequency disturbances and allows low frequency control of the orbiter during experiment operations. It must also restrain the payload during periods of high dynamic loads. Therefore, it must consist of a flexible coupling that can be rigidly locked during periods of high loads. Figure 24 illustrates a suspension/retention method considered by MSFC during its studies on SEM. It uses standard orbiter active sill and keel trunnion fittings for rigid attachment. The flexible suspension incorporates a linear actuator to lift the payload out of the trunnion fittings during operation. This approach has the advantage of using standard orbiter fittings during the critical flight periods. MSFC has considered other concepts for active retention devices which have the advantage that the SEM can be released without requiring a linear displacement of the payload.

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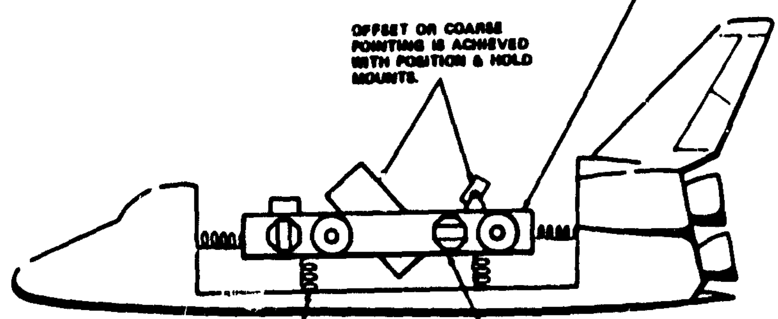
CONCEPT



PRINCIPLE OF THE SEM

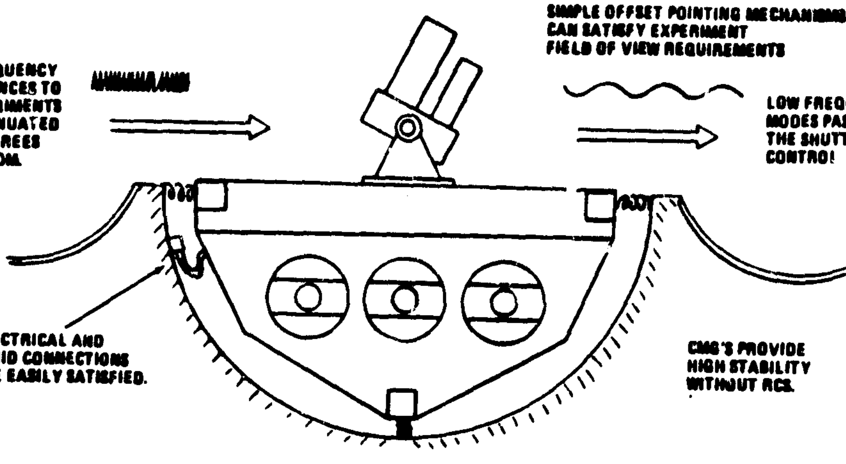


FIGURE 23 SEM CONCEPT AND PRINCIPLE

Helical springs are shown for the flexible isolator in Figure 24, but other options have been considered. These include: elastrometric isolators, gas filled bellows, wire rope helical springs, and the solid wire helical spring, illustrated in Figure 24. Gas filled bellows and solid wire helical springs are the prime candidates for the flexible isolator, since they exhibit a more linear "load versus deflection" characteristic.

The SEM can be built using Skylab type Double Gimbal CMGs, although some modifications to the Skylab CMGs have been considered for a more state-of-the-art configuration. Reaction wheels can also be used in lieu of CMGs, depending on the torque capability required. The CMG system is used to control both the payload and, through the suspension system, the Orbiter. Avoiding Orbiter thruster firings reduces disturbances to the payload and reduces contamination, which is a significant benefit to many payloads.

Momentum saturation is a concern for any momentum based attitude control system. After a period of operation, possibly several orbits, non-cyclic disturbance torques acting on the vehicle, result in net CMG angular momentum build-up. This momentum build-up eventually causes the CMG to become saturated and not capable of compensating for disturbances about its control axis. Desaturation or momentum damping requires that an external torque be applied to the vehicle to absorb the accumulated angular momentum (i.e., realign the CMG momentum vector with its vehicle control axis). Use of the orbiter thrusters is the simplest method of providing this torque. Another possibility is to use gravity gradient torques either by flying an attitude that produces negligible non-cyclic torques or by periodically maneuvering to an attitude where gravity gradient and aerodynamic torques cause an accumulation of angular momentum in a desired direction. In the case of gravity gradient dumping of momentum, offset pointing mechanisms would be required for the payload, since the Orbiter attitude would not be compatible with the payload viewing direction.

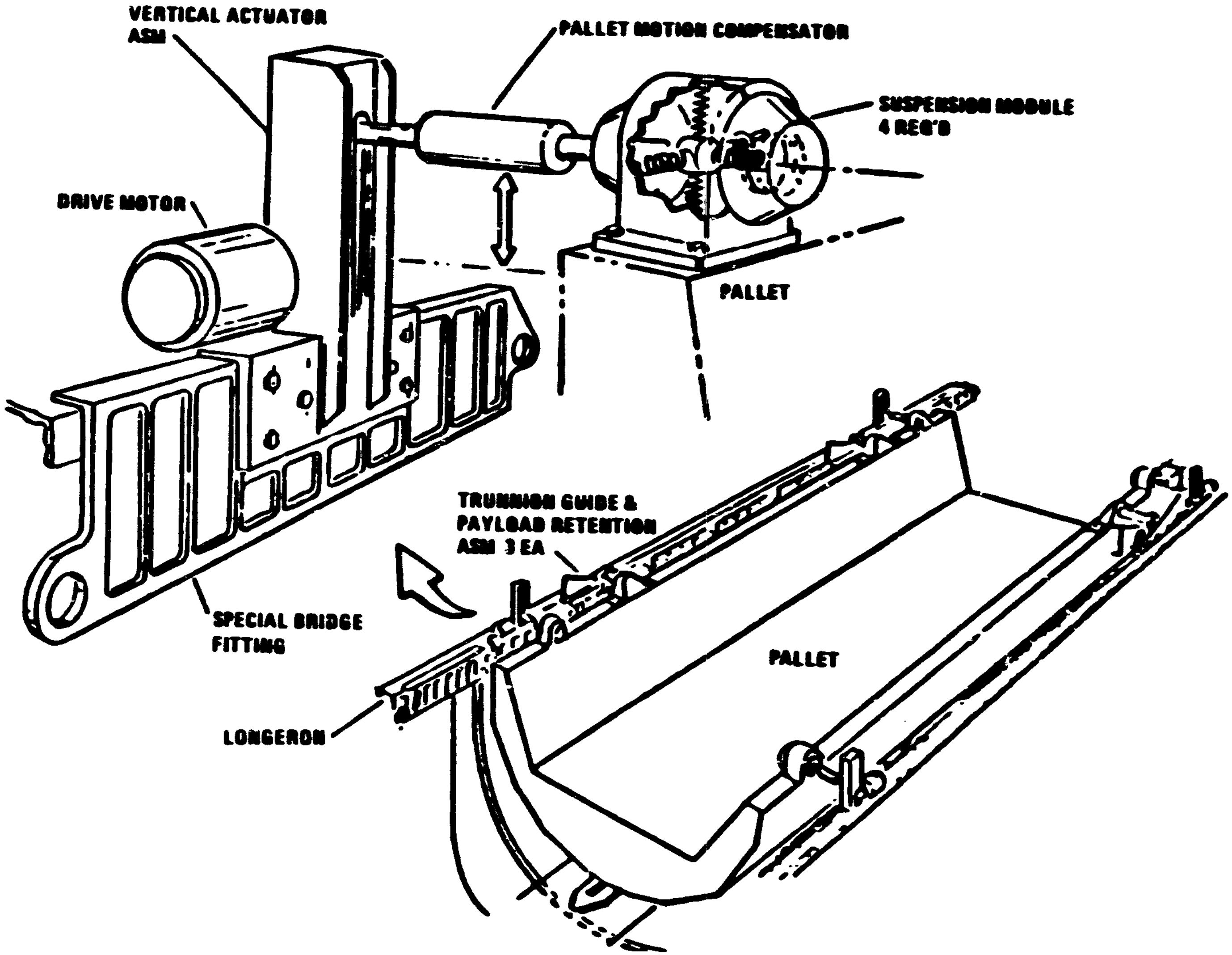


FIGURE 24 SEM RETENTION AND SUSPENSION CONCEPT

The SEM is intended to provide stabilization and isolation, not offset orientation, so the free motion of the suspension system only needs to accommodate the disturbance motions. These should be relatively small. Because of these relatively small motions of the SEM with respect to the Orbiter, it should be easy to provide power, communications, fluids, etc. from the orbiter to the payload. Therefore, it provides the payload with the pointing accuracy and disturbance isolation of a free-flying spacecraft and retains advantages associated with being attached to a base body.

As shown in Table 10, the total weight of a complete SEM for the Orbiter/Spacelab application is 2090 Kg. A "One Spacelab Pallet SEM" would be capable of accommodating payloads up to 1675 Kg. A two-pallet train could accommodate up to 3550 Kg of payload.

TABLE 10. SEM WEIGHT SUMMARY

SPACELAB PALLET WITH CMG'S AND SUSPENSION SYSTEM

	WEIGHT Kg	(LB)
1 Experiment Support Structure	390	(840)
1 Suspension System	100	(220)
3 CMG's plus CMGEA's	570	(1257)
3 CMGIA's	69	(152)
3 Rate Gyro's	16	(35)
1 Computer & Electronics	32	(70)
1 Experiment Microprocessor	32	(70)
1 Star Tracker Opto-Mech	18	(40)
1 Star Tracker Electronics	15	(32)
1 Signal Conditioner	14	(30)
1 Power Controller	15	(32)
1 Control & Monitor Panel	7	(15)
1 Recorder & High Rate Multiplexer	71	(156)
1 CRT Keyboard System	14	(30)
Subtotal	1363	(2999)
Pallet Equipment Attachment Hardware	70	(154)
Total	1433	(3153)
1 Spacelab Pallet	657	(1444)
Mission Total	2090	(4597)

3.10 TALON GOLD GIMBAL (TGG)

The Talon Gold Gimbal (TGG) is a three-degree-of-freedom, micro-processor controlled pointing system, which was developed by LMSC for DOD as part of the Talon Gold Experiment. The experiment, designed to be operated from Shuttle, was to demonstrate target acquisition, pointing and tracking capabilities. The system, illustrated in Figure 25, included the instrument sensor, a vibration isolation pointing system (VIPS) and the TGG. The system was required to provide a pointing accuracy of 0.2 u rad (.04 arc sec).

The experiment was mounted on the VIPS, which was developed by the Sperry Flight Systems Division. The VIPS is a magnetically suspended vernier pointing stage, which provides non-contacting isolation and fine positioni.., of the payload. The system is similar to the magnetic suspension stage used in the ASPS system previously described. Using a quality sensor, provided as part of the experiment, the vernier stage could achieve a pointing accuracy of C.1 arc sec and a stability of 0.01 arc sec. The Talon Gold Gimbal provided the coarse pointing capability for the experiment.

The pointing, tracking and stabilization control system utilized five Motorola MC68000 micro-computers which were distributed across the system's architecture. One MC68000 was included at each gimbal joint, one was l- -ted on the instrument mounting plate, and one on the mounting interface with the Shuttle. The use of these micro-computers which have an internal data bus structu-- rovided a self-contained system which required no additional electronics or compute. support from Shuttle. Attitude sensors include precision rate gyros and star trackers mounted on the payload side of the gimbals. Outputs from an accelerometer package, also mounted on the instrument-mounted surface, were used in feed forward control loop to compensate for disturbances before they could propagate to the instrument. The precise pointing and stability was ultimately achieved by using an error signal generated by a fast steering mirror system located internal to the payload.

The TGG itself is a precise payload pointing system, which incorporates many state-of-the-art features to achieve low jitter, high accuracy, quick response, and near-zero friction.

The gimbal joints use gearless direct drives and are supported by flexure bearings nested inside rolling element bearings. This eliminates problems associated with backlash in

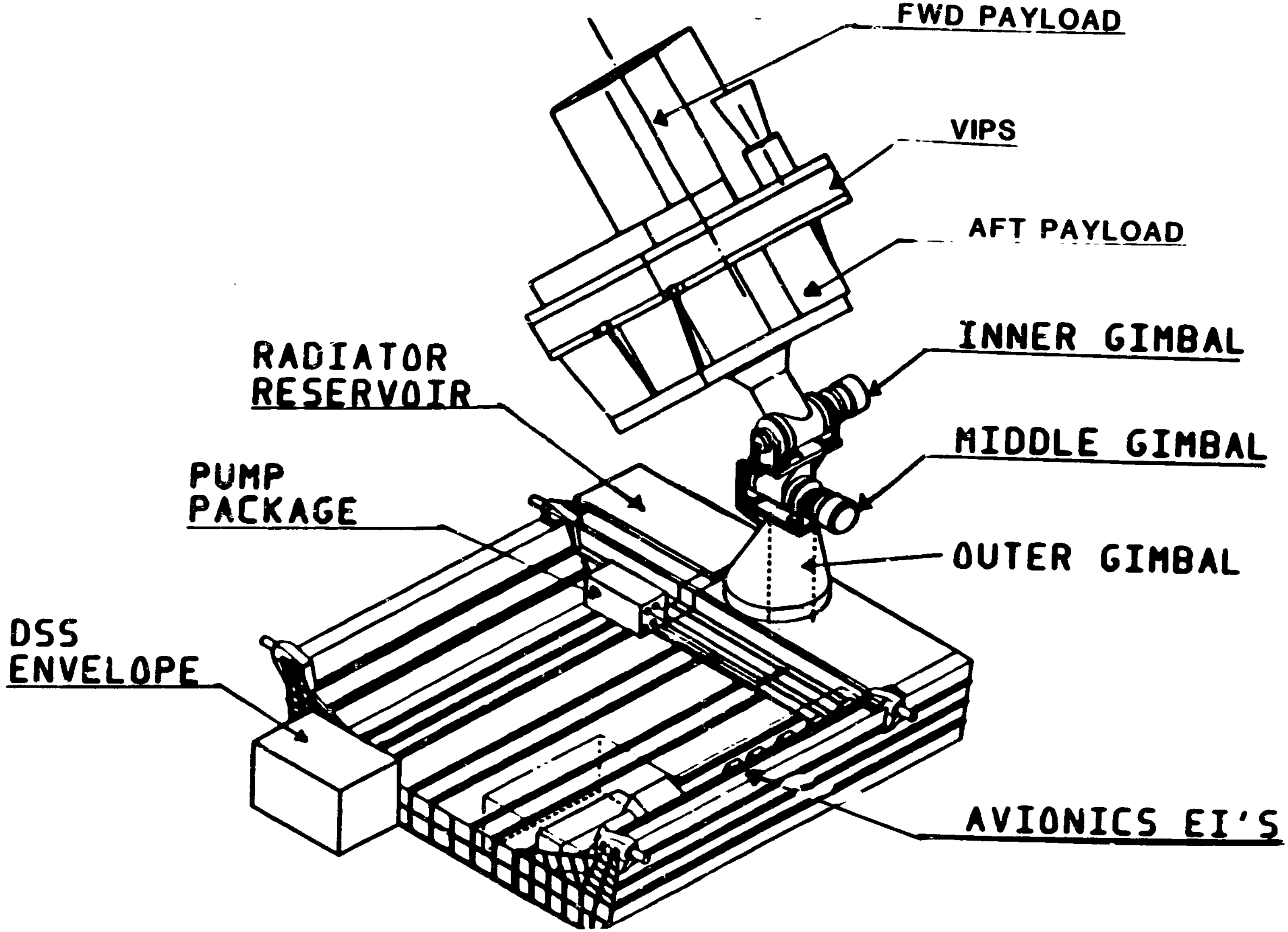


FIGURE 25 TALON GOLD SYSTEM

geared systems. The use of flexure pivots improved stability and vibration isolation relative to other systems. The gimbal joints contain rotary transformers, rotary fluid couplers, capacitive couplers and encoders mounted on the axis of rotation. Electrical power, instrument signals, fluids, angle and rate data are transferred across the joints without cables.

The rotary transformer can transfer up to 5 KW of 120 V at a frequency of 10 KHz to 30 KHz across the gimbals. Use of this device provides several advantages: 1) no wear or wear products, 2) no arcing, 3) no friction, 4) high electrical performance.

The rotary fluid coupler is a two part assembly having its outer housing fixed and inner spool rotating. Fluid enters the outer housing through a Swaglok fitting and is ported into an annular chamber that interfaces with the inner spool outer surface. The outer housing annular chamber edges are sealed from the rotating inner spool by O-rings to prevent axial flow on the spool surface. An opening in the spool surface permits radial flow into one of several cylindrical chambers adjacent to the spool axis. The fluid exits this spool chamber through a Swagelok fitting, flowing in the axial direction. The return flow is a path parallel to the input flow, but in the opposite direction. A rotary coupler has the advantages of eliminating line flexure fatigue failures, reduced fluid line bulk, weight, cost and unlimited rotation.

The fluid coupler contains four ports, two for pressure and two for return. Weight is 2.7 kg (6 lbs); construction is steel. The torque required to rotate the spool inside five seals is approximately 1.02 N-m (9 lb-in). The pressure drop at 11.4 liters/minute (3 gallons/minute) flow is less than 6.89 kPa (1 psia), with 60% ethylene glycol and 40% water.

Rotary capacitive couplers are used to transfer signals across the gimbal joints. They provide advantages of minimum wear, low disturbance torque, and high reliability. A digital line driver transmits a differential charge to a pair of driver capacitor plates. An opposing differential charge appears on a pair of receiver plates located close to the driver plates. The two plate pairs make up a pair of parallel capacitors with properties that are independent of the plates relative rotary motion. A digital line receiver detects the differential signal transmitted via the capacitor charges. The capacitor plates are matching conductive rings on a pair of circular printed circuit boards. There are four matching rings which make four capacitors per channel. Two ring pairs transfer the differential signal. While the other two ring pairs are grounded guard rings on each side of the signal capacitors. The guard rings isolate these signal capacitors from the signal capacitor of adjacent channels to minimize cross- talk. The signal channels are further isolated by interleaving primary and redundant channels, the primary and redundant channels never being used simultaneously. A twelve inch diameter circular printed circuit board pair provides up to ten signal channels. Electronics are mounted on the back sides of the capacitor boards. The transmitted signals are digital, with thresholds set to make the coupler insensitive to plate spacing. Analog signals are converted to digital for transmission, and then reconverted to analog at the output.

The gimbal encoders generate angles with 21 bits (1/2 arc sec) resolution, and analog rate with dynamic range equivalent to 14 bits resolution. The rate signal is derived directly from the encoder optical detector analog waveform, using an analog circuit and discrete digital logic. The rate signal lag is much lower and the resolution is much finer than a rate that is computed digitally by differencing the 21 bit digital angle signal. This fine rate analog signal is instrumental in gimbal low jitter control.

The encoder angle signals are generated by light emitting diode illumination passing through rotary disk windows onto light detectors. Stationary apertures overlay the detectors to outline their fields of view precisely. There are 9 inner gray code rings of windows on the disk and one outer analog signal ring. The analog ring produces 2 triangular signal waveforms that are 90 degrees out of phase.

Absolute angles are assembled from the straight center segments of the analog waveforms and the gray code. An automatic gain control (AGC) analog loop controls the illumination to effect coarse regulation of the optical transmission gain. The angle assembly is done by a microcomputer , which also regulates the fine AGC.

Rate signals are derived from the triangular waveform analog displacement signals, using an analog rate circuit. For constant rate, the waveforms are square. A demodulating sampling circuit converts the signals to "dc" rate. The uC controlled fine AGC is implemented by the digital to analog (DA) and the multiplying digital to analog (MDA) converters.

The TGG can accommodate payloads up to 2400 Kg, and moment of inertia of 11,400 Kg-M^2 about the gimbal axes. The approximate dimensions of the Talon Gold experiment was 10' x 7' x 8' (3m x 2.1m x 2.4m). The pointing system control bandwidth is 0.5 Hz. The TGG can achieve a pointing accuracy of 3 arc sec and a stability of 0.6 arc sec. Torque capability is 70 ft-lbs (95 n-m) per axis when primary and redundant motors are used simultaneously in emergency slewing. The TGG can slew payloads up to 3 degrees/sec and provide acceleration up to 400 arc sec/sec^2.

3.11 HUBBLE SPACE TELESCOPE

The Edwin P. Hubble Space Telescope (HST) is included here since it includes the highest precision pointing system developed to date. It demonstrates that the technology required for highly precise pointing systems is available. In addition it represents one of the free flyers that may utilize the S.S. for servicing. The ST pointing and control system (PCS) uses a combination of rate gyros, reaction wheels, star trackers and interferometric fine guidance sensors to slew the ST from target to target and accomplish precise pointing and stabilizing accuracy. The PCS points the ST at the target object to 0.01 arc seconds and is required to achieve pointing stability of 0.007 arc seconds over a 24 hour period.

The anticipated mission operational lifetime is fifteen years. The scientific instruments and some of the electronics are modular design; i.e., orbital replaceable units (ORUs), so that they can be replaced with improved or different models. Periodic orbital maintenance and altitude reboost via the Shuttle is planned. For these reasons, the facility could operate in space for decades. If necessary, it will be possible to retrieve the ST, return it to earth, perform maintenance and repair, and relaunch the facility.

Figure 26 shows the general operational configuration of the ST. It weighs 11,600 Kg, is 14.3 m long, and 4.3 m in diameter. An exploded view of the ST is shown in Figure 27. Major elements of the ST are: the Support System Module (SSM), the Optical Telescope Assembly (OTA), Science Instruments, Science Instrument Control and Data Handling, and the Solar Array (SA).

