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VOLUME

~~NASA CR~~ - 152571

CONTRACT NO. NAS8-31690
27615-6007-RU-06
DR-MA-03A

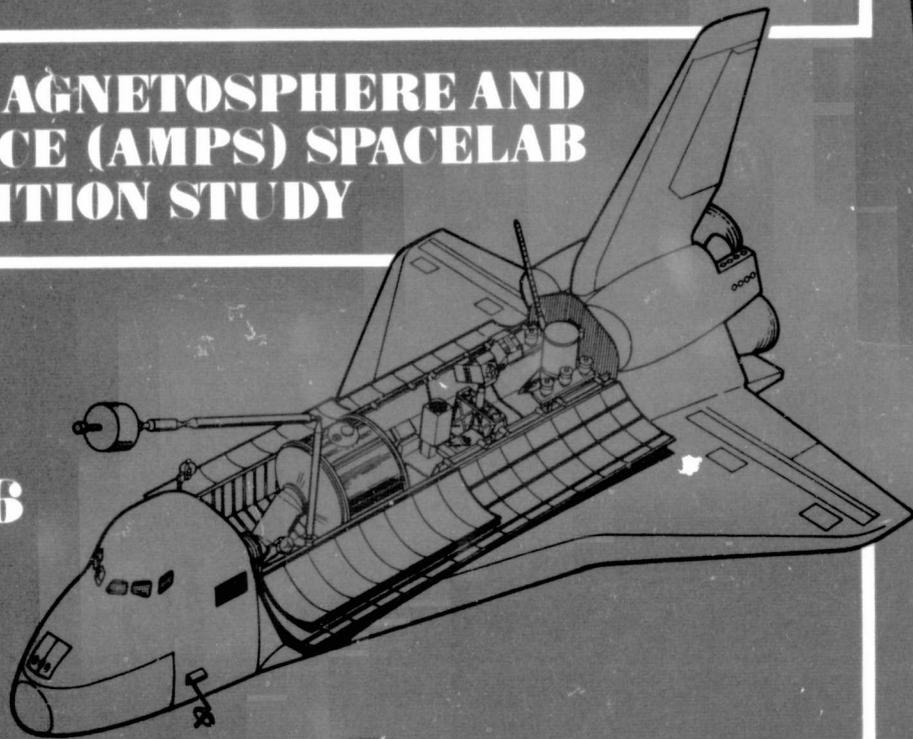
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TECHNICAL SUMMARY

(NASA-CR-152571) ATMOSPHERE, MAGNETOSPHERE AND PLASMAS IN SPACE (AMPS). SPACELAB PAYLOAD DEFINITION STUDY. VOLUME 5: TECHNICAL SUMMARY Final Report (TRW Defense and Space Systems Group) 309 p HC A14/MF A01 G3/12	N77-28174 Unclas 40003
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ATMOSPHERE, MAGNETOSPHERE AND PLASMAS IN SPACE (AMPS) SPACELAB PAYLOAD DEFINITION STUDY

**Final Report
November 1976**



Prepared for
National Aeronautics
and Space Administration
Goddard Space Flight Center
Greenbelt, Maryland 20771



TRW
DEFENSE AND SPACE SYSTEMS GROUP

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FINAL REPORT

VOLUME V
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Approved by:

W. F. Rector, III

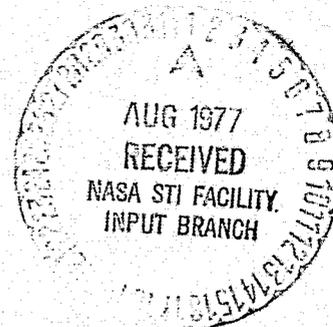
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Prepared for
National Aeronautics and Space Administration
Goddard Space Flight Center
Greenbelt, Maryland 20771
Contract No. NAS8-31690

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1. INTRODUCTION

This document summarizes the technical results of the AMPS Phase B Study sponsored by the NASA Goddard Space Flight Center. It represents the culmination of an extended series of scientific, engineering and operational studies directed towards definition of a Spacelab facility for the conduct of atmospheric, magnetospheric and plasma science in near-earth orbit during a series of flights in the Spacelab element of the Shuttle Transportation System (STS). Five "strawman" AMPS flights have been the basis for performing this study and are summarized from a scientific, experiment and instrument viewpoint in Section 2. The remainder of the document is devoted to the details for joint Flights 1 and 2 of AMPS.

This document covers all engineering and operational facets associated with the implementation of the first two AMPS flights. In Sections 3 and 4, the payload is described including all systems and subsystems. In Sections 5 and 6, the mission planning and flight operations are described. Following this are the important subjects of payload integration, ground operations and logistics. Finally, Section 9 covers certain key supporting analyses and mass properties.

In this document the term Labcraft has been utilized. This, in general, describes the interactive capability between AMPS payload instruments and Spacelab. This, in some cases, involves additional flight support equipment (FSE) which can possibly be utilized for other payload disciplines.

The AMPS Payload design data, performance estimates and conclusions presented herein are based heavily on Spacelab and Orbiter design and performance data available during the final phase of the Phase B Study. They may become invalid as Spacelab and Orbiter design data and performance estimates further evolve. Specific reference is made to the distribution of ESA SLP-2104 Spacelab Payload Accommodation Handbook, Review Issue PDR-B 1976, at the time of publication of this Technical Summary. This heavily revised document incorporates many changes which directly impact the results of this study.

2. PROGRAM DEFINITION

The AMPS program defined as the basis for this Phase B Study was derived from five "Strawman" payloads generated by the NASA Goddard Space Flight Center. These payloads were derived from the results of mission goals developed by the AMPS Science Definition Working Group (ASDWG).

The system and subsystem engineering analyses performed to date were based on ensuring a smooth evolution of the five flights in terms of integration and design of flight support equipment. The accommodation of each payload into Spacelab takes full advantage of existing Spacelab and Shuttle resources such as data handling and communications equipment. Modularity has been applied at both the systems and pallet levels for economy of installation, integration, and refurbishment and to minimize the development costs.

Techniques have been devised for assembling, integrating, and testing the payloads, stressing low cost and safety while ensuring flight readiness. Flight operations have been designed to conform with the constraints and capabilities of the STS. Lowest possible cost has been a prime consideration throughout the development of the concept for achieving AMPS scientific goals. This will continue to be a vital criterion for the program.

The scientific objective of the AMPS program is to assist in developing a comprehensive understanding of the region surrounding the earth; see Figure 2-1. Based on laboratory and space experimentation, several techniques for meeting this goal have been proposed. These techniques involve observing the effects produced by natural processes or by disturbances that are purposely imposed. The gas in the region is in a plasma state, which means that the molecules are ionized, can conduct electricity, and are affected by magnetic fields. Thus, understanding the region involves studying the earth's electric and magnetic field system, energetic particle and electromagnetic wave interactions, physical processes associated with the motion of bodies in rarefied plasmas, and the chemistry and dynamics of the upper atmosphere. AMPS will allow investigators to perturb the near-earth environment in a controlled and systematic manner. For the first time, a full, manned laboratory of instruments can be carried into a region of interest to conduct remote and insitu investigations in a structured, evolutionary program.

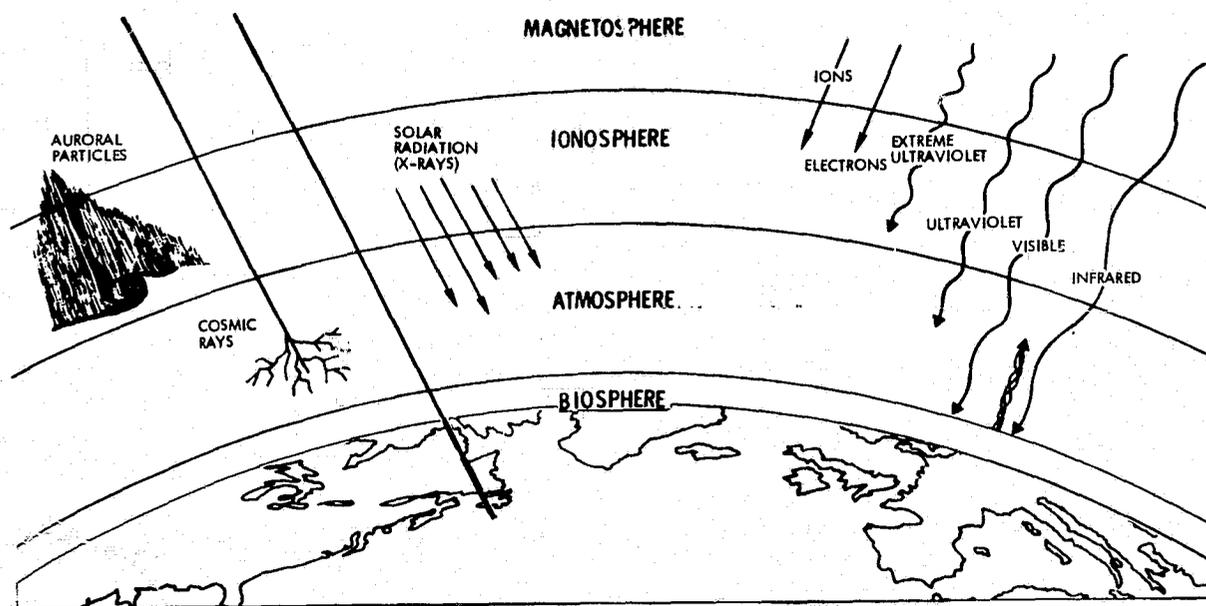


Figure 2-1. The Environment of AMPS Science

The AMPS Spacelab payload is planned by NASA as a laboratory in space for international use. In concept and operation, AMPS will be similar to the national and international laboratories now used by scientists of many countries for nuclear research and for radio and optical astronomy.

The AMPS program uses various techniques and instruments to explore four specific disciplines: particle interaction, plasma interaction and flow, atmospheric science, and wave phenomena. Many of the techniques and instruments will apply to investigations in several disciplines.

2.1 PARTICLE INTERACTION

A broad range of energetic particle interaction experiments is planned for all five flights. On Flight 1, the first electron beam propagation experiments will be performed with a modest accelerator of up to 40 keV and 2 amperes current; see Figure 2-2. Initial beam plasma interactions will be studied. On later flights, accelerator current and voltage will be increased to permit echo, atmospheric interaction, plasma wave excitation, and electric field investigations.

A magnetoplasmadynamic (MPD) arc accelerator will be introduced on Flight 4 to study the interaction of a high-density plasma stream with the natural plasma environment. The high current (up to 10^5 amperes) can be used for conductivity modification, magnetized plasma interaction,

and plasma instability studies. On Flight 5, an ion accelerator will be added. More sophisticated techniques will be used to study parallel electric fields, two-stream instability, and very-low-frequency wave generation by beam modulation.

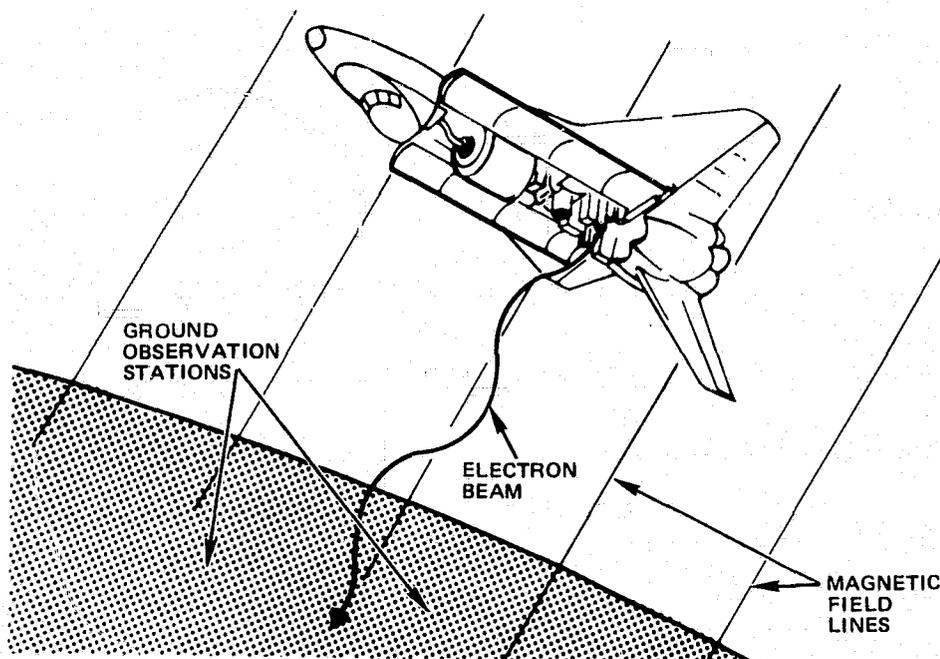


Figure 2-2. Artificial Aurora

2.2 PLASMA INTERACTION AND FLOW

The AMPS laboratory will use chemical and gas releases to generate controlled clouds for a broad class of studies in plasma interactions, see Figure 2-3. Many investigations can be performed using various materials, masses, and techniques. In each case, the orbital velocity provides a large energy source that cannot be duplicated with rocket flight releases. For example, Flight 1's orbital velocity will provide energy to a neutral gas which will create artificial gravity-driven waves. In this experiment, onboard observations and data gathering will be done in conjunction with ground-based systems. In all experiments, the actual gas release or chemical ignition will take place well away from the Orbiter.

From an inert gas release on Flight 1, the program evolves to barium-thermite releases on Flight 2, and to barium shaped-charge deployments on Flights 4 and 5.

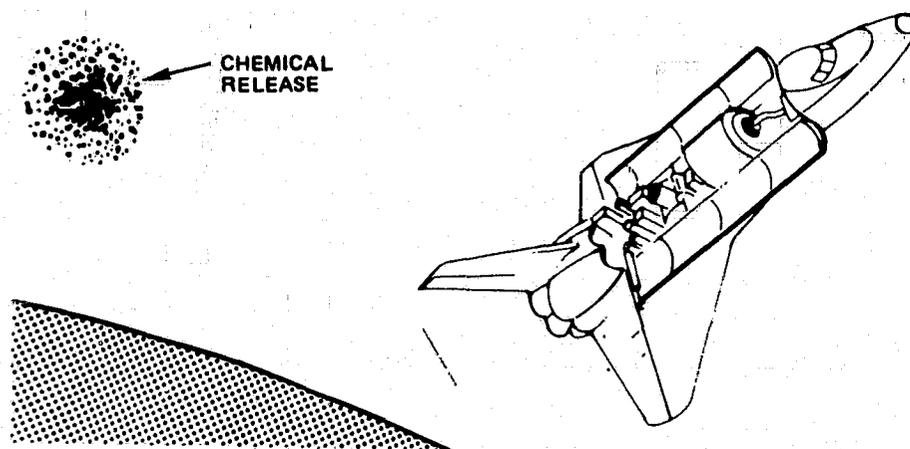


Figure 2-3. Chemical Release Experiment

2.3 ATMOSPHERIC SCIENCE

Observations and studies of the upper atmosphere will be performed in three types of experiments. In the first type, instruments will be pointed at the earth's limb while vertical or height-scanning is performed using spectrometers with various sensitivities, resolutions, and wavelengths, depending on the constituents and lines studied. In the second type, instruments will be pointed downward to study specific atmospheric events or features. In the third type, a lidar will actively probe the atmosphere, see Figure 2-4.

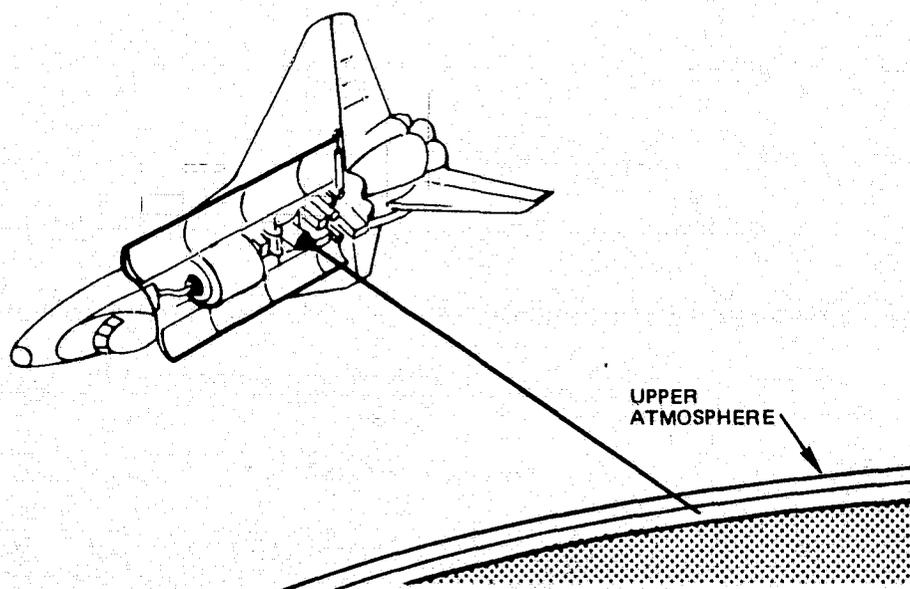


Figure 2-4. Lidar Probing of the Upper Atmosphere

Flight 1 includes limb observations in the infrared to determine minor constituent density (and possibly temperature regimes), observations of aurora by the Optical Band Imager and Photometer System (OBIPS), and lidar studies of ozone densities. In later flights, instrumentation in the visible and ultraviolet areas, as well as a Fabry-Perot interferometer, will be added to support studies of temperature profiles, wind velocities, and constituents such as the hydroxyl radical. An observing facility will be available for detailed studies of particular problems such as chlorine chemistry. The lidar experiments will expand from initial ozone measurements on the first flight to detailed studies of ozone and minor atmospheric constituents such as nitric oxide and chlorine oxide, then to the study of winds and diffusion, and finally to composition measurements using long-path absorption with a detector or retroreflector on a subsatellite.

2.4 WAVE PHENOMENA

Electromagnetic wave transmission experiments will begin at radio frequencies of 30 kHz to 20 MHz to provide information on the natural plasma density, temperature, and composition and to initiate the investigation of wave-particle and wave-plasma instabilities in the ionosphere; see Figure 2-5. To avoid Orbiter-produced electromagnetic interference (EMI), and to make measurements at remote locations, receivers will be placed on Environmental Sensor Packages (ESP's), inexpensive non-recoverable satellites, and maneuverable subsatellites. In later flights, transmitting powers will be increased, transmitting frequencies decreased to cover the range below 1 kHz and more sophisticated controls and diagnostic instruments introduced. Local ionospheric heating, artificial particle dumping, and pulse propagation phenomena will then be studied.

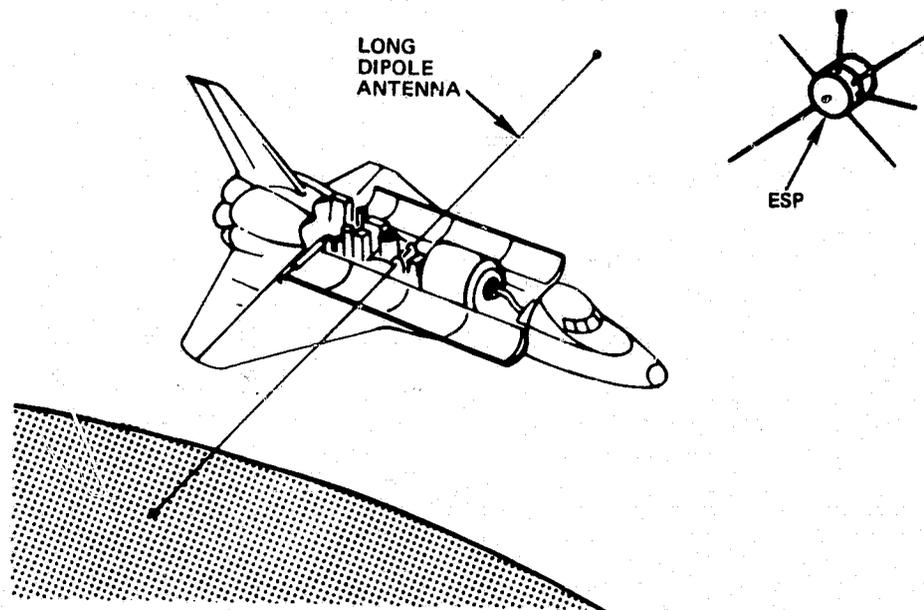


Figure 2-5. Wave Injection and Wave Particle Phenomena

2.5 AMPS FIVE FLIGHT PROGRAM

The initial phases of the science activities covered in the previous sections can be accommodated effectively into a five Orbiter flight program that employs the Spacelab module and three pallets. These flights are characterized by the evolution from relatively simple experimentation on the first flight to highly refined experiments on Flights 4 and 5. During this sequence both instruments and support equipment evolve to increased capabilities. This evolutionary buildup has the effect of deferring the increased cost of equipment refinements until they are needed for experiments. On completion of the five-flight AMPS program covered in the next sections, the physical resources will exist for the continuation of the program into other areas of AMPS experimentation.

2.5.1 Flight 1 Experiment and Instrument Complements

The specific experiments planned for AMPS Flight 1 for the four disciplines are shown below:

<u>Discipline</u>	<u>Experiments</u>
Particle Interaction	Beam Characteristics Spacecraft Neutralization Beam Plasma Interaction

Plasma Interaction
and Flow

Wave Phenomena

Atmospheric Science

Acoustic Gravity Wave

Gas Cloud Dynamics

Environmental Observations (EMI)

Minor Constituents

Ozone Density

Solar Radiation

A set of core instruments has been defined for AMPS. These instruments, initially including the lidar, accelerator, atmospheric array, gas release system and associated diagnostics, are required for performing many different experiments on multiple AMPS flights. Figure 2-6 shows the experiment instrument complement matrix for the flight. In the following sections the core instruments for Flight 1 will be discussed.

INSTRUMENTS	EXPERIMENTS BEAM CHARACTERISTICS	SPACECRAFT NEUTRALIZATION	BEAM-PLASMA INTERACTIONS	ACOUSTIC GRAVITY WAVE	GAS CLOUD DYNAMICS	ENVIRONMENTAL OBSERVATIONS	MINOR CONSTITUENTS	OZONE DENSITY	SOLAR RADIATION
ACTIVE INSTRUMENTS									
GAS RELEASE				•	•				
ELECTRON ACCELERATOR	•	•	•						
SUPPORTING INSTRUMENTS									
VECTOR MAGNETOMETER	•	•	•						
LEVEL 1 DIAGNOSTICS	•								
LEVEL 2 DIAGNOSTICS	•	•	•						
QUARTZ CRYSTAL MICROBALANCE						•			
ATMOSPHERIC ARRAY									
LIDAR								•	
OPTICAL BAND IMAGER	•	•		•	•				
IR RADIOMETER							•		
IR INTERFEROMETER							•		
ENVIRONMENTAL SENSOR PACKAGE	•	•	•			•		•	
SOLAR FLUX MONITOR									•

Figure 2-6. Flight 1 Experiment Complement Matrix

2.5.1.1 Electron Accelerator

Investigations with the electron accelerator on Flight 1 begin with studies of the beam properties and Shuttle/Spacelab neutralization

mechanisms; see Figure 2-7. A beam diagnostics program is the first step. Later in the flight, electron beam induced plasma instabilities will be studied.

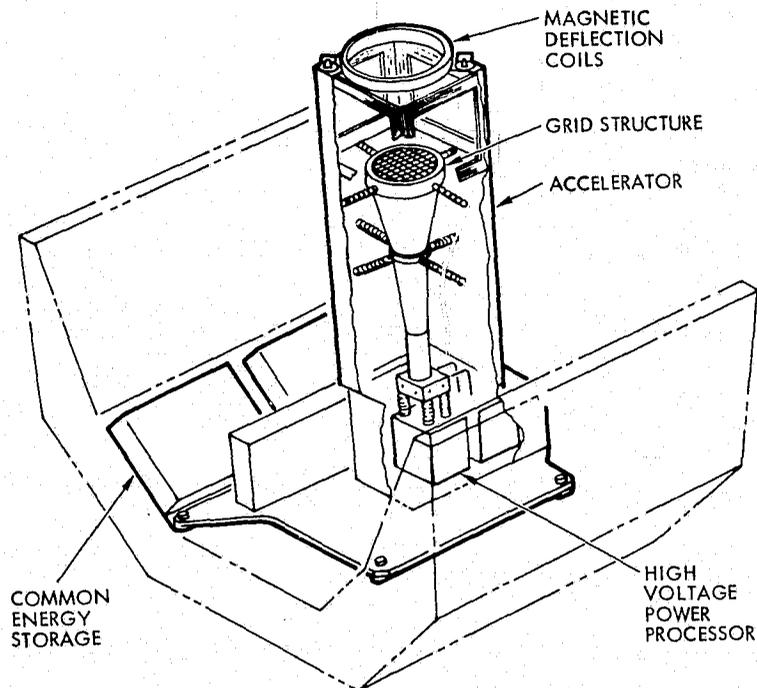


Figure 2-7. Electron Accelerator

The Flight 1 electron accelerator is the prototype for later, more sophisticated and powerful accelerators. The accelerator consists of an electron gun with its associated grid structure, a drift tube, and high voltage power processor, all operating at levels up to 40 kV. The optional magnetic deflection coil system and final grid operate at ground potential. Pulse operation of the electron accelerator required instantaneous peak powers in the multikilowatt-to-megawatt range. To provide this power, a common-energy storage system, consisting of a power processor and a 40-kj capacitor bank is mounted adjacent to the accelerator over the pallet cold plates. Commands to and data from the accelerator are processed by NIM-CAMAC (standardized digital processing equipment) in the Spacelab module. The conceptual design of the accelerator incorporates modularity on several levels to allow for evolutionary growth of capability. During later flights an ion gun and an MPD arc gun will be added while evolving to a very large output energy capacity.

The electron beam characteristics are measured in two ways. First, nitrogen gas fluorescence is detected by the Optical Band Imager and Photometer System (OBIPS) to determine beam shape, intensity profile, and directional characteristics. Second, an electrostatic analyzer is swung rapidly across the beam to measure electron energy and beam resolution.

The electron accelerator is fired first at low potentials and especially at low currents. Beam properties are measured and vehicle neutralization is assured after each pulse. The current and voltage are gradually increased to the maximum of 40 KeV and 2 amperes, assuming no neutralization problems are encountered. If neutralization at high currents cannot be achieved, a greater return current-collecting area will be required for later flights.

After the initial beam property experiments have been performed, the study of beam-plasma interactions starts. Instabilities in this interaction may develop, leading to the emission of electromagnetic radiation. To measure this radiation, high frequency electric and magnetic field detectors are carried in the Environmental Sensor Package (ESP). This package, attached to the tip of the Remote Manipulator System (RMS), is swept throughout near-Orbiter space to perform these measurements.

2.5.1.2 Lidar

The ultimate scientific requirements for the lidar system are so severe that it will be necessary to develop the system progressively over several flights; see Figure 2-8. The lidar consists of two dye lasers, data processor, and a Cassegrain telescope modularly mounted in a unitized structure and hard-mounted directly to the pallet. Each dye laser can exchange dyes using a separate dye reservoir. The data processing system processes incoming backscatter signals and produces the required scientific data for transmission to the module or to the ground. In Flight 1, development emphasis will be placed on preparing a telescope with a 1-square meter collecting area and a field of view with a cone angle of less than one milliradian. The optical finish of the surface is less critical than for astronomical use because the system measures total intensity, not image intensity. This same telescope can be used throughout subsequent flights without major modification.

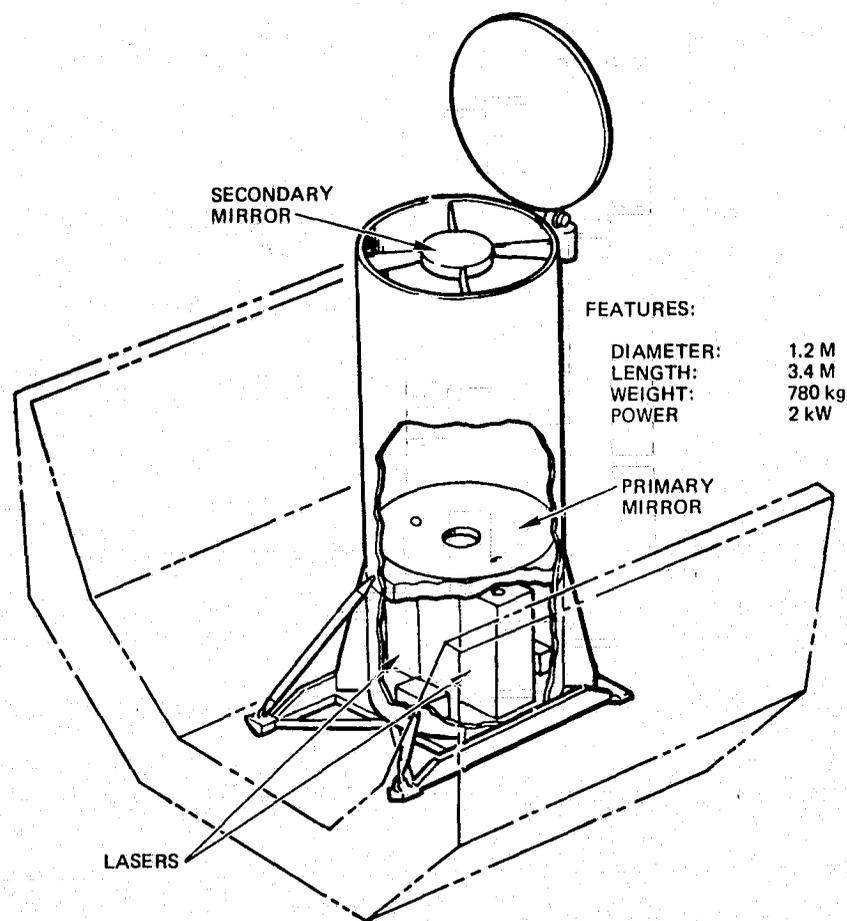


Figure 2-8. The AMPS Lidar

The lidar makes differential absorption measurements of atmospheric ozone using two frequency-doubled dye lasers, each operating in the UV near 300 nanometers. Range-resolved measurements are made in the altitude range from 10 to 50 km with sufficient density and altitude resolution to obtain maps of atmospheric ozone. Since this is a survey experiment, lidar operation is almost automatic. The payload specialist tunes the laser to the proper lines, and checks data output periodically. During this flight, more than 100,000 ozone density profiles will be obtained to chart a worldwide map of the earth's ozone density in the latitude range from 57°N to 57°S.

2.5.1.3 Atmospheric Array

Two infrared instruments, a radiometer and an interferometer spectrometer, perform limb scanning measurements to provide a worldwide map with better than 100-km earth surface resolution for a large number of

molecular species in the upper atmosphere, see Figure 2-9. In particular, maps will be made of the molecular concentration and temperatures of chlorine oxide, nitric oxide, and the hydroxyl radical that are involved in the production and destruction of ozone.

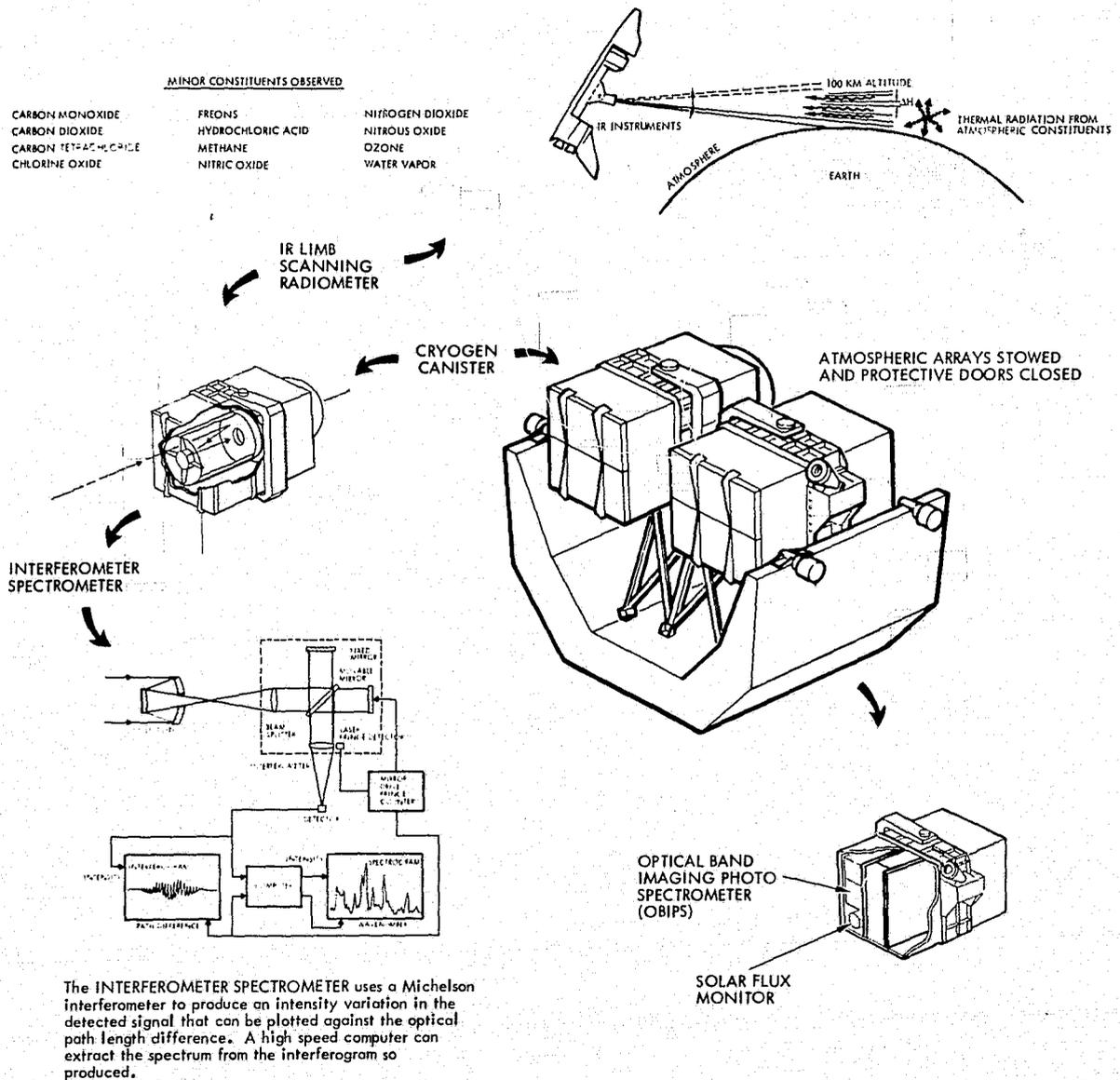


Figure 2-9. Atmospheric Array

The radiometer examines wideband regions with maximum sensitivity and low wavelength resolution. The interferometer makes continuous wavelength scans with high wavelength resolution and consequently lower sensitivity.