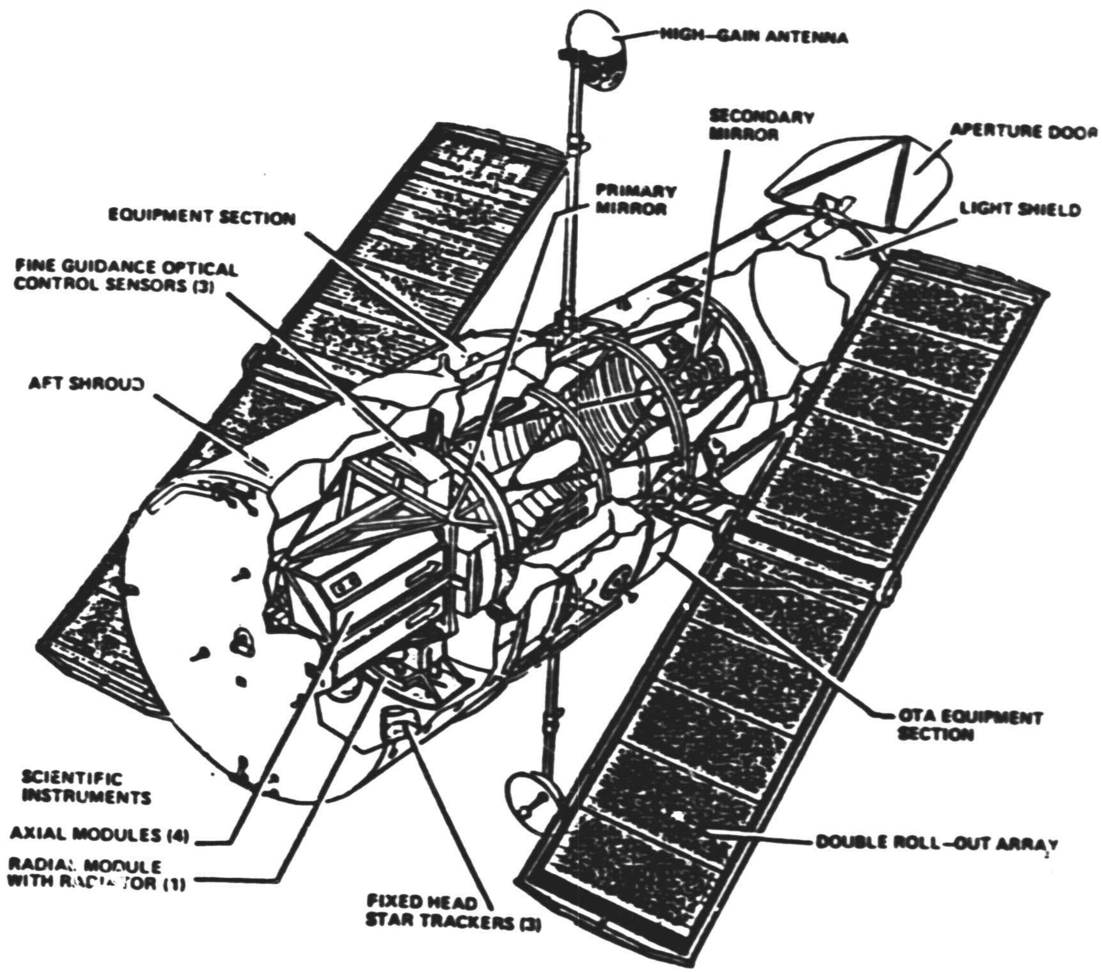


FIGURE 26 HST GENERAL CONFIGURATION

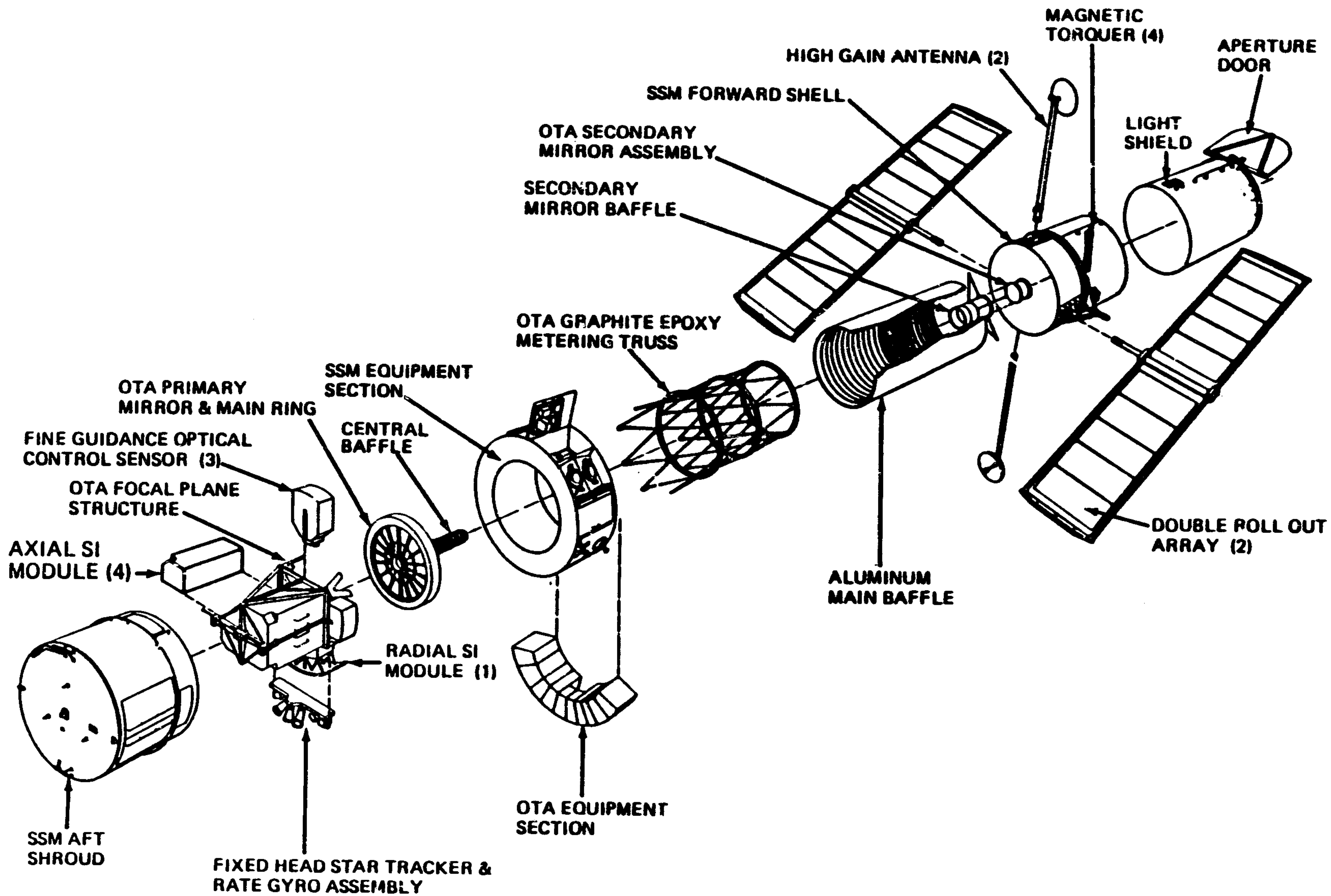


FIGURE 27 HST EXPLODED VIEW

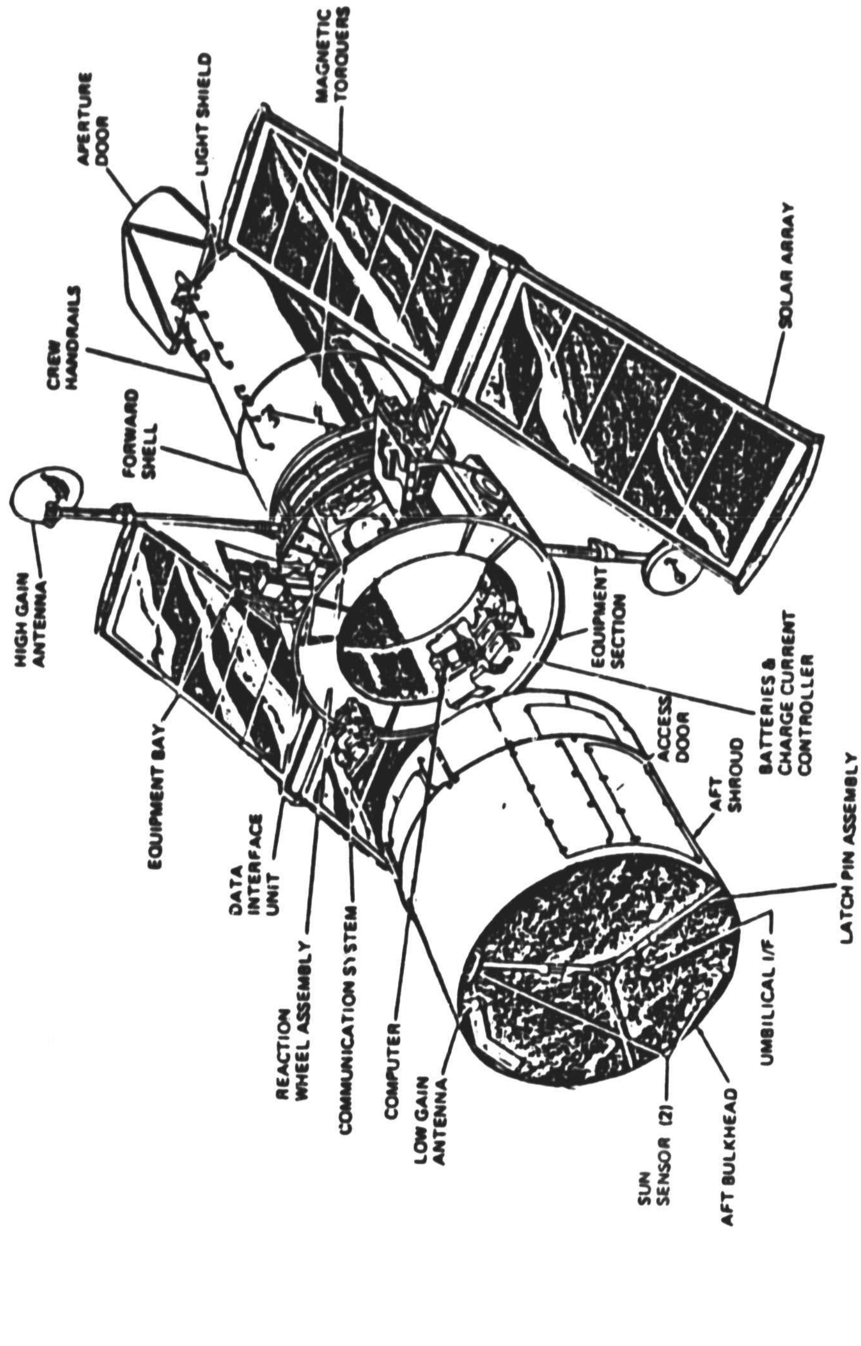


FIGURE 28 HST SUPPORT SYSTEM MODULE

The SSM shown in Figure 28 encloses the OTA and the SIs, supports the Solar Array (SA) and High Gain Antenna (HGA) and provides all the interfaces with the Shuttle Orbiter. The module contains a very precise pointing and stabilization control system, thermal control system, data management system. Instrumentation and Communication (I&C) system, and electrical power system. The electrical power to operate the telescope is provided by batteries which are charged by the two SA panels during the sun side of its earth orbit.

The PCS provides the ST with the capability of pointing at designated targets with sufficient accuracy and for the necessary period of time to achieve mission objectives. A PCS functional diagram is shown in Figure 29. The PCS operates in a digital mode and uses software modules resident in the flight computer and hardware items physically located throughout the ST.

The PCS equipment (Figure 30) can be divided into three categories: sensors, computer, and actuators.

There are five different types of sensors that record ST attitude and rate measurements. These sensors are: Fine Guidance Sensor (FGS), Fixed Head Star Tracker (FHST), Coarse Sun Sensor (CSS), Magnetic Sensing System (MSS), and the Rate Gyro Assembly (RGA).

The flight computer calculates the control law, attitude updates, momentum management law and the command generators. To limit structural mode acceleration, the command generators shape the acceleration and incremental angle commands to the control systems in an accelerate, coast and decelerate pattern.

Finally, actuators turn the control signal into torques that move the vehicle. There are two kinds of actuators: the Reaction Wheel Assembly (RWA), and the Magnetic Torquer (MTs).

Redundant hardware elements are provided to allow a timely resumption of science operations following a detected anomaly in system behavior.

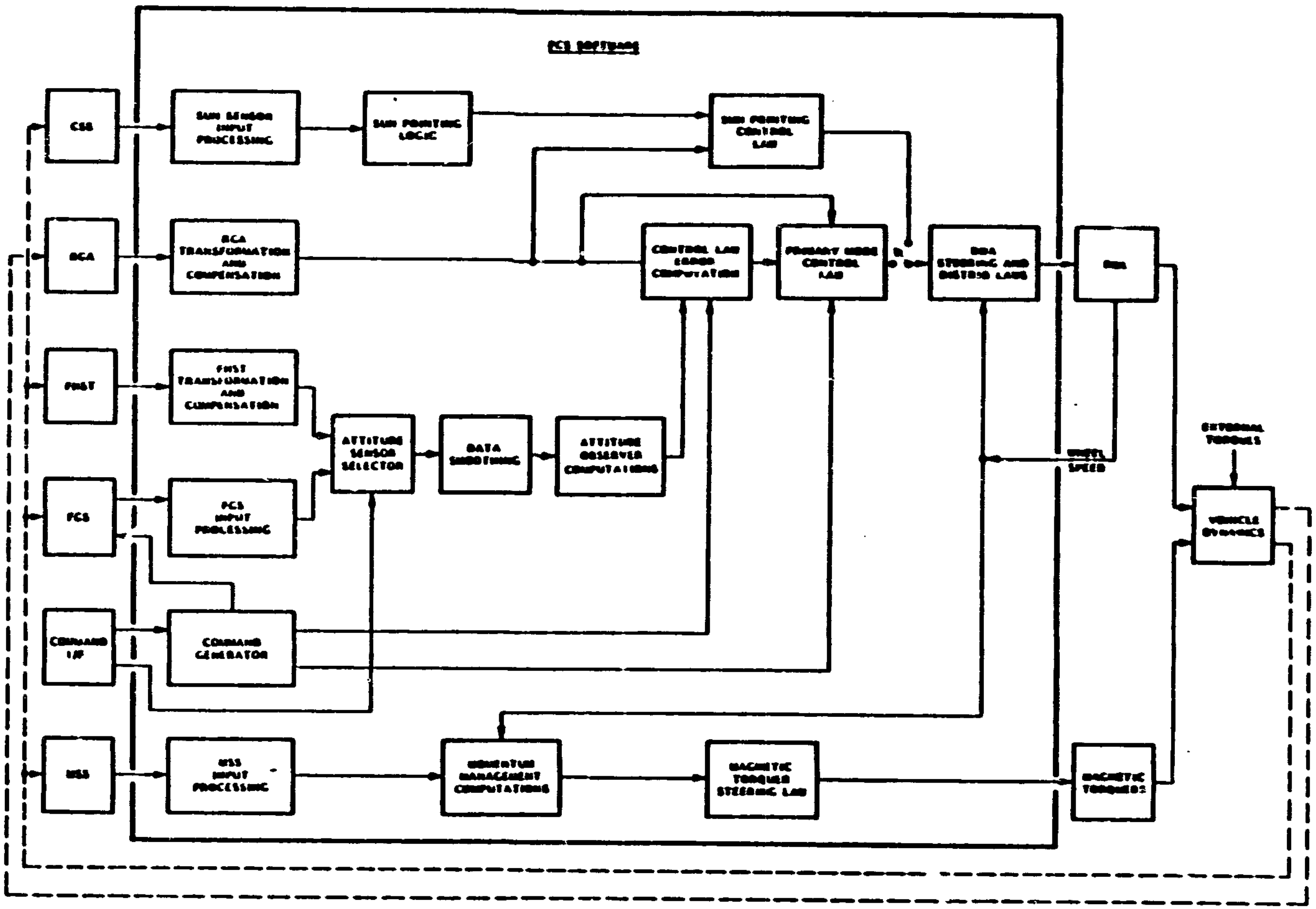


FIGURE 29 HST PCS FUNCTIONAL DIAGRAM

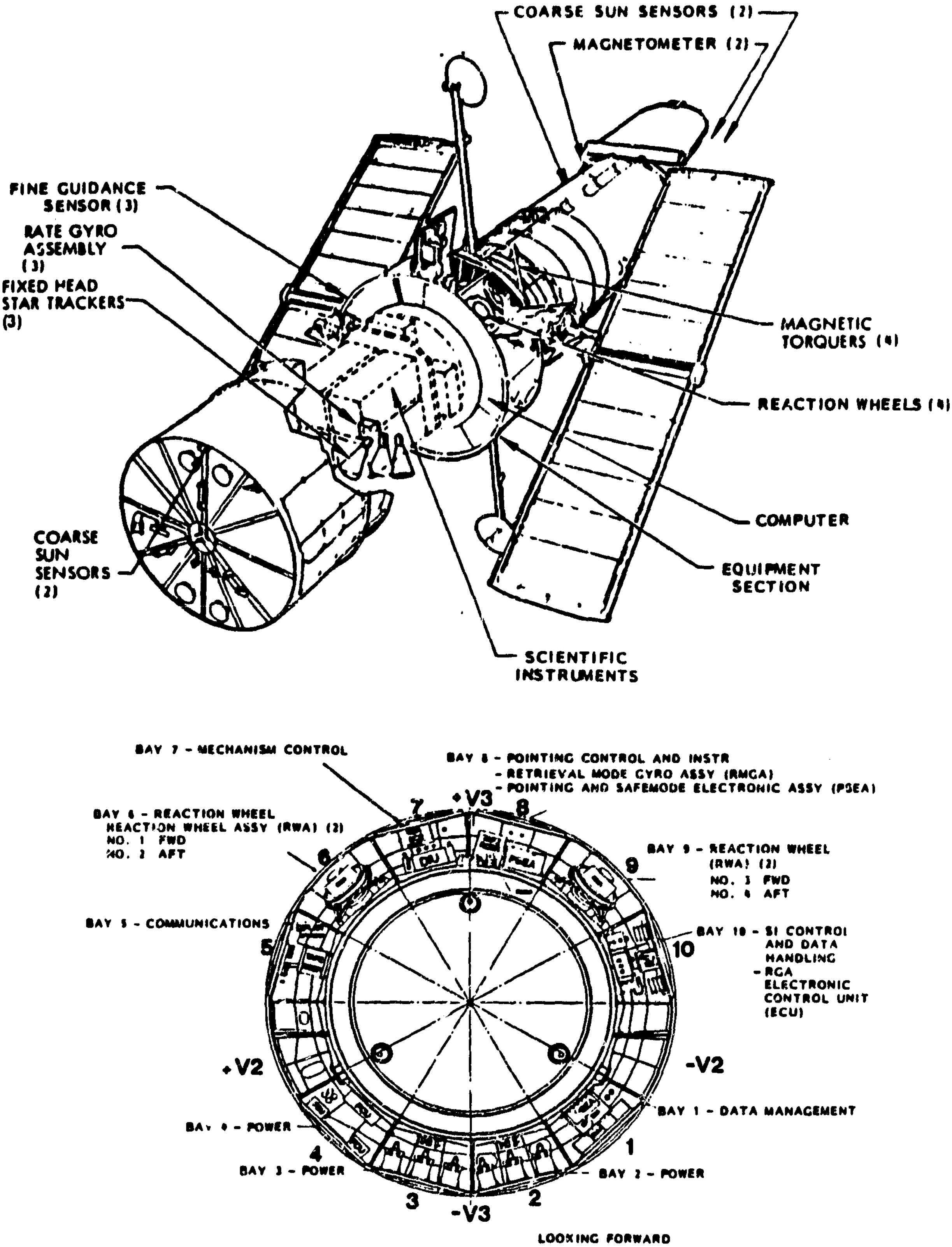


FIGURE 30 HST PCS EQUIPMENT

The components used in the design of the ST PCS represent the leading edge of todays technology. Particularly noteworthy are the Fine Guidance Sun Sensors (FGSs) and the Rate Gyro Assembly (RGA). The FGS measures the angular position of the stars and is the most accurate and the most complicated of the sensors. It uses the principle of interferometry to obtain an error signal that is accurate to fractions of an arc sec. Two of the three FGSs are used by the PCS to perform stellar pointing or low rate scanning. These sensors are used for attitude correction of the telescope Line of Sight (LOS) with rate gyros used for rate and short term attitude information. Control about the LOS is provided by rate integrating gyros with information from the two FGSs being used to provide attitude corrections.

The FGSs share the telescope Field-of-View (FOV) with the scientific instruments. Each FGS has a FOV of about 60 arc min squared. Four photomultiplier tubes are used for both acquisition and fine pointing. For fine pointing, the output of the tubes is differenced for use in an interferometeric mode. Because of the limited dynamic range of the interferometer (approximately ±.02 arc sec), the FGS uses the sum output of the photomultiplier tubes to achieve initial star acquisition after the PCS maneuvers the ST from one target star to another. The FGS contains star selectors, which are rate control servos with rotating prisms, to allow the 5 arc sec squared instantaneous FOV to be maneuvered over the total 60 arc min. squared FOV of the FGS.