Each instrument measures the emission spectra of the atmosphere in the altitude region from 10 to 100 km on the limb with a height resolution on the order of 1 km.

The interferometer, with its wavelength resolution of 0.1 cm^{-1} , requires data handling rates of about 5 Mbps. Realtime fast Fourier transformations are performed with a special instrument-dedicated micro-processor capable of performing a complete Fourier transform in 0.1 second. Thus, the spectrum will be available as "quick-look" data for comparison with the spectral line data obtained with the radiometer.

2.5.1.4 Solar Flux Monitor

A solar flux monitor will be flown on all AMPS flights to provide a short time period calibration of solar instruments being flown on other spacecraft programs. Although not part of the atmospheric array, it is mounted in the same canister to be pointed by SIPS; see Figure 2-9. The typical instrument is an Ebert-Fastie four-channel spectrometer, sensitive in the wavelength range from 30 to 350 nm. It takes data for approximately 30 minutes each day as close to local noon as possible. With a field of view of 1 degree, an instrument pointing accuracy of 0.25 degree is required.

2.5.1.5 Environmental Sensor Package (ESP) Instruments

On Flight 1 a set of instruments will be mounted in the ESP for environmental measurements around the Orbiter and also to support the beam plasma, spacecraft neutralization and beam characteristics experiments. The ESP will be maneuvered around the Orbiter using the remote manipulator system (RMS). The instruments mounted in the ESP for this flight are:

- 1) Medium energy electron detector
- 2) DC electric field meter
- 3) E and B receivers and antennas (ac electric and magnetic field detectors)
- 4) Langmuir probe
- 5) Fluxgate magnetometer (dc magnetometer)

- 6) Neutral mass spectrometer
- 7) Ion mass and distribution analyzer.

2.5.1.6 Vector Magnetometer

A vector magnetometer will be mounted on a 5-meter mast to be located on the aft pallet during Flight 1 in support of the accelerator experiments. This 5-meter displacement of the instrument from the Orbiter is required to minimize the effects of Orbiter magnetic fields.

2.5.2 Flight 2 Experiment and Instrument Complements

The following experiments are planned for AMPS Flight 2.

<u>Discipline</u>	<u>Experiments</u>
Plasma Interaction and Flow	Conductivity Modification Wake Studies
Wave Phenomena	RF Sounding Techniques Long Delay Echoes
Atmospheric Science	Minor Constituents D-region Composition Changes Oxygen Variation Neutral Composition Ozone Changes Solar Radiation

The Flight 2 experiment-instrument complement is shown on Figure 2-10. It should be noted that there is a natural evolution of instrument accommodation from Flight 1 to Flight 2. The electron accelerator is not carried on this flight but will be used extensively during later flights.

2.5.2.1 Radio Frequency Sounder

The study of the propagation properties of high intensity electromagnetic waves in the ionosphere will begin on Flight 2 with the introduction of the RF sounder.

For this flight, two classes of wave experiments will be performed. The first class consists of investigations that use the natural ionospheric plasma as a plasma physics laboratory with a collision-free,

INSTRUMENTS	EXPERIMENTS									
	LOW ALTITUDE CONDUCTIVITY MODIFICATION	PLASMA FLOW	HIGH FREQUENCY WAVE INTERACTIONS	LONG DELAY ECHOES	MINOR CONSTITUENTS	D-REGION CHANGES	OXYGEN COMPOSITION	NEUTRAL VARIATION NEAR AURORA	OZONE COMPOSITION HIGH LATITUDE	SOLAR FLUX CHANGES AT CALIBRATION
ACTIVE INSTRUMENTS										
RADIO FREQUENCY SOUNDER			•	•						
CHEMICAL GAS RELEASE	•									
TEST BODY		•								
SUPPORTING INSTRUMENTS										
MAGNETOMETER	•	•	•	•						
LANGMUIR PROBE	•	•	•							
NEUTRAL MASS SPECTROMETER		•	•							
ION MASS AND DISTRIBUTION	•	•	•							
ATMOSPHERIC ARRAY										
LIDAR				•	•		•	•		
OPTICAL BAND IMAGER	•	•					•	•		
UV-VIS SPECTROMETER				•	•	•	•	•		
CRYO LIMB SCANNER				•			•	•		
CRYO IR INTERFEROMETER				•	•		•	•		
ENVIRONMENTAL SENSOR PACKAGE	•		•	•		•	•	•		
SOLAR FLUX MONITOR										•

Figure 2-10. Flight 2 Experiment Instrument Complement

boundary-free, homogeneous plasma on a very large scale. Wave dispersion, wave particle and wave-wave interactions, plasma turbulence, and plasma wave instabilities will be studied using the RF sounder operating at continuous powers up to 100 watts and peak powers up to 10 kW. In particular, wave propagation at frequencies from 30 kHz to 20 MHz will be studied as a function of magnetic field orientation, ionospheric density, and frequency. An attempt will be made to saturate the transmission properties of the ionosphere so that plasma instabilities are generated. To generate these instabilities, it may be necessary to operate the transmitter up to its full power capability. The wave transmission properties require that the transmitting antenna be mounted at least 15 meters from the Shuttle conducting surfaces. Since the plasma waves generated by the interactions are of very low intensity, the RF receiver will be remotely deployed on

an ESP subsatellite to remove it from the high EMI Shuttle environment during all phases of the plasma wave experiments.

The second class consists of investigations using the RF transmitter in a search for, and detection of, long delayed echoes. In this experiment, sounding echoes that are delayed for 10 seconds or more are deflected from the ionosphere and the echoes will be detected with the receiver, on the ESP.

The RF transmitter used for these investigations is capable of both pulsed and continuous operation, with a wide range of programmable pulse characteristics (width, shape, repetition frequency). The antenna length is variable. The receiver output, with controls for frequency, bandwidth, gain and sweep characteristics, will be processed and displayed using harmonic analysis filtering and conventional ionogram display.

2.5.2.2 Plasma Wake Measurements

During Flight 1, extensive measurements of the plasma environment of the Shuttle will be made. In particular, the wake of the Shuttle will be mapped in the near-Shuttle region. In Flight 2, the first attempts will be made to create and measure the wake from a symmetrical conducting test body which can be biased to any electric potential relative to the local plasma. The test body is a 10-meter diameter spherical balloon made of aluminized mylar or other conducting material. It will be mounted on a mast so maneuvering of the Orbiter and the group of diagnostic instruments will make it possible to map the properties of its wake at distances from a fraction of the radius up to several radii from the surface of the test body, at nearly all azimuthal angles from the plasma flow velocity vector. This will produce a three-dimensional map of the electron and ion densities and temperatures in the wake region.

2.5.2.3 Chemical Release

The Flight 2 chemical release experiment will investigate the effect of gross changes in ionospheric conductivity by injecting a large quantity of ionizable material in the auroral zone. The increase in conductivity will modify the natural ionospheric and magnetospheric currents. This

leads to enhanced field-aligned and perpendicular currents, enhanced particle precipitation, and the possible triggering of an aurora or sub-storm. Observations of the effects will be made by a suitable ground station (such as Ft. Churchill), by Spacelab-mounted diagnostic sensors, and by remotely located sensors.

A 300 kg barium release satisfies the experiment requirement; see Figure 2-11. This quantity is ejected from the Orbiter as 20 thermite canisters of 16 kg each. The ignition occurs at about 180 km altitude and should take place across the magnetic field lines for a distance of about 20 km. Following ignition, the barium, as a hot gas, expands and is ionized either by solar radiation or by interaction with the stationary atmosphere.

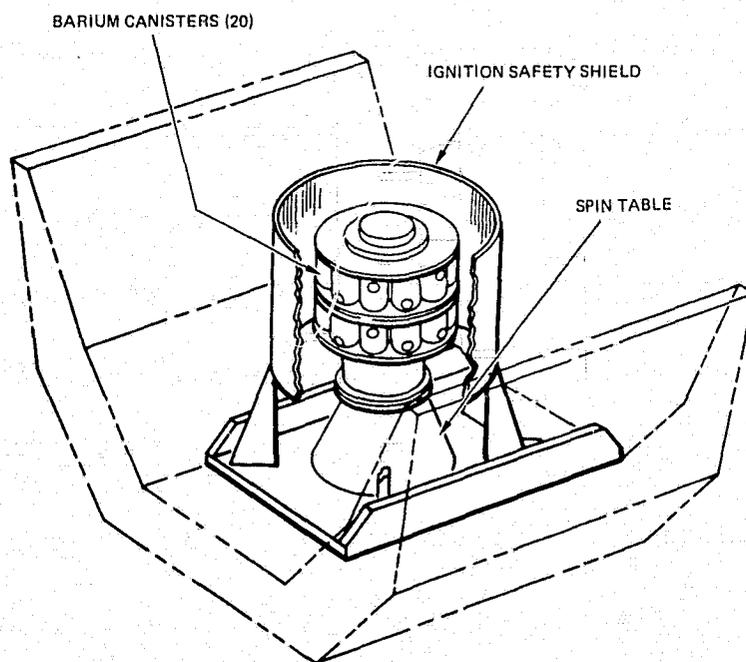


Figure 2-11. Barium-Thermite Release

Measurements of electric fields, particle densities and energies, and possibly OBIPS observations of auroral effects are made from the Shuttle. Particle and field measurements are made from deployed ESPs.

The results from the Flight 1 gas expansion experiment allows an assessment of the probability of ionization of the barium by high-velocity collision with the neutral atmosphere. If such ionization is practical,

the Flight 2 release can take place at any time of day or night; otherwise, it is necessary to use solar photoionization of the barium and the time of the release is restricted to daytime or twilight.

2.5.2.4 Atmospheric Measurements

Flight 1 tests the lidar system and performs limb scanning measurements of atmospheric constituents in the infrared. Flight 2 uses a lidar system with improved temporal resolution and wavelength range. It can make more detailed soundings at special geographic locations that have been discovered during Flight 1. The increased wavelength range of the tunable lasers allows the measurement of both OH and NO as well as ozone by the differential absorption technique. The infrared red scanning instruments are the same as they were on Flight 1, but ultraviolet and visible spectrophotometers are added to the array to extend the number of species that can be observed; see Figure 2-12. This increased capability allows more detailed observation of the processes that control the ozone density in the atmosphere.

In addition to the lidar and limb scanning surveys, Flight 2 will be prepared to make observations of certain intermittent natural phenomena that contribute to the ozone balance of the atmosphere. The experiments that are intended to observe these intermittent events (targets of opportunity) may or may not be done on any given flight depending on whether the natural phenomenon occurs during the course of that flight. An example of this type of experiment is the study of high latitude changes in ozone density caused by the bombardment of the polar cap atmosphere by energetic solar protons during a severe proton event. These solar proton events are relatively rare, usually less than one per month, but during the event, it is thought that very large amounts of ozone are destroyed in the polar cap regions. Another target of opportunity investigation is measurement of changes in the ambient density of major atmospheric constituents caused by high latitude energetic particle bombardment.

Target of opportunity observations will be made with an array of bore-sighted instruments that can be pointed independently of the limb scanning IR instruments. These bore sighting instruments, consisting of the OBIPS and ultraviolet/visible spectrophotometers, provide an observing package covering the wavelength range from 110 to 1000 nanometers. In

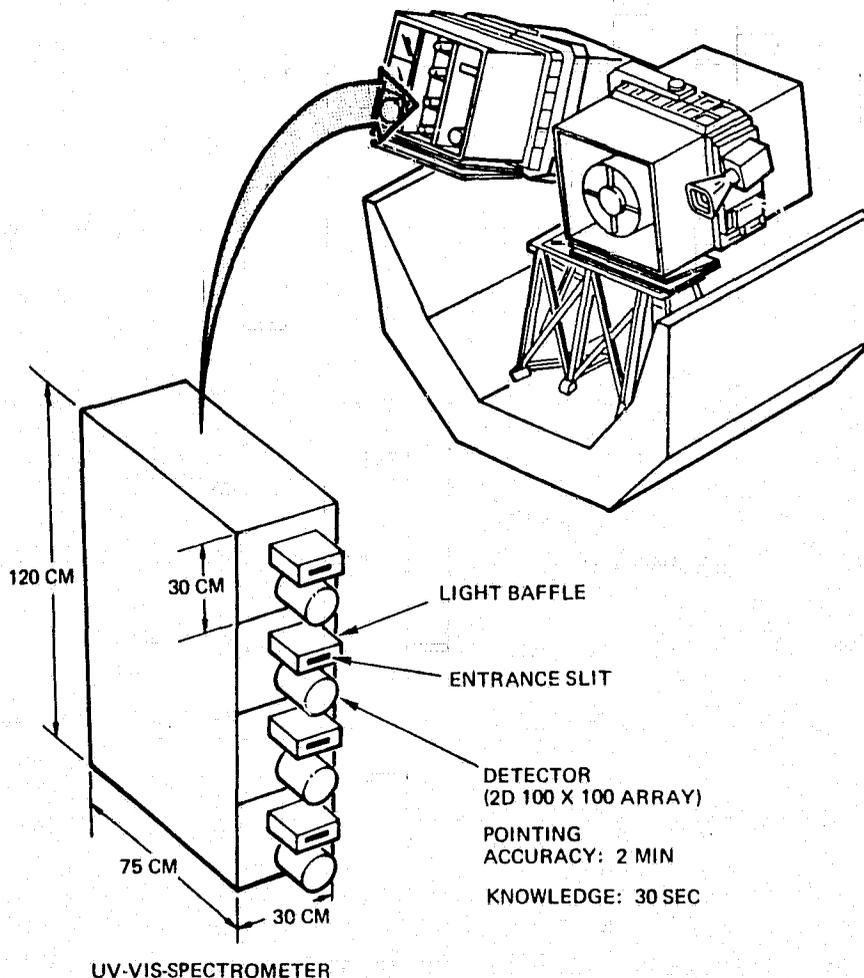


Figure 2-12. SIPS for Flight 2

the target of opportunity mode, the instrument package must be able to track a ground target within an accuracy of ± 0.25 degree.

2.5.2.5 Flight 2 Environmental Sensor Packages

During Flight 2 there are two ESP's. One is a re-flight of the Flight 1 ESP. The second ESP is ejected from the Orbiter and employed as a support for the RF Sounder, Long Delayed Echo, Low Altitude Conductivity, and HF/Wave Particle Interaction experiments.

2.5.3 Flights 3, 4, and 5 Experiment and Instrument Complements

The experiments to be conducted during these flights represent an extension and growth from the set of experiments conducted during Flights 1 and 2. Extensive use is made of the core instruments that were developed earlier. Many require an evolutionary growth in capability

in order to perform the later experiments. In addition, some new instruments and flight support equipment are required. In Table 2-1 the experiments that are indicated for these flights are summarized along with the devices, instruments, and equipment required to perform them.

The evolution of payload and equipment during these flights is demonstrated on Table 2-2. Several new items of equipment, namely, barium-shaped charges, the maneuvering subsatellite, the tethered balloon current generator, and the lidar mounted on a pointing platform could be required during the later phases of the AMPS Program.

Table 2-1. Flights 3, 4, 5 Experiments

Flight	Experiment Objectives	Additional Instruments
A. Atmospheric Experiments		
3	Same as for Flight 2 plus: Detailed temperature profiles Mesospheric winds Study of atmospheric dynamics	Fabry-Perot interferometer Near IR interferometer
4	Same as for Flight 3 plus: Thermospheric composition by solar occultation Studies of auroral EUV emissions Meteoritic material composition OH vibrational distribution	UV occultation spectrograph
5	Same as Flight 4 plus: Chemical and dynamical studies Using gas releases	Photoionization device
B. Plasma Wave Experiments		
3	VLF transmitter studies Initial VLF wave propagation Particle beam VLF generation	3.5-kW VLF transmitter 300-m antenna ESP receiver Electron accelerator
4	Quadrupole probe studies Group delay Natural VLF wave spectra VLF-RF wave interaction Pulse transmission	VLF transmitter RF transmitter Quad probes Subsatellite diagnostics
5	VLF precipitation of particles Particle beam VLF generation General VLF studies	VLF transmitter Electron accelerator Ion accelerator Subsatellite diagnostics

Table 2-1. Flights 3, 4, 5 Experiments (Continued)

Flight	Experiment Objectives	Additional instruments
C. Release Experiments		
3	Atmospheric and electron chemistry High frequency propagation Transport properties, airglow Density diffusion Radio astronomy lens	SF ₆ canister 300 to 500 kg
4	Trace magnetic field lines	BA shaped charge at 500 km
5	Trace magnetic field lines Plasma-neutral gas interaction (critical velocity)	BA shaped charge at 500 km BA Thermite - 50 kg
D. Accelerator Experiments		
3	Preliminary atmospheric interactions Magnetic field line tracking Beam excitation of plasma waves E// B studies	Electron accelerator 50 kW, 10 A, 50 kV, 60 kJ Beam bending, pulse modulation capability
4	Atmospheric interaction studies Electron echoes Plasma wave excitation MPD ARC studies	Electron accelerator MRD ARC 5 mW, 10 ⁴ A, 500 V, 120 kJ
5	Two-beam generation of VLF waves Interaction of dense hot plasma with ionosphere Ion interaction with atmosphere	Electron accelerator MPD ARC 50 mW, 10 ⁵ A, 500 V, 120 kJ Ion accelerator 20 kW, 2 A, 10 kV

Table 2-2. Flights 3, 4, 5 Payload Inventories

	Flight 3	Flight 4	Flight 5
Atmospherics	SIPS with atmospheric instrument complement 10-joule lidar	SIPS with atmospheric instrument complement 10-joule lidar on pointing platform*	SIPS with atmospheric instrument complement 10-joule lidar on pointing platform
Diagnostic Masts	5-meter magnetometer mast 15-meter mast with diagnostics	5-meter magnetometer mast 15-meter mast with diagnostics and RF sounder transmitter antenna (100-m)	5-meter magnetometer mast 15-meter mast with diagnostics
Deployable Subsatellites	One ESP	One ESP Maneuvering subsatellite* Wake mapper*	Reflector subsatellite* Maneuvering subsatellite Tethered balloon current generator*
Accelerators	60-kilojoule electron accelerator*	120-kilojoule electron, and MPD arc accelerators*	120-kilojoule electron, ion, and MPD arc accelerators*
Gas and Chemical	300-500 kg SF ₆ release*	Barium shaped-charge release*	Barium shaped-charge release Gas canister release
Special			Gas photoexcitation and photoionization device*
VLF transmitter Transmitter	Structure hard-mounted to pallet - 300-m dipole*	Same	Same
Common Energy Storage	600 kilojoules, 500 volts, 5 farads	120 kilojoules, 500 volts, 10 farads	120 kilojoules, 500 volts, 10 farads

*Significant development required

3. PAYLOAD DEFINITION AND PRELIMINARY DESIGN

3.1 PAYLOAD SYSTEMS

This section is a system overview of the AMPS Flight 1 and 2 payloads. The configurations and overall payload performance are discussed first, followed by discussions of interfaces with Spacelab and Orbiter. Data acquisition and handling, controls and displays, and software are covered in Sections 3.1.4, 3.1.5, and 3.1.6.

3.1.1 Flight 1 Payload Configuration and Performance

The set of core instruments for this payload were described briefly in Section 2.5.1. This section describes the accommodation of these instruments. This accommodation is achieved using several items of support equipment which must also be developed. Some of the major items are:

- 1) Small instrument pointing system (SIPS)
- 2) Environmental sensor package (ESP)
- 3) Five-meter mast
- 4) Gas release device.

When instruments are integrated into these items of support equipment, the resulting assembly can be integrated as a unit with Spacelab. The term "functional assembly" is used in this document to describe this integrated instrument and support equipment. In addition, there are other items of equipment (such as controls and displays, integration hardware, and cables) that will be discussed in other sections of this document.

3.1.1.1 Atmospheric Array

The IR interferometer, IR radiometer, OBIPS, and solar flux monitor require precision pointing. To satisfy these requirements and minimize the number of pointing platforms, these instruments are mounted on the SIPS; see Figure 3-1. The IR interferometer and IR radiometer require cryogenic cooling; they are housed in one specially designed canister on the port yoke. The OBIPS and solar flux monitor are mounted in the starboard side canister. A more detailed description of the SIPS pointing and control is given in Section 3.2.1.

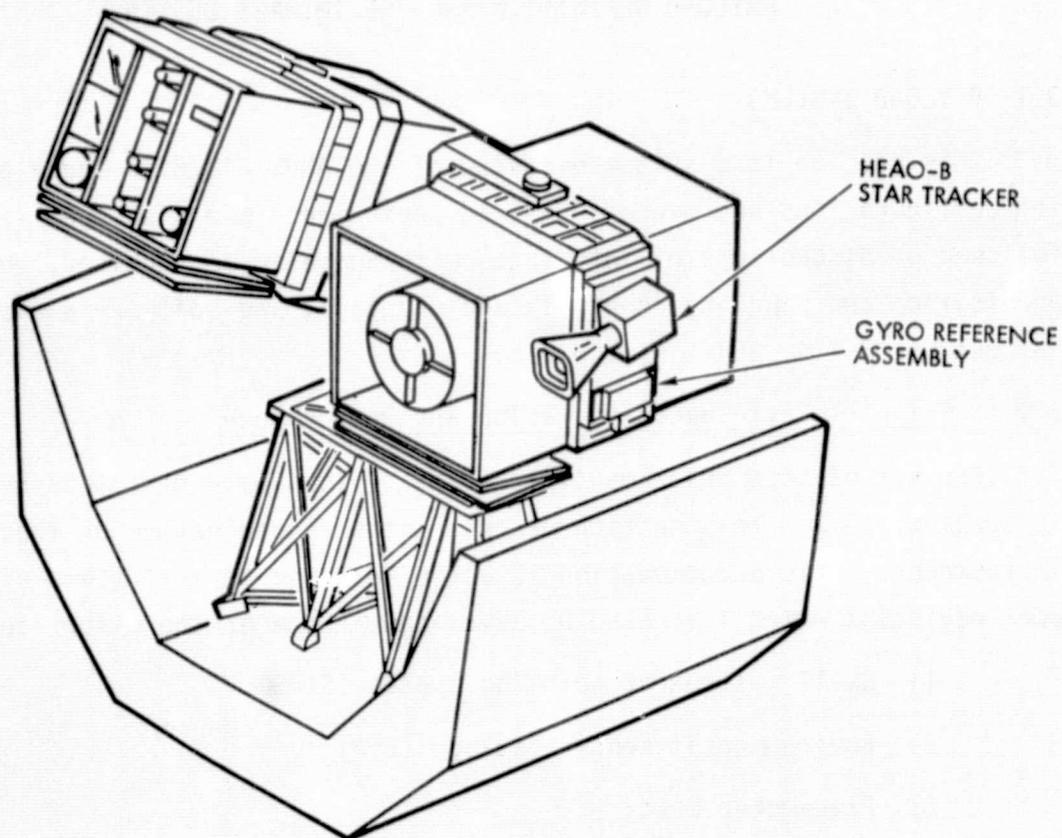


Figure 3-1. Small Instrument Pointing System

3.1.1.2 Environmental Sensor Package (ESP)

The ESP is a versatile, low-cost sensing package designed for multiple applications, including environmental sensing near the Orbiter and remote measurements related to several AMPS experiments. One ESP, which is not released, is used during the first AMPS flight to measure the EMI environment near the Orbiter. The Orbiter Remote Manipulator System (RMS) gives the ESP its support and maneuvering capability. The ESP is described further in Section 3.2.2.

3.1.1.3 Five Meter Mast

The vector magnetometer is mounted at the tip of an extendable/retractable mast on the aft pallet, which carries the device beyond the Orbiter magnetic field. The magnetometer is calibrated shortly after orbital operations are started to compensate for steady-state magnetic

fields. An alignment measuring device compensates for thermal movement of the mast. This piece of Labcraft equipment is designed for use on all five AMPS missions. In addition to the vector magnetometer, the mast also accommodates a quartz crystal micro-balance for particulate contamination measurements. The design of the five-meter mast is discussed further in Section 3.2.3.

3.1.1.4 Gas Release Device

Five gas releases are planned for the first flight. Xenon gas is used; five canisters are ejected into space by small spring mechanisms. The five canisters are mounted on the port and starboard sills of the aft pallet. The xenon is released by timers 1-1/2 to 3 hours after ejection from the Orbiter. These devices are described in Section 3.2.5.

3.1.1.5 Flight 1 Payload Layout and Performance

The design of the Flight 1 payload was driven by many considerations. Among these are the evolutionary buildup of instruments and AMPS equipment; safety; cost-effective utilization of Orbiter and Spacelab resources; maximum usage of multiple mission support equipment and commercial hardware; and application of innovative low cost design approaches. The equipment employed on the Flight 1 payload has been described in previous sections and is shown in Figures 3-2 and 3-3, which depict the major items of Labcraft equipment and instruments in the Spacelab short module and three-pallet configuration. The payload is designed for the management of experiments from the Spacelab module and from the Orbiter aft flight deck. Placement of the payload on the three pallets is the result of many interactive considerations. These are summarized as follows:

- a) The SIPS atmospheric array is mounted in the center pallet to allow a maximum flexibility for pointing the SIPS canisters over a wide field during the conduct of experiments.
- b) Heavier equipment is located toward the aft of the payload bay. The lidar which is the heaviest single item of the equipment is located on the aft pallet. The accelerator which is lighter than the lidar is located on the front pallet.

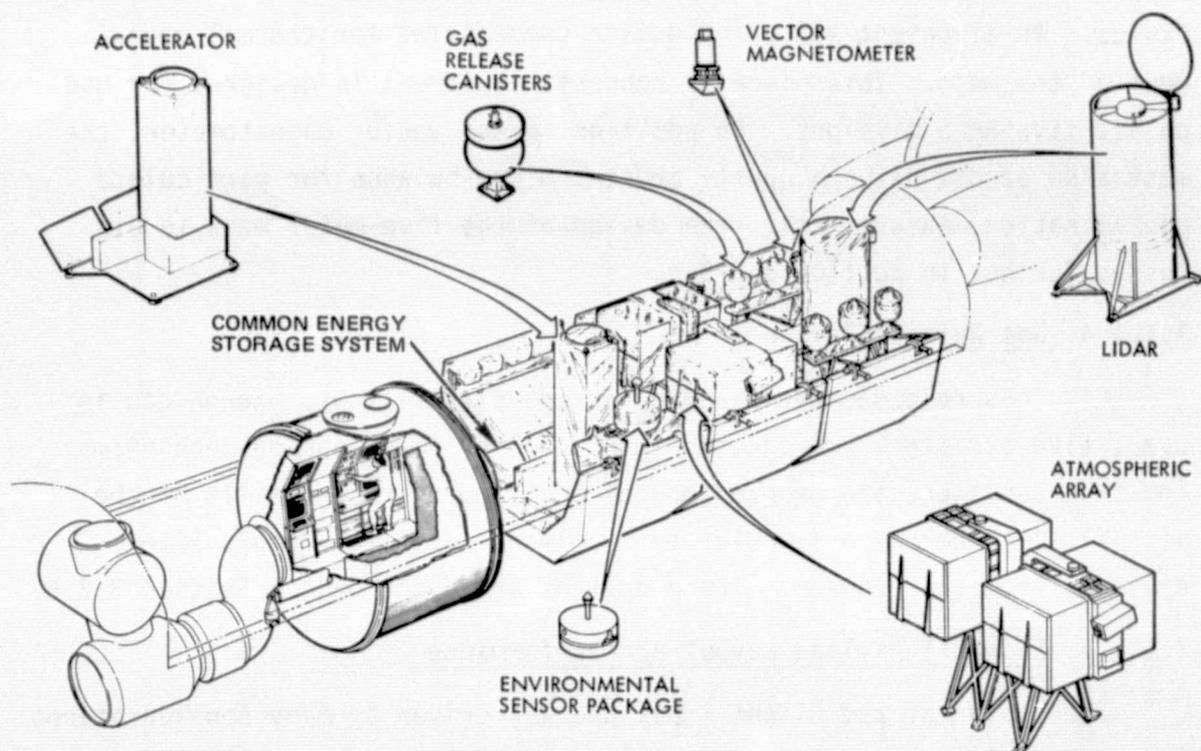
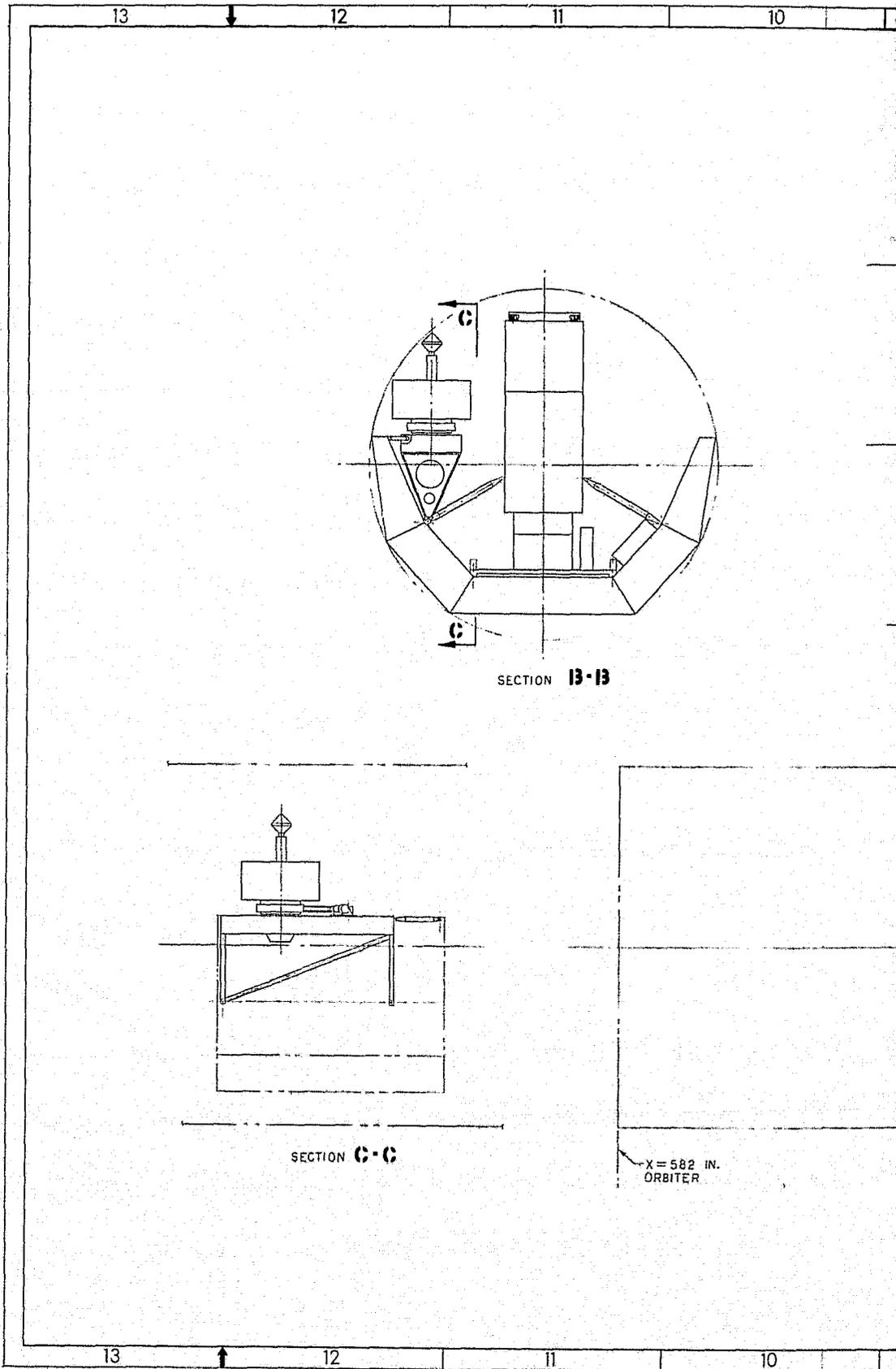
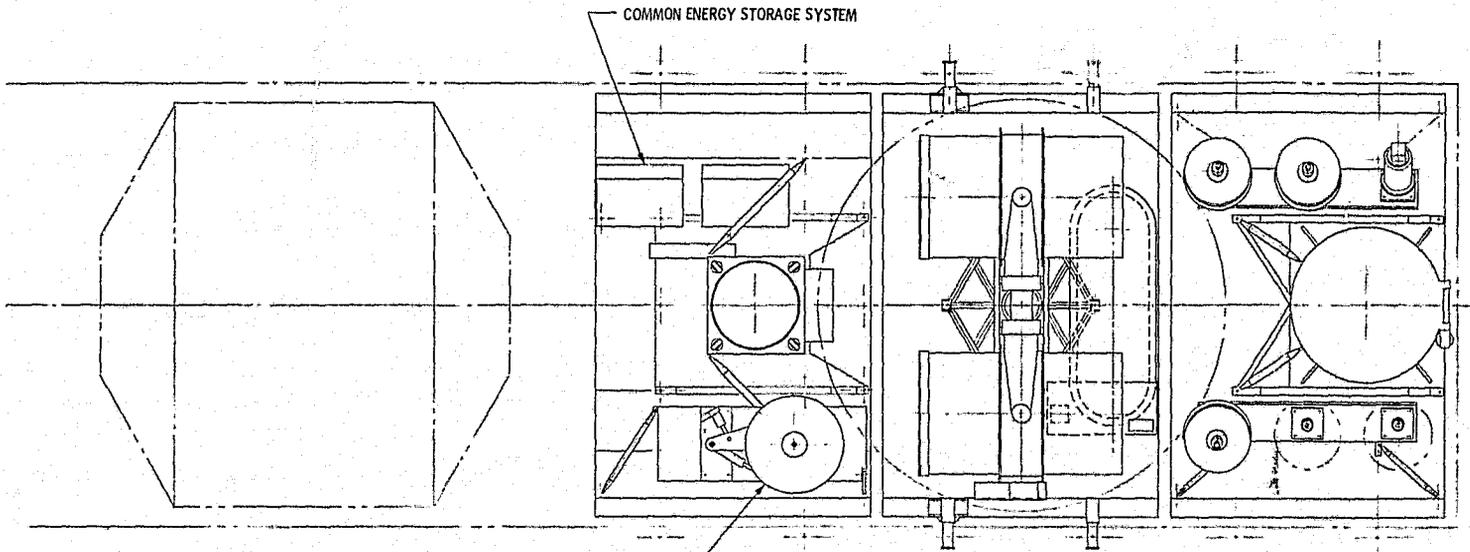
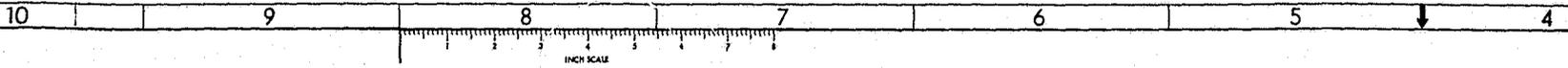


Figure 3-2. TRW AMPS Flight 1 Payload Configuration

- c) Large items of equipment are located along the bottom surface of each pallet. Smaller items such as the gas releases, the magnetometer mast, and the environmental sensor package are located along the sills of the pallets.
- d) Certain items of equipment require a direct contact with cooling plates. These, for example, include the common energy storage system shown on the first pallet and mounted along the pallet cold plate.
- e) The environmental sensor package is designed to be maneuvered using the Remote Manipulator System (RMS). The ESP is located on the left side of the first pallet for easy access by the RMS.
- f) The vector magnetometer is used in conjunction with accelerator operations yet it must not be so close as to be influenced by the accelerator. It is, therefore, mounted on the aft corner of the third pallet, a maximum distance from the accelerator. In its extended position, the magnetometer mast is approximately 5 meters from the Orbiter minimizing magnetic field disturbances by the Orbiter.
- g) The gas releases are along the sills of the aft pallet for easy deployment during flight.



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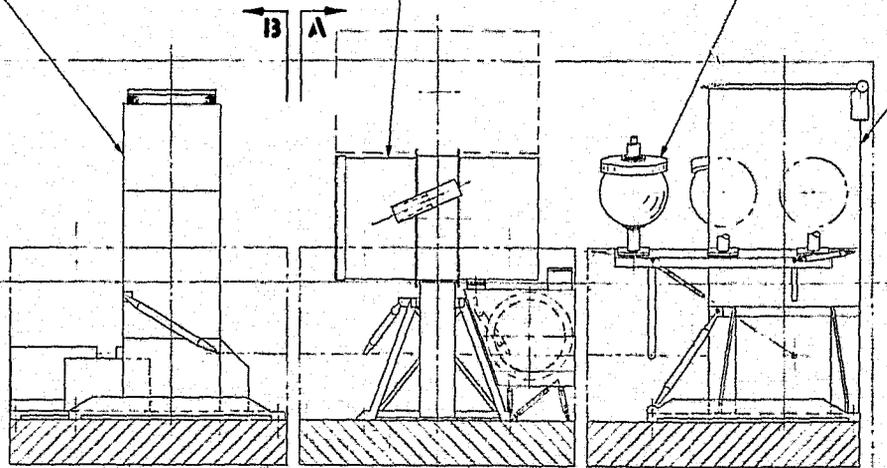
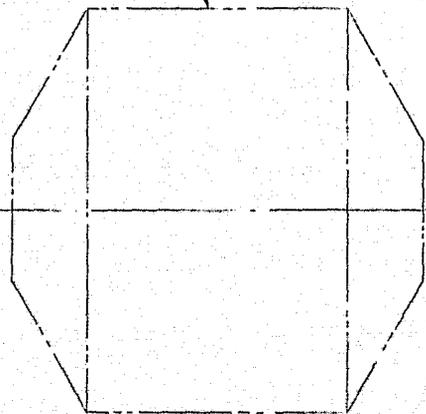
SHORT MODULE SPACELAB

ACCELERATOR SYSTEM

ENVIRONMENTAL SENSING PACKAGE

SIPS POINTING CONTROL SYSTEM WITH
ATMOSPHERIC INSTRUMENT CLUSTER

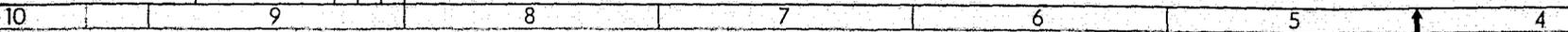
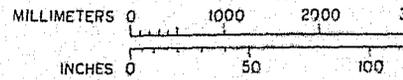
GAS RELEASE SYSTEM (5)



MAXIMUM PAYLOAD ENVELOPE

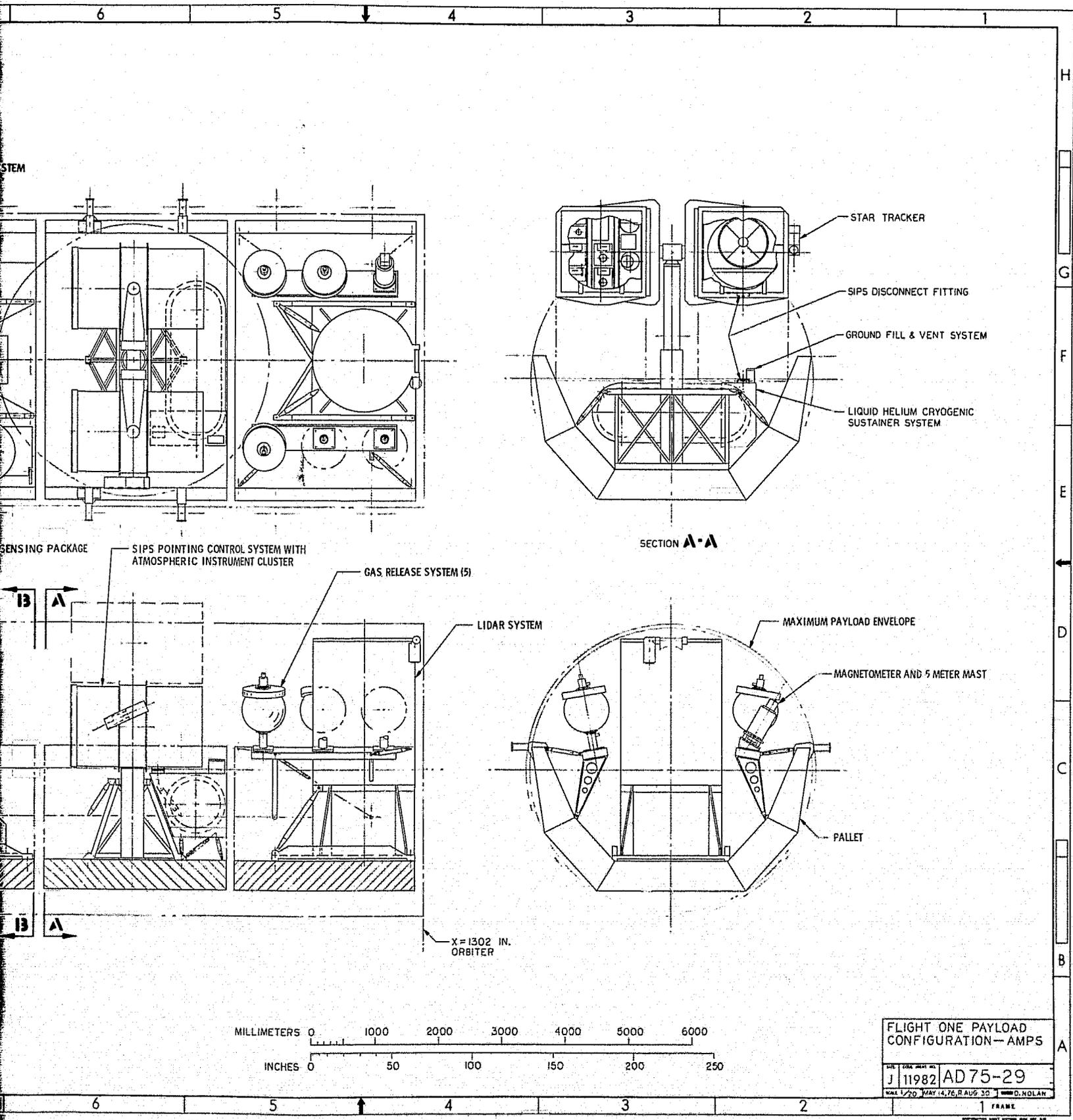
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Figure 3-3. Flight 1 Layout

Analysis of the payload performance during the Flight 1 has been performed and the results summarized in Table 3-1. In all cases the payload can be operated within allowable margins. However, in certain cases the margins are less than desirable at this stage of development of the payload. However, flight operations can be adjusted to increase some margins. Table 3-2 gives the payload mass summary for Flight 1 payload. The margin of 29 percent at this point in the development is adequate to allow for future growth. The mass properties of the payload are discussed further in Section 9.1.

Table 3-1. Resource Requirements for AMPS Flight 1

	UTILIZED	ALLOWABLE	PERCENT MARGIN
1) PAYLOAD MASS (LANDING) (KG)	4388	6222	29
2) CENTER OF GRAVITY	(SPECIFIED ENVELOPES)		
3) RCS PROPELLANT (KG)	1829	2851	36
4) OMS PROPELLANT (KG)	9814	10530	7
5) ELECTRICAL ENERGY (KW-HR)	194	369	47
6) PEAK ELECTRICAL POWER (KW)	3.2	7.4	57
7) MAXIMUM SUSTAINED POWER (KW)	3.0	3.4	12
8) HEAT REJECTION (KW)	3.5	5.3	66
9) MAXIMUM COMPUTER LOAD	62.5K	65K	3.8
10) MAXIMUM COMPUTER-OPERATIONS	<100K	320K	>69
11) MAXIMUM DATA RATE (MBPS)	2.6	50	95

3.1.2 Flight 2 Payload Configuration and Performance

The Flight 2 payload uses some of the instruments and flight support equipment that were developed for Flight 1. The new instruments required are discussed in Section 2.5.2. The new flight support equipment, specifically, the 15-meter mast and the chemical gas release are described briefly below.

3.1.2.1 Fifteen-Meter Mast

The 15-meter mast is designed to accommodate equipment requiring extension away from the Orbiter payload bay. On Flight 2 this includes

Table 3-2 The AMPS Flight 1 Weight Statement
(Based on JSC-07700 Allowances)

	Launch Mass (kg)	Landing Mass (kg)
Pallet segment No. 1	783	779
Pallet segment No. 2	1,446	1,357
Pallet segment No. 3	1,477	937
Module	290	290
Aft flight deck	<u>80</u>	<u>80</u>
Total payload mass	4,076	3,443
Mission dependent equipment	736	736
Baseline Spacelab equipment	7,562	7,162
Payload specialist	<u>210</u>	<u>210</u>
Total payload mass	12,583	11,550
Allowable	29,484	14,515
Margin	16,799	2,965

the radio frequency sounder and the test body. The RF sounder consists of a transmitter and a 100-meter dipole antenna. The operating range is 30 kHz to 20 MHz. The power range is 100 watts to 10 kW.

The test body is a 10-meter diameter balloon made of a conducting material. It is used in conjunction with the ESP mounted to the RMS to study the plasma wake characteristics in orbit. The 15-meter mast is described further in Section 3.2.4.

3.1.2.2 Chemical Gas Release

A barium-thermite release system is used, consisting of a cradle-ejection subsystem mounted to a pallet and a spin-stabilized bus carrying up to 20, 16-kilogram canisters. The bus and ejection system are Labcraft equipment. The trajectory of the canister away from the Orbiter is controlled by orientation of the Orbiter at ejection. On receipt of a radio command, the 20 canisters are ripple-fired to disperse the barium ions. The barium-thermite release system is discussed in Section 3.2.6.

3.1.2.3 Flight 2 Payload Layout and Performance

The Flight 2 payload uses many of the items of Labcraft equipment instruments previously flown on Flight 1. In addition there are new items of equipment which have been described in the previous section. The Flight 2 payload is shown on Figures 3-4 and 3-5. The SIPS pointing and control, the lidar, the magnetometer mast, and the nondeployed ESP are located in the identical positions employed during Flight 1. In addition, the 15-meter mast and barium-thermite release system have been accommodated on the forward pallet. Finally, the deployed ESP is located on the aft pallet sill. It is of the same basic design as the nondeployed ESP located on the forward pallet. In this case, however, ESP is separated from the Orbiter and operates nearby the Orbiter for periods of up to 4 days. The ESP design will be described further in Section 3.2.2.

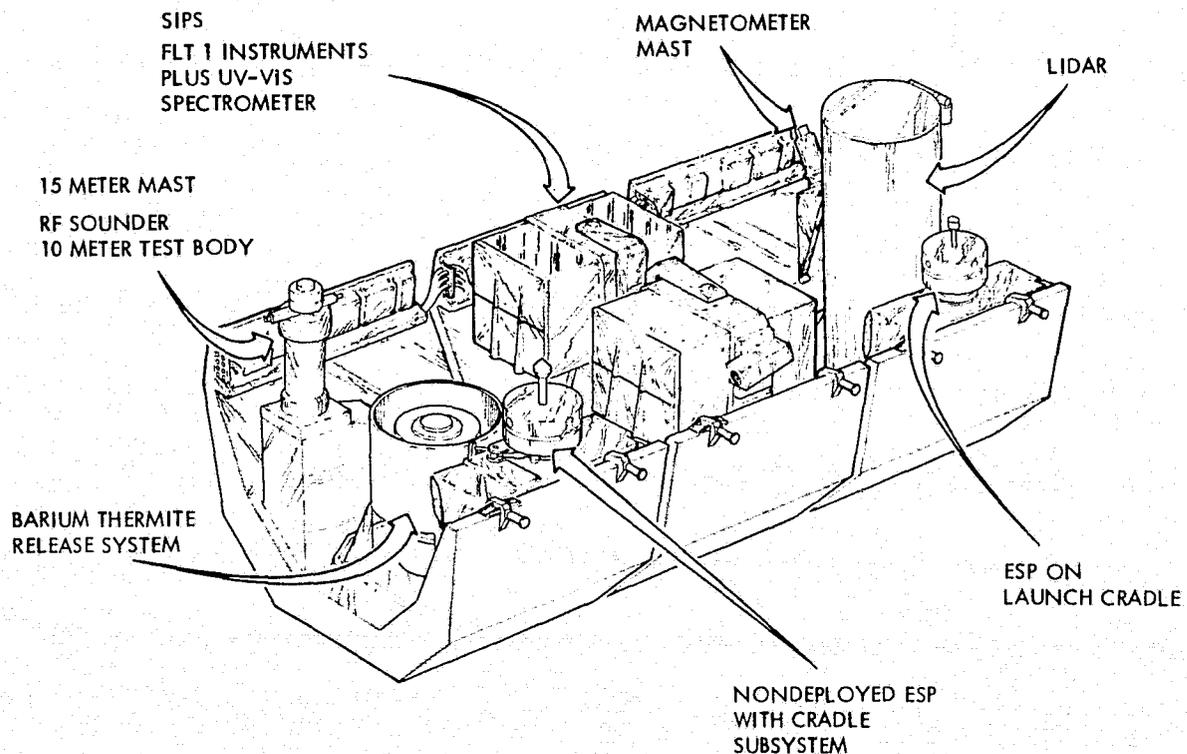


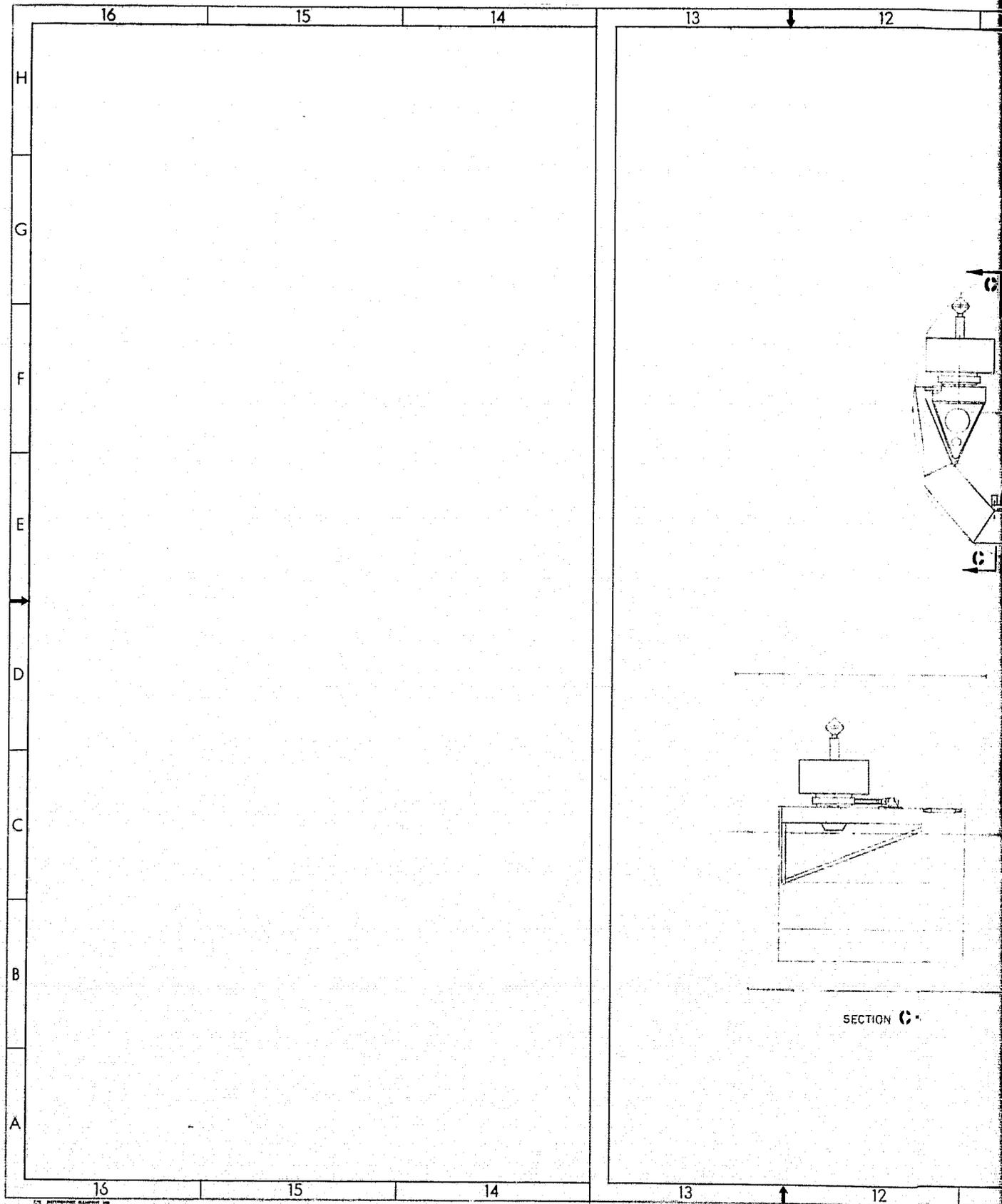
Figure 3-4. Flight 2 Payload (Less Module)

The same basic considerations dictate the layout of this payload and the Flight 1 payload. Heavier equipment is located toward the aft of the payload bay in order to maintain center of gravity limits. Note that all equipment is covered with thermal blankets to maintain temperature control. The barium-thermite release system has a flash barrier located around it to minimize the hazard in the event of an inadvertent ignition of a thermite release. The performance of this payload during flight is summarized in Table 3-3. This payload, as was the case of Flight 1, has ample margins in most cases. However, certain items (specifically maximum sustained power, maximum computer load, and maximum computer operations per second) have small margins which may be controlled to some extent by mission operations but are sources of potential design problems. Table 3-4 is the Flight 2 payload mass statement. Again the margin of 33 percent during the landing phase is adequate for this phase of the development of the payload. For a more detailed mass statement, refer to Section 9.1.

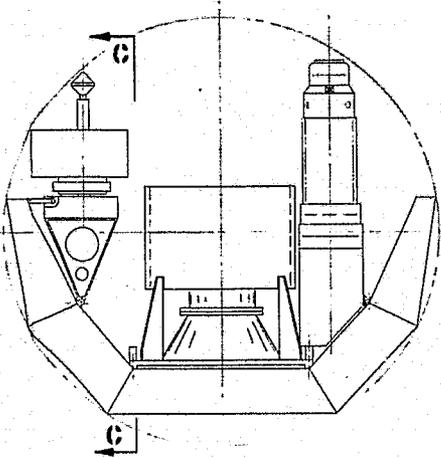
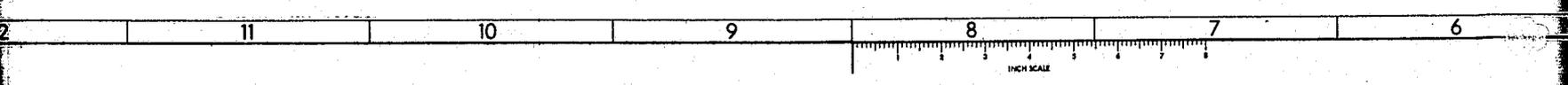
Table 3-3. Resource Requirements for AMPS Flight 2

	UTILIZED	ALLOWABLE	PERCENT MARGIN
1) PAYLOAD MASS (KG)	4166	6222	33
2) CENTER OF GRAVITY	WITHIN LIMITS		-
3) RCS PROPELLANT (KG)	2054	2851	28
4) OMS PROPELLANT (KG)	8946	10530	15
5) ELECTRICAL ENERGY (KW-HR)	261	369	29
6) PEAK ELECTRICAL POWER (KW)	3.6	7.4	51
7) MAXIMUM SUSTAINED POWER (KW)	3.2	3.4	6
8) HEAT REJECTION (KW)	2.7	5.3	49
9) MAXIMUM COMPUTER LOAD - WORDS	57K	65K	12*
10) MAXIMUM COMPUTER OPERATIONS PER SECOND	320K	320K	0
11) MAXIMUM DATA RATE (MBPS)	6.4	50	87

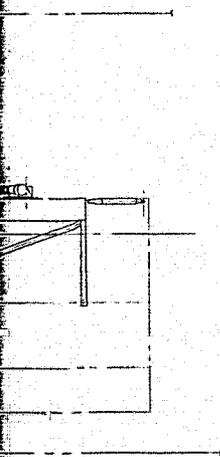
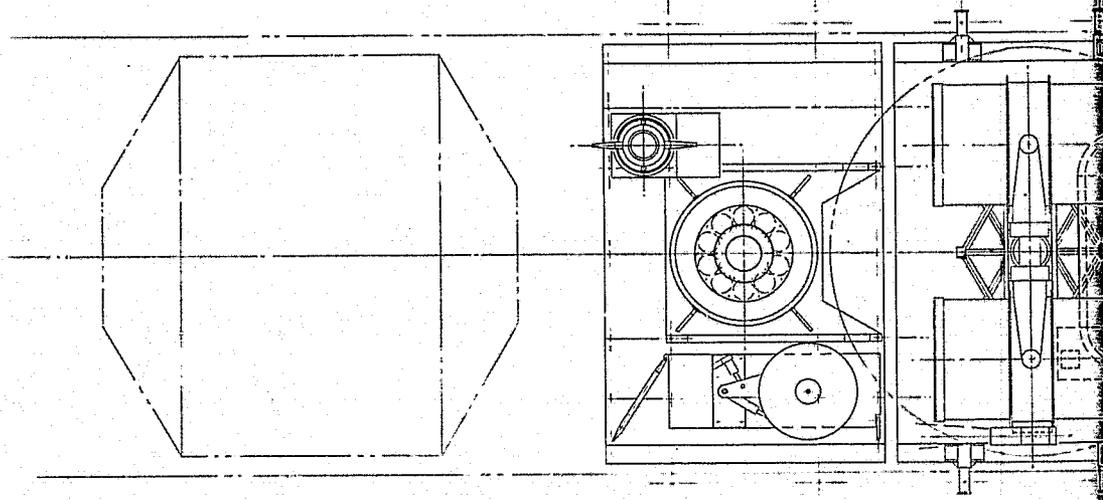
* MAXIMUM COMPUTER LOAD MAY BE REDUCED BY SECTIONING AND OVERLAYING EXPERIMENT LOAD



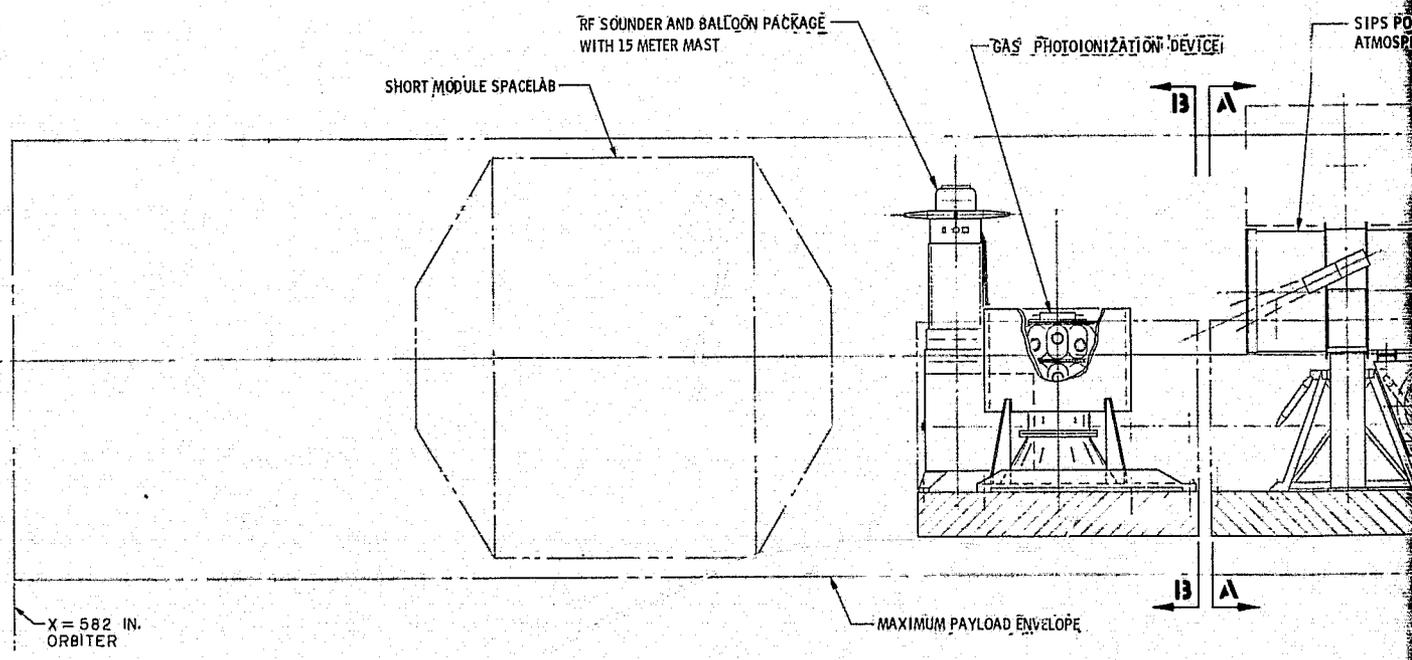
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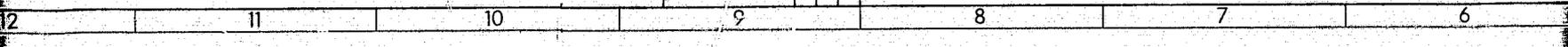
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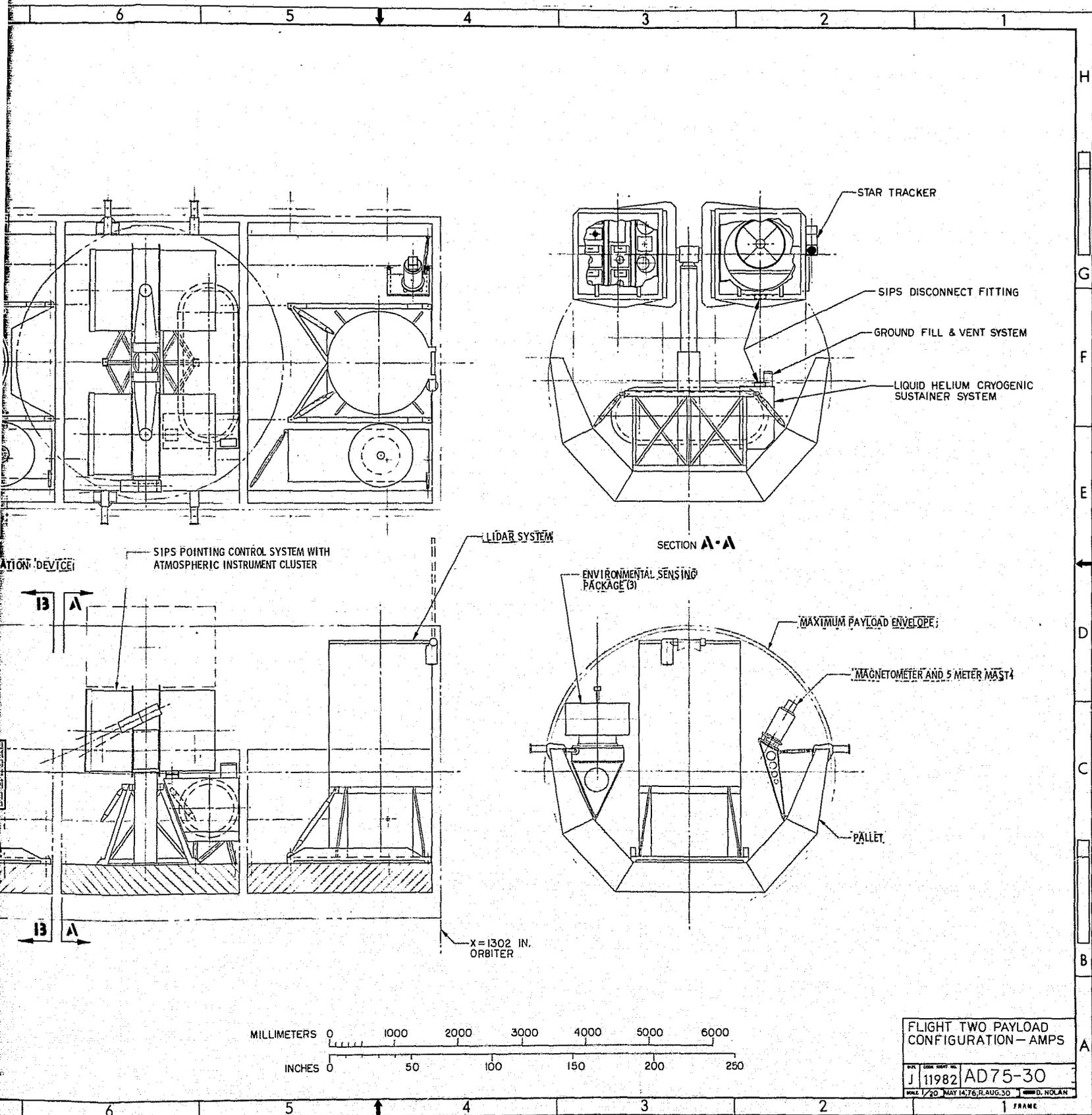


Figure 3-5. Flight 2 Layout

FOLDOUT FRAME 3

Table 3-4. The AMPS Flight 2 Weight Statement
(Based on JSC 07700 Allowances)

	Launch Mass (kg)	Landing Mass (kg)
Pallet segment No. 1	1,153	638
Pallet segment No. 2	1,511	1,422
Pallet segment No. 3	1,049	879
Module	240	240
Aft flight deck	<u>80</u>	<u>80</u>
Total payload mass	4,033	3,437
Mission dependent equipment	697	697
Baseline Spacelab equipment	7,562	7,162
Payload Specialist	<u>210</u>	<u>210</u>
Total payload mass	12,502	11,328
Allowable	29,484	14,515
Margin	17,116	3,187

3.1.3 Spacelab and Orbiter Support

The Flight 1 and Flight 2 payloads receive support from both the Spacelab and the Shuttle Orbiter. In some cases, such as Orbiter orientation/pointing, the support is provided directly by the Orbiter. In other cases, such as electrical power and thermal control, the basic Orbiter service is provided through some Spacelab subsystem. Still in other cases the Spacelab provides support capabilities directly to the payload (e.g., data processing via the CDMS computer).