The RGA is a strapdown, reference gyro package. It senses vehicle motion using two modified, ultra low noise Rate Integrating Gyros (RIG) to provide two-channel digital attitude and analog rate information. During normal vehicle operations, these rate and short term attitude updates are provided by the ST RGAs for fine pointing and spacecraft maneuvers. The RGA system is also used to produce inputs for Pointing Control System (PCS) operation during vehicle safemode.

A single RGA is composed of two separate subassemblies: the Rate Sensor Unit (RSU) which contains the RIG gyros, and the Electronics Control Unit (ECU). The RSU weighs 24 pounds, and the weight of the ECU is 17.5 pounds. Both units are designated as Orbit Replaceable Units (ORU).

The input axis of each single-degree-of-freedom gyro is skewed relative to the ST axes. The RGAs sense vehicle motion and output incremental attitude data to the Data Management System. The data is received by the Data Management Unit (DMU) at 830 micro sec intervals and accumulated into 25-msec samples to be used by the PCS for vehicle attitude control.

The rate gyros have a dual rate range. The high rate scale range is greater than 1800 arcsec/sec, the low rate scale is greater than 20 arcsec/sec. One of the gyro sensor units is redundant and kept in a dormant state unless needed.

4.0 SPACE STATION DESCRIPTION

The Space Station configuration used as a baseline for this study is the "Power Tower" configuration as described in the "Space Station Reference Configuration Description" document [1]. A summary of the IOC SS configuration described in [1] is presented in the following paragraphs. Some aspects of the SS not directly involved with instrument pointing are included. This is intended provide an understanding of the overall environment in which instrument pointing must be performed and to facilitate a subsequent discussion of instrument servicing (Section 6.0).

4.1 SPACE STATION CONFIGURATION SUMMARY

Figure 31 illustrates the Space Station Reference Configuration as described in [1]. It is composed of a set of deployed linear trusses to which pressurized modules, subsystems, and user equipment are attached. The principal structural components are a keel and three booms at right angles to the keel. These are refered to as the upper, transverse and lower booms. The coordinate system is Z parallel to the keel, positive toward nadir; X perpendicular to the keel and booms, positive in the direction of flight and Y parallel to the booms, positive toward starboard. The coordinate system origin (i.e., navigation base) is at the intersection of the transverse boom and the keel. The Attitude Control ASSEMBLY (ACA) houses the navigation base components and is a rigid nine-foot cube at the intersection of the keel and transverse boom. It contains the Control Moment Gyros, Star Tracker and other Guidance, Navigation and Control components needed for control of the station from the initial launch. The Space Station will be incrementally assembled in orbit using the STS Orbiter. It is anticipated that the permanently manned IOC Space Station will become operational in the early 1990's.

The IOC Space Station will have five pressurized common modules.. Two Habitability Modules (HM), two Laboratory Modules, and a Logistics Module. The keel is divided at the bottom to allow installation of the pressurized modules on the centerline of the keel. This maintains the principal axis in the orbit plane while the Orbiter is being berthed. Four of the pressurized modules are arranged in a quadrangle to allow crew movement into or out of all modules, even if one becomes unusable. The quadrangle is also referred to as the "racetrack configuration." The module arrangement is shown in

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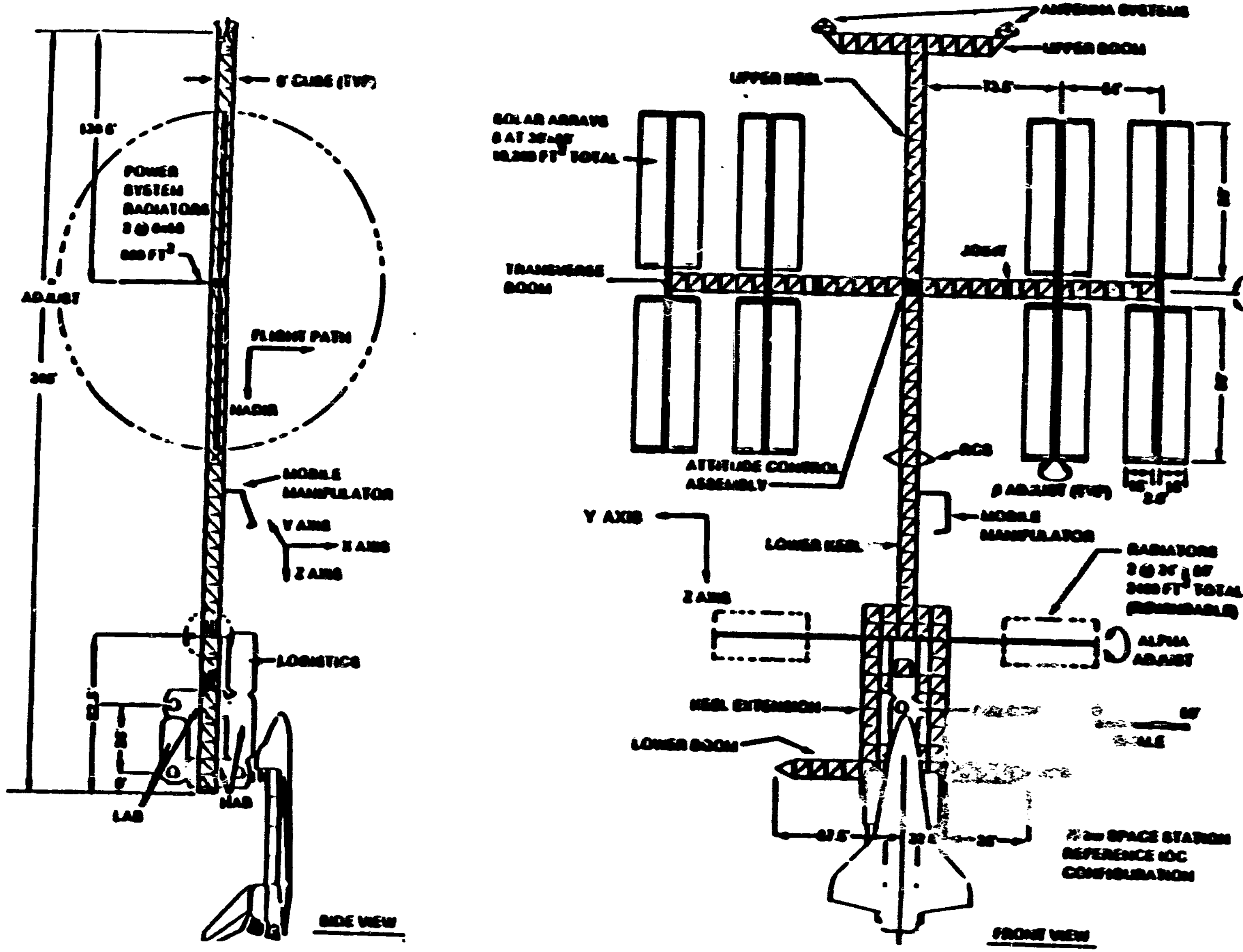

FIGURE 31 SPACE STATION REFERENCE CONFIGURATION

Figure 32. This module arrangement enhances safety in the event of emergency evacuation of any module. The laboratory modules are a Materials Processing Laboratory (LM 2) and a Life Sciences Lab (LM 1). Lab 2 is located at the top of the quadrangle to keep it as close to the center of mass of the station as possible. The HMs are adjacent to each other, to eliminate unnecessary movement by the crew through the Laboratory Modules. A control station for berthing is at the front of HM2 and allows a direct view of operations. The primary Orbiter structural interface is at the end of HM1. An alternate position at the end of Lab 1 is provided for emergency access in case the primary port is inoperable. The logistics module is berthed to HM 2 to permit unloading with minimum disturbances to laboratory operations.

Hatches are provided on all operational ports and crew airlocks. They are hinged so that crew and equipment can traverse through the open port as required. Each may be closed to serve as a pressure bulkhead between modules. Hatch weight has not been defined. The minimum hatch diameter is 50 inches. The logistics module has two end hatches. The other modules have four radial and two end hatches. External airlocks are attached to the port side hatches of HM1 and HM2. The electrical system, including power generation, storage and conditioning, is installed on the transverse boom outboard of the alpha gimbal joints. Each array wing is gimballed individually for beta adjustment to minimize principal axis shifts and simplify assembly and deployment. Primary heat rejection is provided by radiators mounted on the lower keel. The radiators are rotated to maintain an edge toward the sun. They are rewound during the dark portion of the orbit, to avoid a continuously rotating fluid joint. TDRSS and GPS antennas are mounted at the ends of the upper boom to allow a maximum upward view. Tracking and rendezvous radar antennas are mounted on the lower keel under the transverse boom for a clear view along the flight path.

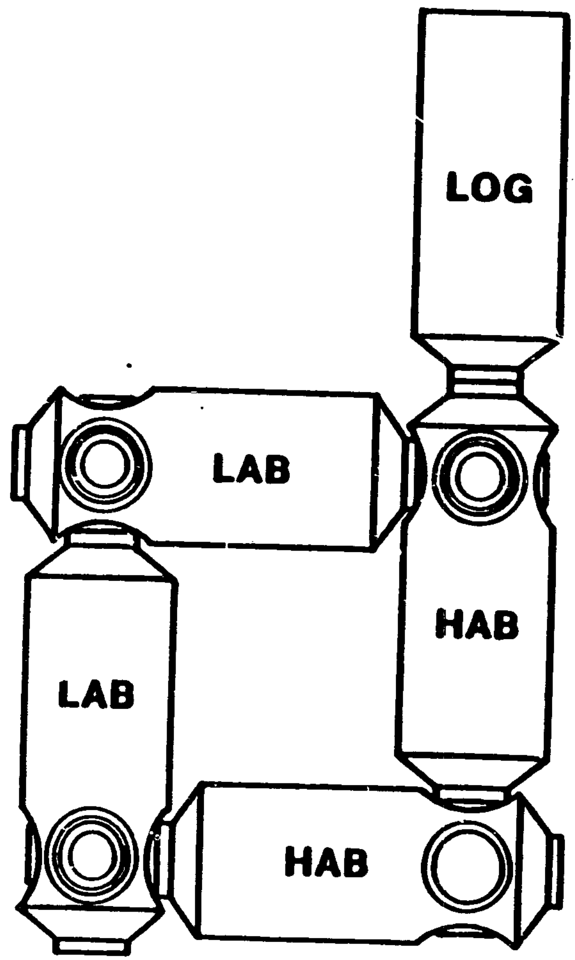

FIGURE 32 RACETRACK MODULE CONFIGURATION

Identical sets of propulsion thrusters are mounted at four locations at the ends of the lower boom and on the lower keel. These are capable of firing forward, aft, and outboard. The lower thrusters (on the lower boom) normally provide all backup attitude control. The upper thrusters normally fire only aft, for orbit maintenance. A Mobile Remote Manipulator System (MRMS) can move on the nodes of the truss to any location on the forward faces of the keel and booms. By moving outboard of the alpha joint and rotating the transverse boom, the MRMS can also move along the aft face of the structure.

4.2 PAYLOAD ACCOMMODATION

Two primary locations are provided for observational users. Solar and stellar instruments are mounted along the upper boom to maximize their view of that part of the sky not occulted by the Earth and Station. Solar viewing is possible on every orbit because the Sun is always within 52 deg. of the orbit plane.

Earth sensors are mounted on the lower boom and the local vertical orientation allows continuous viewing limb-to-limb with no image rotation. Some solar sensors may also be mounted on the transverse boom outboard of the alpha gimbal to simplify pointing if their mass characteristics do not adversely affect the Station's dynamic properties.

Satellite servicing provisions are located along the keel. Two storage/servicing bays are situated parallel to the upper keel. This location minimizes contamination of optical surfaces and sensitive components. Related ORU and tool storage is provided on the transverse boom. A refueling bay is located on the lower keel. Satellite propellants and cryogens are stored at the top of the keel extension near the refueling bay. The OMV and OMV kits are berthed alongside the port and starboard keel extensions.

4.3 SPACE STATION ORIENTATION

In normal operation, the Station is oriented with the keel approximately aligned with local vertical and the Y principal axis (approximately parallel to the transverse boom) held perpendicular to the orbit plane (POP) as shown in Figure 33. A constant pitch attitude is maintained with respect to the local vertical to balance gravity gradient and average aerodynamic torques. CMG's are used for momentum management to compensate for aerodynamic torque variations around the orbit. Long-term (more than a few orbits) variations in mass properties or atmospheric density are accommodated by small adjustments in the flight attitude of the Station.

The solar array is maintained normal to the Sun vector by a continuous rotation about the alpha gimbal at orbital rate and a slow adjustment about the beta gimbal to track the Sun's motion relative to the orbit plane. The radiators are also rotated at orbital rate, but are "rewound" during the dark part of the orbit to avoid continuously rotating fluid joints.

The Space Station orbit characteristics were defined in the Space Station Definition and Preliminary Design Request for Proposal [2]. These parameters are identified in Table 11 for the Space Station and its associated co-orbiting and polar platforms.

TABLE 11. SPACE STATION ORBIT PARAMETERS

ELEMENT	NOMINAL ELEMENT	INCL.	DESIGN ALTITUDE RANGE HIGH	LOW
SPACE STATION	500KM (270NM)	28.5°	555KM (300NM)	463KM (250NM)
CO-ORBITING PLATFORMS	500KM (270NM)	28.5°	555KM (300NM)	463KM (250NM)
POLAR PLATFORMS	705KM (381NM)	98.25°	900KM (486NM)	400KM (216NM)
POLAR PLATFORM SERVICING ALTITUDE IS 276KM (149NM)				

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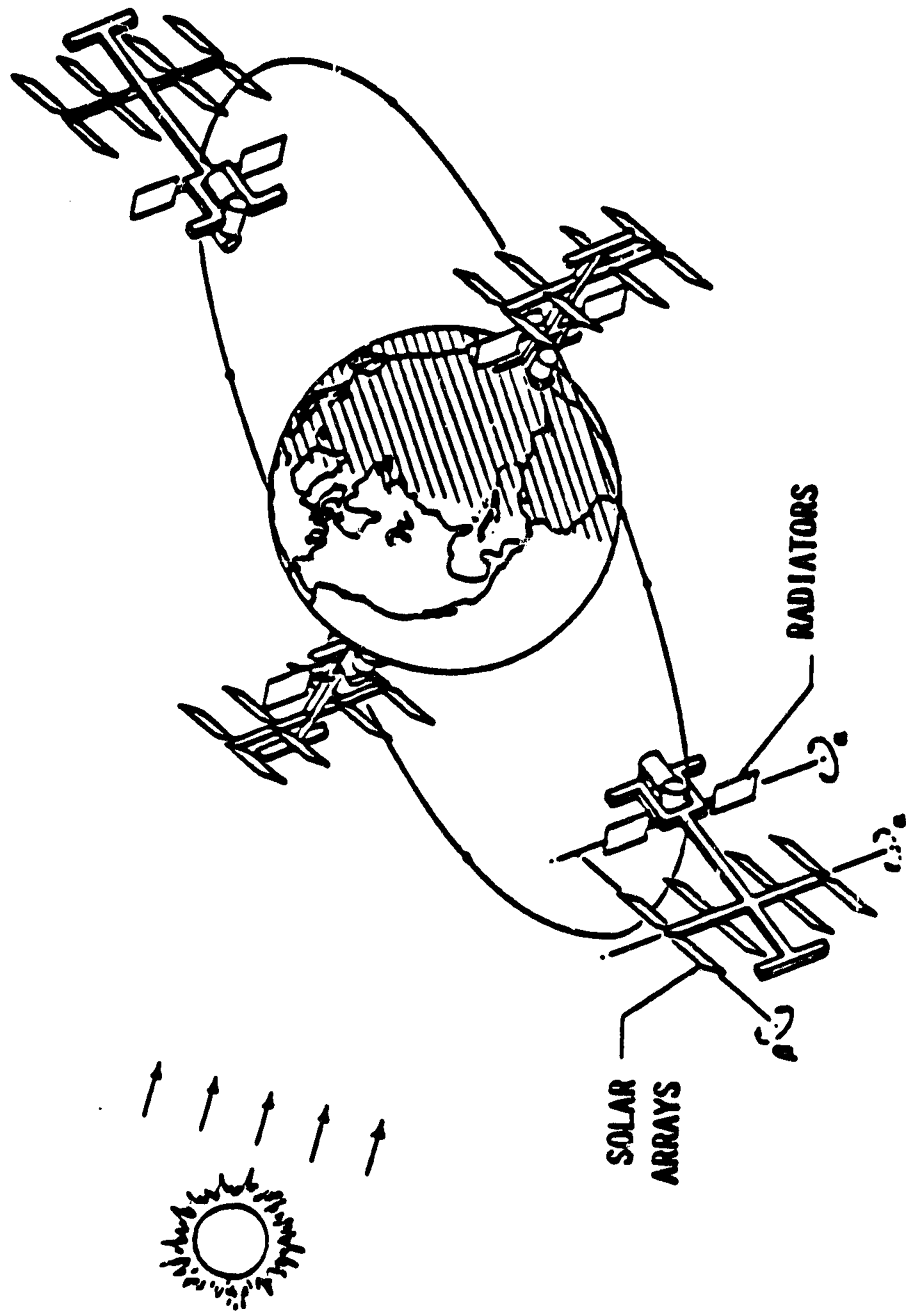


FIGURE 33 SPACE STATION ORIENTATION

Other orientations for the station are also under evaluation. Reference [42] describes an "airplane mode" or "power arrow mode" in which the longitudinal axis is aligned with the velocity vector as shown on figure 34. The preliminary analysis contained in [42] shows that the "airplane mode" offers advantages over the baseline orientation in terms of the momentum storage requirement, reboost control, and the level of "micro g" environment available for payloads. With respect to astrophysics instruments located on the upper boom it will have the effect of decreasing observation time and increasing instrument contamination.

4.4 ATTITUDE CONTROL SYSTEM DESCRIPTION

The Space Station ACS is an inherent part of the Guidance, Navigation and Control Subsystem (GN&C). The GN&C subsystem controls the attitude and orbital position of the Space Station, its payloads and appendices. In addition, it supports proximity operations and traffic control during all mission phases. The GN&C design is responsible for accomplishing:

A) Attitude estimation and control of the Space Station.
B) Momentum Management with aerodynamic variations.
C) Navigation base/payload relative alignment.
D) User interface with the Space Station crew.
E) Payload disturbance isolation.
F) Solar Array, antenna and radiator pointing.
G) Navigation and control during rendezvous and docking.
H) Stability and control during build up and growth.

"POWER TOWER" ORIENTATION

"AIRPLANE" ORIENTATION

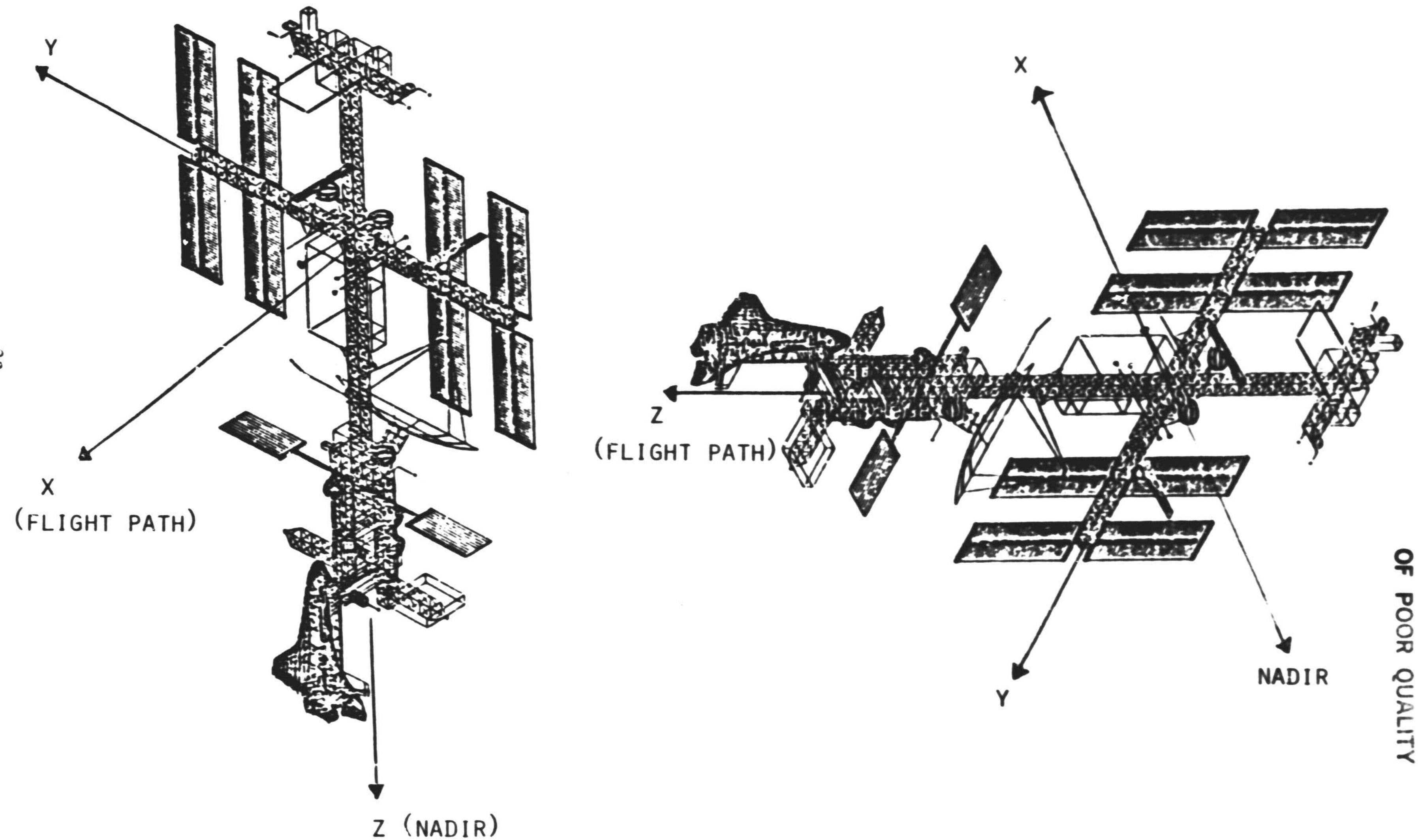


FIGURE 34 AIRPLANE MODE ORIENTATION

Specific performance requirements, such as; 0.01 deg. attitude determination accuracy at the navigation base, and measurement of the payload relative alignments to .002 deg. accuracy with respect to the navigation base, are major design drivers.