3.1.3.1 Equipment Mounting

Payload equipment can be mounted in any of the following three locations.

- 1) Spacelab pressurized module
- 2) Spacelab pallet
- 3) Orbiter Aft Flight Deck (OAFD).

Each of these locations has established constraints on allowable volume, size, and weight for payload equipment. The Space Shuttle System imposes additional weight and center of gravity constraints on payloads as noted in Section 9.1 of this document.

Both the pressurized module and the pallet can accommodate a variety of mounting concepts. The payload equipment proposed for the Flight 1 and Flight 2 payloads is well within the allowable mounting and volume constraints of the Spacelab and Orbiter.

3.1.3.2 Electrical Power

All electrical power aboard the Shuttle Orbiter originates from three fuel cell powerplants located in the Orbiter. Each fuel cell is capable of providing 7 kW maximum continuous and 12 kW peak power (for a maximum duration of 15 minutes every 3 hours) at a nominal voltage of 28 volts dc.

The Orbiter provides (all three fuel cells operating) a dedicated fuel cell with sufficient reactants to generate 890 kWh of energy to the Spacelab and its payload. In addition, up to 750 watts of power are available on the Orbiter AFD for Spacelab subsystems and payload equipment. Additional energy (840 kWh) can be provided by supplemental energy kits weighing approximately 1632 kg (wet); these energy kits are chargeable to the payload weight budget (759 kg landing weight).

The Spacelab Electrical Power and Distribution Subsystem (EPDS) takes the basic Orbiter fuel cell output and controls, converts, and distributes it to payload equipment. The Spacelab EPDS provides the basic fuel cell 28 Vdc output plus up to 2.25 kVa maximum continuous power at 115 Vac, 400 Hz, three phase.

3.1.3.3 Thermal Control

Payload thermal control on orbit is accomplished by radiation either directly to space or indirectly via the Orbiter Active Thermal Control Subsystem (ATCS). A water flash evaporator provides thermal

control during the ascent flight phase. For AMPS flights, it is assumed that the Orbiter radiator kit which increases the heat rejection capability to 29,000 Btu/hour is incorporated in the ATCS.

The Spacelab provides a Thermal Control Subsystem (TCS) to couple all Spacelab and payload equipment to the Orbiter ATCS. The Spacelab TCS can transfer 8.5 kW maximum continuously or up to 12.4 kW peak power to the Orbiter.

Cooling of equipment inside the pressurized module can involve one or more of the following loops:

- The cabin air loop. A maximum of 1.65 kW of payload generated heat can be accommodated
- The avionics air loop, with a nominal capacity of 4.5 kW, of which 3.56 kW is available for payloads
- An experiment heat exchanger, which can accommodate about 4W.

All of the above loops couple directly to the Spacelab water loop which feeds the Orbiter-Payload heat exchanger.

Pallet equipment may be thermally controlled by means of a separate Freon loop with an interloop (Freon-water) heat exchanger and a series of pallet-mounted cold plates.

The Orbiter has certain inherent thermal limits which constrain its ability to hold certain orientations relative to the sun. This leads to limited duration attitude holds for solar incidence angles above 60 degrees with respect to the orbit plane.

3.1.3.4 Pointing and Attitude Control

The Orbiter provides a basic navigation, pointing, and attitude control capability to all payloads. The Orbiter Guidance, Navigation, and Control (GN&C) subsystem can point any vector defined in its coordinate system at any known target to within ± 0.5 degree (3-sigma) half-cone angle. To minimize errors involved in transferring between the Orbiter IMU and the payload, the GN&C will accept attitude information from a payload-supplied sensor.

The Orbiter Flight Control System (FCS), utilizing the Reaction Control Subsystem (RCS) provides payload pointing and vehicle stability (deadband) of ± 0.1 degree per axis and a stability rate (or maximum limit cycle rate) of ± 0.01 degree per second per axis. RCS propellant needed to provide these performance characteristics are chargeable to payload weight.

3.1.3.5 Command and Control

The Spacelab, by means of its Command and Data Management Subsystem (CDMS), provides the basic command and control capability for all payload equipment.

All payload command signals are routed to the equipment via the CDMS experiment data bus and remote acquisition units or RAU's. Commands may originate from a manual keyboard or from a Spacelab computer. Commands may be simple on-off commands (64 per RAU) or serial PCM commands.

The RAU can receive and distribute Greenwich Mean Time (GMT) via the serial PCM command outputs. A user clock, useful for time-tagging experiment data, is also provided by the RAU.

Experiment RAU's can be connected to the experiment data bus in the core segment (two stations), the experiment segment (three stations), or on the pallet (two stations per pallet segment). Each station can accommodate two RAU's. See Section 3.1.4 for further details on the CDMS and the RAU.

The CDMS also provides a data display unit (DDU) and an associated keyboard to permit manual control and crew access to payload data.

3.1.3.6 Payload Data

The Spacelab CDMS provides the capability to acquire, store, and process payload data. Further details on this important subsystem are contained in Section 3.1.4.

3.1.3.7 Communications

The Orbiter provides all communications capabilities in support of the AMPS payload. This includes all space-to-ground communications as well as the RF link from Orbiter to deployed packages such as the

ESP. Both S-band links (the PM direct/relay and the FM direct) and the K_u-band relay links will be used to transmit AMPS data to the ground and to receive ground commands. The direct links interface with the STDN ground stations whereas the relay links are routed via the TDRS satellite to a TDRS-compatible ground station. A high rate digital data multiplexer is provided by Spacelab (see Section 3.1.4).

3.1.3.8 Ground Umbilicals

The Orbiter provides several umbilical panels which can accommodate payload fluid and electrical lines. These include two T-0 umbilicals, a T-4 hour (pre-prelaunch) umbilical panel, and a panel designed to service storable upper stages.

3.1.3.9 Remote Manipulator System (RMS)

The Orbiter provides (at no weight penalty to the payload) a 50-foot articulated manipulator arm. Mounted on the left side of the vehicle, the RMS can remove and replace payload elements which are mounted in the payload bay. A second manipulator arm chargeable to the payload may be mounted on the right side of the Orbiter. However, only one arm may be operated at any instant. The manipulator may be used to deploy payloads, to provide lighting in the payload bay, or to inspect payloads by means of a TV camera mounted near the end of the manipulator arm.

3.1.3.10 Orbital Aft Flight Deck (OAFD)

The OAFD provides panel space and other resources (power, thermal control) to accommodate payload equipment. The OAFD contains the basic control stations for activating Spacelab payloads. The OAFD also provides the control station for the RMS, windows for viewing the payload bay, and various displays, including CCTV displays, for monitoring payload operations.

3.1.4 Payload Data Acquisition and Handling

The AMPS payload data acquisition and handling requirements have been thoroughly established; they are accommodated by the Spacelab Command and Data Management Subsystem (CDMS). These CDMS services include data acquisition, data processing, data formatting, data transmission to the Orbiter, recording, monitoring, display, command and control capability for experiments and subsystems, audio communications, caution and warning, and provisions for closed circuit television.

Figure 3-6 shows a functional block diagram of the CDMS and Spacelab/Orbiter interfaces, and the location of the CDMS equipment in the module plus pallet mode for AMPS flights. The CDMS design is based on the concept of remote multiplexing of telemetry data and remote distribution of commands. The Spacelab design incorporates two serial digital data buses (one for experiment instruments and one for Spacelab support subsystems) to route command and telemetry data to and from Spacelab and instrument subsystems. The data bus minimizes interconnect problems and allows sizing of the CDMS to the actual flight requirements on a flight-by-flight basis for modular growth capability.

The CDMS configurations contain three on-board computers. One computer is designated the subsystem computer and interfaces with the subsystem data bus through the subsystem I/O unit. The second computer is the experiment computer which interfaces with the experiment data bus through the experiment I/O unit. The third computer is primarily intended as backup for the subsystem computer; it is also available to experiments in case of experiment computer failure. The backup computer is normally filled with subsystem programs; consequently, before operating as an experiment computer, the core memory must be loaded with appropriate experiment software from the mass memory unit (MMU). Also, programs for control and processing of experiments exceeding the capability of the experiment computer can be loaded at time of execution from the MMU.

The I/O units interface the subsystem computer and experiment computer with the respective data buses, CRT/keyboard, high rate digital multiplexer (HRDM), Orbiter timing data, Orbiter uplink command data, and Orbiter downlink telemetry. The current Spacelab CDMS baseline does

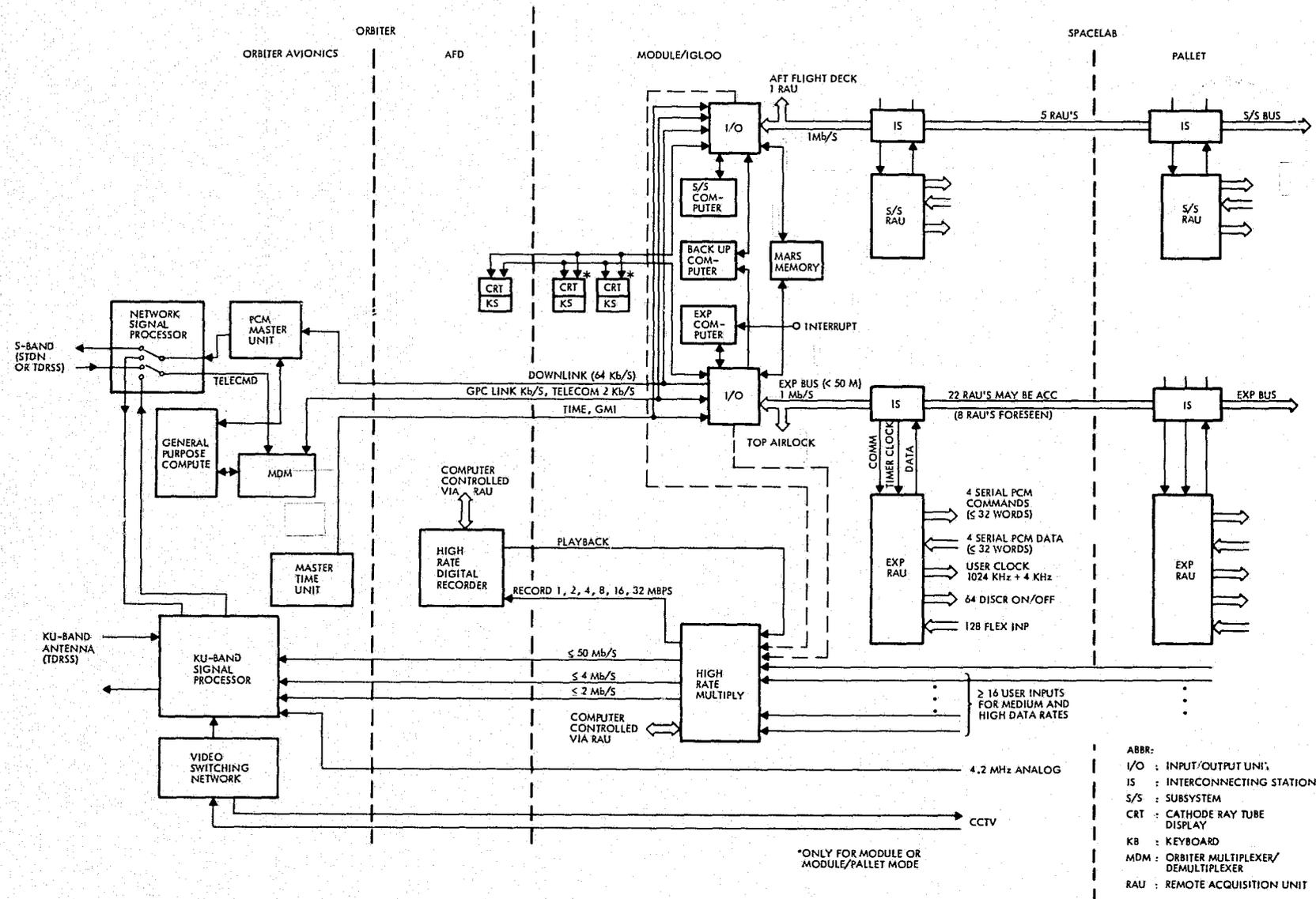


Figure 3-6. Spacelab Command and Data Management Subsystem Block Diagram

not permit data exchange between the subsystem and experiment computers. Under certain conditions, this feature would lead to the reduction of duplicate software and computation by the subsystem and experiment computers when the experiment computer needs data located in the subsystem computer and vice versa. Such a capability could be implemented by a direct connection between the I/O units or as indicated in Figure 3-6 by connecting a subsystem remote acquisition unit (RAU) and an experiment RAU together. These alternatives are currently under review within the Spacelab program. They have not been included in the baseline.

Interactive command capability with the Spacelab computers is permitted through the CRT/keyboards. One CRT/keyboard is located in the Orbiter aft flight deck (AFD) and two are located in the Spacelab module. The CRT/keyboard permits the crew to initiate preprogrammed sequences stored in the computers. It also permits real-time adaptive commands to be generated by the crew on the CRT/keyboard. To ensure proper operation of the instruments, data processed by the experiment or the subsystem computer can be displayed on the CRT's for a quick-look capability.

The RAU's are the prime interfaces for acquisition of low-rate digital data, analog data, and instrument commands. The data exchange between the RAU's and I/O unit is through a 1 Mbps serial digital Manchester II encoded data stream. The RAU's are connected to the data bus at interconnection stations (IS) located in the module and on each pallet. The experiment data bus will accommodate up to 22 RAU's. The RAU is implemented in a modular fashion to satisfy various users (see Figure 3-7).

The smallest unit available to the user is the mini-RAU, consisting of the power supply and the core RAU module. Additional modules are the interface module, which doubles the core module capability; the experiment module, which provides serial PCM inputs and outputs; and the user time clock module.

Medium and high rate scientific digital data is accommodated by the high rate multiplexer (HRM). (See Figure 3-8.) The HRM can multiplex up to 16 data sources and provide the interfaces with the high rate digital recorder (HRDR). Also, the HRM can multiplex recorded data and real-time data into one data stream to maximize use of the Orbiter Ku-band data transmission. The HRM configuration and its interfaces with the HRDR are computer controlled by a RAU.

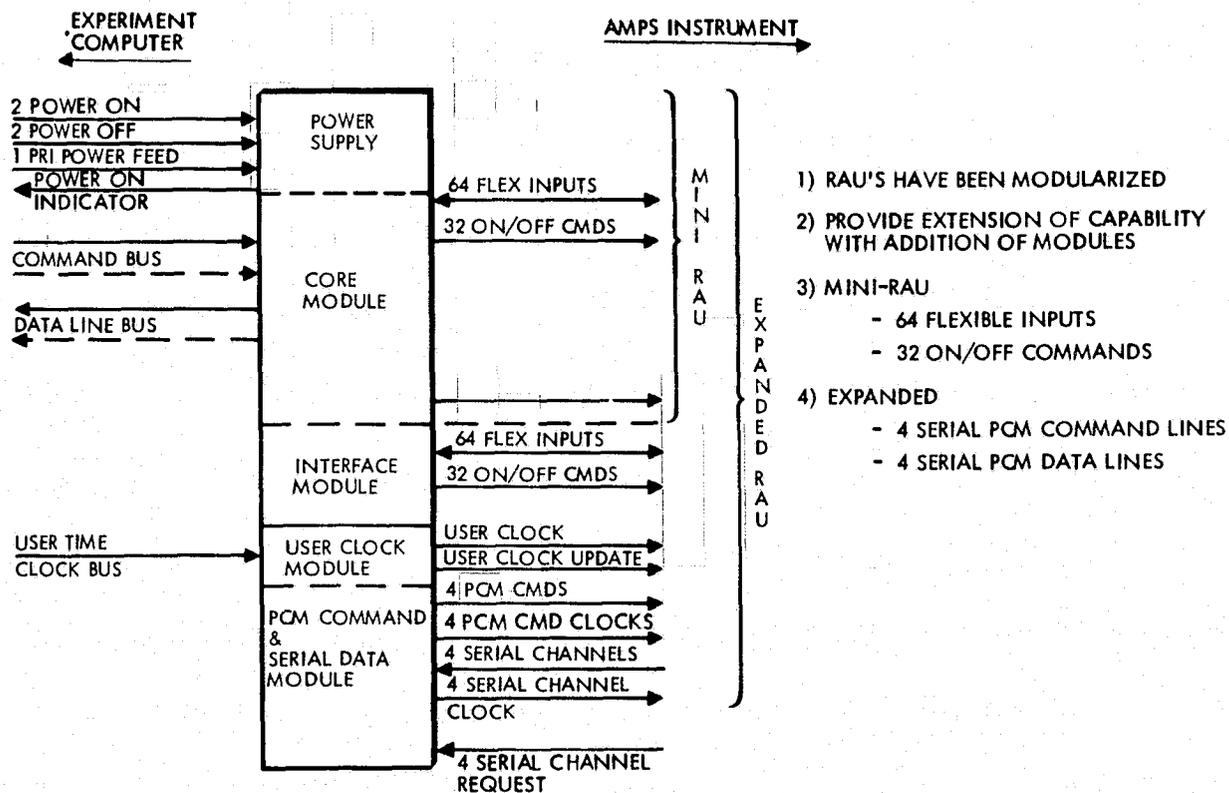
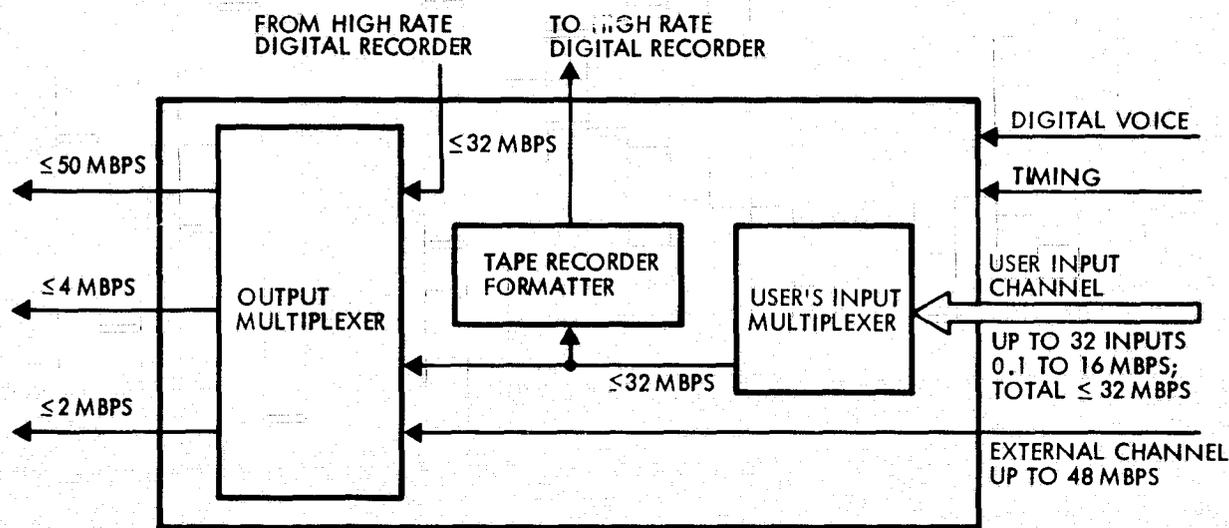


Figure 3-7. Remote Acquisition Unit Capabilities



- 1) PROVIDES HIGH RATE REAL TIME TRANSMISSIONS
- 2) MULTIPLEXES UP TO 32 DATA SOURCES
- 3) INTERFACES WITH HIGH RATE DIGITAL RECORDER
- 4) PROVIDES FOR SIMULTANEOUS RECORDER DUMP AND REAL TIME TRANSMISSIONS

Figure 3-8. High Rate Multiplexer Block Diagram and Capabilities

The HRDR is a variable speed recorder used in conjunction with the multiplexer to provide a data storage capability during non-TDRSS coverage periods in the Orbiter Ku-band system. The HRDR is in the technical definition phase, so is not in the Spacelab CDMS baseline.

The CDMS also provides the necessary electrical interfaces for experiment-provided TV equipment to form an extension of the Orbiter closed-circuit TV system. Space will be allocated in the Spacelab module, center control rack, for an Orbiter TV monitor.

In summary, Table 3-5 lists the overall Spacelab CDMS capability that is available for payload data acquisition and data handling.

3.1.5 Controls and Displays

The major system engineering considerations which influenced the layout of the AMPS controls and displays in both the pressurized module and aft flight deck (AFD) were:

- Crew functions, responsibilities, availability, and capability
- Operational timeline
- Operations efficiency
- Safety requirements
- Compatibility with Spacelab resources and interfaces
- Compatibility with Orbiter (AFD) resources and interfaces
- Integration and test requirements.

Subsystem engineering considerations relating to the definition of controls and displays requirements and preliminary design are discussed in Section 4.4.

The preliminary design and layout of the control and display panels and experiment equipment housed in the module and AFD are well below available rack surface area, volume, weight, power, and thermal limits. The following paragraphs describe those layouts.

3.1.5.1 Short Module

Figures 3-8 and 3-9 show the preliminary layouts of the equipment for Flights 1 and 2, respectively. Less than 50 percent of the available

Table 3-5. Spacelab CDMS Capability Summary

<u>Payload Data Acquisition</u>	
Housekeeping and Low Rate Scientific Data (to computer via RAU's)	
Max number of RAU's	22
Number of flexible inputs (analog or digital) per RAU	128
Analog: resolution of analog/digital conversion	8 bit
Discrete: number of inputs addressable as group	16
Number of serial PCM inputs per RAU	4
Clock rate	1 Mbps
Max number of words transferred per sample	32
Word length including parity bit	17 bit
Max basic sampling rate	100 Hz
Wideband Scientific Data	
Number of CCTV video input channels	1
Number of 4.2 MHz analog channels	1
Number of experiment channels of the high rate multiplexer (HRM)	16
Min data rate of HRM input channels	64 kbps
Max data rate of HRM input channels	16 Mbps
<u>Data Transmission to Ground</u>	
Nominal data rate for housekeeping and low rate subsystem and experiment computer	64 kbps
Max data rate for wideband scientific data (via TDRSS)	50 Mbps
Max data rate of high rate digital recorder (HRDR) bridging TDRSS noncoverage periods	32 Mbps
Storage capability of HRDR	3.5×10^{10} bit
<u>Payload Command Capability</u>	
Telecommand rate from ground via Orbiter	2 kbps
Number of on/off command outputs per RAU	64
Number of serial PCM command channels per RAU	4
Clock rate	1 Mbps
Max number of words per command	32
Word length including parity bit	17 bit
<u>Payload Data Processing and Displays</u>	
Payload data processing:	
Word length	16 bit
Speed	350 kops
Floating point arithmetic	32 (24 + 8) bit
Mass memory	131 Mbit
Payload data display: alphanumeric display screen	12-inch diagonal Tri-color

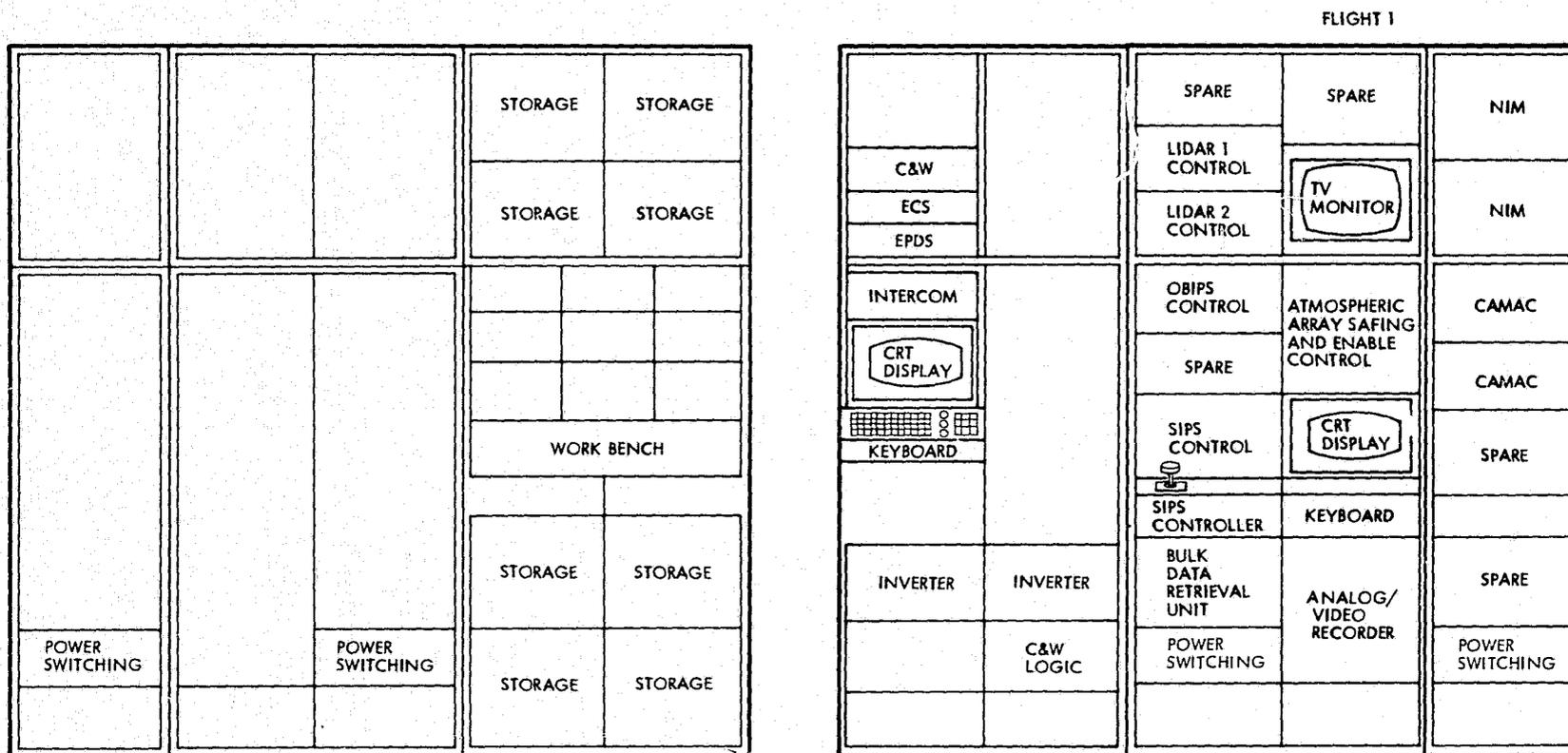


Figure 3-8. Spacelab and AMPS Flight 1 Controls and Displays

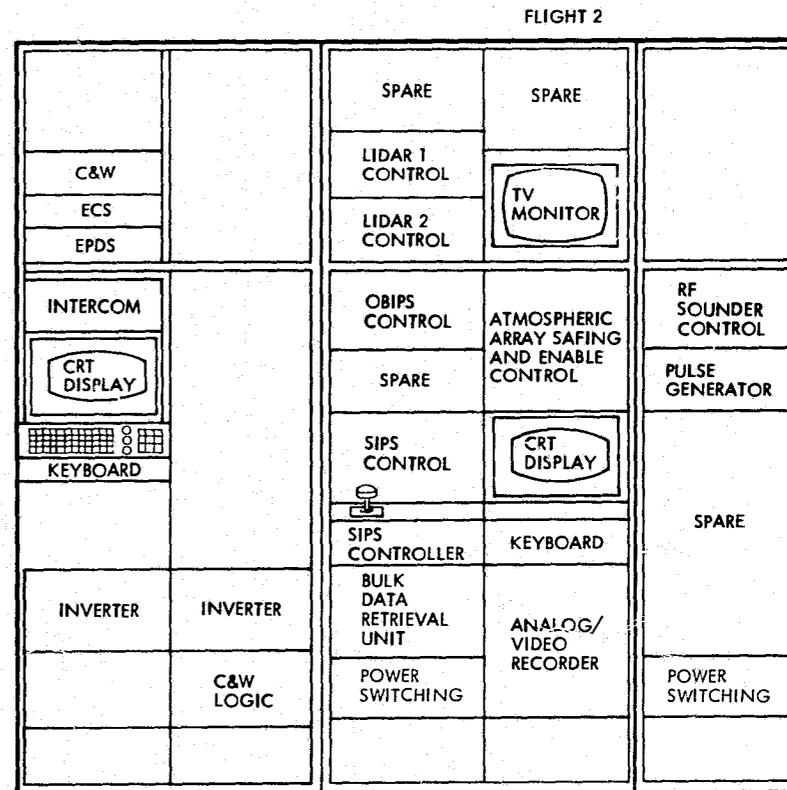
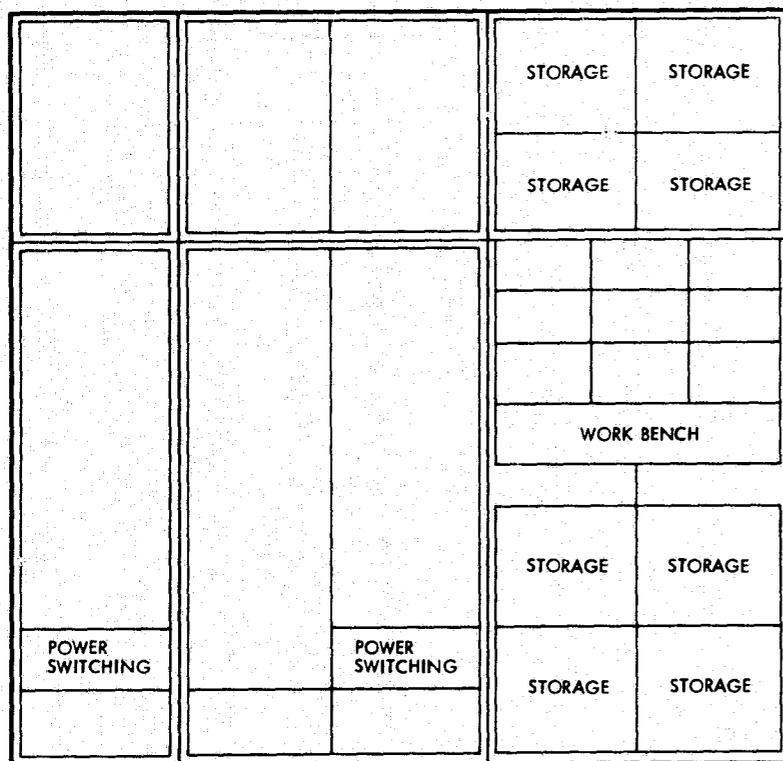


Figure 3-9. Spacelab and AMPS Flight 2 Controls and Displays

rack space for payloads in the short module is required for either payload. The layout anticipates that the proposed change to relocate the second CDMS-DDU/keyboard from the core rack will be implemented. NASA has submitted an additional change request to ESA to route the DDU bus to each double rack in the module. If accepted, both DDU's/keyboards can be relocated to any convenient location.

For operating efficiency and to simplify integration, the atmospheric array control and display is housed in one double rack. The electron accelerator or RF sounder is housed in a single rack. It is recommended that the bulk data retrieval unit (see Section 4.4) be considered as MMSE and be relocated to the core segment double rack.

Scientists evaluated the layouts in our soft mockup by simulating runs of the operation timelines.

3.1.5.2 Aft Flight Deck

Figure 3-10 describes the preliminary layout of the AMPS controls and displays at the AFD-PSS station.

On-orbit experiment operation analyses indicated there are no mandatory requirements for experiment control from the AFD. Exceptions are experiments requiring RMS positioning of the ESP and the requirements to provide the capability for emergency reconfiguration and/or safing of the payloads.

Pointing platform (SIPS) control has been included for flexibility in use of personnel and scheduling. A modified "joy-stick" hand controller enables pitch, azimuth, left/right canister slew, focus/FOV, and cursor control with a single unit. OBIPS control is included as an option to supplement the Orbiter CCTV monitors and provide excellent coverage of the cargo bay. The above controls, coupled with the Spacelab-furnished CDMS-DDU/keyboard, provide all the capability required for both Spacelab subsystems and experiments/instruments monitoring and control from this position.

All controls and displays are readily adaptable to the MSS-PSS CORE concept with a surplus of at least 65 percent controls and displays and panel space.

3-27

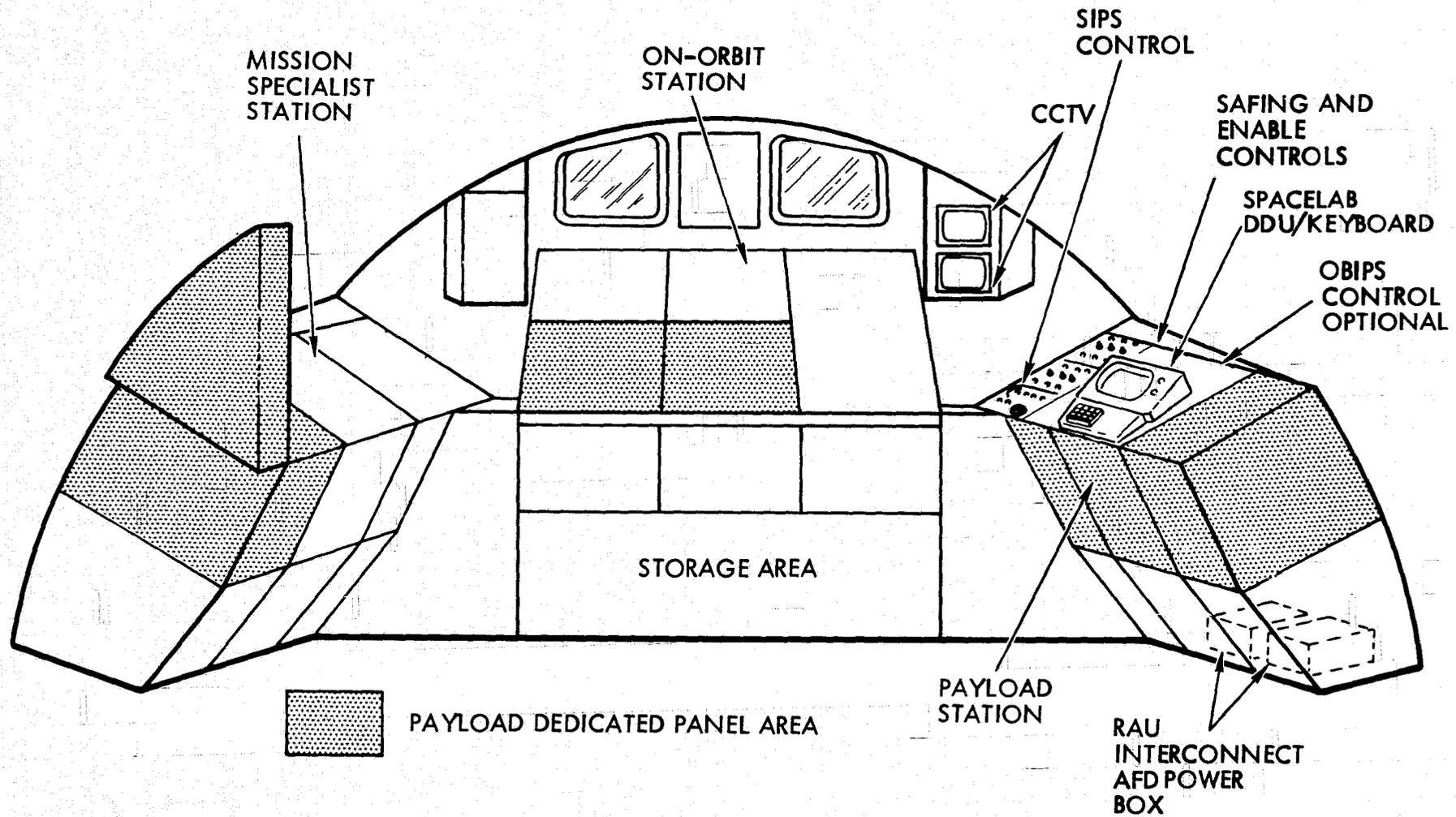


Figure 3-10. Air Flight Deck C&D Utilization

3.1.6 Flight Software Definition

3.1.6.1 Software Requirements Analysis

Early in the study, the software requirements were formulated addressing all software including flight, mission planning, flight operations, ground support equipment, post flight, POC, MCC, Orbiter, training, simulation, data reduction and support. Figure 3-11 summarizes the flight software requirements.

- 1) COMPATIBLE WITH SPACELAB EXPERIMENT COMPUTER SOFTWARE
- 2) FULL PAYLOAD SPECIALIST CONTROL
- 3) EACH INSTRUMENT AND SUBSYSTEM SEPARATELY CONTROLLABLE
- 4) ALL FLIGHT SOFTWARE FUNCTIONS ARE TESTABLE
- 5) SUPPORTS ON-BOARD TEST CAPABILITY
- 6) MINIMIZES FLIGHT-TO-FLIGHT CHANGES BY TABLE DRIVEN, MODULAR DESIGN
- 7) ASSURES SAFE OPERATION BY SOFTWARE INTERLOCKS
- 8) USES NASA-APPROVED PROGRAMMING LANGUAGES
- 9) INCORPORATES EASY-TO-PROGRAM SUBROUTINES FOR DATA ACQUISITION, COMMAND AND DISPLAY
- 10) INTERFACES WITH TEST INPUTS, SIMULATION, AND OTHER SOFTWARE TOOLS TO PERMIT ADEQUATE PREFLIGHT VERIFICATION AND VALIDATION
- 11) CAN ACCEPT GROUND COMMANDS AND DATA

Figure 3-11. Flight Software Requirements

If flight software is properly defined, all other test, simulation, and support software requirements follow. Subsequent work, therefore, focused on flight software. (Flight software requirements are described in the Payload General Specification, Volume IV.) Also, in fulfilling software requirements, there is more to flight software than the CMDS experiment computer software. Some dedicated data processing capabilities resides in microprocessors, small computers, or sophisticated electronics already dedicated to a particular instrument or flight support equipment (FSE).

3.1.6.2 Software Development Process

The sequence of software development and integration steps required for flight operation is illustrated in Figure 3-12. The concept recognizes that the compatibility with the ESA experiment computer operating system and ESA support packages should be maintained by the AMPS software wherever possible. The AMPS flight, on-board test, and ground test software will be developed in an orderly, top-down fashion yielding an integrated flight software package for each mission.

Compatibility between training and POC software with the flight software will be maintained during the development. During software/hardware integration, software changes will be made if incompatibilities are discovered, whether caused by unanticipated hardware characteristics or any other reason.

3.1.6.2.1 Software Facilities. The facilities needed for software development and software integration are shown in Figure 3-13. The software development and integration facility (SDIF) is connected to all AMPS instrument and FSE developers by a time-sharing network. The SDIF computer is a large IBM 360/370 class computer, chosen primarily because of its compatibility with ESA-developed software and simulators. Further, this large computer can accommodate multiple users concurrently, obviating the requirement for a multiplicity of Government-furnished individual software simulators. The SDIF not only supports initial software development by use of the terminals, but also supports checkout of later changes such as those needed in hardware/software integration and maintenance.

3.1.6.3 Payload Specialist - Compatible Software

The software design should accommodate the payload specialist's goal of becoming proficient in the operation of the payload equipment. There is great potential complexity of operation of some of the AMPS experiments. Some experiments, involving several instruments and FSE, can have as many as 60 different adjustable controls. An 8-hour workday of a payload specialist could involve hundreds of control settings. Because of the great expense of each flight, there is an incentive for the payload specialist to perform experimental tasks quickly and efficiently.

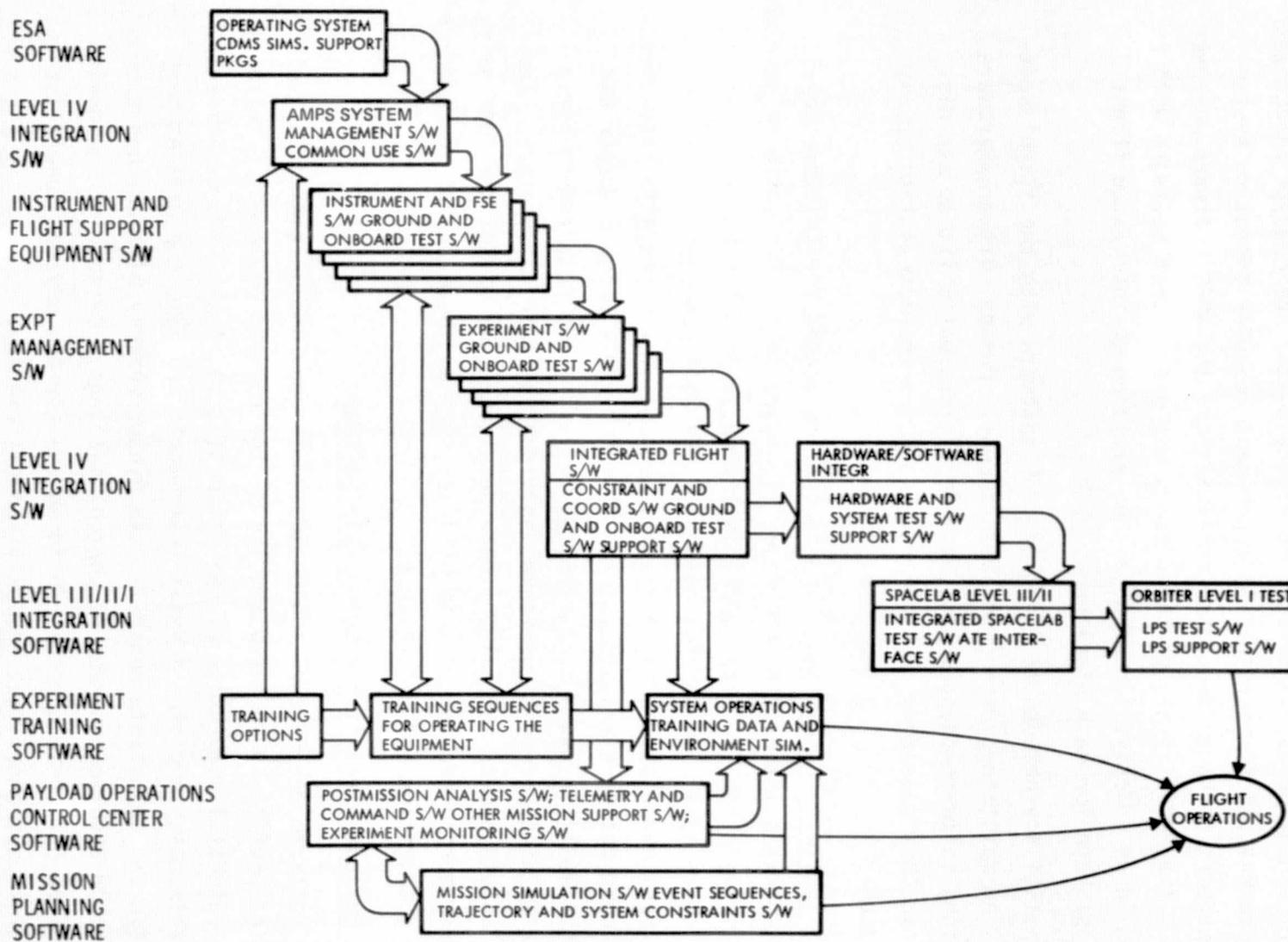


Figure 3-12. AMPS Mission Software

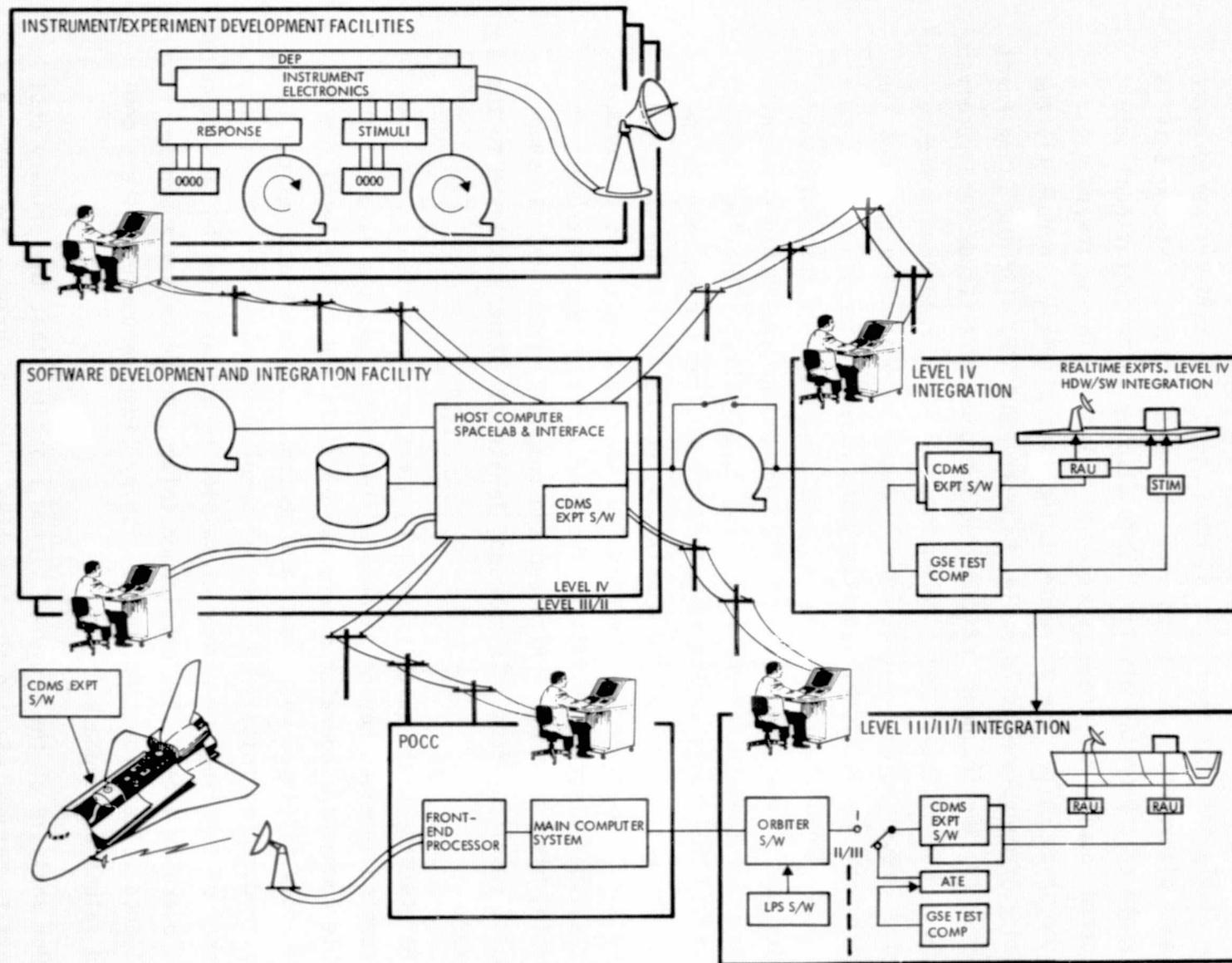


Figure 3-13. Network Concept for the Development of AMPS Software

We examined various display system designs and selected the one that prompts the operator to choose the most promising alternative. A prompted system displays to the operator, at each stage of experiment performance, the legal alternatives he has to control the experiment. Figure 3-14 is an example of a prompted display. This display is an operational data collection showing measurements taken by a lidar pointing toward nadir from the spacecraft. The operator, using this display, controls (1) the frequency of measurements (example, Line 1: GO/STOP), (2) the Fabry-Perot bandpass, (3) the frequency of sampling the density/altitude data received, (4) the number of data samples taken, and (5) whether to downlink or record the received data. (The CDMS computer automatically provides the operator with the latitude, longitude, and MET/GMT time.) Formulating a command is similar to filling out a form. The operator instructs the computer which line number he wants to control and what action he wants to take. For example, Figure 3-14, line 1, illustrates the "GO" command. The operator has typed on his keyboard: "1:GO," as can be seen on the next-to-the-bottom line of the display, a replica of what he has keyed. On Line 1: "measurements," the word "GO" is displayed. This is feedback from the lidar electronics telling the operator that the instrument is confirming his GO command. Although the computer is not yet finished, the data, resulting from this command, is already being displayed to the operator.

An operator could quickly learn to operate such a display. Since each command involves keying in only a line number and the value of the variable commanded, operator errors can be minimized. (The legal formats are directly displayed where they are needed.) Operator actions can be further reduced by installing a set of preplanned values for the bandpass, frequency of sampling, or number of samples to be taken in the CDMS computer. If the operator accepts the preplanned values, he depresses a single "preplanned values" function key to display lidar response to these commanded values as feedback.

Our analysis of the prompted system indicates that the primary cost tradeoff is between placing the prompting words directly on the CRT and incorporating the same information in backup media, such as operators handbooks or microfiche.

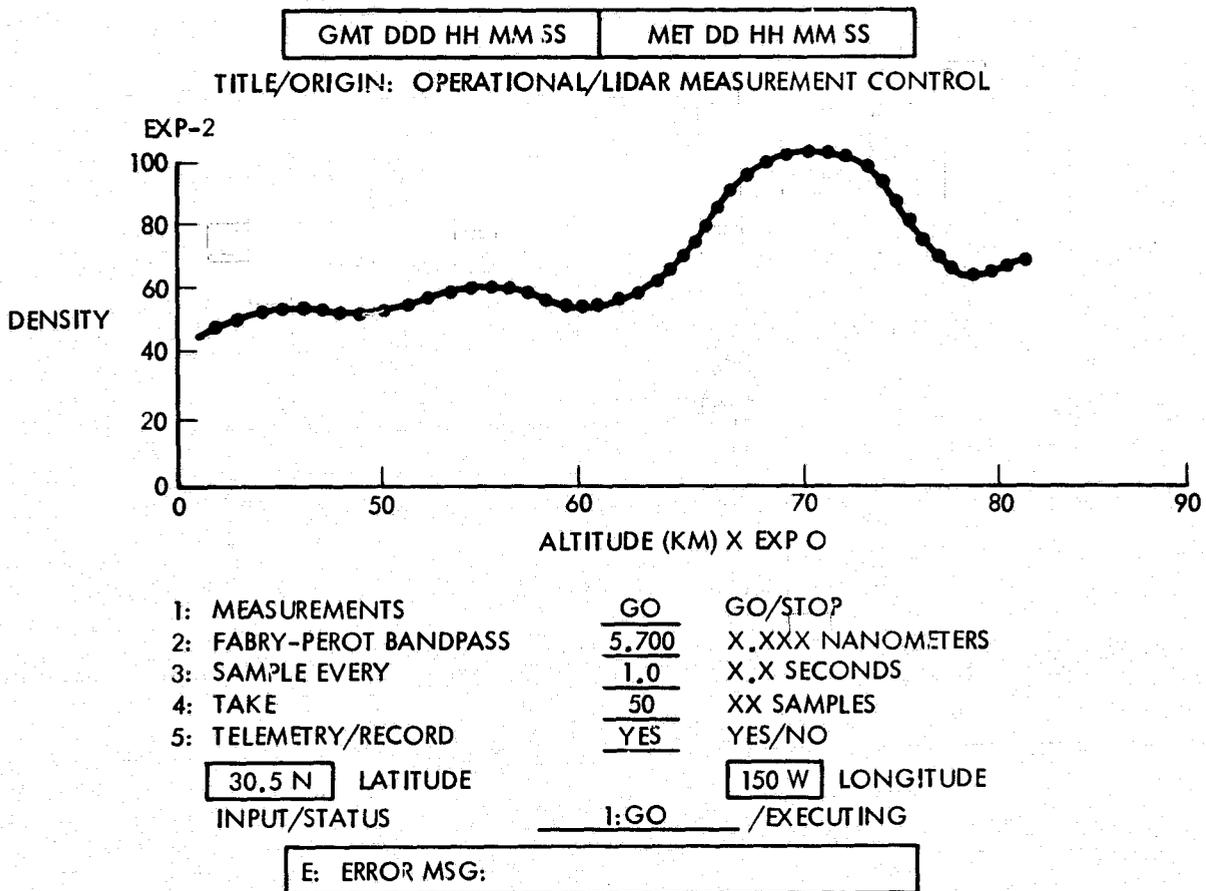


Figure 3-14. Example of a Prompted Display Format

3.1.6.4 Functional Design Concept

Figure 3-15 shows the overall functional block diagram of the AMPS software and the relationship of this software to the CDMS experiment computer operating system (ECOS). The following paragraphs summarize each of the SMPS flight applications software functions shown in the figure.

3.1.6.4.1 AMPS System Management. The AMPS system management package interfaces the AMPS software with the CDMS operating system. It coordinates and integrates the operation of more than one concurrently performed payload activity to resolve conflicts between them, inhibit operations that cause damage or serious delays, and provide common routines and common data elements to the participating experiments. Details are given in the general system specification (Volume IV, Book 2).

3.1.6.4.2 AMPS Experiment Management. The AMPS experiment management packages time and sequence the control variables that control the operation

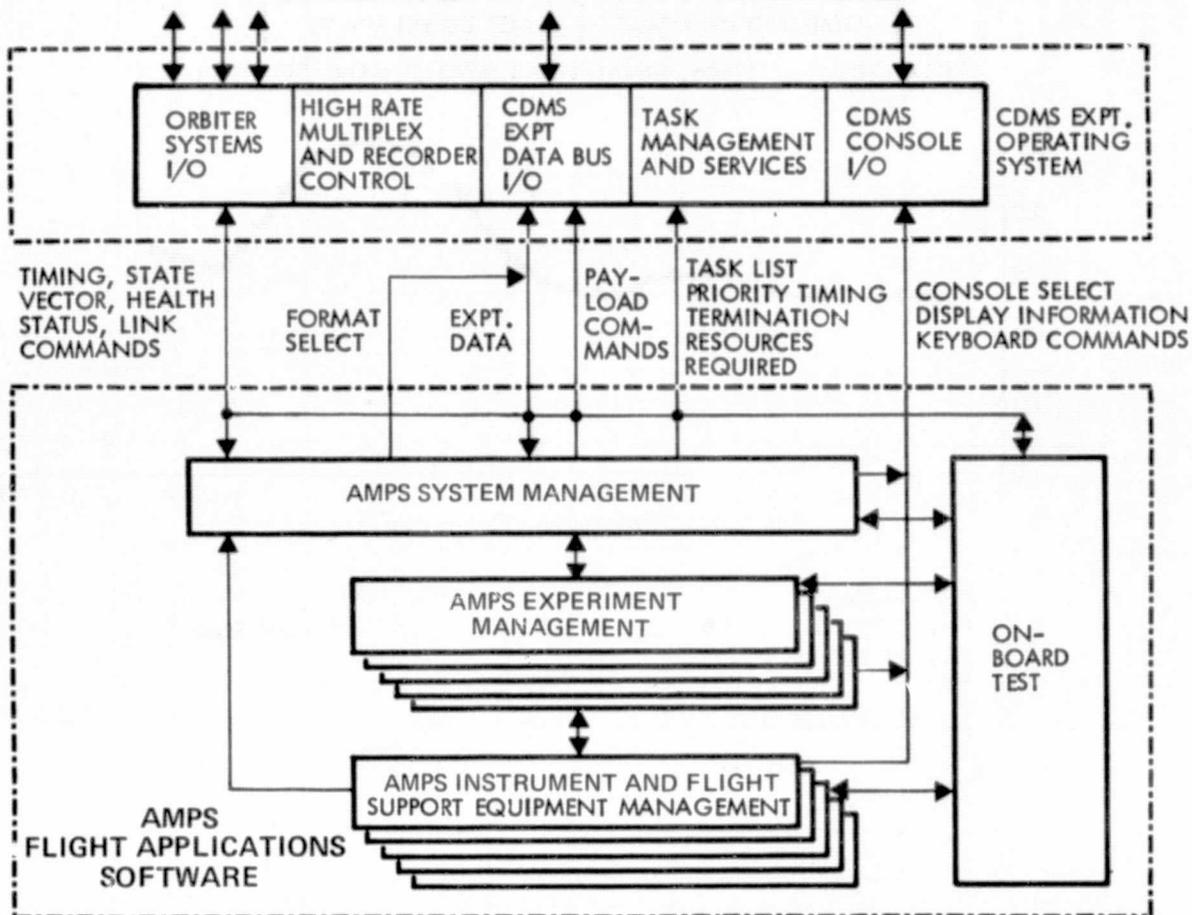


Figure 3-15. Flight Software Flow Diagram

of the instruments and flight support equipment of each experiment, insofar as these are computer controlled. There is one experiment package for each distinct experiment flown in each mission. Each experiment package contains the data and programs to adjust the relevant payload control settings as the experiment progresses. Details are given in the general system specification (Volume IV, Book 2).

3.1.6.4.3 AMPS Instrument and Flight Support Equipment Management. The AMPS instrument and flight support management packages contain all the software modules needed to operate each instrument or flight support equipment that is an independent unit.

There is one instrument and flight support equipment package for each equipment controlled. Each package contains data and programs required by

an instrument or FSE to set up and operate, including the on-off logic, control setting, calibration, alignment, feedback, data processing, and displays.

3.1.6.4.4 On-Board Test Function. The AMPS on-board test package contains all the modules required to test the payload elements that require testing prior, during, or after experiment performance.

There is one on-board test module for each instrument, FSE equipment tested, or other payload-related element tested. The on-board test software exercises and/or measures the performance of payload equipment to determine its command responses; the health status of temperature, voltage, current, and other parameters of critical test points; and the proper operation of error messages and alarms. In addition, the on-board test package is responsible for access to any flight software instructions or data that could require a change as a result of a test or an on-board equipment or software failure or for any other reason.

Details are given in the general system specification (Volume IV, Book 2).

3.1.6.5 Computer Size and Speed Requirements

Because software costs increase appreciably if the computer is too small and/or slow, much of our early study effort was directed toward computer speed and size requirements. The following two paragraphs discuss these requirements.

3.1.6.5.1 Computer Speed Requirements. Early analysis in the Phase B study revealed that the CDMS computer could process only 6 percent of the total information produced by the candidate AMPS science instruments, even if they were operated one at a time. The decision was made to route the raw data to the ground data processing stations through a high-rate multiplexer. The CDMS computer is then off-line, processing only selected samples of the science data produced by the scientific instruments. It also monitors the control parameters of each instrument and FSE. This decision reduced the computational speed requirements of the CDMS computer to a fraction of its 330,000 average operations per second capability. Using general assumptions concerning the computations needed, the speed requirement was still only 18 percent of the average operations per second capability.

3.1.6.5.2 Computer Size Requirements. The computer size requirements impose a more serious constraint on the AMPS software. Early in the study, it was already clear that the sum of all the experiment software would not fit the 65 kilo-word CDMS memory. The software preliminary design was therefore partitioned into elements that could be separately loaded into the CDMS computer.

Partitioning imposes a penalty, since in the current CDMS design the loading of the CDMS computer takes place from the mass memory, which is a tape unit. Driving the tape to the location at which the next partition is stored, could require rewinding the tape and, then, searching forward for the modules required. This could take a few seconds to 2 or 3 minutes. To avoid these delays the number of partitions should be minimized. In addition, the organization of the software gets more complicated with increased partitioning. Because increasingly detailed consideration must be given to assuring that each partition gets all the information needed for its operation, the information could have been developed in another partition. This complication tends to increase costs, especially when changes are made in instruments, FSE, or experiment procedures.

Partitioning options include:

- Option 1. Partitioning the software modules required to do each planned experiment
- Option 2. Partitioning the software as in Option 1 but, in addition, separating the operations of setup and calibration from those of instrument data collection operations
- Option 3. Partitioning the software by modules required to serve small groups of instruments or FSE or to serve each instrument or FSE individually.

Table 3-6 shows our sizing estimates for the AMPS software, based on Option 1. In evaluating the results of these sizing estimates, the total CDMS computer core size is 65K words, while the ECOS operating system occupies about 20K words, leaving 25K words for the application programs sized here. For all practical purposes, the first experiment of Table 3-6, involving vehicle neutralization, beam characteristics, and plasma interactions fills the entire memory. A margin of 19.5K to 29K words appears to exist for all other experiments.

Table 3-6. Estimated Main Memory Occupancy of AMPS Flight One Software
(Using Option 1 Partitioning)

<u>Experiment</u>	<u>AMPS System Management (kilowords)</u>	<u>AMPS Experiment Management (kilowords)</u>	<u>AMPS Instrument and FSE Management (kilowords)</u>	<u>On-board Test (kilowords)</u>	<u>Total Kilowords</u>
1) Vehicle Neutralization Beam Characteristics, Plasma Interactions	13.0	4.0	24.0	3.0	44.0
2) Acoustic Gravity Wave Gas Cloud Expansion Dynamics	11.0	1.0	11.5	2.0	25.5
3) Environment Measurements	10.0	0.5	11.0	-	21.5
4) Ozone Density	10.0	0.5	3.5	2.0	16.0
5) Minor constituents	11.0	1.0	11.5	2.0	25.5
6) Solar Radiation	10.0	0.5	7.5	1.0	19.0

NOTES

- 1) Involves - Electron accelerator, vector magnetometer, mast, SIPS*, Level I diagnostics, Level II diagnostics, OBIPS, ESP I with only E&B receiver, Langmur probe active
- 2) Involves - Gas release/canister, Cryo IR interferometer, OBIPS, SIPS
- 3) Involves - Quartz crystal microbal, ESP I, med energy detector, dc electronic field meter, E&B receiver, vector magnetometer, Langmuir probe
- 4) Involves - Lidar
- 5) Involves - Cryo IR interferometer, cryo LIMB scan radiometer, SIPS
- 6) Involves - Solar flux monitor, SIPS

* SIPS with no Star map, no Kalman filter software

During the software life-cycle it is normal for the software to grow as the flight data approaches. It is probable that Option 1 partitioning will be abandoned for the first experiment of Table 3-6 and possibly for the other experiments in favor of Option 2, as the program matures. This situation shows that computer main memory occupancy needs careful review as new computer applications that produce software size growth are proposed for incorporation into the flight computer.

3.2 FLIGHT FUNCTIONAL ASSEMBLIES

3.2.1 SIPS Pointing and Control

The pointing and control subsystem configuration baselines the Small Instrument Pointing System (SIPS) as the instrument pointing mount. Precision attitude reference is obtained from a stellar inertial attitude reference system employing gyros and a strapdown star tracker, all mounted on the SIPS canister. The majority of the software and electronic data processing functions are performed by programmable digital electronics (PDE). It has been demonstrated that the SIPS can be pointed with 95 percent confidence to within 18.8 arc-sec (2σ). The short-term (10 to 20 seconds) pointing stability error is less than 0.5 arc-sec (2σ).

The Flight 1 instrument pointing requirements are listed in Table 3-7. The cryogenic instrument must point very accurately and must raster scan through 100 km above the earth horizon at a 0.36 deg/sec rate. The configured pointing and control system exceeds the performance requirements of the cryogenic instrument, which has the most stringent pointing and stabilization requirements of all of the SIPS mounted instruments. It is the design driver for the SIPS pointing and control system.

A drawing of a pallet mounted SIPS is shown in Figure 3-16. The SIPS has two instrument carrying canisters, each supported at its center by a yoke which can rotate independently of the other canister in an up-down direction (120 degrees freedom). Each canister in turn is connected to the yoke so as to provide a limited (± 10 degrees) left-right rotational degree of freedom. Both yokes are attached to a common ± 180 -degree azimuth gimbal drive at the base. An optional roll gimbal about the instrument line of sight can be added internally to each canister. AMPS uses such a roll gimbal on the cryogenic instrument side.

Figure 3-16 shows the instruments which comprise the atmospheric array in the right canister and the cryogenic limb scanner instrument in the left canister. A strapdown star tracker and a gyro reference assembly, providing a precision inertial attitude reference for the cryogenic instrument mount to the right of the canister. This attitude reference is transferred to the less demanding instruments in the other

Table 3-7. Flight 1 Instruments Pointing Requirements

INSTRUMENT NAME	INSTRUMENT GROUPING OR LOCATION	POINTING REQUIREMENTS					
		REFERENCE	FIELD OF VIEW	OVERALL ACCURACY	ATTITUDE DETERMINATION ACCURACY	STABILITY OR STABILITY RATE	TRACKING RATES
CRYO LIMB SCANNER	SIPS (B)	EARTH LIMB	±90°	120 SEC	20 SEC	20 SEC	0.36°/SEC
CRYO IR INTERFEROMETER							
SOLAR FLUX	SIPS (A)	SUN		15 MIN	90 SEC	90 SEC	
OBIPS	SIPS (A)		HEMISPHERE	0.1°	0.1°	1°/SECONDS	
LIDAR	PALLET	NADIR		4°	2°	6°/SECONDS	
CHEMICAL GAS RELEASE	PALLET			3°	2°	1°/MINUTE	
ELECTRON ACCELERATOR	PALLET	B VECTOR	±60° TO B	3°	2°	1°/SECONDS	
VECTOR MAGNETOMETER	PALLET 5M BOOM	B VECTOR TO ACCELERATOR COORDS			2°		

SIPS (A) = PORTSIDE

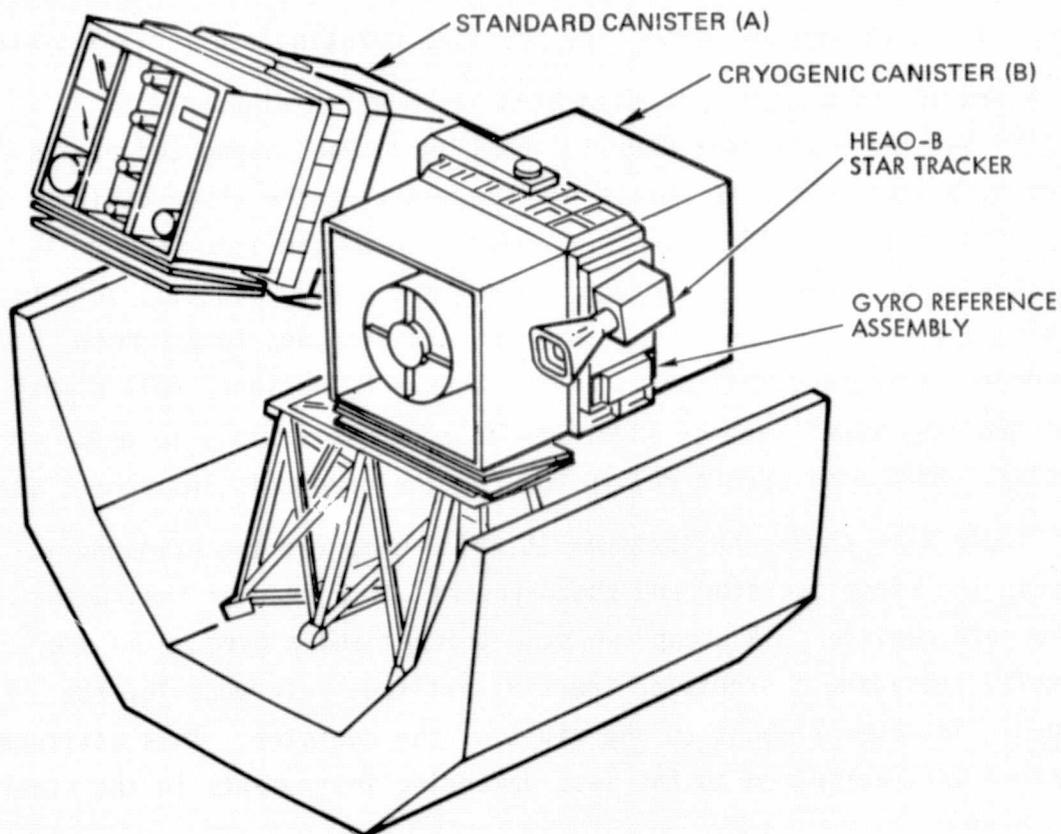


Figure 3-16. Small Instrument Pointing System (SIPS)

canister and to the pallet through the gimbal angle resolvers; attitude transfer between pallet mounted instruments is accomplished by an optical alignment measurement system. Orbiter ephemeris data is required to go from the inertial attitude reference to earth referenced pointing.