The baseline control system approach per reference [1] is a centralized system of co-located attitude sensors and CMG Control Actuators, as shown in figure 35, which provide inherent stability of the flexible body. The navigation system utilizes Inertial Sensor Assemblies (ISA) composed of rate gyros and accelerometers to provide attitude, attitude rate, position and velocity. Star Trackers will be used to update attitudes, and GPS satellites will be used to update position and velocity. The navigation processing will be performed in the Navigation and Traffic Control Computer.

The primary attitude control actuators will be CMGs, which may be supplemented by magnetic torques to perform continuous momentum management. The use of magnetic torquers will probably be eliminated [43,44] in favor of a torque equilibrium attitude (TEA) which uses the gravity gradient torque to minimize CMG momentum buildup. The overall momentum management strategy will be to reduce the roll/yaw disturbances to near zero by placing the vehicle principle axis (Z) in the orbit plane. A Z-axis offset (pitch) within the orbit plane will nominally balance aerodynamic bias torques with the gravity gradient torque.

The RCS will be used to perform all translational maneuvers of the Space Station and to backup the CMG system when the momentum or torque exceeds the capability of the CMG system. The RCS will also be used for primary attitude control (with reduced pointing) in the event that the CMG system becomes inoperative.

The RCS and CMG systems will have the capability to operate simultaneously or individually. An attitude transfer system will be used to determine the relative alignment between the navigation base and the mounting surface of the pointing instruments.

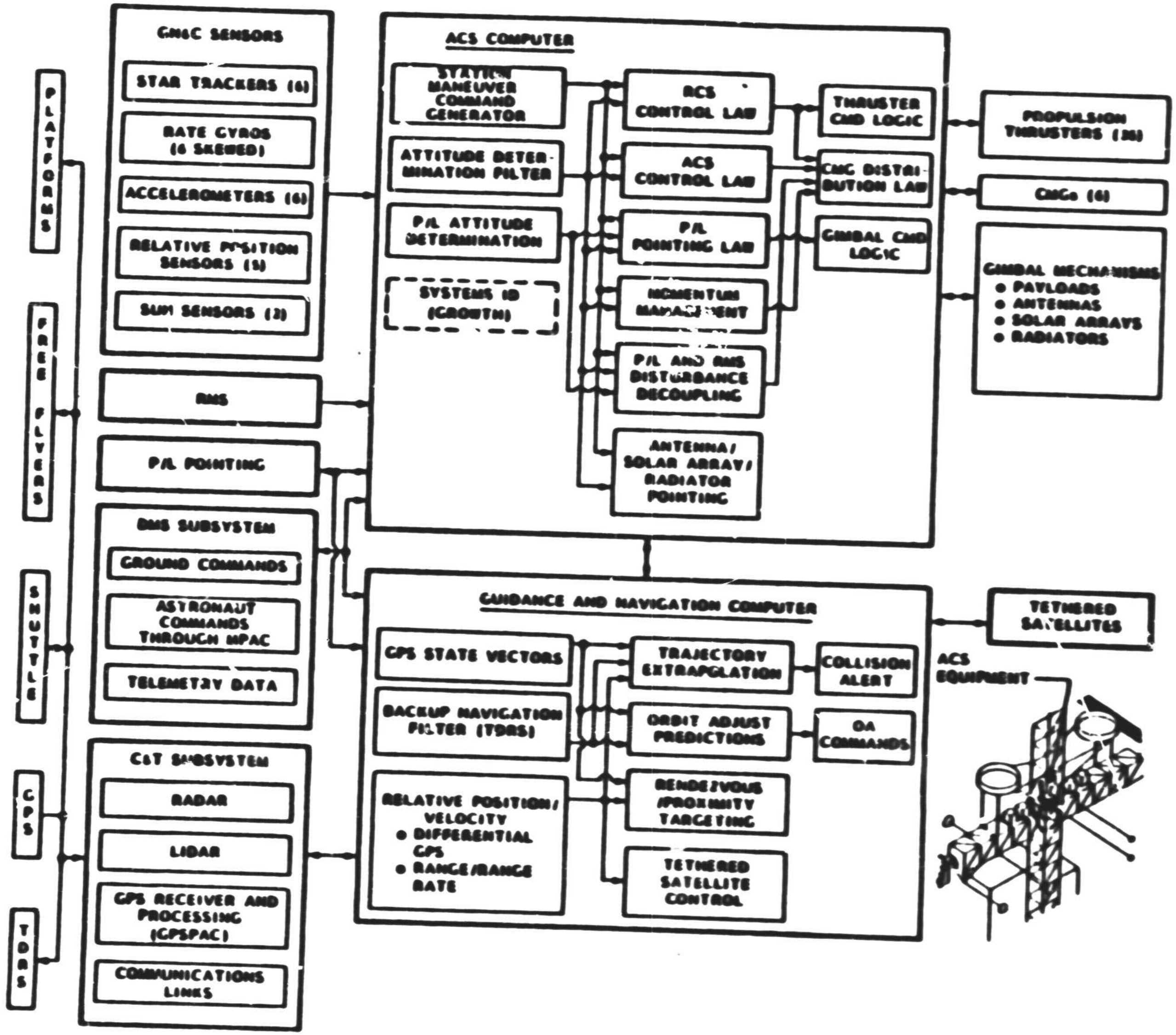


FIGURE 35 GN&C SUBSYSTEM BLOCK DIAGRAM

The GN&C equipment will be located as follows:

The CMGs and the Inertial Sensor Assemblies will be located in the Attitude Control Assembly, which in turn is located at the intersection of the keel and the transverse boom. The Star Tracker triad location has not been finalized, but it will be located either at the upper boom or in the ACA.

If used, the magnetic torquers will likely be on the main keel and/or on one of the booms. The solar array and thermal radiator drive unit electronics will most likely be placed at the location of the drivers themselves.

The GN&C processors and the interface devices will be located in the ACA.

The navigation/traffic control processors and the other miscellaneous GN&C electronics will be located in HM2. Each interface device (ID) that supports remote sensors will be located with the sensor itself.

4.5 SPACE STATION DYNAMICS

The Space Station as described in reference [1] is a quite flexible structure having frequencies in the range of .0965 Hz to 0.948 Hz. The frequency of the fundamental vibration mode of the IOC SS is approximately 0.1 Hz. A review of the deflection characteristics at the intersection of the keel and the upper boom reveals that a pointing system with three degrees of freedom (two translational and one rotational) will be required to decouple the payload from disturbances, propagating through the flexible body. Response of the upper boom in terms of peak accelerations and settling times due to Shuttle berthing, RCS Reboost firings, MRMS operations and crew motions are also given for various operational configurations. Peak acceleration at the end of the upper boom when the MRMS is located there, is found to be 0.0218 g (without payloads attached). This implies a significant control problem for the station during MRMS operations on flexible portions of the SS. With payloads attached to the upper boom, the peak acceleration is reduced to 0.0003 g's.

Peak acceleration levels at the upper boom root with payloads attached, due to orbiter berthing and RCS reboost firing, are 0.0028 g's and 0.0062 g's respectively. It responds to a crew motion (translational kick-off) with a peak acceleration of 1.42 x 10-4 g's. A pointing mount required to provide stability in the arc sec or sub arc sec range will need to be isolated from the effects of the various disturbances on Space Station.

4.6 DISTURBANCE SOURCES

The Space Station environment from an instrument pointing system point of view will encompass both high frequency and low frequency disturbances. Disturbances arise both from internal and external sources as shown in Table 12. The effect of these disturbances on the pointing system will, in large part, be determined by the structural dynamics (flexibility) of the base body. The latest technical information with respect to Space Station disturbances is contained in Reference [1,45,46].

4.6.1 ENVIRONMENTAL DISTURBANCE SOURCES

The major environmental torques that will effect the Space Station attitude, as listed in Table 12 are:

A) Aerodynamics torque caused by the motion of the Space Station through the upper atmosphere.

B) Gravity gradient torques caused by the small difference in gravitational attraction from one end of the Space Station to the other.

C) Geomagnetic torques, due to the interaction between the Space Station magnetic field and the earth's magnetic field.

D) Solar radiation torques, due to both the electromagnetic radiation and particles, radiating outward from the sun.

The relative strength of the various torques will depend on both the Space Station orientation and the form and structure of the space station itself.

TABLE 12. DISTURBANCE SOURCES

INTERNAL	EXTERNAL
• RANDOM DISTURBANCES	• ENVIRONMENTAL
- MAN MOTION	- GRAVITY GRADIENT
- OUTGASSING/VENTING	- AERODYNAMIC FORCES AND TORQUES
- FLUID SLOSH	- GEOMAGNETIC FORCES AND TORQUES
	- SOLAR RADIATION PRESSURE
	- SOLAR HEATING
• ARTICULATED ELEMENTS	• DOCKING/BERTHING
- MOBILE REMOTE MANIPULATOR SYSTEM	• POST DOCKING OPERATIONS
- SOLAR PANELS, RADIATORS	
- ANTENNAS	
- PAYLOADS	
• ORBIT MAINTENANCE	
• ATTITUDE CONTROL	

The largest environmental disturbances with the Space Station in its baseline orientation are caused by gravity gradient and aerodynamic torques. Figure 36 shows an orbital torque profile, due to the combined effects of aerodynamic and gravity gradient torques, which will act on the Space Station reference configuration. The torque profile assumes a principle axis of inertia offset from the local vertical, causing the yaw and roll cyclic gravity gradient and pitch bias gravity gradient torque. The large pitch double bulge is caused by aerodynamic torques on the solar arrays. These torques will be overcome by careful vehicle re-orientation, using the attitude control system CMG's to generate the momentum profile, which is also shown in the figure. Depending on the control philosophy implemented, the momentum shown can be absorbed entirely by the CMG's resulting in no disturbance at the upper boom (experiment platform).

Figure 37 illustrates an example of a several degree sway about the Y-axis, which can occur under specific aerodynamic density conditions. The amount of sway varies and is dependent on the number of CMG's (momentum capacity) resident on the Space Station. The figure shows that a complement of CMG's capable of 19,000 nms of momentum storage will control the station with zero sway under all anticipated conditions, while a lesser momentum capacity will result in an associated vehicle sway, as shown. For example, a 12,000 nms CMG momentum capacity will result in a 1.5 degrees, once per orbit sway, if the diurnal bulge (variation of the aerodensity over the sunlit and dark portions of the orbit) is 50% considering a .67 x 10 kg/m3 aerodynamic density. This is a low frequency disturbance to the astrophysical experiments. In addition, the Space Station is already rotating at 4 degrees/minute to maintain the local vertical orientation.

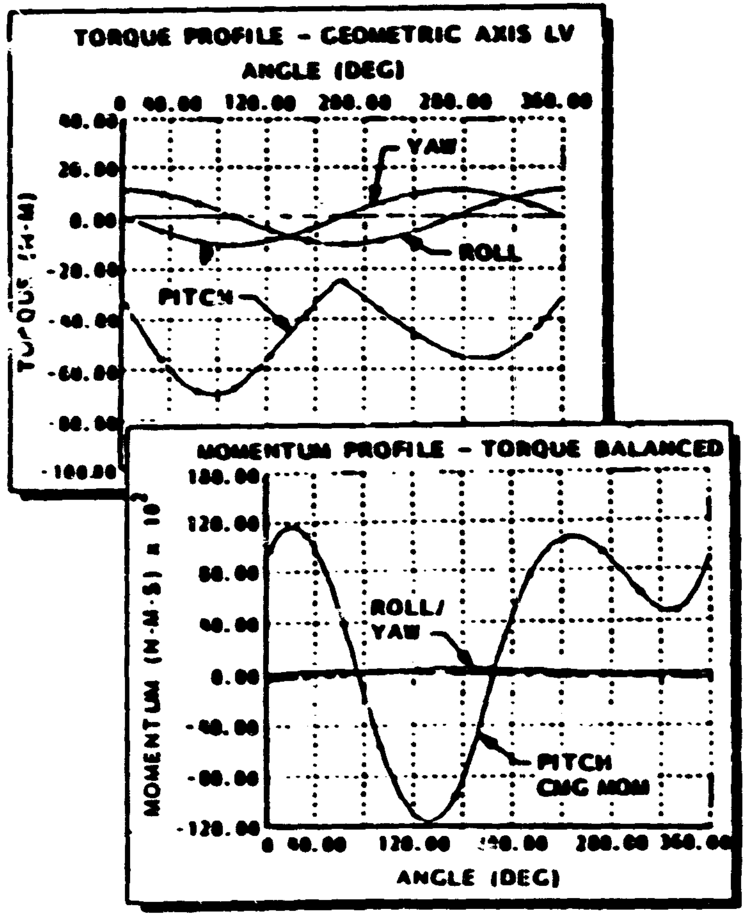


FIGURE 36 NATURAL ENVIRONMENT DISTURB NCES TORQUE PROFILE

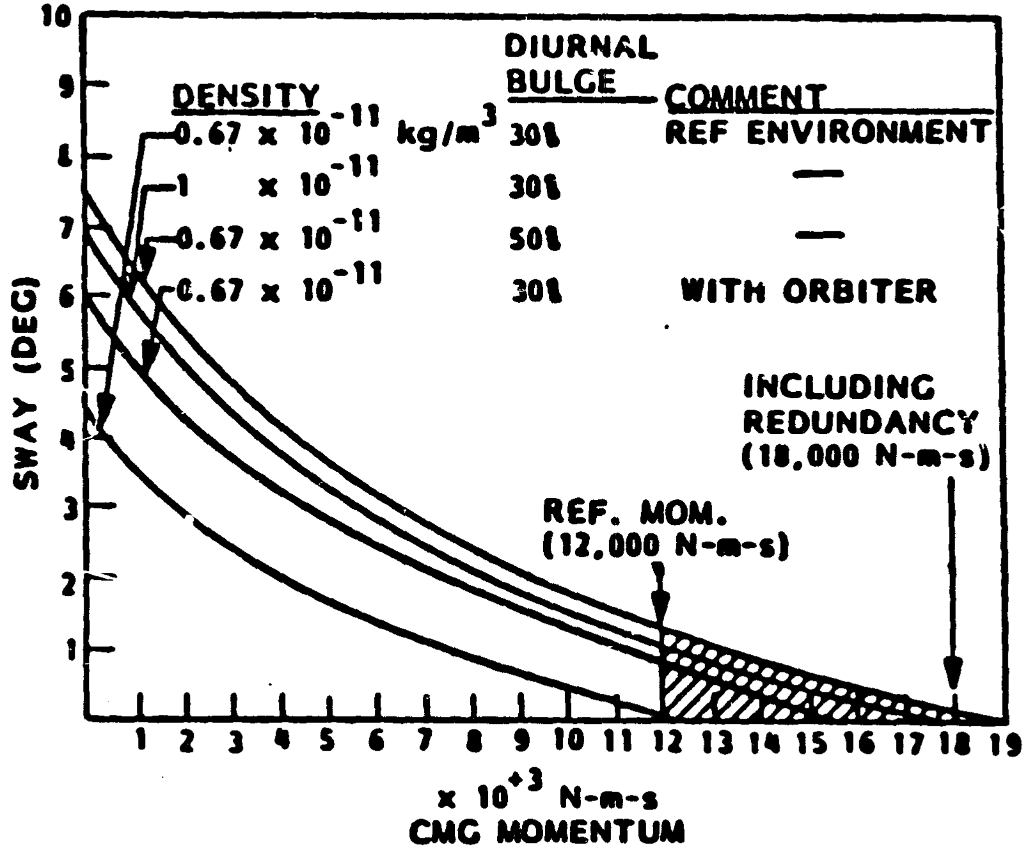


FIGURE 37 SWAY ABOUT THE Y-AXIS

4.6.2 DISTURBANCES DUE TO OPERATION AND INTERNAL SOURCES

In addition to the environmental disturbance sources, disturbance from sources internal to Space Station or arising from Space Station Operations, will also effect the pointing system performance. These include:

A) Shuttle Docking/ Berthing Operations

B) Post Docking Operations

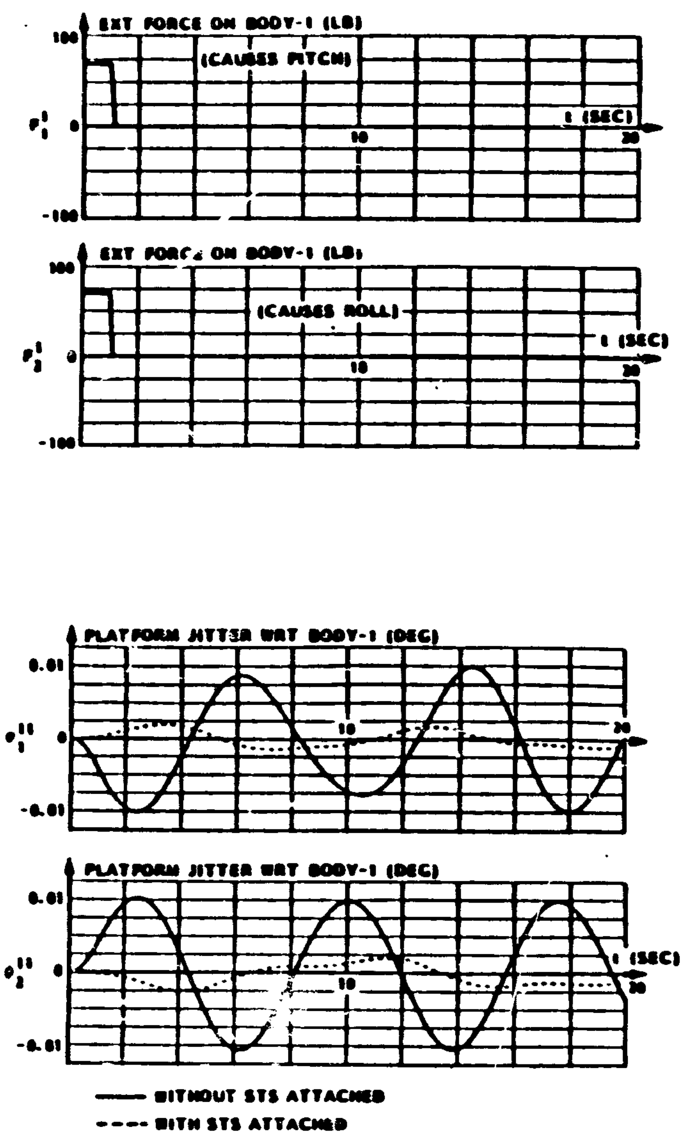
C) Motion of articulated elements, such as the Mobile Remote Manipulator System (MRMS), motion of the solar arrays, radiators, antennas, and other payloads.

D) Random events, such as astronaut motion, outgassing/venting, fluid slosh, vibrations from pumps, CMGs, etc.

E) Space Station Orbit Maintenance and Attitude Control.

4.6.2.1 EFFECT OF ASTRONAUT DISTURBANCE AND ORBITER ATTACHMENT

The response of the Space Station to disturbances will differ as a function of its mass, stiffness and inertia. Figure 38 illustrates the effect of a typical astronaut disturbances of 70 lb. of force applied at the Space Station Habitability Module 1. Two forces are shown. ${F^1}_1$ is a 70 lb force applied in the flight direction, causing the Space Station to pitch. ${F^1}_2$ is a force in the - y direction and causes the Space Station to roll. The effect of these forces at the upper boom (experiment platform) is shown for two cases, with and without the orbiter attached. The effect of orbiter attachment reduces the motion of the upper boom from ±.01 degree to ±.0025 degree as shown. This analysis was performed by the Lockheed Missile and Space Company, during the Space Station Phase B Proposal, using their Terflex 20-body dynamic simulation.