The gyro and star tracker data are processed by programmable digital electronics (PDE) consisting of function dedicated microprocessors. The gyro-sensed rates are integrated to provide a high bandwidth, stable attitude reference. The star tracker data is processed by a filtering algorithm that provides optimal attitude and gyro bias updates assuring long term attitude reference stability. In addition, the gyro data is processed to provide gimbal rate information for the SIPS gimbal servos.

The servo control laws are also implemented by programmable digital electronics. The programmable digital electronics communicates with the Spacelab experiment computer and the payload specialist's station through a remote acquisition unit (RAU) and the 1 Mbps data bus. The data bus will mainly carry command level and status type information. This approach to the pointing and control subsystem unburdens the Spacelab computer, makes the SIPS as autonomous as possible, and keeps the interfaces simple. Figure 3-17 is a functional block diagram of the system. More detailed block diagrams expanding on the PDE functions are presented in Section 4.1, along with predictions of system performance.

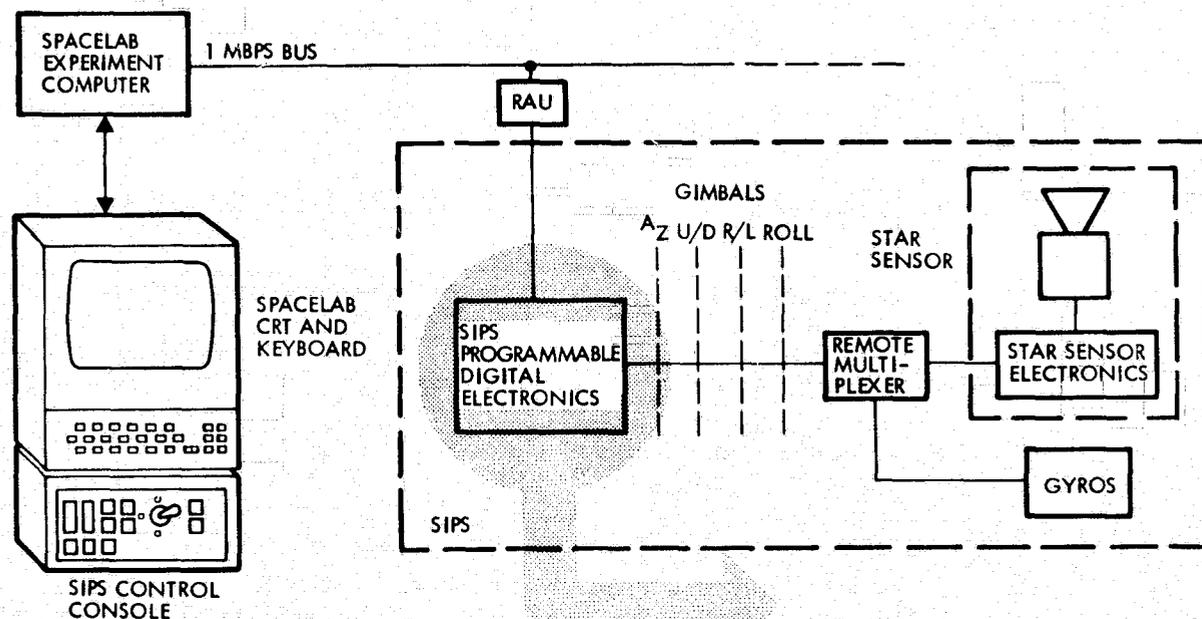


Figure 3-17. SIPS Pointing and Control Functional Block Diagram

3.2.2 Environmental Sensor Package

This section summarizes the ESP design requirements.

3.2.2.1 Operational Requirements

During the course of the Phase B study, operational requirements were identified for ESP's to be flown on four of the AMPS flights and two of the Orbiter Flight Tests (OFT). The instrument complements for these flights are summarized on Table 3-8. ESP's were identified as having two basic operational requirements:

- 1) RMS Mode. The ESP is attached to the end of the Remote Manipulator System (RMS) during orbital operations for the purposes of making experiment support measurements close to the Orbiter and environmental measurements also in the vicinity of the Orbiter. A typical experiment requiring this type of ESP support is the beam plasma interaction experiment during which the ESP instruments are oriented toward the accelerator.
- 2) Deployed Mode. The ESP is ejected from the Orbiter for experiments in the near vicinity of the Orbiter and at distances up to 100 km from the Orbiter.

The AMPS experiments requiring ESP support during the five strawman payloads are shown on Table 3-9. In all cases when the ESP is deployed, a slowly spinning mode with the spin axis perpendicular to the orbit plane is acceptable.

3.2.2.2 Design Requirements

The design requirements for the ESP were driven heavily by the following considerations:

- a) Low Cost. The low cost requirement, of course, is an inherent part of the AMPS program. The short life and trade-off between recoverable subsatellites and throw-away subsatellites caused this factor to be dominant during the design of the ESP. The result is an approach that makes use of capabilities that will be supplied by Spacelab/Orbiter.
- b) Dimensional Constraints. Two important considerations affected the sizing of the ESP: 1) the subsatellite must be large enough to accommodate the required instruments and 2) small enough so that excessive pallet/area is not required.
- c) Safety. The ESP must meet the safety requirements of JSC 11123 Space Transportation System Payload Safety Guidelines Handbook.
- d) Instrument Accommodation. Maximum flexibility must be allowed to accommodate the instrument groupings on Table 3-8.

Table 3-8. ESP Instrument Complement OFT 4 and 6, AMPS 1-4

	OFT-4	OFT-6	AMPS-1	AMPS-2*	AMPS-3	AMPS-4
1) Medium Energy Electron Detector			•	•	•	
2) DC Electric Field Meter			•	•	•	
3) E and B Receivers and Antennas (ac electric and magnetic field detectors)	•	•	•	•	•	
4) Langmuir Probe	•	•	•	•	•	•
5) Fluxgate Magnetometer (dc magnetometer)	•	•	•	•	•	•
6) Neutral Mass Spectrometer			•	•		•
7) Ion Mass and Distribution Analysis			•	•		•
8) High Energy Ion/Electron Detectors	•	•				
9) Retarding Potential Analyzer	•	•				
Total (maximum values)						

Source Material: ESP's for OFT Flights - Environmental Science Package (ESP) Definition Study - SE-012-050.2H

AMPS Flights Strawman Payload - GSFC

*AMPS-2 -- two ESP's used on this flight.

Table 3-9. ESP Applications AMPS Missions

Flight No.	Experiment No.	Experiment	Comments
1	A-4	Beam-Plasma Interactions	RMS Mode
2	A-1	Low Altitude Conductivity Modification	Deployed Mode
2	A-2	High Frequency/Wave Particle Interaction	↓
2	A-3	Long Delayed Echoes	↓
2	A-4	Plasma Flow	RMS Mode
2	B-5	High Latitude Changes Ozone Density	Deployed Mode
3	A-2	Second Generation Electron Accelerator	↓
3	A-4	Electron Echo	↓
3	A-5	E B Studies	↓
3	A-6	Beam Excitation of Plasma Waves	↓
3	A-7	ELF/VLF Antenna Development	↓
4	A-3	MPD Arc Injection Studies	Deployed Mode

Specific design requirements that resulted from analysis of experiment/instrument requirements resulted in the following:

- Required Pointing Accuracy. RMS mode: $+1^{\circ}$ nominal, spin mode ± 5 to 7 degrees with respect to the orbit plane
- Maximum Range. 100 km (a compromise based on communication capability)
- Data Rate. 16 kbs (compromise based on communication capability)
- Position Control. With respect to Orbiter, ± 1 to 2 meters at close ranges.
- Life in Orbit. Up to 4 days
- Energy Requirements. 96 hours at 25 percent; duty cycle at approximately 90 watts.

3.2.2.3 Design Analysis

The basic configuration for the ESP is a cylinder 1.0 meter in diameter and 0.56 meter high. (See Figure 3-18.) Antennas and sensor are deployed after the ESP is attached to the RMS or after ejection from the Orbiter. The estimated mass is 171 kg. The spin rate is approximately 10 rpm before antenna deployment.

- a) Attitude Control. Passive spin stability is achieved by a spin stable ellipsoid of inertia combined with a passive nutation damper. During the 4-day life of the ESP, the spin axis moves with respect to the normal to the orbit plane through an angle dependent upon the regression of the line of nodes of the orbit plane. For example, at an altitude of 300 km and an orbit inclination of 55 percent, the maximum angular movement is approximately 14.7 degrees. This maximum angle can be reduced by one-half by compensation at ejection of the ESP.
- b) Position Control. The ESP is not maneuverable because the Orbiter has an adequate translation capability.
- c) Telemetry and Command. The payload interrogator on board the Orbiter will be employed for communication with the ESP whenever it is separated from its cradle. This minimizes system cost but limits data rates to 16 kbs, constrains the ESP design, and affects orbital operations.

- d) Tracking. The Ku-band radar is used to range of 20 km (skin track)
- e) Electric Power. Low-cost primary or secondary battery system is used
- f) Differential Drag Effects. The ballistic coefficient of the ESP is greater than or less than the Orbiter depending on the Orbiter's attitude. (See Table 3-10.) The smaller the ballistic coefficient, the faster the orbit will decay. Therefore, when the Orbiter is in its normal mode of operation, mode A, the ESP will move downward and in front of the Orbiter. The relative motion of the ESP with respect to the Orbiter is given on Figure 3-19. Since the Orbiter must expend RCS propellant to maintain station with the ESP, it becomes costly to maintain below approximately 250 km. The effect of orbit altitude on differential drift is shown on Table 3-11.

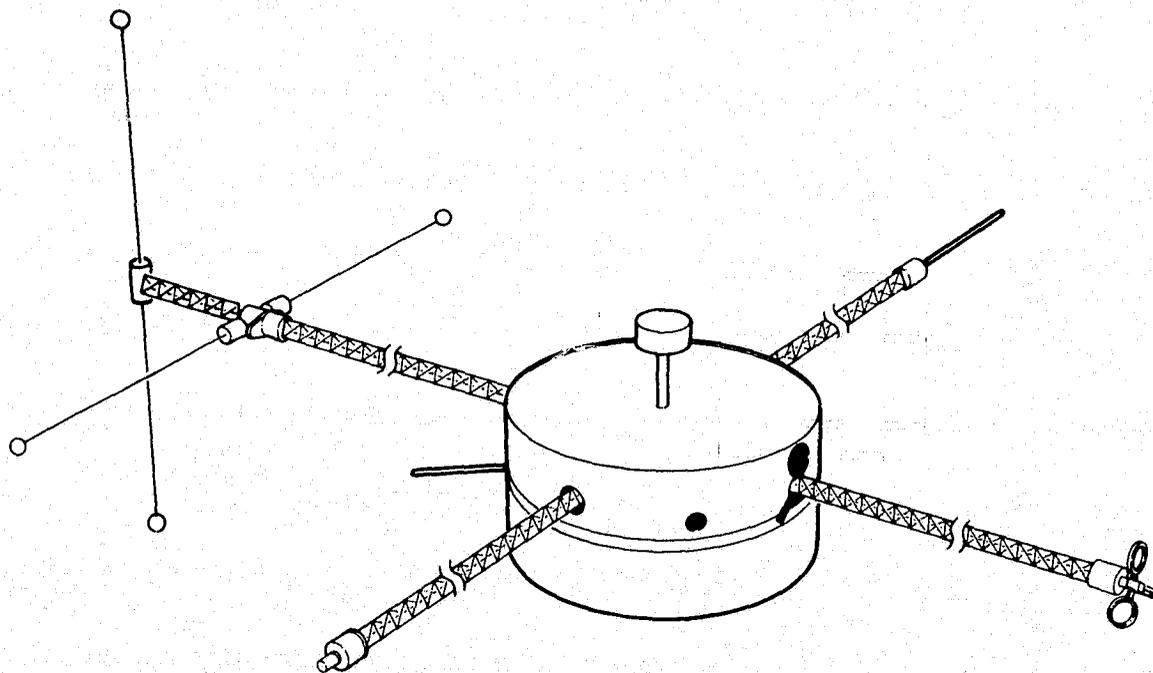


Figure 3-18. Environmental Sensor Package Configuration

Table 3-10. ESP/Orbiter Ballistic Coefficient Comparison

Orbiter			ESP
Mode A	X-Axis	Forward - 472 kg/m ²	Spin axis perpendicular to orbit plane - 160 kg/m ²
Mode B	Y-Axis	Forward - 142 kg/m ²	
Mode C	Z-Axis	Forward - 82 kg/m ²	

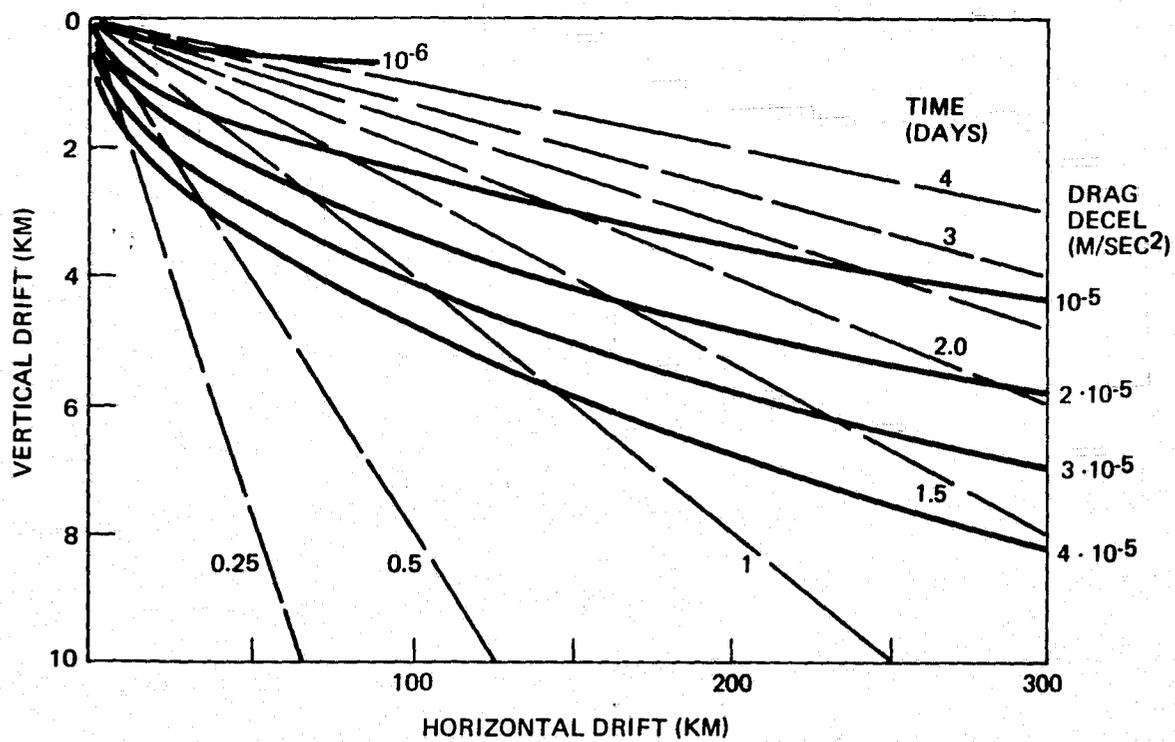


Figure 3-19. Horizontal and Vertical Drift Distances Relative to Drag-Free Trajectory

Table 3-11. Differential Drag Effects (Orbiter X-Axis Forward)

Altitude (km)	Differential Drift (ΔR) - 24 Hr (km)	Drag Compensation (ΔV) (m/sec)
350	7.9	0.17
200	231	5.1

3.2.2.4 Preliminary Design

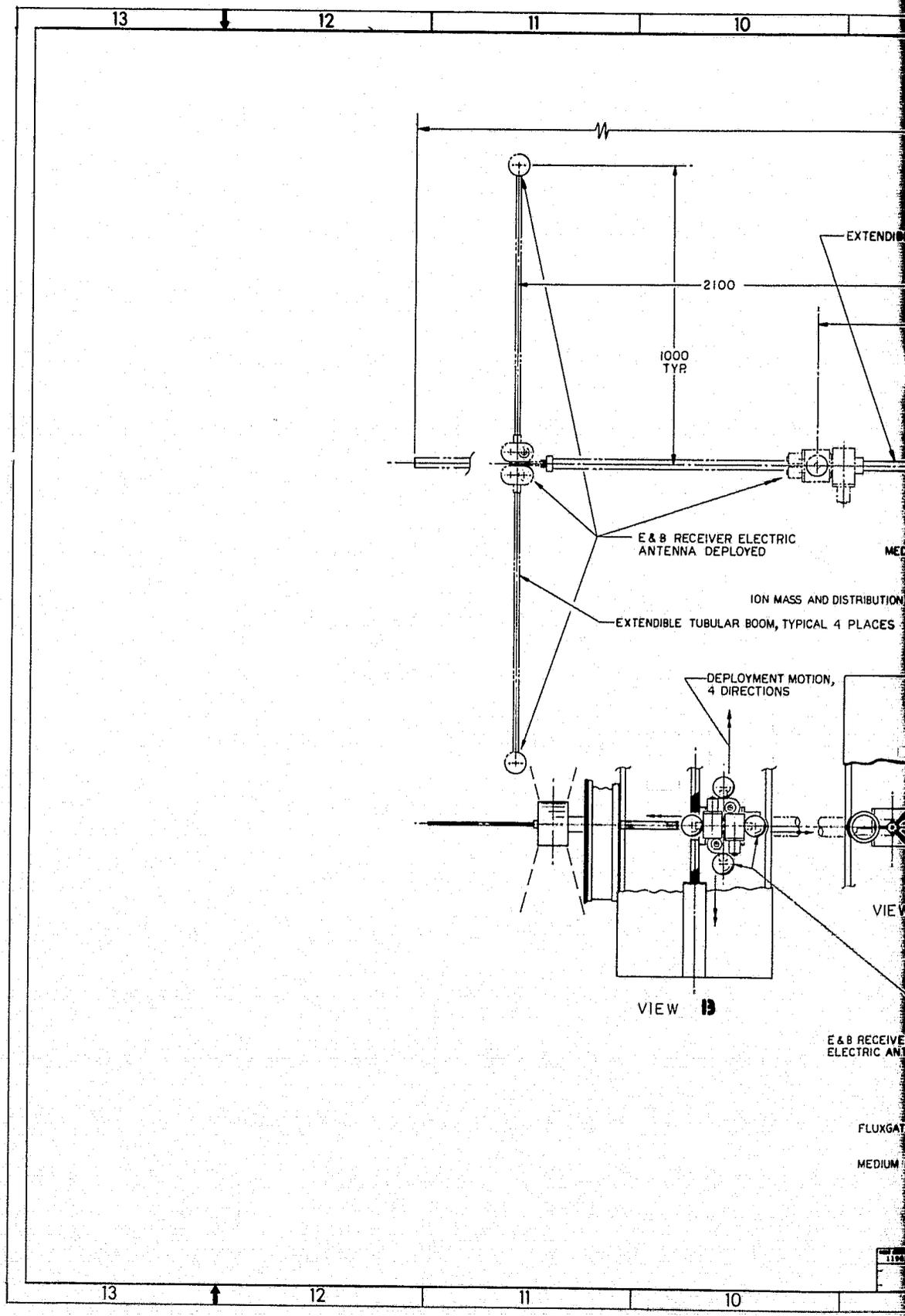
The ESP is comprised of a standardized ESP bus and instrument complement. (See Figure 3-20.) The bus is designed to accommodate a selection of instruments tailored to specific flights. The bus provides a supporting and handling structure, electrical power, tracking, telemetry and command, and thermal control of all instruments.

Specific design characteristics of the LSP are:

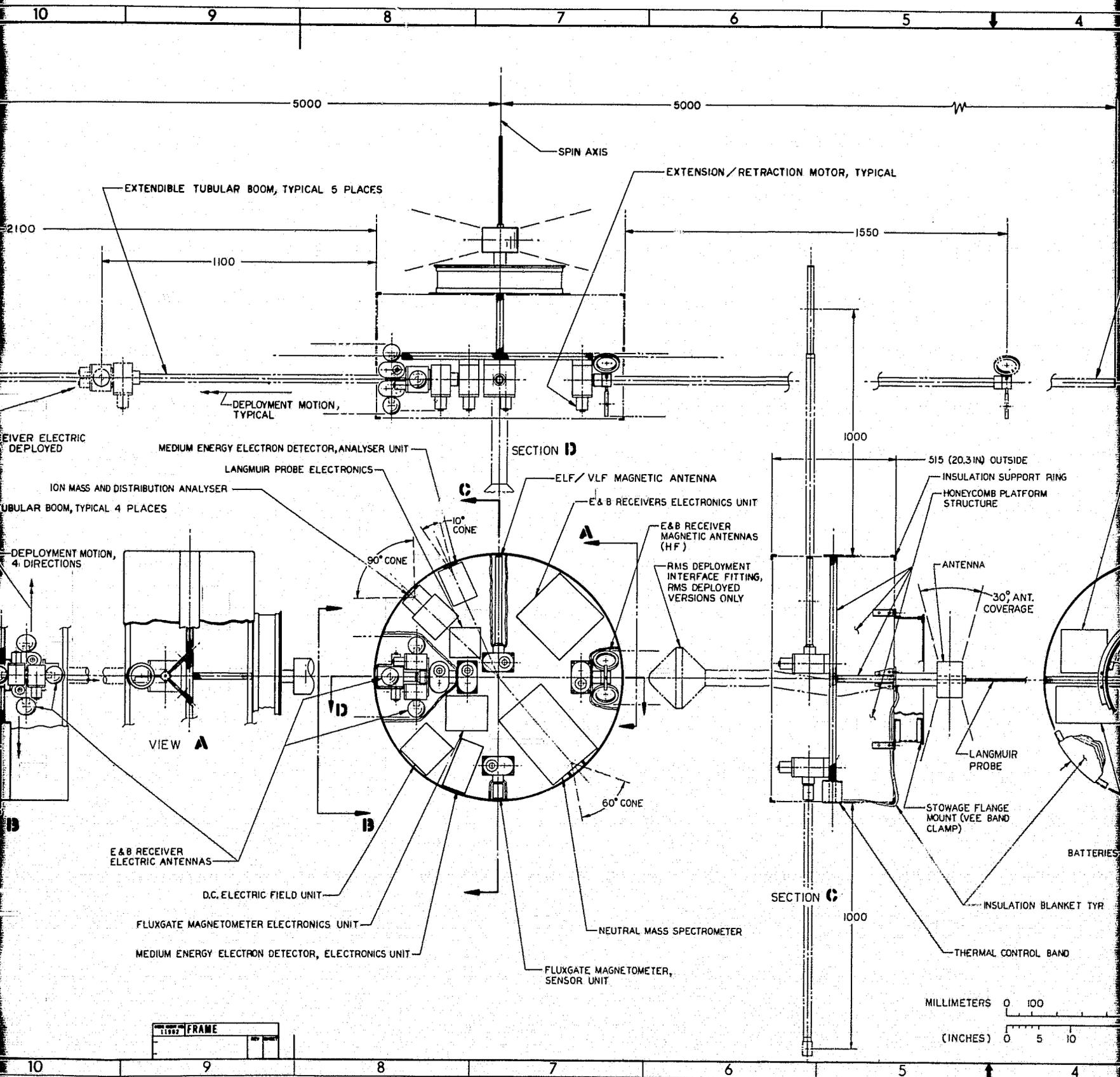
- a) Mass Properties. The mass properties of the ESP are shown in Table 3-12.
- b) Electrical Power. Electrical power and distribution is summarized in Table 3-13.
- c) Telemetry, Tracking and Command System. The S-band TT&C system will be compatible with the Orbiter payload interrogator system. It will provide the following capabilities:
 - 2 kbps maximum command rate
 - 16 kbps maximum telemetry rate
 - 100 km maximum range at 16 kbps data
 - ESP requires EIRP of 7 dBW
 - ESP antenna provides 3 dB antenna gain with a toroidal beam antenna with 18-degree beamwidth.

A system block diagram is provided in Figure 3-21.

- d) Thermal Control. During deployed orbital operations a passive temperature control can be employed if a carefully controlled sequence of instrument operations is maintained. The nominal duty cycle is "On" 15 minutes, "Off" 45 minutes. During instrument "On" periods, energy dissipation causes a temperature increase. During "Off" period, power is reduced to a standby level with only the command receiver in operation. The rate of temperature decay is a function of the heat dissipation characteristics of the thermal blankets surrounding the subsatellite. The rate of heat dissipation is controlled by the width of a circumferential white strip. (See Figure 3-22 and 3-23.)
- e) Ejection Scheme. The ejection method for the ESP requires orienting the Orbiter so that the ESP spin-axis orientation is in the direction required after separation nominally perpendicular to the orbit plane. The ESP is then ejected at a $\Delta V = 0.1$ mps, resulting in an initial ESP motion very close to that of the Orbiter. The Orbiter thrusters then maneuver the Orbiter relative to the ESP.



FOLDOUT FRAME



11002	FRAME

FOLDOUT FRAME 2

FOLDOUT FRAME 3

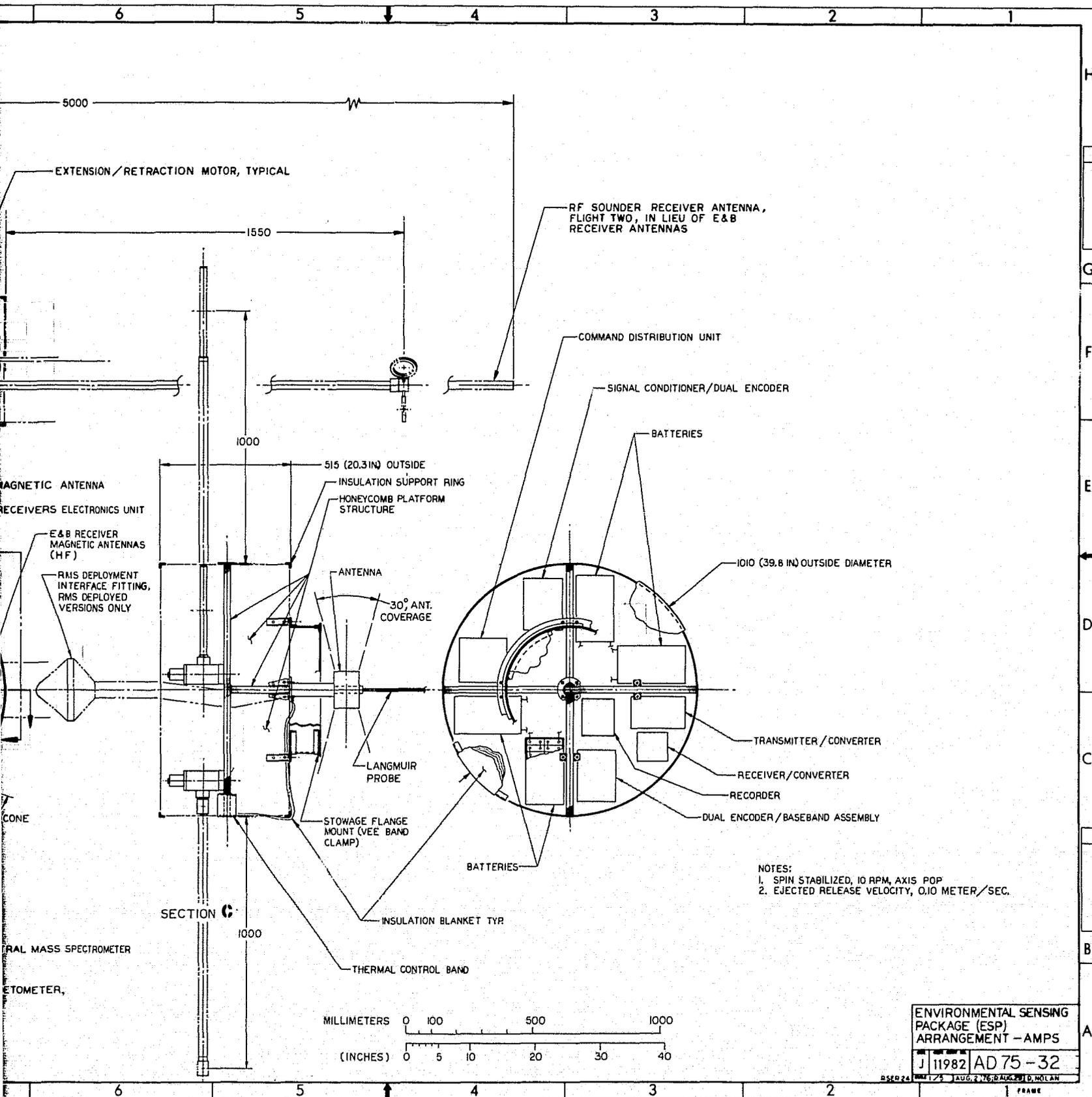


Figure 3-20. Environmental Sensing Package (ESP) Arrangement

FOLDOUT FRAME 3

Table 3-12. Mass Properties (kg)

Instruments (Maximum)	48
TT&C	9
Power and Distribution	46
Nutation Damper	1
Beacon	1
Structure	32
Total	137
Contingency (25 percent)	34
Grand Total	171

Table 3-13. ESP Electrical Power Summary

<u>Electrical Power and Distribution</u>		
Voltage:	28 vdc	
Energy:	120 amp-hr	
Battery:	Nickel-cadmium, no recharge capability in flight	
Duty Cycle:	130 watts steady state and 63 watts operating at 25 percent duty cycle	
Maximum Operating Life:	96 hours including 72 hours standby and 24 hours at standard duty cycle	
<u>Power and Energy Summary</u>		
<u>Active (25 percent Duty)</u>	Power (W)	Energy (W-hr)
Instruments	90	2,160
TT&C	36	850
Subtotal	126	3,010
<u>Standby</u>		
TT&C	5	470
Total Energy Requirements (24 hours 100 percent operation)		3,480
120-amp-hr battery		3,360
Margin (W-hr)		(-120)

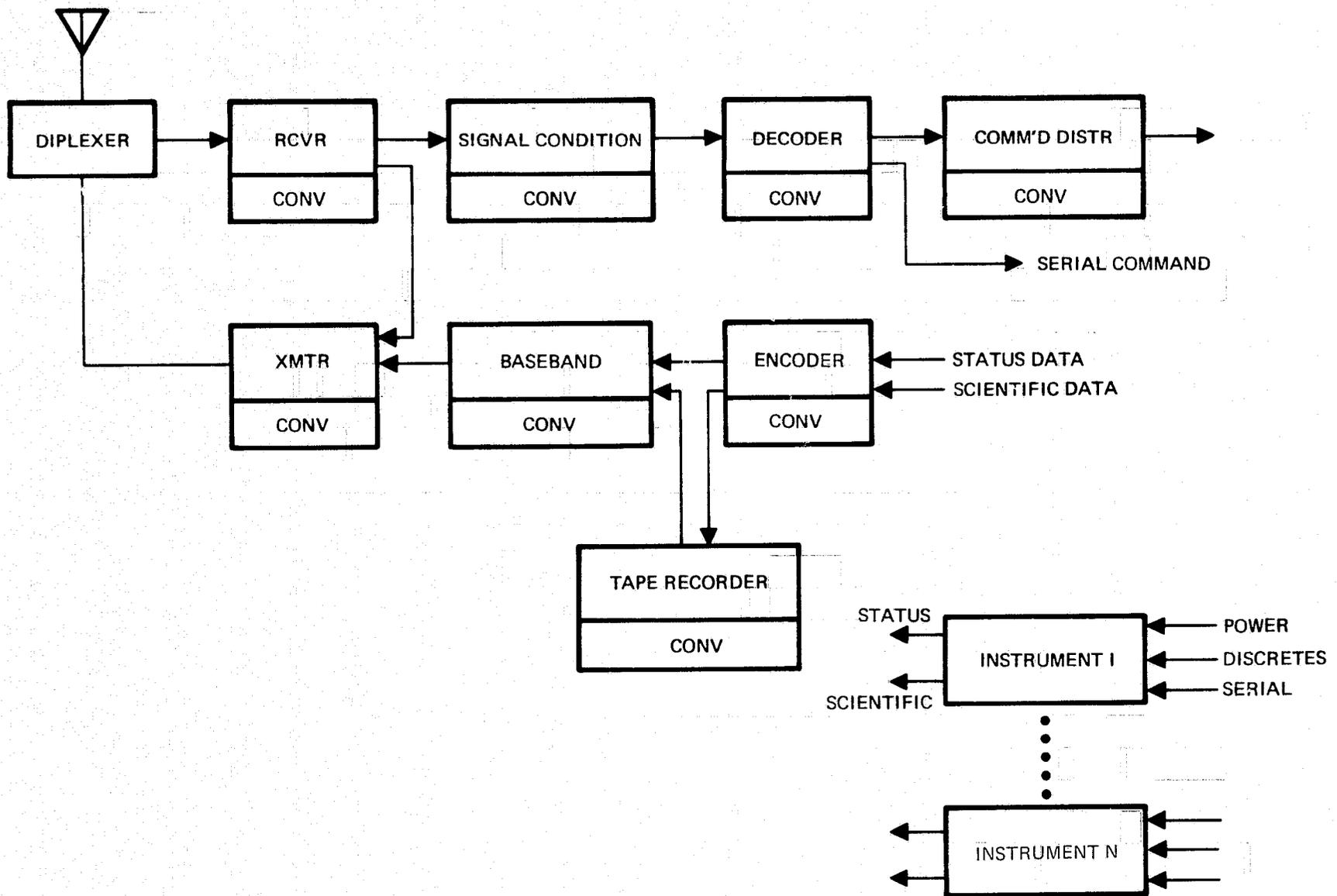
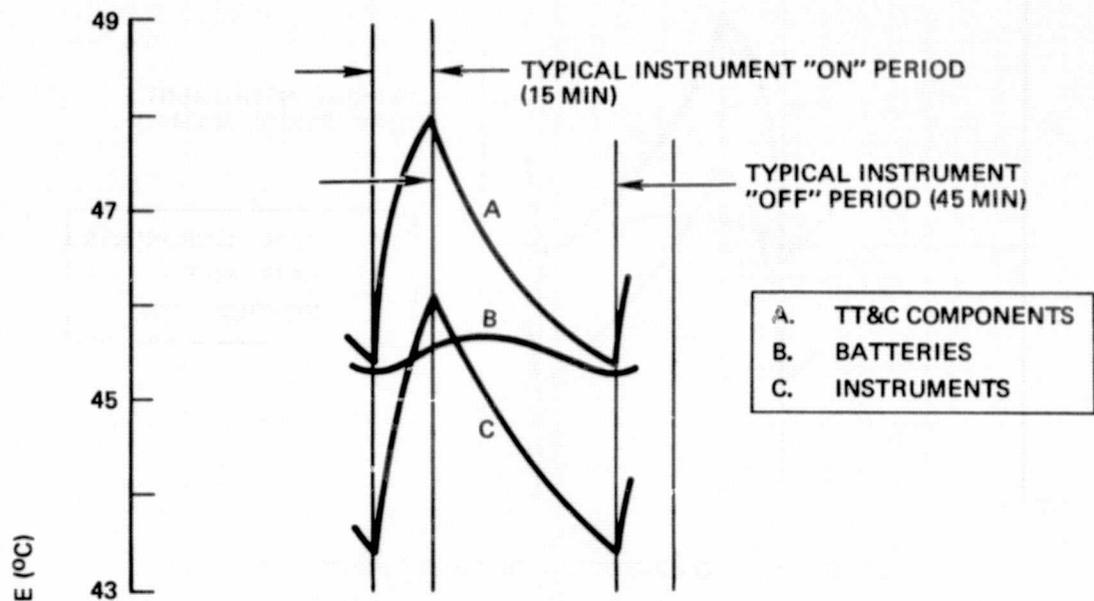


Figure 3-21. Telemetry Tracking and Command System

0.6 FT² WHITE CIRCUMFERENTIAL STRIPE
 BATTERY DISSIPATION 15% OF TOTAL LOAD
 BOOMS DEPLOYED

HOT CONDITION (SOLAR AND EARTH INPUT 9.6 WATTS)



COLD CONDITION (NO SOLAR OR EARTH INPUT)

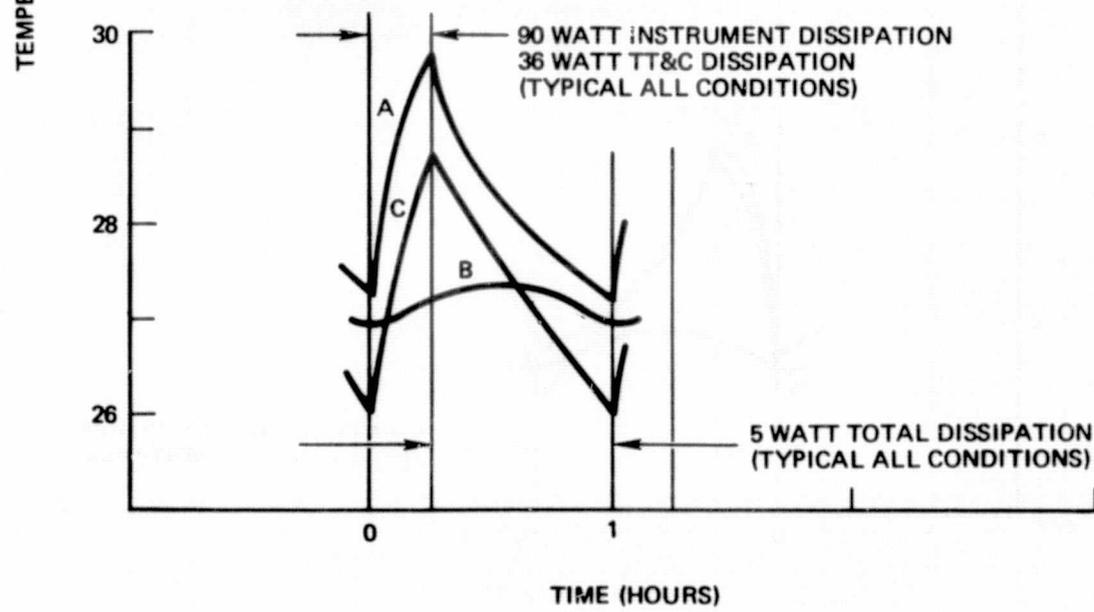
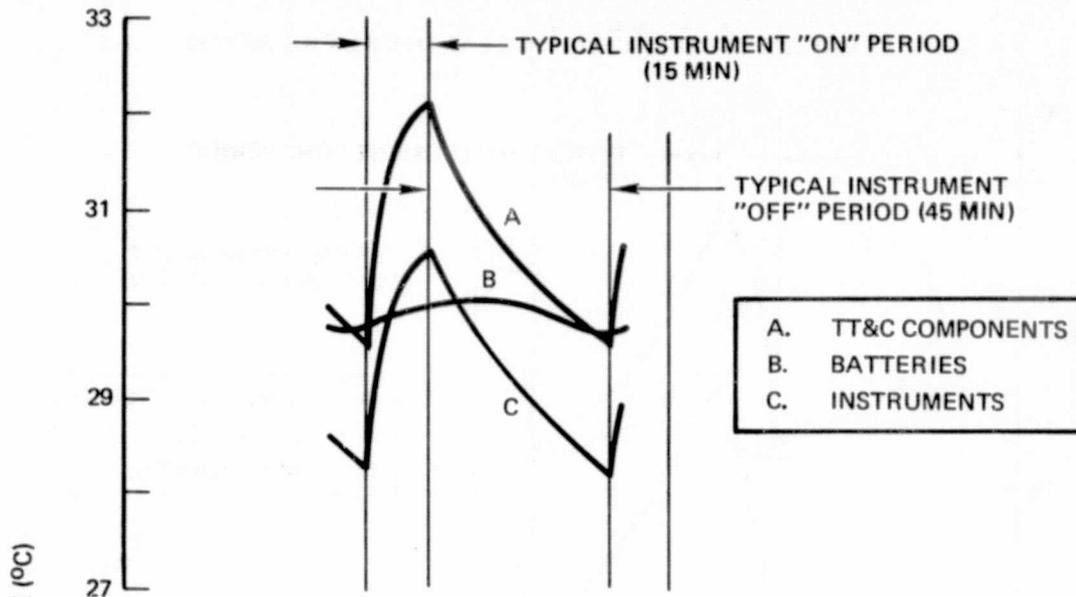


Figure 3-22. ESP Typical Operating Temperatures - A

1.2 FT² WHITE CIRCUMFERENTIAL STRIPE
 BATTERY DISSIPATION 15% OF TOTAL LOAD
 BOOMS DEPLOYED

HOT CONDITION (SOLAR AND EARTH INPUT 19.2 WATTS)



COLD CONDITION (NO SOLAR OR EARTH INPUT)

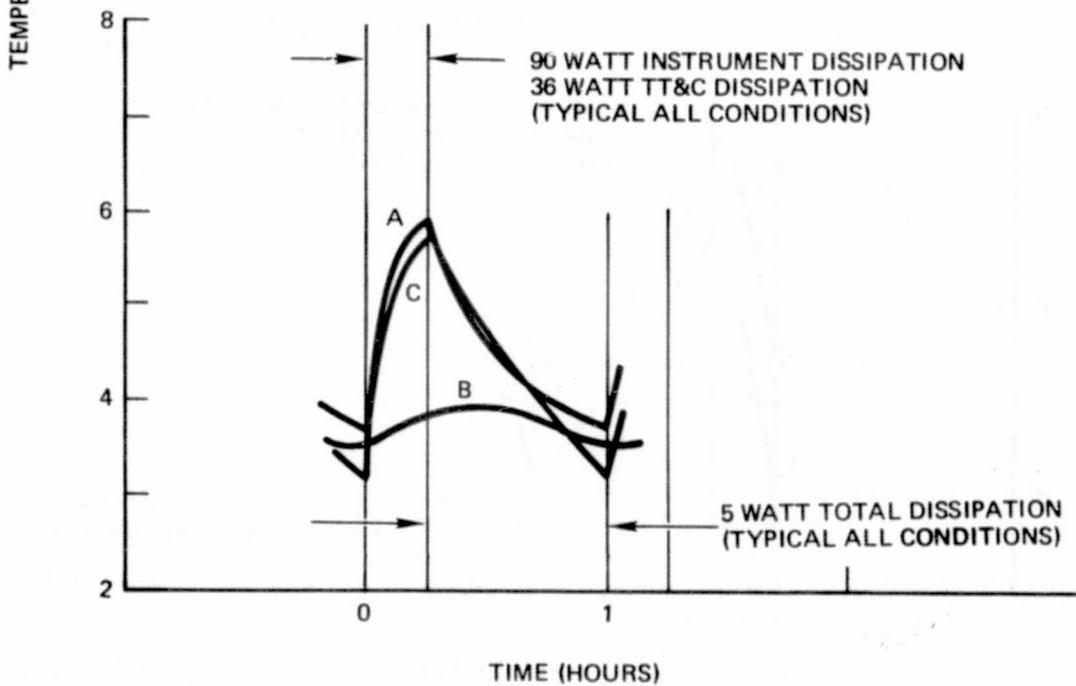


Figure 3-23. ESP Typical Operating Temperatures - B

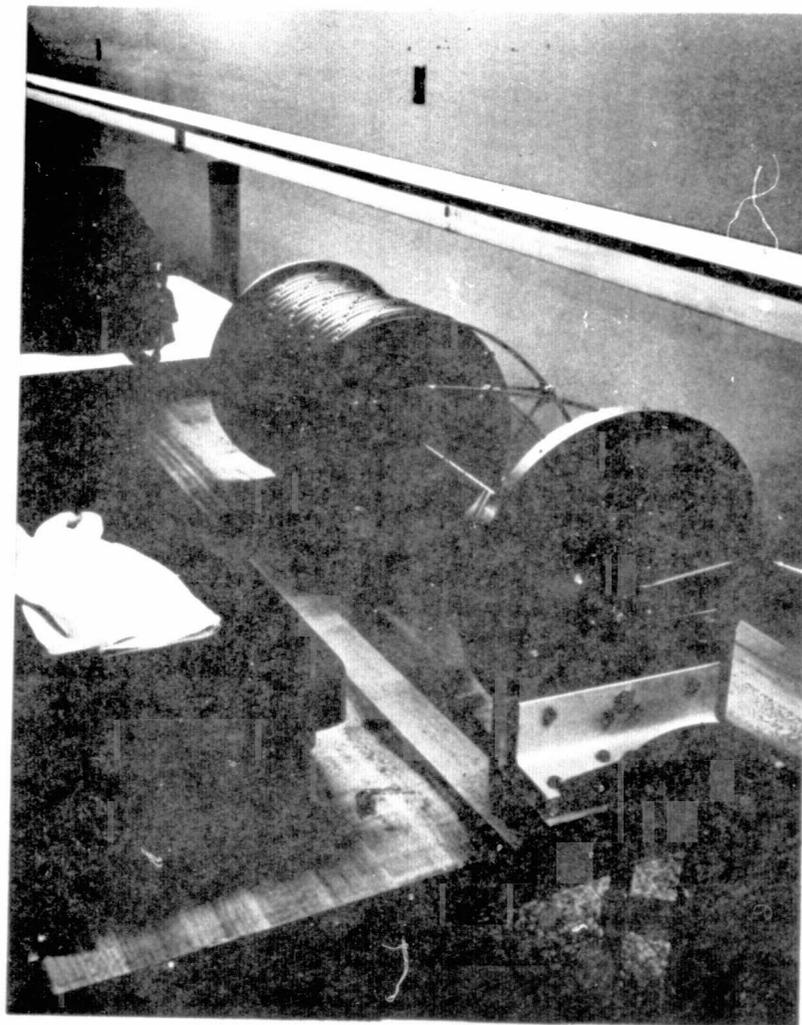
3.2.3 Magnetometer Mast

Extendible and retractable masts are required on all AMPS flights to support instruments outside of the payload bay. The magnetometers which support many of the AMPS experiments must be deployed 5 meters. A trade study of various magnetometer masts is reported in Section 4.2. In summary, the magnetometer mast which was selected is a continuous-longeron Astromast developed by Astro Research Corporation of Santa Barbara, California. It was selected because of the advanced technological status demonstrated by actual use on existing spacecraft programs. The continuous-longeron type of Astromast proposed for AMPS is identical (except for being 8 meters shorter) to the system being produced for NASA's Mariner Jupiter-Saturn spacecraft. The lattice structure of fiberglass rods, shear-stiffened by diagonal cables, is retracted by forcibly twisting it about its axis. This twisting causes the horizontal batten members to buckle and shorten. The mast is retracted by means of a motor driven lanyard. This lanyard also provides restraint as the boom is deployed by strain energy stored in the three coiled continuous longerons. The lanyard was selected over an available canister deployment option because of its lighter weight, less expense, and greater accuracy in angular positioning of the deployed boom. The system has sufficient structural strength to withstand the Orbiter RCS action and possible jettison forces even in a partially deployed state. A photograph of the NASA Marine Jupiter-Saturn spacecraft mast is shown in the partially deployed and fully deployed state in Figure 3-24. As with all devices which extend beyond the payload dynamic envelope, the 5 meter mast system incorporates a jettison device at the base as a last measure of safing the payload.

To minimize thermal twisting of the Astromast, the mast is fabricated with a pretwist over its length. This feature is quite apparent in Figure 3-24. The pretwist prevents adverse sun angles from occurring except over a small portion of the boom length, thereby reducing any thermal distortion effects.



Magnetometer Mast - Deployed



Magnetometer Mast - Partially Deployed

Figure 3-24. NASA Mariner Jupiter Saturn Mast

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3.2.4 Fifteen-Meter Mast

On flight 2, the logical grouping of instruments which required a remote location from the Orbiter or other cooperating instruments resulted in the need for a relatively strong mast 15 meters long. At the base of the 15-meter mast system is an antenna rotator and separation system.

The main mast utilizes an articulated-longeron type of Astromast developed by Astro Research. Although other beam designs are available, the unique retractability and partially deployed load carrying capability of this Astromast makes it a logical choice for the main mast.

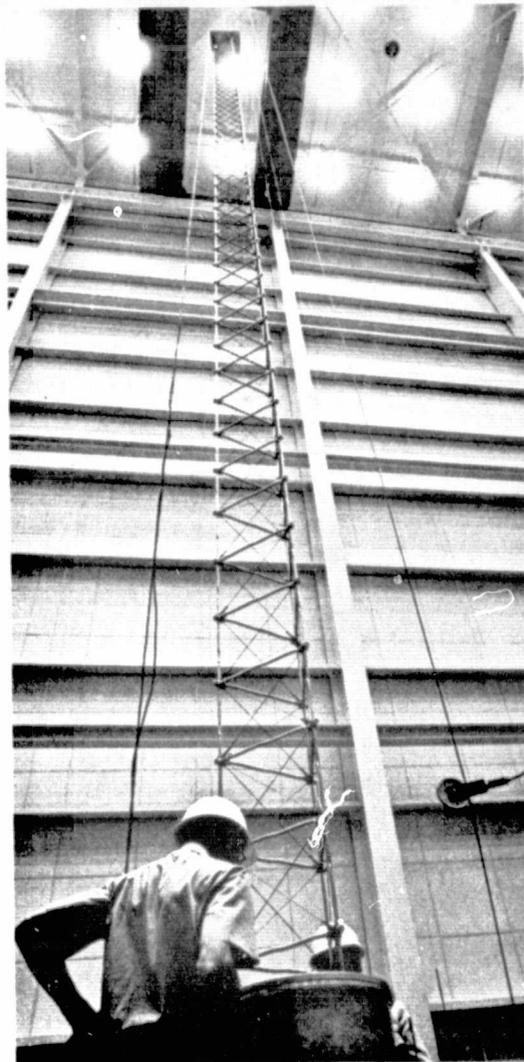
In this type of application, the longerons are in segments and connected with hinge points to the batten frames. The assembly is stiffened in shear and torsion by diagonal cables extending across each rectangular face in the lattice structure. Three of the six tension cables terminate at one end (bottom of the bay) in an "over-center" toggle-joint locking mechanism.

Deployment and retraction of the mast takes place from the cylindrical canister/actuator. The upper external portion of the canister is a rotatable nut with three sets of inward facing rectangular threads. The internal stationary part of the canister supports three vertical guide slots. Rollers attached to the batten corners, are guided by tracks to be simultaneously engaged between the vertical guide slots and lands of the threaded nuts. The beam deploys or retracts into the canister when the nut is rotated by electric motors. Cams for latching and unlatching the diagonal linkages of the beam are fixed to the inner canister wall just below the rotating nut. Photographs of this type of device are shown in Figure 3-25.

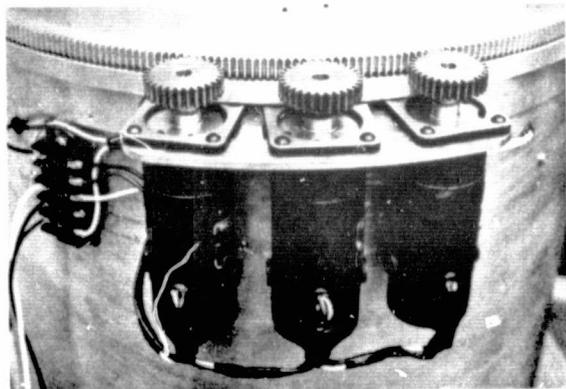
The instrument package, mounted at the outboard end, contains a number of instruments and a wake and sheath test body. From a design standpoint, the most difficult accommodation is that of the RF sounder antenna. This dipole antenna must be unfurlable and has a length of 100 meters tip to tip. The design solution identified for this antenna is to utilize a flight-proven interlocking tubular extendable boom. A photograph of this device is shown in Figure 3-26.



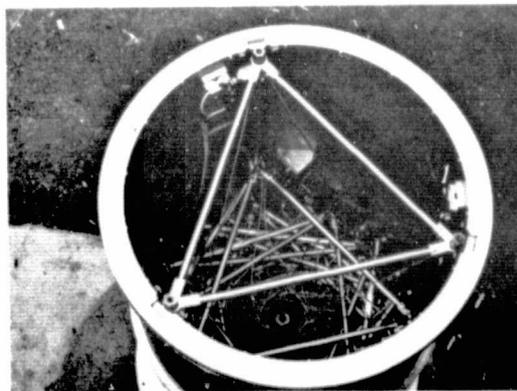
Mast Deploying Automatically



Fully Extended Mast



Deployment Motors



Retracted Configuration
400 lb Mast plus Canister

Figure 3-25. Fifteen Meter Mast

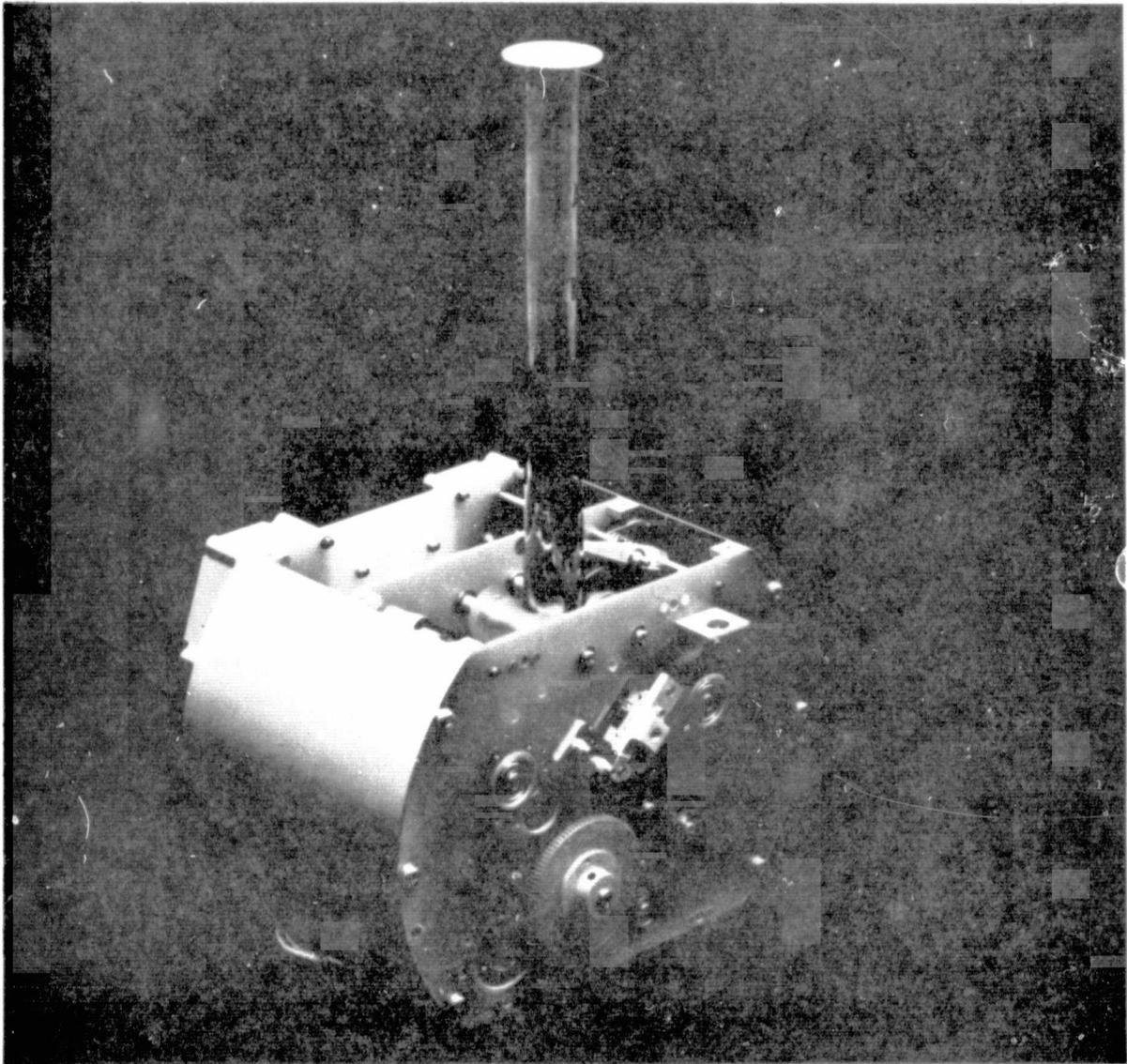


Figure 3-26. RF Sounder Boom

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3.2.5 Gas Release Functional Assembly

The requirements for this equipment were generated by the Acoustic Gravity Wave Experiment (EOR-29) and the Gas Cloud Expansion Dynamics Experiment (EOR-12). Both experiments involve the rapid expansion of gases in the vicinity of the Orbiter and over pre-determined ground sites. During Flight 1, up to seven 50 to 100 kg releases were projected. In the following sections, the design requirements imposed directly by the experiments and indirectly by Spacelab/Orbiter requirements are summarized.

3.2.5.1 Operational Requirements

The operational requirements for this equipment are:

- a) Explosive release of 50-100 kg of gas traveling at orbital velocity at an altitude of 150-300 km at a pre-programmed location over Arecibo, Puerto Rico, and in the vicinity of the Orbiter.
- b) Observation of the release process using the Optical Band Imager Photometer System located on SIPS.
- c) Capability to perform this experiment on more than one orbital pass over Arecibo during Flight 1.
- d) The selected gas spreads rapidly to provide maximum acoustic acoustic gravity wave excitation.

3.2.5.2 Design Requirements

The design requirements for this equipment are:

- a) The selected design meets the approved safety standards of Space Transportation System Payload Safety Guidelines Handbook.
- b) The release system is compact and can be accommodated along the pallet sills.
- c) Released gas shall not contaminate the payload or Orbiter, whether released by deliberate expulsion, leakage, or accident.
- d) For accurate placement, the maximum time from separation of the canister from the Orbiter to release of the gas should not exceed one orbit (1-1/2 hours).

3.2.5.3 Design Analysis

Specific design characteristics of the gas release functional assembly are:

- a) Release materials — See Table 3-14 for an analysis of release materials. The only acceptable materials are stored in gaseous form because energy must be added to vaporize liquids. Since there are several candidates that are acceptable, the release container must handle a range of selections. For design analysis of Flight 1, xenon was selected.
- b) Release container — The container consists of titanium material designed to a 2-to-1 safety factor over standard design practice. The leak-before-burst concept was used.
- c) Container maximum pressure — Design pressure of xenon is 50 atmospheres and quantity stored is limited to 50 kg to prevent cargo bay over-pressure in the event of high-rate leakage.
- d) Gas release mechanization — High-speed valves on opposite sides of the tank are used to minimize tank impulse during release. The tank design has no parts ejection during release.
- e) Separation distance at release — Releases are at 1.0 meter per second from the Orbiter so a separation distance of 15-20 km exists at gas expulsion.
- f) Separation method — The low separation velocity of 1.0 meter per second allows the use of a spring separation mechanism. Pressurized gas and solid propellant separation were discarded after cursory examination.
- g) Number of releases — During the 7-day Orbiter flight period, its orbit can be adjusted for a total of six passes over Arecibo. Due to the pressure of other activities, however, a maximum of five separate releases is practicable. The design, therefore, incorporates five release containers.
- h) Release timing — Timing of the gas release over Arecibo can be accomplished by radio command from the Orbiter or by a timer installed on the release itself. A command receiver is obviously more costly than a timer. Therefore, if the latter is sufficiently accurate, it should be selected unless variable timing is necessary for experimentation. For this application, the timing accuracy should be ± 2 seconds in 1-1/2 hours or about one part in 10^4 , a value easily achievable with a mechanical or digital clock timing mechanism.

Table 3-14. Release Material Analysis

Release Material*	Comments	Mass gm/mols	Acceptable
Liquid N ₂	Probably will not gasify rapidly	28	No
N ₂	Relatively low molecular weight, medium high gamma, would require high volume or pressure	28	Yes
CO ₂	Relatively low molecular weight, medium gamma, would require high volume	48	Yes
Water	Probably will not gasify rapidly	18	No
CO ₂ - Solid	Probably sublimates too slowly	48	No
Explosive Material	On-board explosive hazard, not total gasification, possible low gamma constituents, interacting gas is hot	50	No
Barium Thermite	10% to 70% gas production, interacting gas hot, high mass + high gamma on-board explosive hazard, may ionize	137	No
SF ₆	High mass, low gamma, may ionize quickly	146	No
WF ₆	Very high mass, low gamma may need heating to remain gaseous	298	Yes
Xe	Inert, high mass, high gamma	131	Yes

* Other materials were considered but were quickly discarded as being unfeasible, such as H₂, He, and Ne, because of safety, containment, or cost.

3.2.5.4 Preliminary Design

The gas release functional assembly consists of gas release canisters and a cradle support and launcher subassembly.

3.2.5.4.1 Pressure Vessel. Geometrically, the pressure vessel configuration consists of two hemispheres welded together along with a relatively large through-tube located on the polar axis which contains the operational gas release valve mechanism. (See Figure 3-27.) One end of the through-tube is adapted to engage the mounting support and operational launch cradle assembly.

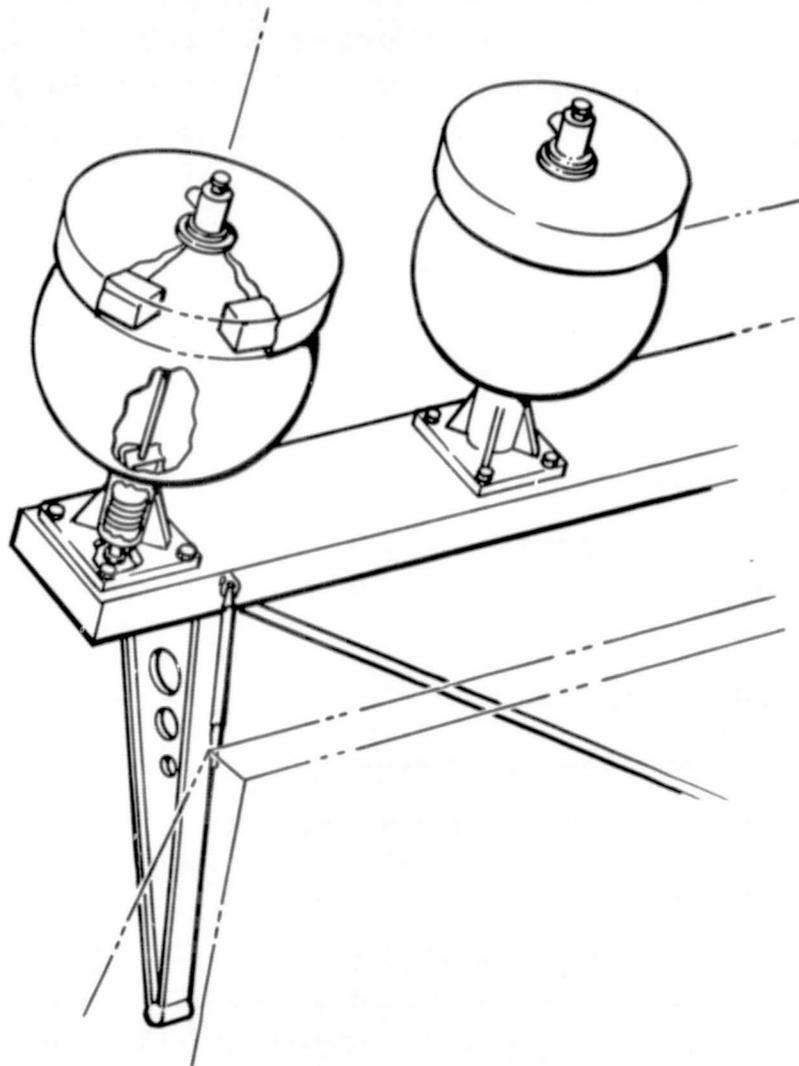


Figure 3-27. Gas Release Assembly

Particular attention was paid to the problem of the complex deflection behavior under pressure which results from incorporating the through-tube.

The pressure vessel is equipped with a pressure relief valve, a remotely controllable pressure dump valve, and a full range pressure transducer so that the pressure can be monitored. The pressure vessel is designed for 735 psig working pressure.

3.2.5.4.2 Release Valves. The operational gas release valve assembly consists of two poppet type valves located symmetrically and opposite to each other and within the through-tube in the pressure vessel, see Figure 3-28. The valves are held closed against each other by a preloaded tie rod which is secured by an ordnance operated separation nut, such as manufactured by Hi-Shear Corporation. Gas release is accomplished by operating the separation nut which releases the tie rod, permitting both valves to be driven open by the internal gas pressure. The gas escape routes end in directions tangential to the pressure vessel so that the resulting rotation contributes, to a small degree, to the behavior uniformity in the forming of the gaseous sphere.

An independent safety device is provided that prevents catastrophic release of gas if the separation nut is accidentally operated.

3.2.5.4.3 Equipment Platform. An equipment mounting platform and enclosure is attached to the pressure vessel. The equipment complement consists of:

- a) Battery power supply
- b) Ordnance power and relay unit
- c) Timer (or command receiver and antenna)
- d) Beacon.