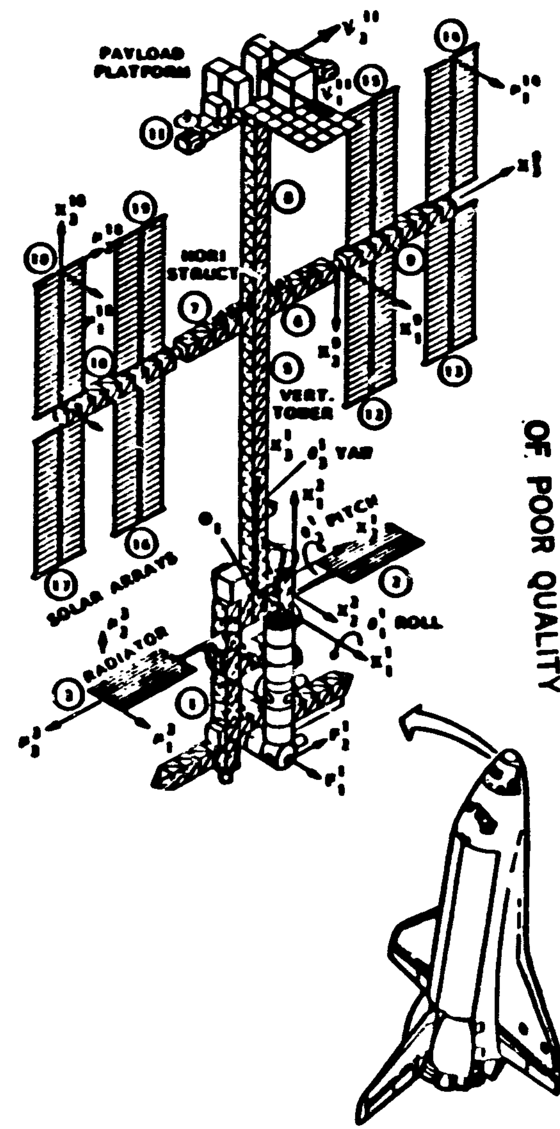
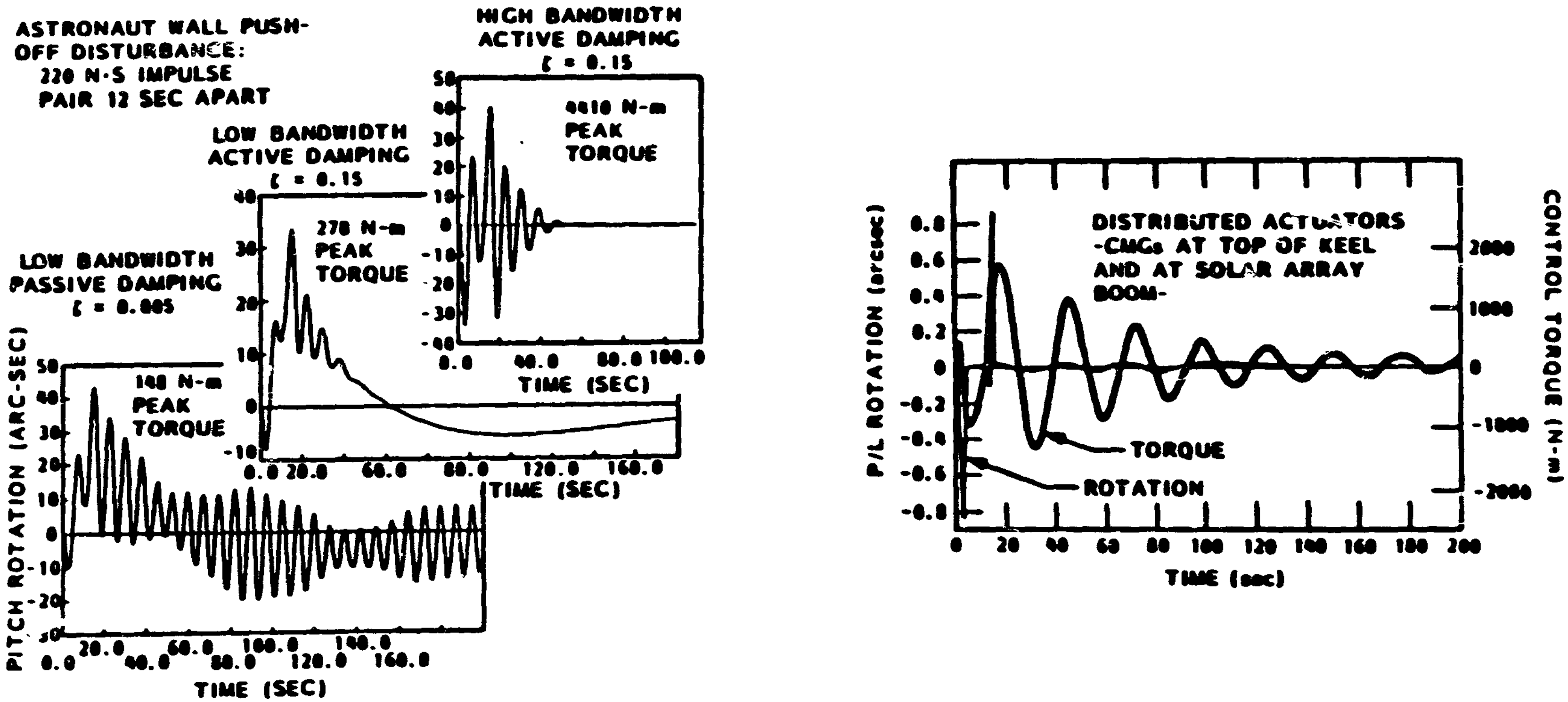


FIGURE 38 EFFECT OF ASTRONAUT DISTURBANCES AND ORBITER ATTACHMENT

Early studies for Space Station control indicate that, with a centralized control system, large upper boom base motion results from astronaut disturbance. This is true, as shown in Figure 39 for both low bandwidth and high bandwidth systems, even when active control is implemented. Figure 39 shows that, when distributed actuators are employed, motion at the payload is reduced. In this case, however, the control torque is large and requires single gimbal CMGs. A study is required to trade the complexity of the Space Station Control System versus the complexity of the astrophysical platform configuration as a function of control.



CENTRALIZED

DISTRIBUTED

FIGURE 39 EFFECT OF CENTRALIZED VS DISTRIBUTED CONTROL ON PAYLOAD POINTING

5.0 POINTING MOUNTS FOR SPACE STATION

The Space Station is required to simultaneously accommodate instruments that have Earth, anti-Earth, Solar and Stellar viewing requirements. Satisfaction of this requirement from the Space Station flying in the baseline attitude orientation (LVLH) will require a gimbal system. In the baseline orientation, the Z-axis is aligned to the local vertical and the station rotates about the Y-axis, which is perpendicular to the orbit plane, once per orbit. The Space Station phase B RFP [2] recognized the requirement. It provides for up to 9 attachment points for externally mounted payloads. Some of these nine (9) attachment points will be articulated attachments which would provide structural support, positional control, electrical, thermal and data interfaces for the attached payloads. These coarse pointing/tracking gimbals would be designed such that attitude control would be between ±1 to ±5 degrees and attitude determination to ±.01 degrees. Maximum angular rates would not exceed ±.02 degrees per second. In addition, the Space Station would provide knowledge of the instrument mounting surface relative to the navigation base within ±0.002 degrees. Users or customers with instruments that require higher pointing accuracy will be required to provide their own pointing mounts.

Conceptually, as shown in Figure 40, the payload attach points could incorporate Alpha and Beta gimbals at the upper boom similar to those used to point the solar arrays and radiators to provide the general pointing guaranteed by SS. Pointing systems, such as the IPS, would then fine point the instruments, provided that the disturbance environment is within its operational capabilities.

An alternate concept, shown in Figure 41, is to provide an attach point with a Beta gimbal on the transverse boom outboard of the Alpha gimbal. A pointing mount could then be attached at the Beta gimbal for fine pointing.

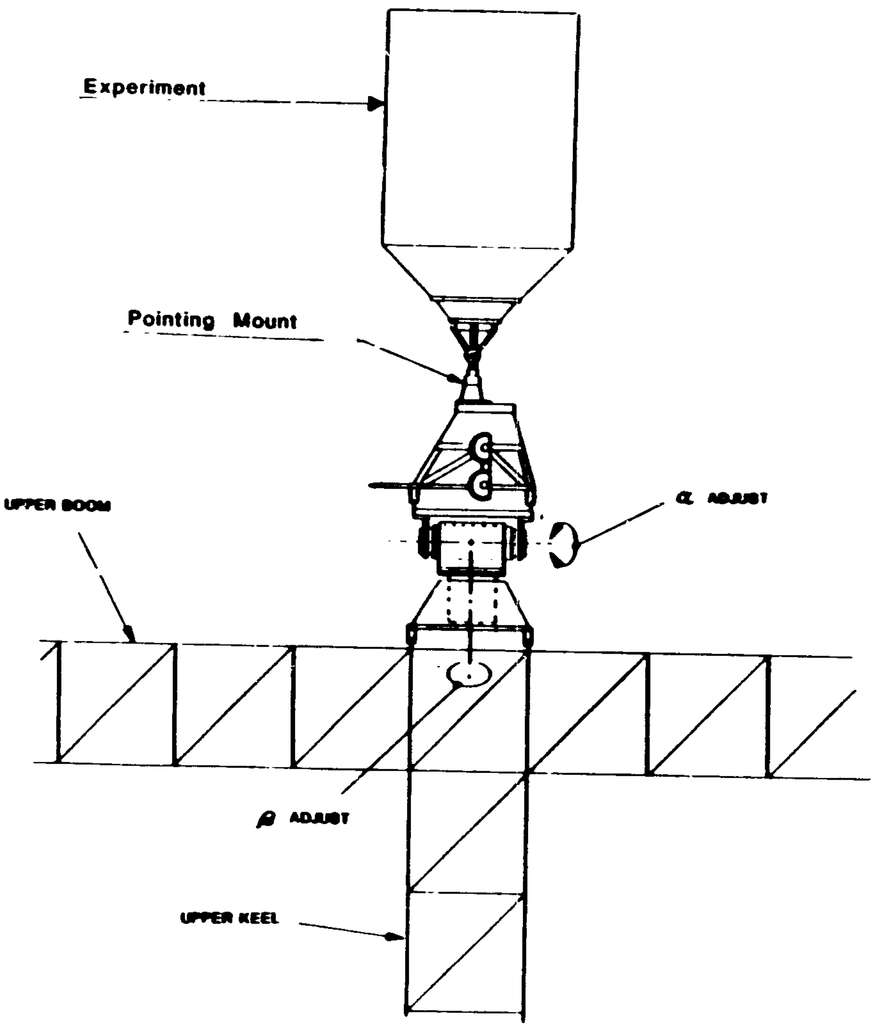


FIGURE 40 ALPHA/ BETA JOINTS FOR COARSE POINTING

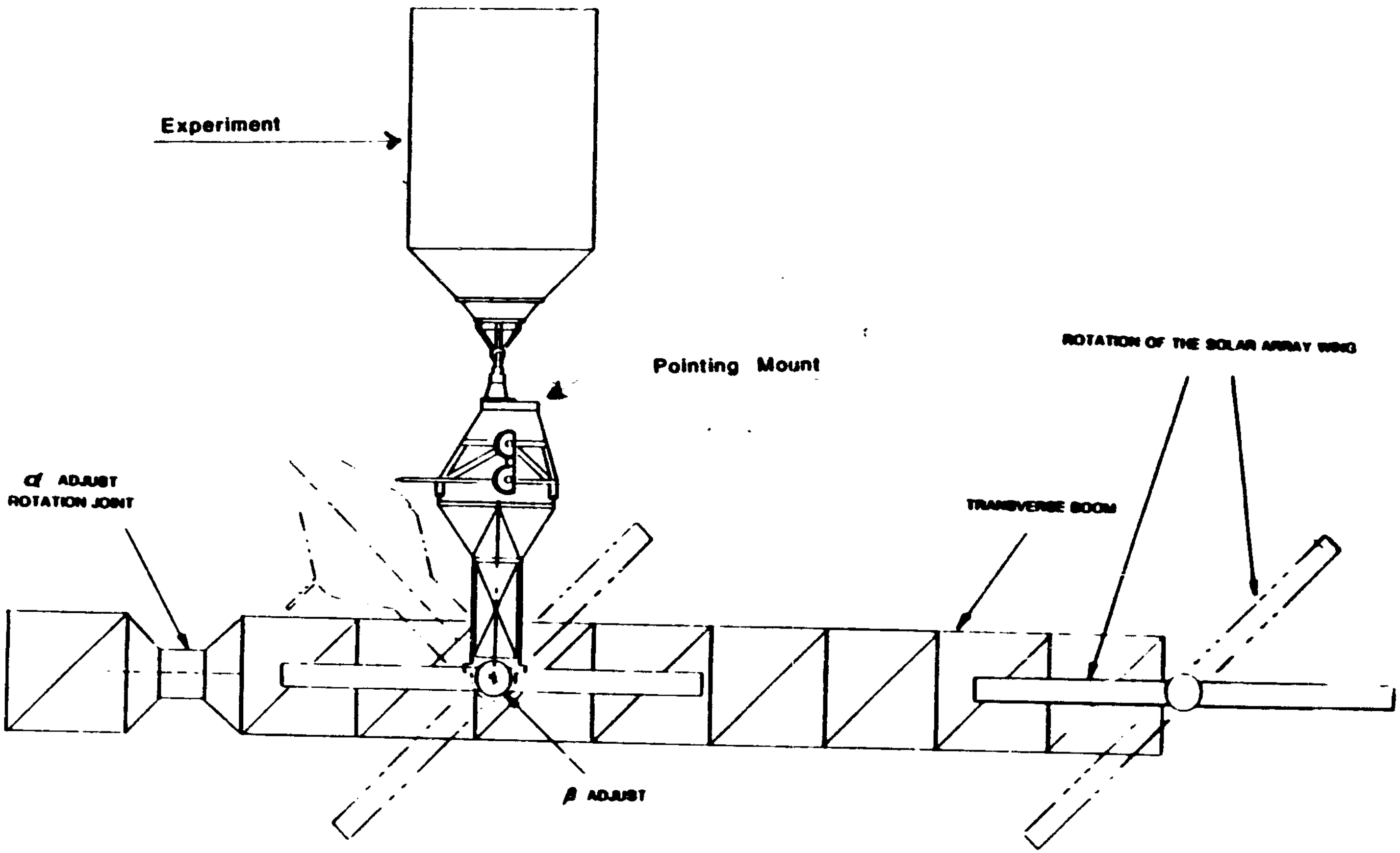


FIGURE 41 BETA JOINT FOR CCARSE POINTING ON TRANSVERSE BOOM

The RFP [2] identified several levels of pointing accuracy required by the various instruments identified in the SSP Mission Data Base [3] and the percentage of customers satisfied with each level of pointing accuracy. These data are listed in Table 13.

TABLE 13. POINTING CAPABILITY VS. CUSTOMER SATISFACTION

LEVELS OF POINTING ACCURACY	ATTITUDE ERROR (PLACEMENT) (ARC SECONDS)	STABILITY (ARC SEC)	JITTER (ARC SEC)	SATISFIED CUSTOMERS (PERCENT)
1	100	20	2	40
2	30	6	0.6	55
3	10	2	0.2	60
4	3	0.6	0.06	85
5	1.0	0.02	0.02	95
6	0.15	0.002	0.002	100

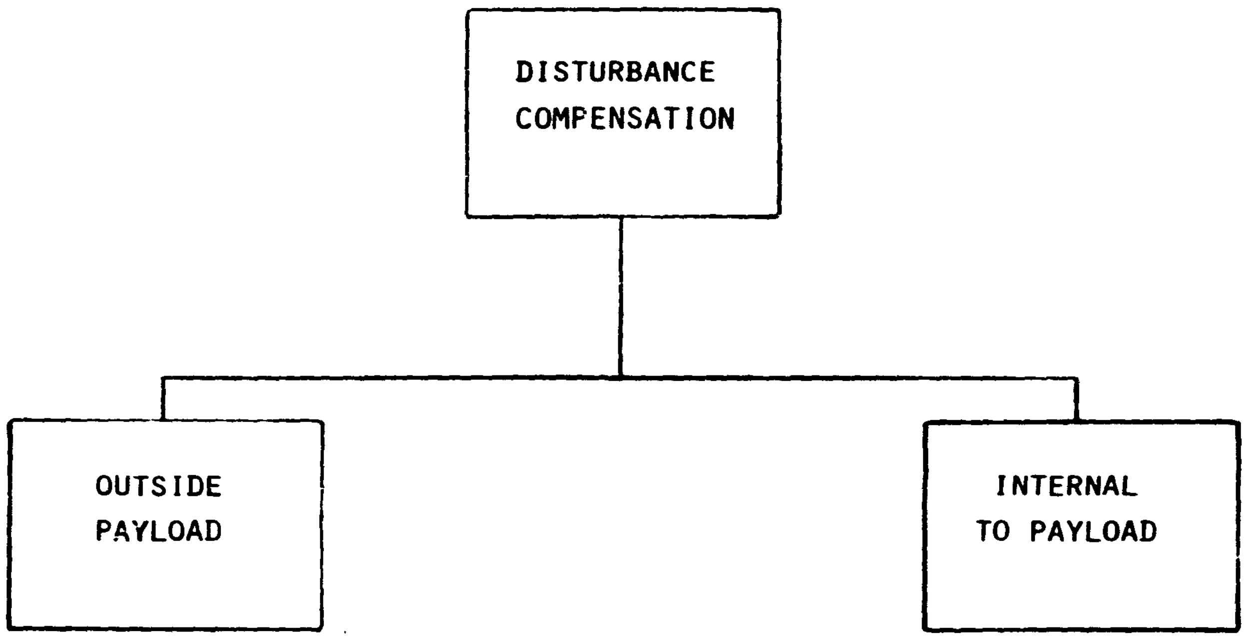
The coarse pointing/tracking gimbal to be provided by SS would satisfy less than 40 per cent of the customers. The survey of existing pointing mount technology indicates that 100% of the pointing error requirement and up to 35% of the stability requirement could be met by existing technology if the environment was comparable to the Shuttle environment for which these systems were designed. The ASPS could satisfy up to 100% of the attitude error (accuracy) requirement and 85% of the stability requirement for payloads less than 600 Kg. The AGS could satisfy 100% of the pointing accuracy requirement and up to 60% of the stability requirement for payloads up to 7200 Kg. The IPS could satisfy up to 90% of the pointing accuracy requirement and up to 80% of the stability requirement. The ST capabilities illustrate that over 95% of the attitude error and stability requirements can be achieved by a free flyer. As shown previously, a generic pointing mount capable of satisfying the ASO requirements would have to accommodate a 7000 Kg payload, provide a pointing accuracy of 1.0 arc sec and a stability of 0.1 arc sec. Such a system would satisfy up to 90% of the customers. The remaining customers would then have to include additional pointing capability within their instruments for higher pointing accuracy or stability.

The coarse pointing/tracking provided by the SS per the RFP [2] would require that over 60% of the customers provide their own pointing mount. This may not be cost effective, since existing technology indicates that it is possible to develop a generic pointing mount that will achieve a much higher per cent of customer satisfaction.

Current or near term pointing systems like the IPS were developed for shuttle and their performance is predicated on operation in the Shuttle environment. The Space Station environment is expected to be significantly more noisy and complex than the shuttle environment. It is not likely that current systems will maintain their performance if operated without design modifications on Space Station. Development of a pointing mount insensitive to or isolated from the effects of disturbances is the key to providing a generic pointing mount for Space Station.

Figure 42 identifies approaches which have been taken previously to disturbance isolation or compensation. Disturbance isolation or compensation can be incorporated either on the pointing mount outside the payload or by the payload using image motion compensation. Image motion compensation, whether by image data processing, or by movable optical elements is usually tailored to a specific instrument and what works with one instrument may not work with another. This limitation may be overcome. MSFC has developed an IMC system to provide more stringent pointing stability for two of the Astro-1 telescopes than could be provided by the IPS [47]. The MSFC IMC system consists of an Astro Star Tracker, IMC Electronics, and the DRIRU II. These components replace the IPS computers and sensors.

The IMCS, in conjunction with the telescope secondary mirror actuators, provides the ability to improve pointing stability by sensing the IPS motion due to external disturbances and issuing commands to the secondary mirror actuators. The DRIRU senses the high frequency (up to 15 hz) errors and the AST is used by the IMCS software to correct for drift of the DRIRU II. The IMCE provides for a dedicated HRM channel interface to down link IMCS house keeping and pointing performance data. The hardware complement can be considered portable, since it can be easily adapted to other applications.



- FEED FORWARD CONTROL
- MAGNETIC SUSPENSION
- FLEXURES
- SOFT MOUNTS
- FLOATED PALLET

- IMAGE DATA PROCESSING
- IMAGE MOTION COMPENSATION

FIGURE 42 - DISTURBANCE COMPENSATION METHODS

A generic pointing mount, which provides a high degree of pointing stability, will reduce the requirement for payloads to employ these or similar techniques.

Consider providing disturbance compensation outside the payload on or near the pointing mount. Reference [48] developed fundamental stability and disturbance rejection characteristics for simple and two stage gimbal systems such as the IPS, ASPS and Gimbalflex, as well as a soft mount inertially reacting concept.

The simple gimbal system considered in [48] is shown in figure 43. The two stage system considered was a model of the ASPS which has a magnetically isolated vernier stage on top of a gimbal system. The soft mount inertially reacting concept is described as a payload mounting body, equipped with both actuators and sensors connected by a soft interface to the base body. The actuators could be reaction wheels or CMGs and the sensors would be star trackers and gyros. An inertially referenced proportional plus derivative control law was assumed. Conclusions derived in [48] are directly applicable to selection of a pointing concept for SS and are reiterated here.
Simple gimbal systems will always have the problem of dynamic interaction with the base body. There is a frequency range determined by the base body natural frequencies in which the pointing system control frequency must not lie in order to maintain system stability. This is an unacceptable restriction for SS since there are many flexible modes below 1 Hz.

Simple gimbal systems with end mounted payloads also will be affected by base body translational motions at frequencies above the control bandwidth. If the pointing system is offset from the base body center of mass it will also be affected by high frequency rotational motions.

Systems which employ an isolation or soft suspension stage on top of the gimbal will perform well from a rigid base body such as Shuttle. However, they may still have the same dynamic interaction problems as simple gimbal systems when operating from a flexible base body like SS.

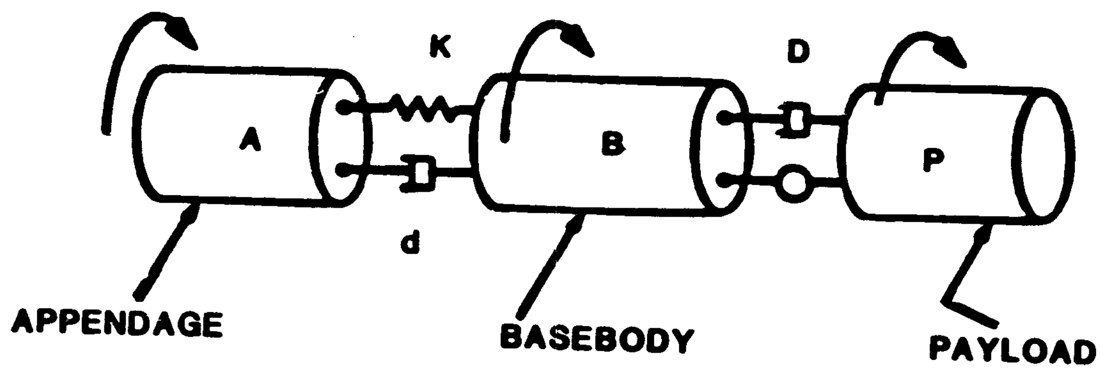


FIGURE 43 A SIMPLE GIMBAL SYSTEM

A soft mount, inertially reacting system will minimize dynamic interactions and approach the performance of a free flying spacecraft. It could still obtain required resources from the base body.

The soft mount, inertially reacting concept is equivalent to the SEM which has been under study at MSFC for several years. Figure 44 identifies its potential advantages over other concepts. Hardware is available to implement the pointing and control portion of the SEM but implementation of the suspension system needs to be studied. The concept also has applications to other payloads which require low acceleration environments. The interaction of the SEMs control system with the Space Station ACS must also be considered.

6.0 INSTRUMENT FOCAL PLANE SHIRTSLEEVE ACCESS

Maintenance, servicing and repair of attached payloads, free flyers or instruments on co-orbiting platforms, will be accomplished by EVA and MRMS operations. The primary maintenance technique will be by changeout of ORUs supplemented by a capability to perform some troubleshooting and repair at an elemental level [1].

A major find of the "Task Force Study on Utilization of the Space Station" (TFSUSS) was:

"The most important new capability to be provided by the Space Station Program for astronomy and astrophysics is that for routine maintenance and repair of co-orbiting facilities."

The "TFSUSS" also commented that:

"In addition, a large hangar-like module providing a 'shirtsleeve' environment would greatly augment the maintenance capability and be very helpful in the assembly, test, and checkout of instruments delivered to the Station."

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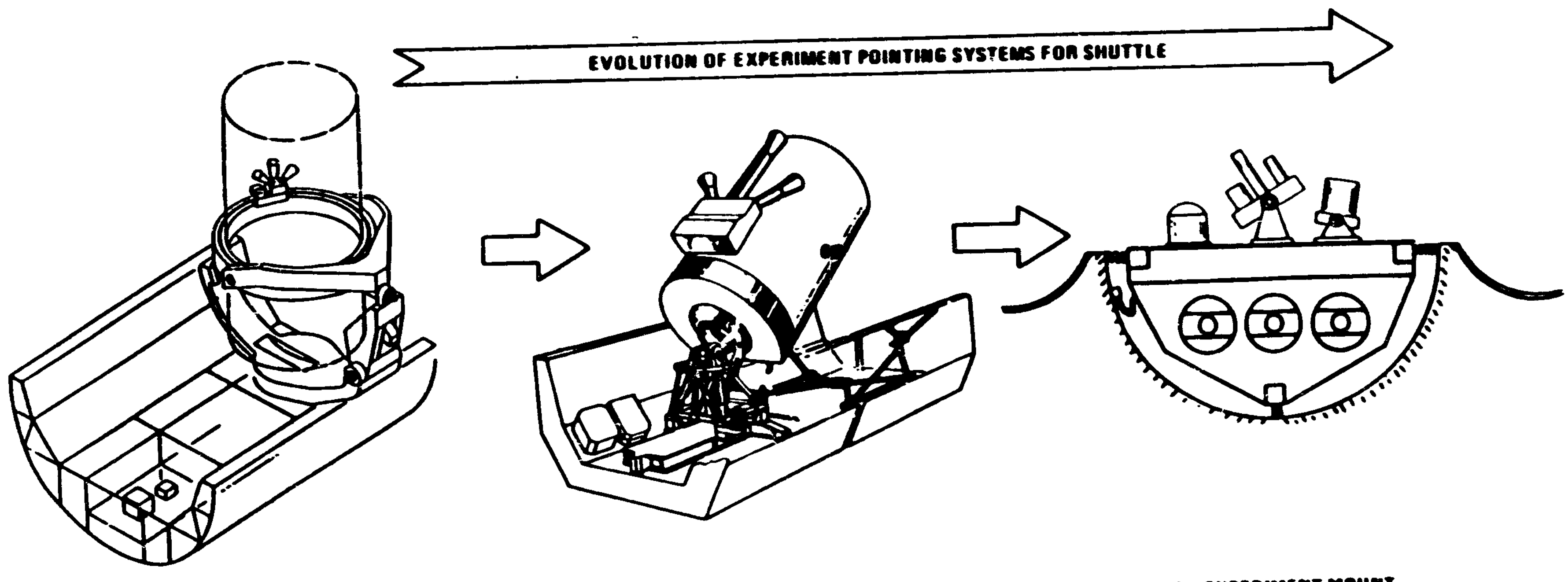

CONVENTIONAL GIRTH RING POINTING SYSTEM

- FIXED GIRTH RING DIAMETER
- FIXED C.G. LOCATION
- ELECTRICAL (AND FLUID) LINES ACROSS LOW TORQUE GIMBALS
- MULTIPLE INSTRUMENTS/SINGLE VIEWING DIRECTION
- INTEGRATION WITHIN NARROW POSITION, VOLUME, C.G. CONSTRAINTS

NASA/ESA INSTRUMENT POINTING SYSTEM

- BASE PLATE MOUNT
- LARGER C.G. ENVELOPE
- ELECTRICAL (AND FLUID) LINES ACROSS MEDIUM TORQUE GIMBALS
- MULTIPLE INSTRUMENTS/SINGLE VIEWING DIRECTION
- INTEGRATION RELAXED
- ONLY TWO IN INVENTORY AT HIGH COST
  - LIMITED FLIGHT OPPORTUNITIES

SUSPENDED EXPERIMENT MOUNT

- INDEPENDENT EXPERIMENT MOUNTING
- NO C.G. ENVELOPE CONSTRAINTS
- ELECTRICAL AND FLUID LINE FEED TO PALLET EASILY SATISIFIED
- MULTIPLE INSTRUMENTS/MULTIPLE VIEWING DIRECTIONS
- INTEGRATION UNCONSTRAINED BY POSITION, VOLUME, CONTAMINATION FROM THRUSTER FIRINGS, ASTRONAUT MOTION, ETC.
- CAN BE DEVELOPED USING EXISTING EQUIPMENT/TECHNOLOGY AT RELATIVELY LOW COST
  - LARGE EXPERIMENT MOUNTING AREA GIVES INCREASED FLIGHT OPPORTUNITIES

FIGURE 44 SEM ADVANTAGES

In the sense used, the term "shirtsleeve environment" would require a pressurized volume or module in which an astronaut in normal work attire could perform hands-on focal plane instrument change-out, repair or maintenance. The payloads to be serviced would include the attached payloads, free- flyers and instruments located on co-orbiting platforms.

6.1 IOC SPACE STATION "SHIRTSLEEVE ENVIRONMENT"

As described previously, the IOC Space Station will have five pressurized modules. Provisions for maintenance and repair of equipment have been provided in HM2. The module contains a maintenance work station which will provide the capability for some level of troubleshooting and repair. The work station will be outfitted with controlled lightning and magnification capabilities. A dedicated microprocessor with a printer and display capability will serve as a diagnostic tool supplemented by hand-held instruments. Electrical test equipment, a maintenance tool kit and a soldering kit will be provided. The maintenance station will be equiped with various small item restraints: a particle control system, a measurement kit (tape, micrometer, gauges, etc.), an adhesive kit (tape, cement, etc.), and a fastener kit (nuts, bolts, washers, etc.). If it is desired to perform troubleshooting and repair at this work station, the unit would have to be transported to it. In the case of the astrophysics payloads, an EVA would be required to obtain the unit requiring servicing and transport it back to the HM2 work station. The MRMS may be used to perform or assist in this operation if possible. The instrument or component size will be limited by the module hatch diameter and available space within HM2.

If moving a unit to the work station is not practical, an alternate scheme is provided. A portion of the maintenance work station in HM2 is removable and can convert into an EVA compatible work station. It can be transported, along with necessary equipment to perform troubleshooting and repair, to any location on the Space Station requiring servicing. It will be designed so that it can be positioned and rigidly restrained at the work site.

6.2 IOC SPACE STATION SERVICING FACILITY

An external Servicing Facility will be provided on board the manned IOC Space Station. It will be used to service free- flying satellites, co-orbiting platforms and attached payloads. The Servicing Facility will also provide for the storage of ORUs, instruments and tools needed for use during servicing. It will also provide for the maintenance of the OMV, OMV kits and storage of payloads, OMV and OMV kits.

These operations are performed in two separate bays. One bay is used for servicing operations and another for refueling operations. These two separate bays were determined to be necessary to prevent degradation of instrument optics by contaminants, which may be present due to refueling operations. This concern with potential contamination also dictated that the servicing bay be located away from the pressurized modules and the Orbiter berthing area.

6.3 PRESSURIZED HANGAR DESIGN CONSIDERATIONS

The servicing bay for the IOC S.S. is estimated to be a cylindrical volume, which is 30 ft. in diameter and 70 ft. long. This would allow for the berthing of a 15 ft. diameter by 60 ft. long satellite, with clearances all around for movement of an EVA crew and the placement of work stations.

A pressurized hangar, in which to perform services or provide access to an instrument focal plane, would be of similar dimensions. Consider the HST, its dimensions are approximately 15M x 5M x 5M (49 ft. x 16 ft. x 16 ft.). Approximate dimensions of the SOT are 7.5M x 4.4M dia. (25 ft. x 15 ft. dia.). The SOT attached to the IPS, along with a frame to secure them, would have approximate dimensions of 13.4M x 4.7M dia. (44 ft. x 15 ft. dia.). It is not unreasonable to assume that a hangar of the dimensions cited will be required to contain these large payloads and still provide working room, especially if more than one payload requires servicing at the same time.

Consider the location of the Hangar. Concerns with respect to instrument contamination would dictate that it be located on the upper keel above the transverse boom. This location would either require crew EVA for entrance/exit and the associated requirements that go with EVA, or alternate means of entrance/exit which preclude EVA. A pressurized tunnel running from one of the pressurized modules on the lower keel to the hangar could provide entrance/exit; so could a pressurized "elevator," which would traverse from a module on the lower keel to the Hangar. If crew access to the Hangar is the driving concern, then the Hangar should be located near the pressurized modules where a short crew transfer tunnel or module to Hangar interconnection would provide access. Regardless of the location, the Hangar, if incorporated into S.S., would have an effect on the ACS. The mass, size, shape and location of the Hangar would certainly force adjustments in the ACS so that it could orient and control the S.S. as required. The effects on the S.S. capability to provide required micro-g environment and point payloads must also be considered. The Hangar would have to have a large access port to enable the payloads to be brought in and out. Allowing the gas, required for a shirtsleeve environment, to escape each time the payload access port is opened, is unacceptable both from a contamination and resource point of view. A system will be required that will pump down the Hangar before the door is opened and repressurize it after the door is closed. The hangar environment itself should be controlled to some acceptable level to prevent contamination of the instruments . Obviously, the Hangar will have a major impact on the Environmental Control and Life Support System (ECLSS). It will require a separate ECLSS or be interfaced with the Space Station ECLSS, potentially requiring re-sizing of this system. Incorporation of the Hangar would have a major impact on most SS subsystems, including: Power, Thermal Control, Attitude Control, Environment and Life Support, information and data management, propulsion, structures and mechanisms, and crew systems. The Hangar would have to be assembled in orbit because of the orbiter constraints on cargo size and weight. Figure 45 illustrates the envelope for the orbiter cargo bay [49]. The envelope is of cylindrical shape with a diameter of 15 feet (4.572 M). Length available for the cargo would be 47' (14.266 M). The orbiter has the capacity to place up to 29,484 Kg (65,000 lbs.) into orbit.

Maximum cargo landing weight is 14,515 Kg (32,000 lb.). In essence, the cargo weight is constrained to 14,515 Kg, although the orbiter can return and land with cargo weight up to 29,484 Kg (65,000 lb.) under abort conditions.

These considerations identify that incorporating a module on Space Station which would provide a shirtsleeve environment would have many impacts. Incorporating such a facility would require a major program commitment.

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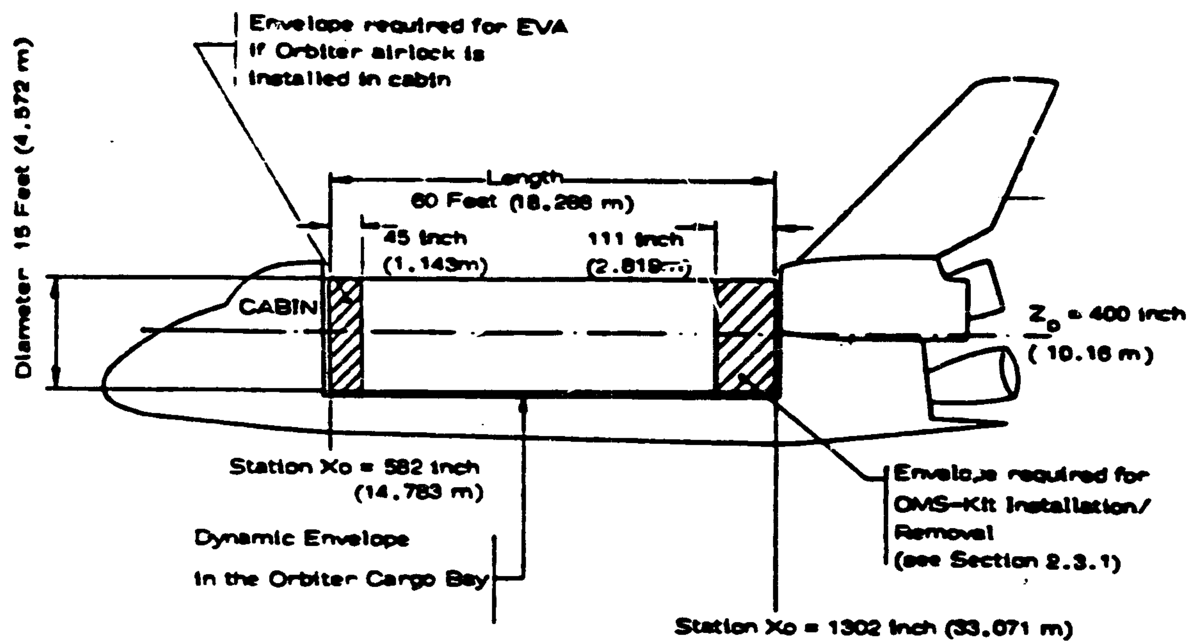

FIGURE 45 ORBITER CARGO BAY ENVELOPE

6.4 PAYLOAD DESIGN CONSIDERATION FOR SHIRTSLEEVE SERVICING

The performance of on-orbit servicing tasks will impact the design of the hardware requiring service. On-orbit servicing tasks would include:

- observation and inspection
- test, fault detection and isolation
- instrument modification or change-out
- component repair or replacement
- alignment and calibration
- surface restoration

The requirement to perform these tasks should be included as part of the system or hardware design and performance specifications. These specifications would incorporate the requirement for accessibility to the portions of the hardware which will require servicing. The hardware design should facilitate access to removal and installation of components, which are planned to be replaced on orbit. It should provide test points for connection of Electrical Support Equipment (ESE) required to perform fault isolation and detection.

From a mechanical or packaging point of view, the design should consider:

- crew visual and manual access
- tool insertion, engagement and movement envelope
- electrical cable identification, and positioning
- electrical connector location, indexing/ keying and ease of mate/demate
- access to guides, rails, alignment aids
- access to fasteners, hold-down and release devices, clamps, etc.
- placement of grounding straps
- hardware corners and edges
- protection or safing of any sensitive items
- placement of hand-holds or tethers
- effect of crew-induced loads on the hardware
- crew safety
- placement, attachment, and thickness of any protective blankets, wrappings or covers

Electrical designs should facilitate on-board testing to validate payload/system interfaces and performance of fault detection and isolation.

Since there is a potentially large demand for on-orbit servicing, the designs of the payloads should be coordinated to the degree possible to maximize the benefits of the servicing facility. Designs should be modular to allow easy removal and replacement of components. Designs should incorporate the use of common or standard mechanical and electrical interface devices. These include the use of standard latches, fasteners, electrical connectors, electrical circuits, data bus, electrical power and utility connectors to simplify the performance of on-orbit servicing. Achievement of modular payload designs using standardized connections, mounting provisions and configuration restrictions, will influence payload designs in terms of integration compatibility with Space Station, as well as enhance the on-orbit servicing capability. These designs should reduce the requirement for provision of payload unique special support equipment.

Some payloads, the HST and AXAF, for example, have or are incorporating these design considerations, since they are required to be serviced on-orbit via EVA. Other payloads and systems have not been specifically designed for on-orbit maintenance. The IPS maintenance concept, for instance, is to remove and replace to the functional line replaceable unit (LRU) level, either on a scheduled or unscheduled basis [25]. A LRU is defined as a combination of parts, a component, a sub-assembly or an assembly manufactured and/or assembled into one unique unit, or so arranged that, together the combination is common to a single mounting, and in addition provides a complete function to the higher level equipment within which it operates. Examples of possible LRU's are: fuses, harness, circuit boards, electrical units, thermal blankets, mounting struts, integrated assemblies, control panels, etc.

Previous demonstrations of on-orbit EVA servicing and repair (e.g., Solar Max Repair Mission) indicate that on-orbit maintenance of systems like the IPS in a shirtsleeve facility sould be possible to some extent, depending on the spares, tooling and test equipment provided.

7.0 CONCLUSIONS AND RECOMMENDATIONS

The Space Station "local-vertical local-horizontal" orientation is not conducive to performance of solar or stellar pointing experiments without additional pointing mechanisms. The Space Station RFP [2] recognized this and included the requirement to provide articulated attachments for pointing instruments. The station would provide two-axis pointing with an accuracy of ±1 degree and stability of ±0.02 deg/sec. Customers, who require more precise pointing, would have to provide their own pointing mounts. The study indicates that the technology required to provide precise pointing for payloads exists or is rapidly becoming available. It is recommended that a trade study be performed to determine the cost effectiveness of requiring the individual instrument to provide precise pointing versus the Space Station providing a generic precise pointing capability.

The dynamic characteristics and the disturbance environment of Space Station are the challenges with respect to pointing mount design. Existing systems and concepts were designed for operation from the Space Shuttle and it is unlikely that they will sustain performance when exposed to the Space Station disturbance environment. Development of a disturbance insensitive pointing mount is the key to providing a precise generic pointing mount for Space Station. An inertially reacting pointing system is less sensitive to dynamic interactions with the base body than gimbal systems. It is recommended that the Suspended Experiment Mount concept be investigated for use as a potential pointing mount for Space Station.

The Spacelab IPS is an existing flight system. It represents a large investment of time and money both from ESA and NASA. It should be considered for use as a generic pointing mount for Space Station. It is recommended that a study be made to determine if the Spacelab IPS can be modified for use as a generic pointing mount on Space Station and to determine the nature and extent of any required modifications. The Talon Gold Gimbal is a mature system, which provides for a high degree of disturbance rejection in the gimbal drives. Its ability to provide power, signal, and fluids across the gimbals without the use of cables or hoses also enhances its performance. It is recommended that the Talon Gold Gimbal be studied for potential use as a Space Station pointing mount. The technology incorporated into Talon Gold should also be considered for use in other Space Station mechanisms.

Availability of a module which will permit instrument change-out, repair and maintenance, is particularly desirable from a customer's view point. Inclusion of a shirtsleeve environment module would impact most Space Station subsystems. Addition of a facility of this type would require a major program commitment. It is recommended that a study be performed to determine the cost impact of providing a "shirtsleeve environment" facility versus the benefits gained by the customer with respect to longer orbital lifetime, increased observing time and less complicated EVA tasks.

APPENDIX 1

ASTROPHYSICS PAYLOADS

OBJECTIVES AND DESCRIPTIONS

SAAX0001 SPECTRA OF COSMIC RAY NUCLEI

OBJECTIVE:

To help explain the characteristics and distributions of galactic cosmic ray propagation through interstellar space.

DESCRIPTION:

CRNE exists. It was developed through the spacelab program: a one to two year exposure on the Space Station will enhance the scientific return by two orders of magnitude. Power, telemetry, contamination, and pointing requirements will be easy to satisfy on the gravity gradient stabilized station. The flight duration could be made variable, dependent on data gathered, but a two year exposure is baselined.

SAAX0004 SHUTTLE INFRARED TELESCOPE FACILITY

OBJECTIVE:

To conduct definitive high-sensitivity infrared photometric and spectroscopic studies of a wide range of astrophysical phenomena.

DESCRIPTION:

The infrared platform will carry the Space Infrared Telescope Facility (SIRTF), which is being planned as a 15-year mission with refurbishment at 2-year intervals. Consideration is being given to two options for the infrared platform: one as a platform derived from the Space Station's subsystems and the other as a platform derived from other astro platform subsystems. In either case, the infrared platform and its SIRTF payload will be serviced at the Space Station. This entry in the data base contains information delineating requirements on the platform, as well as requirements for servicing.

SAAX0005 TRANSISTION RADIATION & ION CALORIMETER

OBJECTIVE:

To observe electrons, protons, and helium nuclei in order to determine if there is a major change in the composition of cosmic rays at very high energy.

DESCRIPTION:

The transition radiation and ionization calorimeter experiment is an out-growth of a spacelab experiment. Long observation times improve its capability by a factor of ten. It would make a good candidate for a station resident experiment, since pointing accuracy and contamination are not primary factors; some preference for high inclination, but not critical.

SAAX0006 STARLAB

OBJECTIVE:

To investigate the visual, ultra-violet, and near infrared properties of extra galactic space, the milky way galaxy, and the solar system with high spatial resolution and wide angle imaging.

DESCRIPTION:

The first Starlab mission is to be by shuttle.

SAAX0007 HIGH THROUGHPUT MISSION

OBJECTIVE:

To investigate high energy processes in astrophysical systems, including the very compact sources associated with black holes and neutron stars.

DESCRIPTION:

HTM is an X-ray telescope which obtains good resolution by trading off mirror quality for quantity. Duration can continue beyond year 2000.

SAAX0008 HIGH ENERGY ISOTOPE EXPERIMENT

OBJECTIVE:

To search for rare nuclei and exotic particles like magnetic monopoles and to measure the composition of ultra-heavy nuclei.

DESCRIPTION:

If the HEIE is selected as a polar platform payload, it will be placed in a 57 deg. (or higher) orbit by the shuttle.

SAAX0009 PINHOLE OCCULTER FACILITY

OBJECTIVE:

To study non-thermal phenomena of plasma dynamics in the solar corona and to observe the acceleration of nonthermal particles in solar flares and in coronal disturbances with both x-ray and coronagraphic instruments.

DESCRIPTION:

The POF consists of a 32 M boom that separates an occulating mask from an array of detectors and telescopes. The name 'pinhole' derives from the use of a remote occulter (for coronagraphic studies), containing an array of small apertures, to obtain high angular resolution of hard x-radiation.

SAAX0010 SOLAR CORONA DIAGNOSTIC MISSION

OBJECTIVE:

To determine the cause of solar corona heating.

DESCRIPTION:

The SCDM has been envisioned as a possible free flyer. However, the SCDM should be studied as a station resident experiment, operated and maintained by the station personnel. Also seen as the third user of a common explorer bus, replacing Fuse. Replacement effected at the Space Station.

SAAX0011 ADVANCED SOLAR OBSERVATORY

OBJECTIVE:

To carry individual instruments capable of examining solar phenomena that can be pointed to regions of interest on the solar disc or throughout its atmosphere.

DESCRIPTION:

The ASO consists of the extreme ultraviolet telescope, the solar soft x-ray telescope, pinhole/occulter, and the solar optical telescope operated attached to the Space Station.

SAAX0012 HUBBLE SPACE TELESCOPE

OBJECTIVE:

To learn of the evolution of stars, of ours and other galaxies, and to explore quasars, pulsars, gas clouds and other planets.

DESCRIPTION:

The Hubble Space Telescope (HST) will be in service for about 5 years before a station based OMV will be available for servicing. Although not in the planned baseline for HST, it could be returned to earth for updating and refurbishment and replaced in orbit for service by the OMV. From then on, the OMV could be used for periodic servicing and reboost. HST Space Station support operations are assumed to begin in 1992.

SAAX0013 GAMMA RAY OBSERVATORY

OBJECTIVE:

To make the next major step in gamma ray astronomy by providing the first opportunity for comprehensive observations covering over five decades of energy from 0.05 to 30,000 MEV.

DESCRIPTION:

The observatory will be a three axis stabilized free flyer, capable of pointing to any celestial target for long periods of time. The attitude control accuracy is 0.5 degrees and the attitude determination accuracy is 2 arc minutes. All telemetry and command will be done by TDRS. GRO will consist of 4 experiments that collectively span the photon spectrum from 0.05 to 30,000 MEV.

SAAX0014 X-RAY TIMING EXPLORER

OBJECTIVE:

Measure temporal variability of x-ray objects for time scales ranging from microseconds to years.

DESCRIPTION:

Free-flyer explorer-class spacecraft, which houses a large area proportional counter and a wide field optical camera. Could be first of a series of replaceable telescopes on a single free-flyer bus. Replaced at Space Station by Fuse in 1993 in this scenario.

SAAX0016 SMALL SOLAR PHYSICS

OBJECTIVE:

Investigate causes and effects of solar flares.

DESCRIPTION:

MMS with telescopes covering UV to gamma rays.

SAAX0017 ADVANCED X-RAY ASTROPHYSICS FACILITY

OBJECTIVE:

X-ray astronomy research in the areas of source location and structure, spectroscopy, polarimetry, and temporal behavior.

DESCRIPTION:

X-ray platform will carry the Advanced X-ray Astrophysics Facility (AXAF). The X-ray platform will have spacecraft and instrument modules that will be replaced at regularly scheduled intervals at the space station. AXAF will be a national x-ray telescope facility with advanced capabilities in energy range, sensitivity, angular resolution and mission lifetime. AXAF instruments have been selected, and the platform is in Phase B definition. AXAF is a candidate for a payload on a space station platform.

SAAX0018 VERY LONG BASELINE INTERFEROMETER

OBJECTIVE:

To provide maps of compact celestial radio sources with finer resolution, less ambiguity, and more efficiency than earth-bound VLBI techniques.

DESCRIPTION:

The very long baseline interferometer mission requires correlation of measurements with the ground. The orbiting portion is placed by the shuttle. Co-habitation with the station is a possibility, if pointing and contamination requirements can be met.

SAAX0019 FAR UV SPECTROSCOPY EXPLORER

OBJECTIVE:

To carry out high and low resolution spectroscopy of distant stars, galaxies and intersteller matter in the 90-120 NM spectral range.

DESCRIPTION:

The FUSE is being planned for a shuttle launch and injection into geo-orbit. Another plan is to make fuse the second in a series of explorer missions, which use a single spacecraft bus. The replacement would take place at the Space Station. It follows XRTE and preceeds SCDM in this scenario.

SAAX0020 LARGE DEPLOYABLE REFLECTOR

OBJECTIVE:

To conduct infrared astronomical investigations for the survey of a wide variety of astrophysical phenomena throughout the infrared spectral region.

DESCRIPTION:

LDR will be brought to the Space Station on two shuttle loads. It will be assembled and tested at the station. It gets placed in operational orbit by the OMV and is periodically serviced by the OMV.

SAAX0021 SUPER CONDUCTING MAGNET

OBJECTIVE:

Antimatter and energetic isotopes contain information on the propagation of cosmic rays through the interstellar medium.

DESCRIPTION:

A superconducting magnet and charged particle track detectors form a facility in which the charge to mass ratio of particles can be measured. Associated detectors depend on the specific objective and are changed for different experiments. A bucking coil cancels the large scale field. A liquid helium cryostat is located between the two coils, attached to the top of a gravity-gradient stabilized space station. Power, commands, and data will be handled automatically through the station.

SAAX0022 THROUGH SAAX0025, SPACE STATION SPARTAN PLATFORM

OBJECTIVE:

Low cost astrophysics and space science payloads to perform scientific investigations and test new instruments and concepts. Low cost, minimum turn around time is a prime objective.

DESCRIPTION:

The Space Station based Spartan will be an evolution from the shuttle Spartan program. Scientific payloads will be integrated with the spartan carrier at the space station. The Spartan will be then transported into an appropriate orbit by the OMV and operate essentially independent of the Space Station for up to 3 months before being retrieved by the OMV, de-integrated on the station, serviced and re-deployed with a new payload for another 3 months.

SAAX0026 LEASED PLATFORM 1 (EXPLORER)

OBJECTIVE:

Exploxer missions are low-cost missions for special astrophysics, space plasma physics, and atmospheric investigations from space. The first payload would be XRTE.

DESCRIPTION:

The leased platforms will provide spacecraft services to the explorer payloads. Specific payloads are to be selected by OSSA as part of the approved explorer program. XRTE uses a large-area proportional counter, a scanning shadow camera, and a hard x-ray timing experiment to learn about the physics of neutron starts, black holes, and other compact celestial objects through variations in their x-ray output. Payload has been selected and the instruments are being defined. Payload will be replaced by another explorer after 2.5 years.

SAAX0027 LEASED PLATFORM 2 (EXPLORER)

OBJECTIVE:

Explorer missions are low-cost missions for special astrophysics, space plasma physics, and atmospheric investigations from space.

DESCRIPTION:

The leased platforms will provide spacecraft services to the explorer payloads. Specific payloads are to be selected by OSSA as part of the approved explorer program. Payload could be the FAR Ultraviolet Spectroscopy Explorer (FUSE).

SAAX0028 LEASED PLATFORM 3 (EXPLORER)

OBJECTIVE:

Explorer missions are low-cost missions for special astrophysics, space plasma physics, and atmospheric investigations from space.

DESCRIPTION:

The leased platforms will provide spacecraft services to the explorer payloads. Specific payloads are to be selected by OSSA as part of the approved explorer program. First payload could be the Solar Corona Diagnostics Mission.

SAAX0029 LEASED PLATFORM 4 (EXPLORER)

OBJECTIVE:

Explorer missions are low-cost missions for special astrophysics, space plasma physics, and atmospheric investigations from space.

DESCRIPTION:

The leased platforms will provide spacecraft services to the explorer payloads. Specific payloads are to be selected by OSSA as part of the approved explorer program.

SAAX0030 SPACE STATION HITCHHIKER 1

OBJECTIVE:

Very specialized science and applications investigations will be conducted with small payloads attached to the outside of the Space Station. Specific missions are to be determined by OSSA.

DESCRIPTION:

Space Station Hitchhikers are small, low-cost investigations attached to the outside of the Space Station. Payloads would fit into the shuttle Hitchhiker envelope -- gas can, sill and bridge carriers. Power, command, and data will go through the station to the ground. This Hitchhiker describes a get-away-special cannister with Hitchhiker power and telemetry.

SAAX0031 SPACE STATION HITCHHIKER 2

OBJECTIVE:

Very specialized science and applications investigations will be conducted with small payloads attached to the outside of the Space Station. Specific missions are to be determined by OSSA.

DESCRIPTION:

Space Station Hitchhikers are small, low-cost investigations attached to the outside of the Space Station. Payloads would fit into the Shuttle Hitchhiker envelope - - gas cans, sill and bridge carriers. Power, command, and data will go through the station to the ground. This hitchhiker describes the MSFC hitchhiker concept for the shuttle.

SAAX0032 SPACE STATION HITCHHIKER 3

OBJECTIVE:

Very specialized science and applications investigations will be conducted with small payloads attached to the outside of the space station. Specific missions are to be determined by OSSA.

DESCRIPTION:

Space Station Hitchhikers are small, low-cost investigations attached to the outside of the Space Station. Payloads would fit into the Shuttle Hitchhiker envelope - - gas cans, sill and bridge carriers. Power, command, and data will go through the station to the ground. This hitchhiker describes a concept developed at GSFC for use on the shuttle.

APPENDIX 2

POINTING MOUNT DESIGN CONSIDERATIONS

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1.0 PAYLOAD POINTING MOUNT DESIGN CONSIDERATIONS

In simple form, an instrument can be accurately pointed from a base body if a pointing system is provided that can compensate for base body motion, slew from object to object, and track the object, as necessary, to compensate for orbital rotation rate when the base body is operated in a local vertical orientation, as in Space Station. These simple tasks rapidly become complicated as the designer asks questions regarding size, shape, accuracy, time, amplitude, frequency, cost and schedule, in order to produce the hardware with the required capability. A list of the various studies and task, that need to be performed to arrive at answers to the pointing mount designers questions, are identified in Table A-1.

TABLE A-1. POINTING SYSTEM DESIGN CONSIDERATIONS/TASKS

• PAYLOAD REQUIREMENTS	• ERROR ANALYIS
• BASEBODY DISTURBANCES	• INTERFACE DEFINITION (ELEC., MECH, ETC.)
• BASEBODY DYNAMICS	• COMMAND FUNCTION
• ENGIRONMENTS	• TECHNOLOGY DEVELOPMENT
• ACTUATOR STUDY	• IN-FLIGHT ADJUSTMENTS
• SENSOR STUDY	• GROUND OPERATIONS
• CONTROLLER STUDY	• COST
• ISOLATION MOUNT STUDY	• SCHEDULE
• ESTIMATOR STUDY	• OTHER

A simple block diagram of a Space Station mounted platform pointing system is shown in Figure A-1.

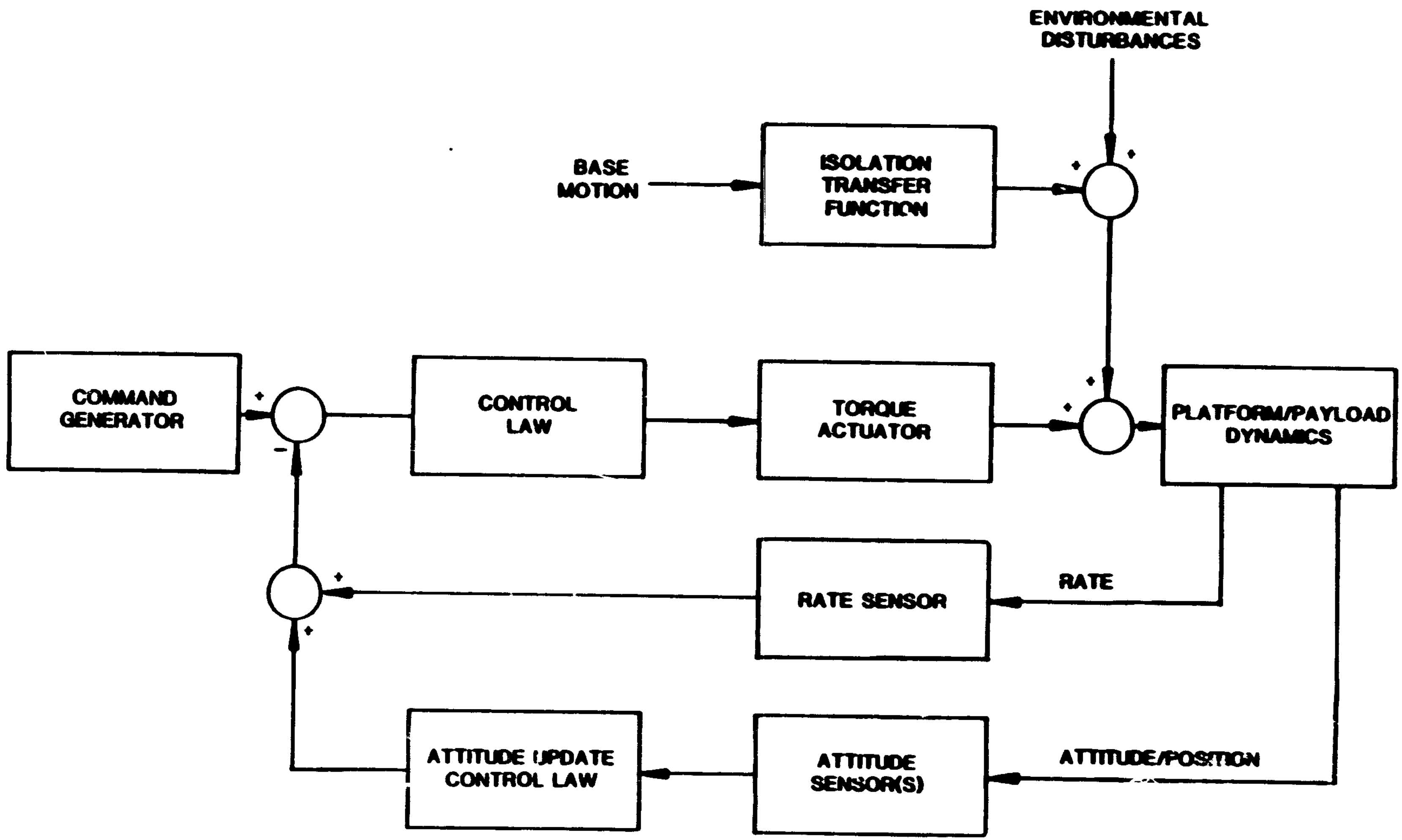

FIGURE A-1 PLATFORM POINTING SYSTEM CONCEPTUAL DIAGRAM

The block diagram is a convenient and well known method for representing a physical system or the set of mathematical equations that characterize its components. The blocks are the transfer elements which represent the functional relationships between the various inputs and outputs. The operations of additions and subtractions are represented by the small circle, called a summing point. The output of the summing point is the algrebraic sum of the inputs, each with an appropriate sign. The platform/payload is the part of the system that needs to be controlled. Inputs to the system come from three sources:

A) the command generator
B) dynamics of the base body (base motion)
C) environmental disturbances.

Ideally, we wish the pointing system to respond only to the command generator. The command generator provides a command profile for moving the instrument line of sight (LOS), either between observations or during observations, to compensate for the station's orbital rate motion due to its local vertical orientation. Inputs resulting from the base body dynamics and the environment are disturbances which produce unwanted effects. Table A-2 lists the major sources of disturbances which arise both from the dynamics of the base body, events occurring on the base body and from the environment and events external to the base body. The effect or these disturbances on the pointing system will, in large part, be determined by the structural dynamics (flexibility) of the base body. Refer to the simple block diagram of Figure A-1. The command generator provides a command to move or maintain the instrument LOS at some desired location/direction. This is compared to the actual location of the LOS, as best can be determined by the inertial sensor assemblies, which may include gyros, accelerometers, gimbal position encoders, and star trackers. The resultant difference or error signal is utilized by the controller, which computes commands according to a control law and issues these commands to a torquing device or actuator. A control law is a principal on which the controller is designed to achieve the overall system performance. The torque actuator which can be a gimbal torque motor, reaction wheel or CMG provides the torque required to move the platform/payload. Platform/ Payload rate, position and attitude data are then fed back and compared to the original input.

TABLE A-2. DISTURBANCE SOURCES	
INTERNAL	EXTERNAL
• RANDOM DISTURBANCES - MAN MOTION - OUTGASSING/VENTING - FLUID SLOSH	• ENVIRONMENTAL - GRAVITY GRADIENT - AERODYNAMIC FORCES AND TORQUES - GEOMAGNETIC FORCES AND TORQUES - SOLAR RADIATION PRESSURE - SOLAR HEATING
• ARTICULATED ELEMENTS - MOBILE REMOTE MANIPULATOR SYSTEM - SOLAR PANELS, RADIATORS - ANTENNAS - PAYLOADS	• DOCKING/BERTHING • POST DOCKING OPERATIONS
• ORBIT MAINTENANCE	
• ATTITUDE CONTROL	

The attitude update control law removes known systematic errors, which are introduced by known or measurable sensor inaccuracies by means of state estimation techniques. These are normally implemented by means of Least Squares Estimators or Kalman Filters.

1.1 KEY PAYLOAD REQUIREMENTS

The payload or instrument requirements, which most directly influence pointing system design, are payload mass, shape, pointing accuracy, stability, acceleration, tracking rate and settling time.

Payload weight and shape affect the pointing system structural design for withstanding both launch and operational loads and, because of its direct influence on moment of inertia, it becomes a major factor in designing mechanical isolators, gimbal bearings and actuators. Pointing accuracy requirements affect allowable mechanical alignment accuracies and thermal distortions. Stability Requirements primarily affect control system design, sensor selections and allowable control system bandwidth. Payload requirements for acceleration, tracking rate and settling time directly influence the selection of mechanical isolation devices, torquer/actuator design and overall control system design.

1.2 PAYLOAD/ POINTING SYSTEM TERMINOLOGY

The definition of several terms used to characterize payload requirements or pointing system capabilities are provided. "Pointing Error" - Defines the total angular error that will be allowed by the Pointing System. Pointing error is normally the sum of pointing accuracy and pointing stability. Statistical combinations are permitted, but the method must be defined.

"Pointing Accuracy" - Defines how close the Pointing System must initially point to the desired target. The numerical value of pointing accuracy is usually set by an instrument field-of-view (FOV). Pointing accuracy is static or at least quasi-static relative to experiment operation time. Examples of errors that contribute to pointing accuracy are sensor readout, etc. Pointing accuracy is often referred to as bias accuracy or bias error. Pointing accuracy numerical values are considered to be 3 sigma values because they set the probability of acquisition.

"Pointing Stability" - Defines how close the Pointing System must stay, during an experiment operation, pointed to the initial point. The initial point is set by the pointing accuracy requirement. Errors that contribute to pointing stability are dynamic in nature, such as sensor noise, structural bending vibrations, limit cycles, and perhaps thermal distortions. Pointing stability is often thought of as two conditions; either quiescent or disturbance. Quiescent stability is the case where there are no external disturbances acting on the Pointing System. The quiescent stability values are considered as 1 sigma. The disturbance stability errors are due to external disturbances acting on the Pointing System. Crew motion and RCS thruster firings from the Space Station are examples of the external disturbances. The disturbance stability errors are considered as peak values.

"Stability Rate" - Alternate definitions for the term are used. It is sometimes defined as the first derivative of the stability error time history. Alternately, it is defined as the maximum total attitude error that can be tolerated during any one (1) second of time When defined as the latter, stability rate is not the time derivative of attitude error. Stability rate is often referred to as jitter or sometimes jitter rate.

"Slew Rate" - Defines the maximum angular rate required during experiment operation or between operations as in slewing to another target. Since slewing is dynamic, a settling time (the time to reach steady state) must be defined. This also implies an acceleration level.

"Settling Time" - Defines the allowable time for the pointing control system to reacy steady state following any slew. When steady state is reached, it is understood that the desired pointing accuracy is met.

"Line of Sight (LOS)" - The LOS is a vector which considers with the payload optical axis. For many payloads, the LOS intersects the pointing system center of rotation (coordinate system origin) and is perpendicular to the payload interface plane.

"Servo Bandwidth" - Bandwidth is defined as the natural frequency of the controller position closed loop transfer function (-3db crossover).

1.3 DISTURBANCE SOURCES

Once the payload requirements have been defined, the disturbance environment in which a pointing system will perform needs to be defined. Disturbances arise both from internal and external sources as shown in Table A-2. The effect of these disturbances on the pointing system will, in large part, be determined by the structural dynamics (flexibility) of the base body.

1.4 POINTING MOUNT ARCHITECTURE

An understanding of current pointing system technology is facilitated by a review of the multilevel architecture currently utilized. Reference [51] provided such a review and it is summarized here. Pointing mount architectural considerations can be placed into three categories:

a) LOS Articulation Options (LOS)

b) Sensing Options

c) Control Methodology Options

LOS Articulation Options refer to the number of stages of isolation between the base body and the payload. Sensing Options refer to the source of the attitude reference, which, in turn, will influence the choice of the system attitude sensor(s). Control Methodology refers to the manner in which the control system loop is closed.

The method used to interface or physically attach the payload to the pointing mount will have an impact on the choice of the basic architectural options.

1.4.1 PAYLOAD MOUNTING

Previous systems and concepts have concentrated on three pointing system configurations with respect to payload mounting or interfacing to the pointing system. These are:

a) Conventional Girth Ring or Center of Mass Mount Pointing System

b) Base Plate Mount, End Mount or Cantilever Mount Pointing Systems

c) A Suspended or Floated Mount Pointing System

Center of Mass mounting systems require the payloads' center of mass to coincide with the center of rotation (COR) of the pointing system gimbal axis. This method of payload mounting reduces the effects of high frequency disturbances, since it decouples the base body translational motions from the payload rotation. Mounting every payload at its center of mass becomes impractical, since the payload mounting system at least, and perhaps the entire pointing system, must then be custom made for each payload. The Skylab Apollo Telescope Mount experiment pointing system represents a typical center of mass mounted payload pointing system.

Base Plate, End Mount or Cantilevered systems provide a more universal mounting interface that is advantageous when a wide variety of payload shapes and sizes are to be accommodated by the same pointing system. Other advantages over the center-of-mass method include a generally larger pointing envelope, a more compact and rigid mounting structure, and simplicity of deployment and stowage. Disadvantages include: an increase in the gimbal torque and torquer size required to provide the same angular acceleration; and difficulty in ground testing [52]. The major disadvantage of end mounted pointing systems (also referred to as cantilever mounts) is that translational motions of the base body at frequencies above the control bandwidth disturb the payload pointing angle. High frequency base body rotational motion (jitter) will also affect payload pointing when the pointing system is offset from the base body center of mass. This is almost certainly to be the case on Space Station. An end mount system then can easily accommodate a large selection of different instruments, but this is subject to increased disturbances from base body motion [48]. The Dornier Instrument Pointing System (IPS) represents an end mounted pointing system.

Another concept, which has been under study at MSFC for several years, is to provide an instrument mounting plate, which is attached to the base body by a flexible suspension system. Base plate stabilization is accomplished through CMGs or Reaction Wheels mounted on the base plate. The resulting system isolates the payload from high frequency disturbance originating from the base body and provides a high degree of stabilization for the payload. The payload may still acquire power, communication, and other services from the base body. The concept has applications to payloads that require a low-acceleration environment (e.g., materials processing experiments) and for payloads that require a high degree of stability. Slewing of payloads could then be accomplished by relatively coarse pointing mounts.

1.4.2 LOS ARTICULATION OPTIONS

LOS Articulation Options refer to the number of stages of disturbance isolation between the payload and the base body implemented to achieve the desired pointing and stability characteristics. In the case of a free-flying spacecraft which is dedicated to the pointing of its payload, there may be no articulating elements external to the payload. The Infrared Astronomical Satellite (IRAS) is an example. High precision is achieved by making the telescope an integral part of the spacecraft, constraining disturbances to a very low level and pointing the whole assembly, using a set of reaction wheels.

LOS Articulation becomes necessary to satisfy multi-payload pointing requirements and to compensate for base body disturbances if the spacecraft base body is noisy. More than one level of articulation allows for a veinier approach where each level of articulation becomes more precise. A three-level system is shown in Figure A-2 . Note that the various levels of articulation can serve distinctly different functions. The coarse gimbal rejects low frequency disturbances, the veinier stage acts as an isolator, attenuating the amplitude of the disturbance, and the Image Motion Compensation (IMC) stage rejects high frequency disturbances. A multi-level articulation architecture provides the designer with the options as to the number of levels, the arrangement of levels, and the component complement at each level.

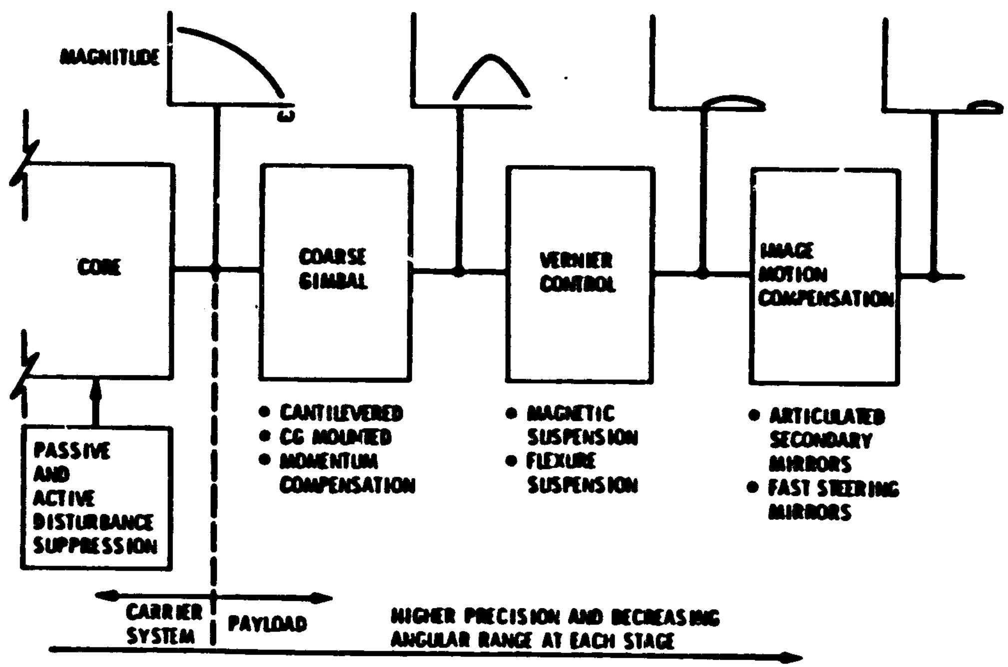

FIGURE A-2 MULTI-LEVEL ARTICULATION ARCHITECTURE

1.4.3 SENSING OPTIONS

Sensing Options pertain to the choice of attitude sensing hardware and to the location of the sensing hardware or source of attitude reference.

1.4.3.1 ATTITUDE SENSING HARDWARE

Hardware used for attitude determination and sensing includes Sun Sensors, Horizon Sensors, Magnetometers, Star Sensors and Gyroscopes (Gyros)[50]. Sun Sensors are the most widely used type; one or more have flown on most spacecraft. This is due to several factors. For low earth orbit operations the sun's angular radius is sufficently small that a point source approximation can be made and the Sun is bright enough to permit the use of relatively simple equipment without discriminating between sources. This simplifies the design of the sensor and the attitude determination algorithim. Most space missions are concerned with the orientation and time variation of the Sun vector in spacecraft body coordinates. The wide variety of applications have led to the development of numerous sensor types with fields of view (FOV) from several square arc minutes to steradians and resolutions of several degrees to less than 1 arc sec.The basic classes of Sun Sensors are Sun Presence Sensors, Analog Sun Sensors and Digital Sun Sensors. Analog Sun Sensors have a continuous output signal that is a function of the sun angle. Sun Presence sensors provide a continuous output whenever the sun is within the FOV. Digital Sun Sensors provide an encoded digital output which is a function of the sun angle.

Horizon Sensors provide a means for directly determining the orientation of a spacecraft with respect to the earth. Earth looking payloads typically require pointing accuracies of 0.05 deg. to 1 arc min. which exceeds the capability of present horizon sensors. Earth oriented spacecraft control systems utilize error signals from horizon sensors with accuracy ranges from 0.5 to 1 deg. Thus, although payload requirements may not be met by horizon sensors, control system requirements are easily satisfied. For the most part, Horizon Sensors are designed to operate in the infrared spectral band.This is done to avoid problems which would be introduced by visible light sources. It avoids triggering of the sensor by visible light reflected off clouds, and sunlight reflected from spacecraft surfaces. They are normally classified with respect to the scanning method used to search the celestrial sphere. These include rotating mirrors, body mounted sensors, and wheel mounted sensors. Body mounted sensors are fixed at a known angle to the structure of a rotating spacecraft. Wheel mounted units are fixed to a momentum wheel and the wheel rather than the spacecraft provides the scanning motion.

Magnetometers sense the direction and magnitude of the earth's magnetic field. They are lightweight, have low power requirements, operate over a wide temperature range, and have no moving parts. They are not extremely accurate since the earth's magnetic field is not completely understood and models to predict the magnetic field magnitude and direction at the spacecraft position are subject to substantial errors. Residual spacecraft magnetic biases restrict the use of magnetometers to altitudes below 1000 KM due to the earth's magnetic field decreasing with distance from earth as $1/R^3$.

Star Sensors measure known star coordinates in the spacecraft frame and provide attitude information when the observed coordinates are compared with the known star locations obtained from a star catalog. Star sensors are the most accurate of attitude sensors achieving accuracies in the arc sec range.On the other hand they are heavy, expensive, and require more power than other sensor types. More extensive computer software is required since measurements must be processed and identified before attitude information is obtained. There performance is affected by interference from the earth , sun and other bright sources and by occultations of the sun and earth. There are three main types: star scanners, gimballed star trackers, and fixed head star trackers. Star scanners use the spacecraft motion to perform the searching and sensing function. Gimballed star trackers use mechanical action for the search and track functions. Fixed head star trackers utilize electronic searching and scanning over a small field of view.

Gyroscopes are instruments that use a rapidly spinning mass to sense and respond to changes in the inertial orientation of its spin axis. Basic types used as attitude sensors are Rate Gyros, Rate Integrating Gyros, and Accelerometers. Rate gyros measure spacecraft anngular rates and are frequently part of the control loop for attitude stabilization. The angular rates may also be integrated via an on board computer to provide an estimate of spacecraft attitude with respect to some inertial reference. Rate integrating gyros measure angular displacements directly.

1.4.2.2 SENSING HARDWARE LOCATION

Payload attitude determination schemes fall into three categories:

A) base body referenced
B) inertially referenced
C) target referenced

In the base body referenced scheme, the attitude sensing hardware is located on the vehicle or spacecraft base body. The payload is responsible for determining its position with respect to the base body. This is normally accomplished by use of potentiometers, resolvers or encoders. The attitude of the payload is measured with respect to the base body. The attitude of the base body is measured by using the attitude sensors located on the base body. The method has certain limitations in that the payload attitude can never be as good as the base body, since sensing the relative position of the payload with respect to the base body introduces errors. It is difficult to sense relative position with high precision. In addition, structural deformations between the base body attitude sensors and the Payload LOS are not sensed and can take on significant values.

An inertially referenced method of attitude sensing utilizes attitude sensors mounted on the payload side of the gimbals. The payload attitude is measured with respect to known star locations. The accuracy of this method is limited by the errors inherent in the sensors themselves and the accuracy of the star catalog being used. Current star catalogs are accurate to about 1 arc/sec and are predicted to be accurate to 0.1 arc sec during the next decade. An inertially referenced attitude determination scheme can satisfy the requirements of most pointing payloads, provided the attitude sensors are rigidly attached to the instrument's focal plane and their alignment to the instrument LOS is known and maintained to the required accuracy.

If an inertially referenced scheme is not sufficiently accurate, target referenced attitude sensing can be employed. In target referenced attitude sensing, the pointing control system uses an error signal provided by the payload's focal plane sensors to close the pointing loop. Errors introduced by structural bending modes and misalignments are bypassed. This method of attitude sensing does complicate interfacing the payload to the pointing system and utilization of the pointing system to accommodate multiple payloads.

1.4.4 CONTROL METHODOLOGY OPTIONS

Control methodology options refer to the manner in which the control loop is closed or the control system design approach. The basic choices in this regard are classical control or modern control techniques. Classical control techniques primarily treat single input and single output systems and numerous frequency domain procedures have been developed to handle such systems. When more than one input or output are present, they are assumed to be decoupled or independent and separate single input and single output systems are provided for each. Modern control techniques are particularly well suited for use in complex systems where coupling effects are important, such as cases where several degrees of freedom are affected by a single actuator or measured by a single sensor. Modern control techniques, primarily formulated in the time domain, utilized the computational capability of digital devices to handle non- linear control problems. Whether a classical or modern approach is taken, the concepts of feed-ahead commands, adaptive control, and distributed control can be used. Feed-ahead or feed-forward commands utilized knowledge of a previously calibrated anticipated disturbance to cancel the disturbance torque before it propagates through the system, resulting in attitude motion, which then must be sensed and counteracted via a low bandwidth control system. Adaptive control systems include a parameter adjustment loop which changes control law parameters, such as: control gains, bending mode filter parameters, and command generator commands to adapt to changes to the configuration of the base body. Adaptive control is based on knowledge of previous input/output history, and a prior knowledge of the system configuration (i.e., an accurate system model or system identification) and the performance requirements. The adaptive control system continually adjusts itself, based on the system model and what it perceives through its sensors.

Distributed control includes systems where two or more control systems having their own sensors and actuators operate in a coordinated manner (i.e., master/slave or hierarchical control) or a system which utilizes distributed sensors and/or actuators which interface with a centralized controller.

Whether a classical or modern control approach is used, an accurate system model (system identification) must be available. Due to the dynamically interactive environment of Space Station, a precision pointing system may have to have a model not only of itself, but of the Space Station as well. Changes to the Space Station's mass and system properties during its evolutionary growth, and due to shuttle docking, will necessitate changes in the control law gains. The new mass properties can be estimated on-board, based on the properties of the indivi. 'al elements added to the station. The controller gains may be updated via ground command, astronaut inputs or autonomously on board.

LIST OF ABBREVIATIONS

AAS	American Astronautical Society
AC	Alternating Current
ACA	Attitude Control Assembly
ACS	Attitude Control System
AEPI	Atmospheric Emissions Photometric Experiment
AGC	Automatic Gain Control
AGS	Advanced Gimbal System
AIAA	American Institute of Aeronautics and Astronautics
APCS	Attitude and Pointing Control System
APM	Articulated Primary Mirror
ASM	Assembly
ASO	Advanced Solar Observatory
ASPS	Annular Suspension and Pointing System
ATM	Apollo Telescope Mount
ATMDC	Apollo Telescope Mount Digital Computer
AXAF	Advanced X-Ray Astrophysics Facility
BEA	Backup Electronics Assembly
BGSD	Bendix Guidance Systems Division
CDMS	Command and Data Management System
CEA	Control Electronics Assembly
C.G.	Center of Gravity
CGA	Coarse Gimbal Assembly
CIP	Coordinated Instrument Package
CMG	Control Moment Gyroscope
CMGEA	Control Moment Gyroscope Electronics Assembly
CMGIA	Control Moment Gyroscope Inverter Assembly

LIST OF ABBREVIATIONS (Cont'd)

Conf.	Configuration
COR	Center of Rotation
CPS	Cycles per second
CRT	Cathode Ray Tube
CSS	Coarse Sun Sensor
C&T	Communications and Tracking
DA	Digital to Analog
DARPA	Department of Advanced Research Projects Agency
dB	Decibel
DC	Direct Current
DCU	Digital Control Unit
DDU	Digital Display Unit
DEA	Digital Electronics Assembly
Deg	Degrees
DEP	Dedicated Experiment Processor
DMU	Data Management Unit
Doc	Document
DRIRU	Dry Rotor Inertial Reference Unit
ECLSS	Environment Control and Life Support System
ECU	Electronics Control Unit
Elec	Electrical
EPC	Experiment Pointing Control
EPCS	Experiment Pointing Control System
EPDS	Electrical Power Distribution System
EPDU	Electrical Power Distribution Unit
EPEA	Experiment Pointing Electronics Assembly
EMC	Electromagnetic Compatibility
ESA	European Space Agency
EVA	Extravehicular Activity
Exp	Experiment

LIST OF ABBREVIATIONS (Cont'd)

FGS	Fine Guidance Sensor
FHST	Fixed Head Star Tracker
FOV	Field of View
FSS	Fine Sun Sensor
FUSE	Far Ultraviolet Spectroscopy Explorer
G	Gravity
GEA	Gimbal Electronics Assembly
GN&C	Guidance, Navigation and Control
GPC	General Purpose Computer
GPS	Global Position Satellite
GRO	Gamma Ray Observatory
GSFC	Goddard Space Flight Center
Gyro	Gyroscope
HEIE	High Energy Isotope Experiment
HGA	High Gain Antenna
HM	Habitability Module
HRM	High Rate Multiplexer
HST	Hubble Space Telescope
HTM	High Throughput Mission
Hz	Hertz
I&C	Instrumentation and Communication
ID	Interface Device
I/F	Interface
IMC	Image Motion Compensation
IMCS	Image Motion Compensation System
I/O	Input Output

LIST OF ABBREVIATIONS (Cont'd)

IOC	Initial Operational Capability
IPS	Instrument Pointing System
IRAS	Infrared Astronomical Satellite
ISA	Inertial Sensor Assembly
IVA	Intravehicular Activity
JEA	Jettison Electronics Assembly
Kg	Kilogram
Kg-M^2	Kilogram $Meter^2$
Kg-M^3	Kilogram $Meter^3$
kHz	Kilohertz
Km	Kilometer
Kw	Kilowatt
Lb	Pound
LDR	Large Deployable Reflector
LLLTV	Low Level Light Television
LM	Laboratory Module
LMSC	Lockheed Missiles and Space Company
LOS	Line of Sight
LRU	Line Replaceable Unit
LVLH	Local Vertical Local Horizontal
m	Meter
MAST	Modified ATM Star Tracker
Max	Maximum
MBA	Magnetic Bearing Assembly
MDA	Multiplying Digital to Analog

LIST OF ABBREVIATIONS (Cont'd)

MDM	Multiplexer Demultiplexer
Mech	Mechanical
MEV	Million Electron Volts
MJA	Mounting and Jettison Assembly
MMC	Mount manual control
MMC	Martin Marietta Company
MPC	Manual Pointing Controller
MPM	Miniature Pointing Mount
MRMS	Mobile Remote Manipulator System
MSFC	Marshall Space Flight Center
MSS	Magnetic Sensing System
MT	Magnetic Torquer
NASA	National Aeronautics and Space Administration
N-M	Newton Meter
N-M-S	Newton Meter Second
NSSC	NASA Standard Spacecraft Computer
OA	Orbit Adjust
OMV	Orbital Maneuvering Vehicle
OPTO-MECH	Optical Mechanical
ORU	Orbital Replaceable Unit
OSP	Optical Sensor Package
OSSA	Office of Space Science and Applications
OTA	Optical Telescope Assembly
OTV	Orbital Transfer Vehicle
OWS	Orbital Workshop

LIST OF ABBREVIATIONS (Cont'd)

PCA	Payload Clamp Assembly
PCE	Power Conditioning Electronics
PCS	Pointing Control System
PDU	Power Distribution Unit
PEA	Payload Electronics Assembly
P/L	Payload
PMS	Payload Mounting Structure
POF	Pinhole Occulter Facility
POP	Perpendicular to the Orbit Plane
RAU	Remote Acquisition Unit
REF	Reference
RG	Rate Gyro
RGA	Rate Gyro Assembly
RIU	Remote Interface Unit
RMS	Root Mean Square
RPM	Roll Positioning Mechanism
RSU	Rate Sensor Unit
RWA	Reaction Wheel Assembly
SA	Solar Array
SAAX	Science and Applications Mission Data Base
SCDM	Solar Corona Diagnostic Mission
SCRN	Spectra of Cosmic Ray Nuclei
SEC	Second
SEM	Suspended Experiment Mount

LIST OF ABBREVIATIONS (Cont'd)

SEUVTF	Solar Extreme Ultraviolet Telescope Facility
SI	Science Instrument(s)
SIPS	Small Instrument Pointing System
SIRTF	Shuttle Infrared Telescope Facility
SOT	Solar Optical Telescope
SS	Space Station
SSM	Support System Module
SSP	Space Station Program
SSXRTF	Solar Soft X-Ray Telescope Facility
ST	Space Telescope
STRUC	Structure
STS	Space Transportation System
SUPERMAG	Super Conducting Magnet
TACS	Thruster Attitude Control System
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TEA	Torque Equilibrium Attitude
TFSUSS	Task Force Study on Utilization of Space Station
TGG	Talon Gold Gimbal
TRIC	Transition Radiation and Ion Calorimeter
TV	Television
Typ.	Typical
UV	Ultraviolet
V	Volts
VAC	Volts, Alternating Current
VCS	Vernier Control System
VDC	Volts, Direct Current
VIPS	Vibration Isolation Pointing System
VLBI	Very Long Baseline Interferometer
VPA	Vernier Pointing Assembly

LIST OF ABBREVIATIONS (Cont'd)

WJU	Wiring Junction Unit
XRTE	X-Ray Timing Explorer

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