3.2.5.4.4 Cradle Support and Launcher. The cradle support and launcher consists of a structural support tube that engages a mating protuberance on the pressure vessel structure. A compressed coil spring, trapped within the tube, furnishes the energy to launch the canister vehicle at 1 meter per second. No attitude stabilization is required; therefore, no spin motion is imparted during the launch stroke. However, if attitude

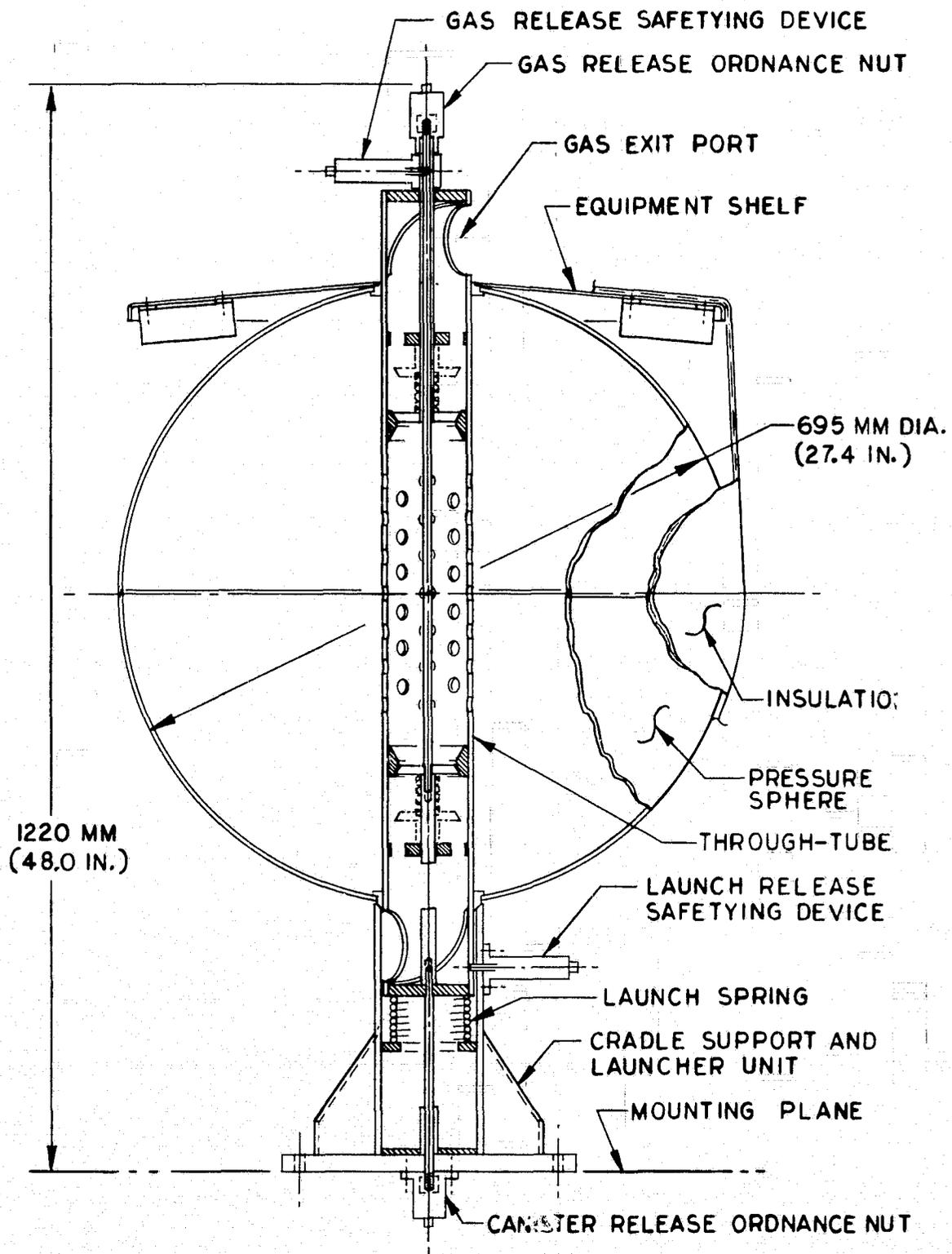


Figure 3-28. Gas Release Canister

stabilization were desired, it could be provided by mechanically imparting the necessary spin rate during the launch stroke and increasing the spring force accordingly.

The canister is retained in the cradle by a preloaded tension rod running through the center of the support tube, secured by an ordnance operation separation nut, such as manufactured by Hi-Shear Corporation.

An independent safety device is provided that prevents release of the canister if the separation nut is accidentally operated. An electrical umbilical is provided that connects the pressure transducer, the gas release valve safety unit, and the battery charge maintenance system. Also, provisions for switching to canister battery power are incorporated in the umbilical.

3.2.5.4.5 Platform Support. A platform support structure is provided that positions the canister relatively high and close to the sill in the Orbiter payload bay, to maximize space available for center-mounted payload items. The structure provides for three canisters and can be placed on either side. Flight 1, however, shows only five canisters with one of the positions being occupied by the magnetometer mast.

The structure consists of a box construction platform mounted on two beams pivoted at the attach points on the pallet. The unit is stabilized with two struts tied to the pallet and necessary stabilizing struts between the support beams.

3.2.5.4.6 Electrical Subassembly. Electrical power for the gas release is supplied through the EPDS and by small batteries located on each gas release. The Spacelab power system supplies all power for checkout monitoring and ejection. A silver-zinc battery supplies power to the timing and valving mechanism after release ejection from the Orbiter.

3.2.5.4.7 Thermal Control. The xenon gas in the release canisters and the electromechanical devices on the release canisters require temperature control. The required gas temperature at separation from the Orbiter is 70°F. This is achieved by use of standby heaters which are part of the platform support structure. (See Figure 3-29.) Xenon canisters and control equipment are covered with multilayer insulation to minimize temperature decay after ejection from the Orbiter. Figure 3-30 gives an estimate

PRIOR TO CANISTER RELEASE

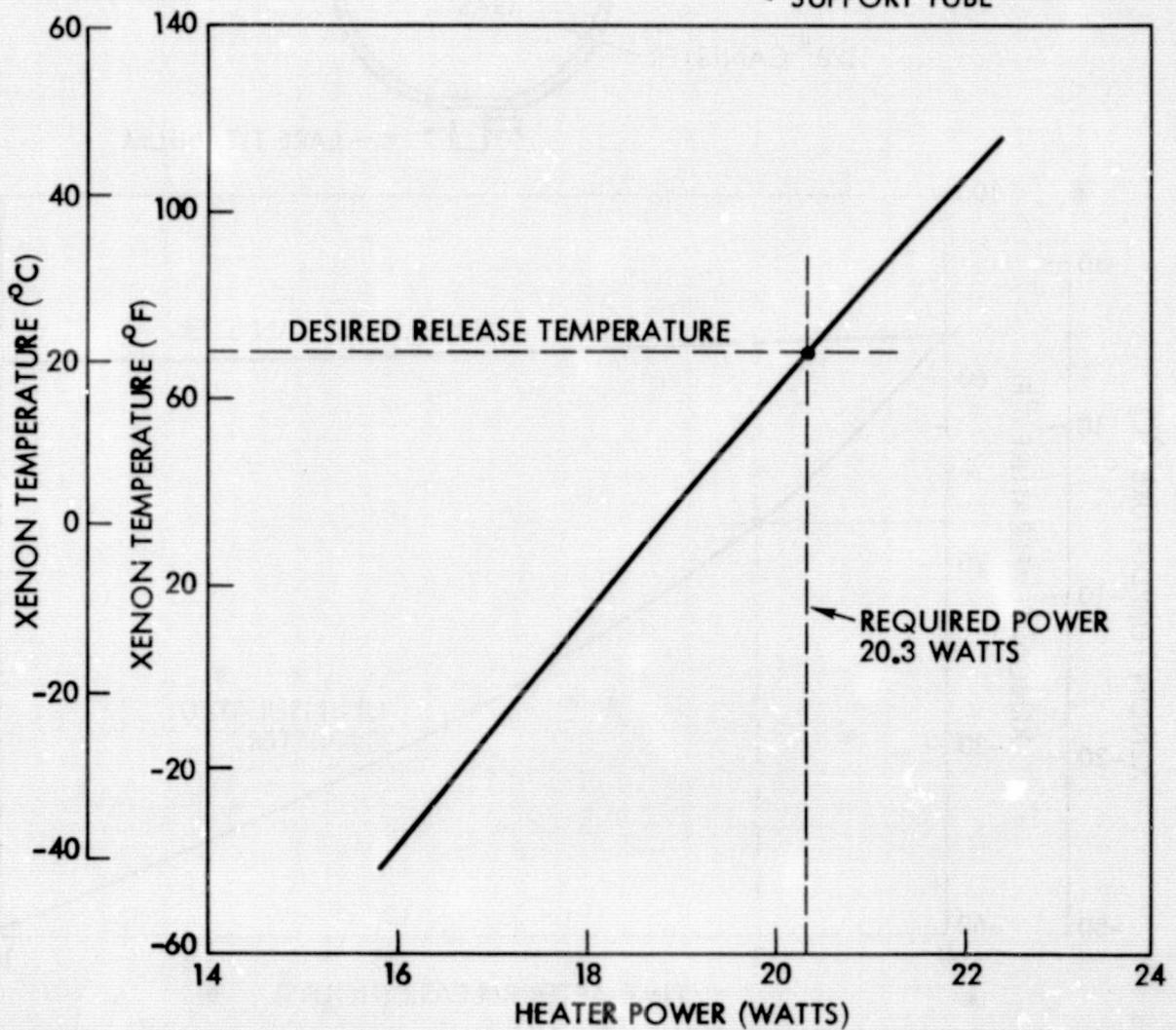
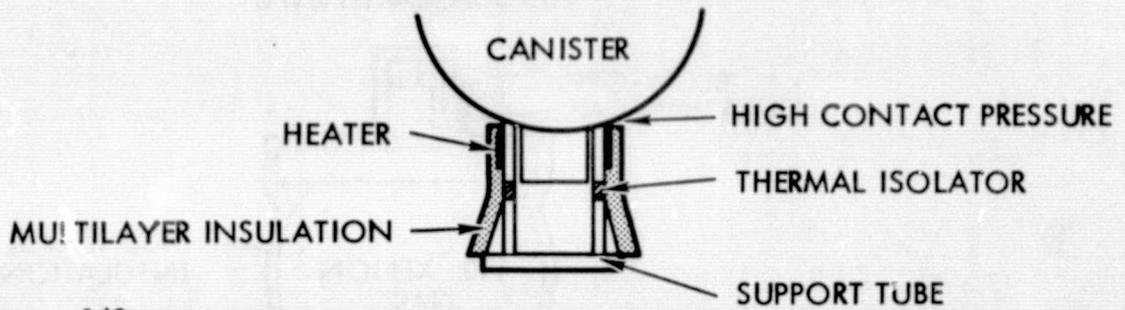


Figure 3-29. Canister Standby Heater Power Requirement

C-2

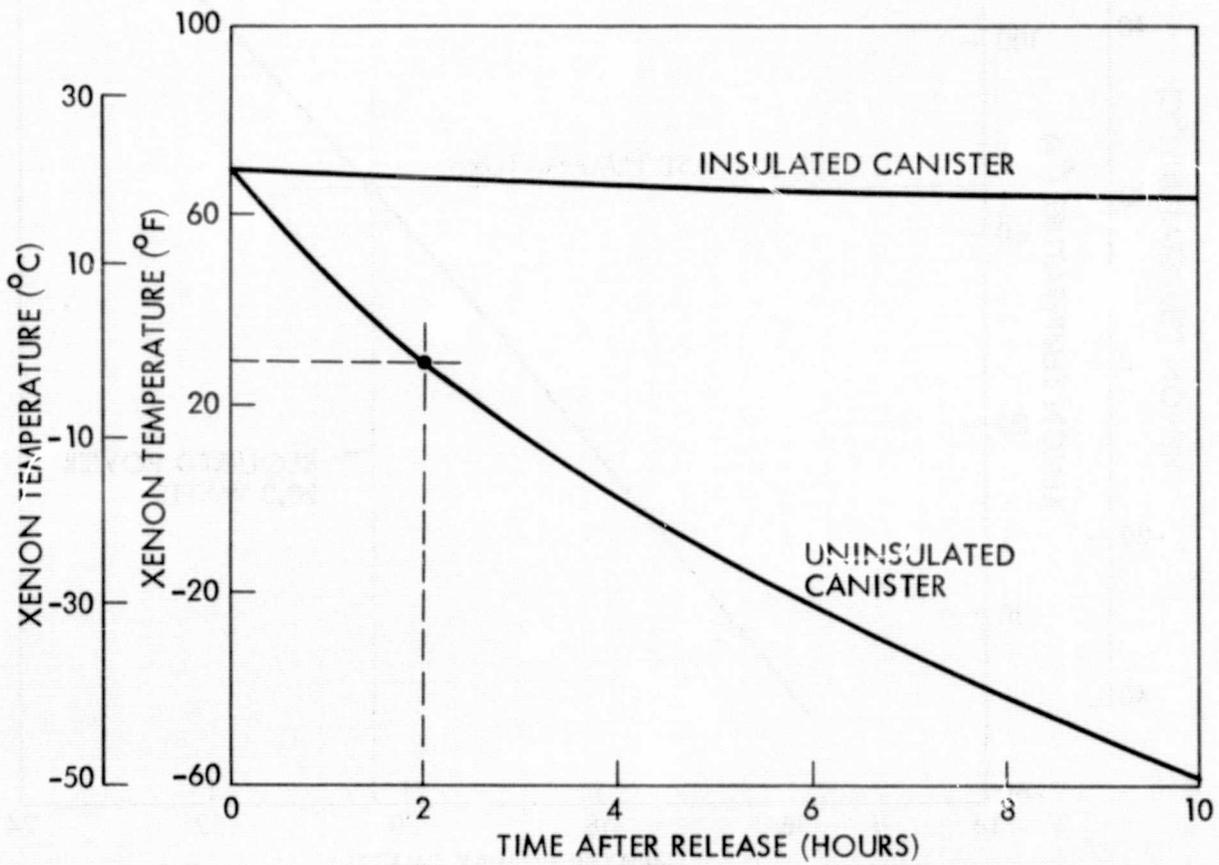
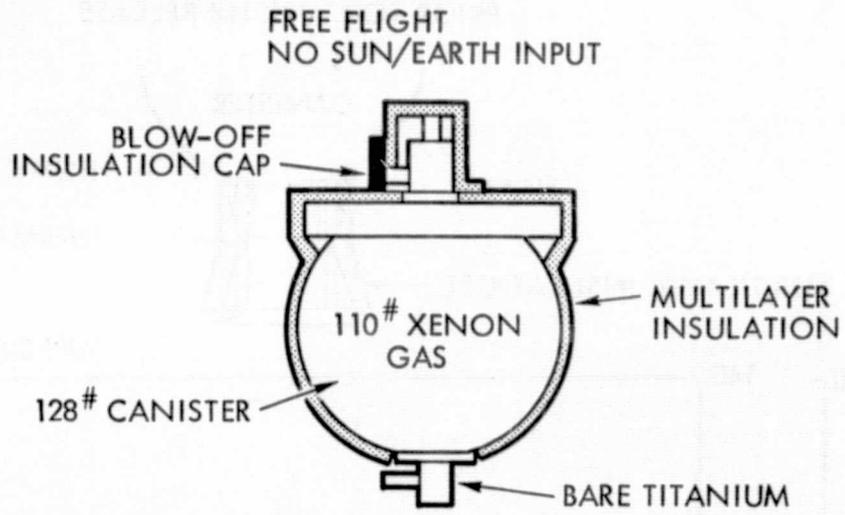


Figure 3-30. Canister Temperature Decay

of the temperature decay after ejection from the Orbiter. After a maximum estimated time of 2 hours from canister ejection to gas release, the xenon temperature has decayed to approximately 30°F.

3.2.6 Barium Thermite Release Assembly

3.2.6.1 Operational Requirements

This release supports the low altitude conductivity modification experiment on AMPS Flight 2. The objective is to release a large quantity of ionizable material that will cause conductivity changes and could lead to artificial aurora, magnetic/electric field changes, or trapped charged particle density changes. Barium thermite canisters are released and ignited. There should be approximately 20-16 kg canisters and they should be ignited in a line approximately 1 km apart to cause an extended cloud. The phenomenon is observed by passage of the Orbiter through the gas cloud 1 to 2 minutes later. Observations from the ground are required. The altitude at release ignition should be 180-200 km.

3.2.6.2 Design Analysis

There are two basically different design approaches to achieving the required separation of the barium thermite canisters from the Orbiter and achieving the 1 km spacing. The first (which can be discarded as impractical) is to eject each individual canister with a sufficient velocity impulse to achieve ignition conditions. In order to achieve the required initiation conditions, each canister must be ejected with a minimum ΔV of 6 mps in the retrograde direction. After 4.7 to 9.4 revolutions, all canisters would then be activated simultaneously. To assure a relative spacing of 0.1 km, the velocity error cannot exceed 9×10^{-4} mps. This value is probably impossible to achieve with mechanical or pneumatic ejectors. A second approach is to house all barium thermite canisters in a bus structure which is spun up and ejected at a low velocity relative to the orbiter. The Orbiter Maneuvering System (OMS) is employed to accelerate the Orbiter into a transfer orbit away from the bus. The canisters are ripple ignited by radio command when the required conditions are achieved.

The design characteristics of the system are given in Table 3-15. Since barium thermite is a hazardous material, every precaution must be taken to prevent accidental ignition and to control the hazard in the event of accident. One safety feature is to employ a thermal shield around the bus on the pallet.

Table 3-15. A Barium-Thermite Functional Assembly Design Characteristics

- | |
|---|
| 1) Spin-stabilized bus |
| 2) 20 canisters plus ripple fire sequencer |
| 3) Released at 200 km altitude |
| OMS retro - burn |
| 1 m/sec spring deployment |
| OMS forward burn |
| 4) Design safety concepts |
| Mechanical safe/arm device |
| Thermal shield |
| Redundant sequencer control
(command plus timer) |
| Remotely activated igniter battery |

3.2.6.3 Preliminary Design

The barium thermite release functional assembly consists of the bus assembly with 20 barium thermite canisters, a command subassembly, and a spin table/cradle assembly for mounting the bus assembly to the Orbiter pallet; see Figure 3-31. The spin table/cradle assembly will mount to the pallet hard points on the horizontal surface of the pallet. The entire functional assembly is designed to occupy less than one-half a pallet. The entire assembly is 1.5 meters high by 1.5 meters in diameter.

The functions performed by the spin table/cradle assembly are:

- a) Attachment of the barium thermite functional assembly to the pallet
- b) Retention of the barium release bus during flight and during landing in the event of an abort
- c) Spin-up and ejection of the bus on command
- d) Provision of a thermal and splatter shield as protection against accidental barium thermite ignition

- e) Thermal control of the bus during orbital operations until ignition
- f) Provision for lifting and handling during integration operations
- g) Interfacing electrical power and signal and control with the Spacelab.

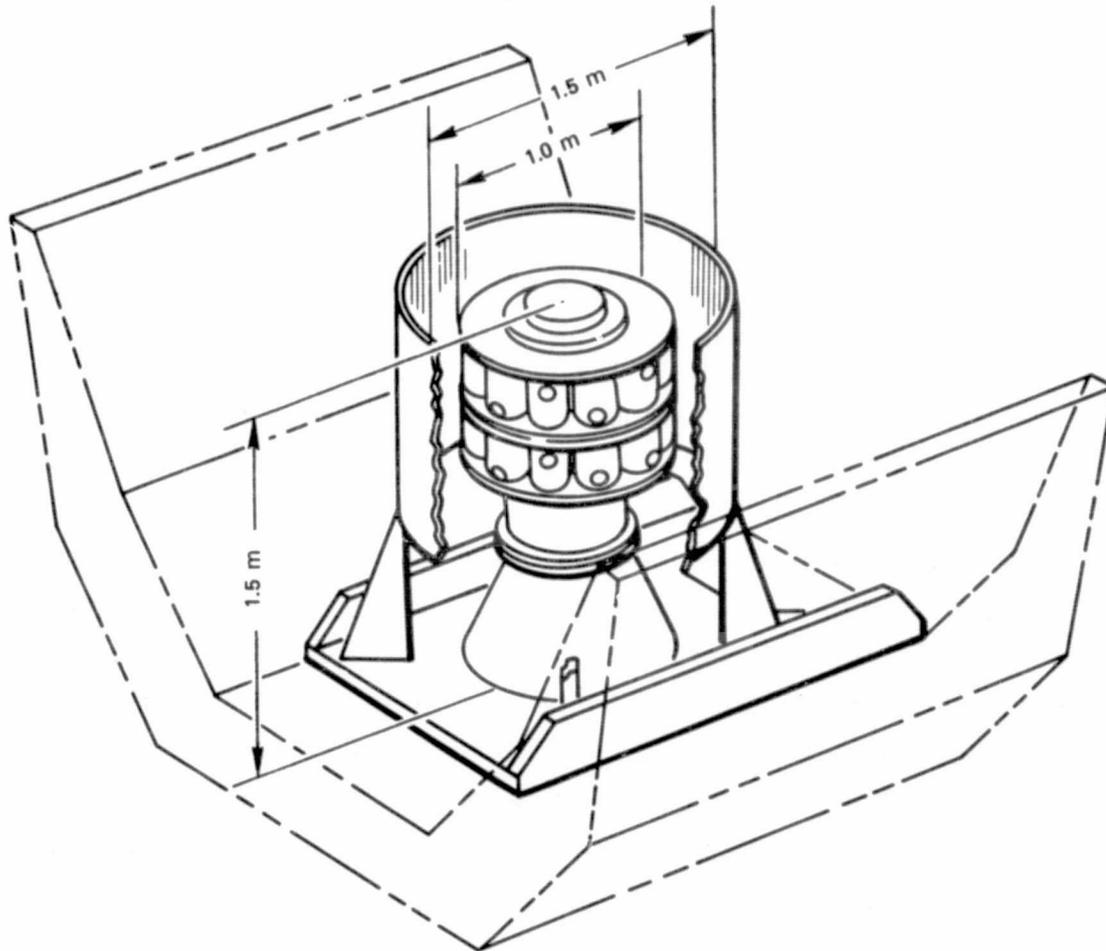


Figure 3-31. Barium Thermite Functional Assembly

The barium thermite bus is of spin-stable design and provides the structure for accommodation of the 20 barium thermite canisters, the command receiver, and ignition and safety equipment. The distance to the Orbiter at ignition of the releases will range from 400 to 1000 km. Therefore, the most practical command source is at the ground observation station. The releases may be ignited simultaneously or ripple fired to provide an elongated cloud.

4. SUBSYSTEM DESIGN

4.1 POINTING AND CONTROL SUBSYSTEM

The pointing and control subsystem encompasses those elements which perform the pointing and stabilization of the AMPS scientific payload. The main function of the subsystem is to control the gimballed experiment pointing mount and to provide attitude transfer between pallet mounted instruments. It consists of attitude sensors, on-orbit alignment devices, control processing software and electronics, and gimbal drive mechanisms. The pointing and control subsystem is of particular importance to AMPS since it must demonstrate the capability to correctly point and stabilize the scientific instruments so that useful experiment data can be obtained.

The design of the pointing and control subsystem must:

- Satisfy instrument pointing and stabilization requirements
- Satisfy the requirements for the gimballed experiment pointing mount to be used for AMPS
- Provide a stable and accurate attitude reference system
- Control the experiment pointing mount in the presence of Orbiter limit cycle motion, crew motion, and hinge friction disturbances.

The pointing and control subsystem configuration baselines the Small Instrument Pointing System (SIPS) as the only instrument pointing mount that is required. Precision attitude reference is obtained from a stellar inertial attitude reference system employing gyros and a strapdown star tracker, all mounted to the SIPS canister. Attitude transfer to and between pallet mounted instruments makes use of the SIPS resolvers and an optical alignment system. The majority of the software and electronic data processing functions are performed by programmable digital electronics (PDE). The configured pointing and control system meets and exceeds the performance requirements of the cryogenic instrument, the design driver. It has been demonstrated that the SIPS can be pointed with 95 per cent confidence to within 18.8 arc-sec (2σ). The short-term (10 to 20 seconds) pointing stability error is less than 0.5 arc-sec (2σ).

The following sections discuss the pointing and control subsystem in greater detail, using the cryogenic instrument as the primary design example since it has the most stringent pointing requirements. Section 4.1.1 provides a general description of the system, Section 4.1.2 focuses on system design problems and their solutions, and Section 4.1.3 presents performance assessment by simulation.

4.1.1 General System Description

The Flight 1 instrument pointing requirements are listed in Table 4-1. The instruments listed in the upper portion of the table are mounted on the gimballed experiment mount, the SIPS, the ones in the lower portion, are hard mounted to the pallets. The cryogenic instrument must raster scan through 100 km above the earth horizon at a 0.36 deg/sec rate. As can be seen, it has the most stringent pointing and stabilization requirements among the SIPS mounted instruments. The design driver for the pallet-mounted instrument is the attitude determination accuracy of 2 degrees of the vector magnetometer relative to the electron accelerator. It imposes the most stringent demands on the attitude transfer system because the accelerometer and the vector magnetometer are located on different pallets.

Three different experiment pointing mounts were considered for AMPS:

- 1) The Instrument Pointing System (IPS)
- 2) The Small Instrument Pointing System (SIPS)
- 3) The Mini-Mount Instrument Pointing System (MIPS).

The IPS is being developed in Europe under ESA's direction and is primarily configured to accommodate large (2000 kg) instruments for stellar-inertial and solar pointing. The IPS is not a balanced mount; that is, the payload (instrument) center of mass (CM) does, by design, not coincide with the gimbal hinge point. This has been done to accommodate large instruments on the IPS pointing platform without suffering spatial interference from the gimbal structure to the CM/hinge point offset, Orbiter limit cycle and crew motion become significant disturbances on the pointing mount. To "filter-out" high frequency content

disturbances (any sudden disturbance contains high frequency components), the IPS has been soft mounted on a spring and damper system (shock mounted). This shock mount cannot be tuned to the requirements of much smaller instruments and tends to adversely affect pointing performance of any instruments with raster and tracking requirements, such as the cryogenic limb scanner on AMPS.

Table 4-1. Flight 1 Instruments Pointing Requirements

INSTRUMENT NAME	INSTRUMENT GROUPING OR LOCATION	POINTING REQUIREMENTS					
		REFERENCE	FIELD OF VIEW	OVERALL ACCURACY	ATTITUDE DETERMINATION ACCURACY	STABILITY OR STABILITY RATE	TRACKING RATES
CRYO LIMB SCANNER	SIPS (B)	EARTH LIMB	±90°	120 SEC	20 SEC	20 SEC	0.36°/SEC
CRYO IR INTERFEROMETER	SIPS (B)						
SOLAR FLUX	SIPS (A)	SUN	HEMISPHERE	15 MIN	90 SEC	90 SEC	
OBIPS	SIPS (A)			0.1°	0.1°	1°/SECONDS	
LIDAR	PALLET	NADIR		4°	2°	6°/SECONDS	
CHEMICAL GAS RELEASE	PALLET			3°	2°	1°/MINUTE	
ELECTRON ACCELERATOR	PALLET	B VECTOR	±60° TO B	3°	2°	1°/SECONDS	
VECTOR MAGNETOMETER	PALLET 5M BOOM	B VECTOR TO ACCELERATOR COORDS			2°		

SIPS (A) = PORTSIDE
SIPS (B) = STARBOARD

The SIPS will be developed in the United States under NASA's direction. It is designed as an Orbiter-borne balanced instrument pointing mount (payload CM and gimbal hinge point nominally coincide), is hard mounted, and accommodates smaller instruments weighting up to about 600 kg. It can raster and track, as well as serve both stellar-inertial pointing missions and earth pointing missions.

The Mini-Mount is basically a minaturized IPS and has been proposed for development by NASA Marshall Space Flight Center. It would use an existing gimbal structure from a Bendix-built star tracker that was used on the Skylab program with the Apollo Telescope Mount. Besides being somewhat too small for accommodating the cryogenic instrument, its unbalanced payload suspension and its associated shock-mounting requirement interfere unfavorably with the raster and earth oriented tracking requirements as they exist on AMPS.

The above considerations led to the decision to baseline the SIPS as the instrument pointing mount to be used on AMPS.

4.1.2 System Design

Figure 4-1 shows a block diagram of the SIPS command and error processing functions performed by the SIPS PDE. The target command and Orbiter ephemeris are the only inputs required from the Spacelab computer. The target command is given in a user convenient coordinate frame that is identified in the command. For the cryogenic limb scanner experiment, the earth pointing orbital reference frame would probably be most suitable. Using the Orbiter ephemeris, the target command is first converted to inertial coordinates. This allows uniform command processing from this point on, because the gyros and star tracker provide an inertial attitude reference, resulting in the cleanest software. It also makes the pointing and control subsystem applicable to any Labcraft mission (not only AMPS) because it does not matter to the subsystem whether the SIPS is commanded to point to the earth limb, whether it is tracking a target fixed to the earth surface, or whether it is pointing inertially. The command preprocessing also computes commanded inertial rates $\dot{\omega}_C$. The commanded angular acceleration is converted to a feed forward gimbal torque command in the control law, thereby significantly improving pointing performance by reducing the required bandwidth for pointing servo.

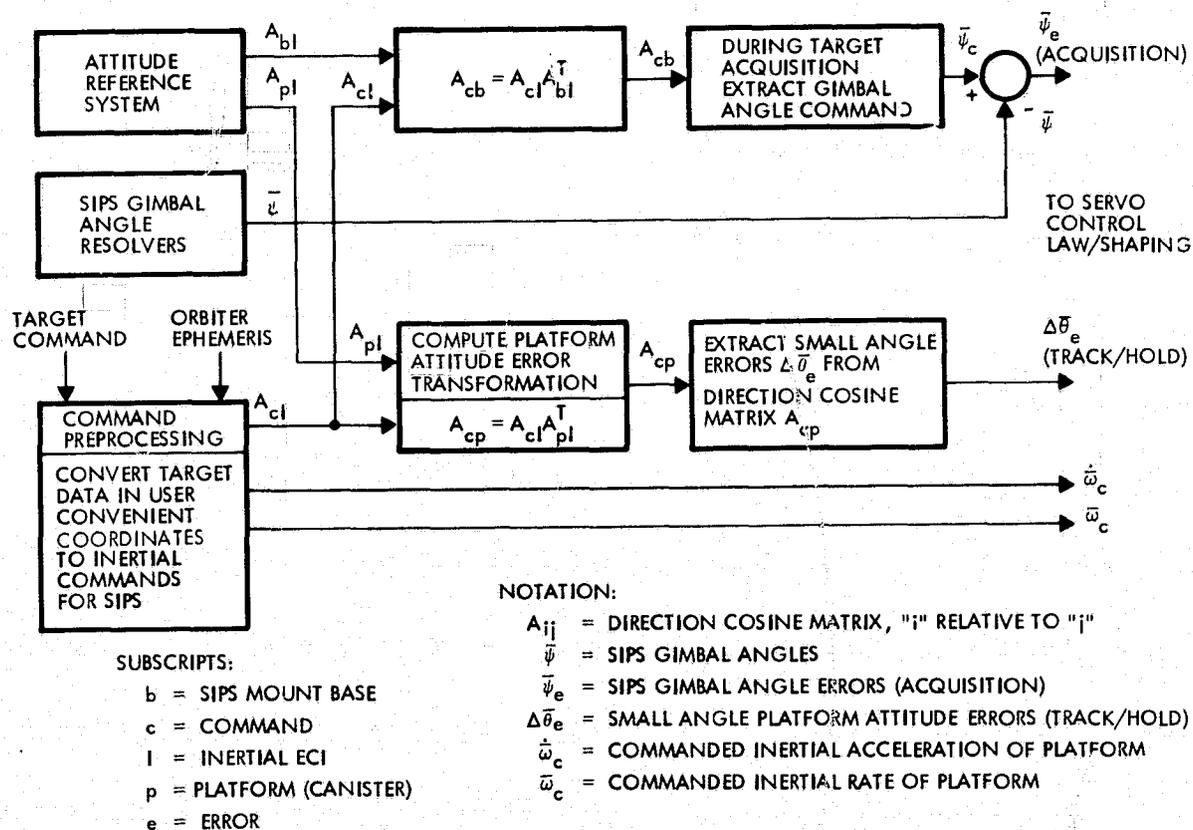


Figure 4-1. SIPS Command and Error Processing

4.1.2.1 Attitude Reference System Design

For the gyro reference assembly, the NASA Standard Inertial Reference Unit (DRIRU-II) has been selected for use on the Multi-Mission Modular Spacecraft (MMS). The unit contains three orthogonally arranged Gyroflex gyros. The Gyroflex is a cantilever suspended, 2-degree of freedom gyro providing two-axis rate information per gyro. Thus, the DRIRU-II gyro three-pack contains complete redundancy and any one out of the three gyros may fail without causing performance deterioration.

The DRIRU-II has been selected because it will be a standard NASA component and it meets the AMPS requirements. According to the NASA specifications* for the DRIRU-II, the long term random drift of the gyro is within 0.003 arc-sec/sec (1σ) and the short term random noise

* "Specification for NASA Standard Inertial Reference Unit," GSFC-S-712-10, May 1976.

variation from sample to sample (noise equivalent angle) is ± 0.167 arc-sec (1σ) relative to the best linear fit (in the least squares sense) to the long term angular drift. These DRIRU characteristics were used in the error covariance analysis that was conducted to assess the performance of the attitude reference system.

The strapdown star tracker has a field of view (FOV) of 2×2 degrees and sensitivity of $M_V = +6$. It is a modified version of the HEAO-B star tracker that is being built for NASA by Honeywell. It is a photon counting star tracker using an image dissector tube, and in its present HEAO-B version has sensitivity of $M_V = +9$. The tracker modification for the AMPS/SIPS application consists of putting a field stop in front of the HEAO-B tracker lens. This reduces the tracker sensitivity from 8th to 6th order magnitude stars and reduces the aperture diameter from 10.6 cm (4 inches) to 2.68 cm (1 inch). The latter is the reason for the modification. The geometry of Figure 4-2 shows that the tracker view of the celestial sphere is highly restricted by the Orbiter payload bay doors, Orbiter wing surface, and the earth. To have an 8×40 degree window on the celestial sphere, the tracker field of view must come within 20 degrees of the earth's limb and 15 degrees of the Orbiter surfaces. Under these conditions, a large tracker aperture in the 3.5 to 4 inch diameter range would require a shade in the 70 to 90 inch range. But a tracker aperture of 2.68 cm (1.055 inch) diameter will require only a 16-inch long shade with a 5.6 by 5.6 inch square front opening. The latter can easily be mounted on the outside of a SIPS canister, while the above mentioned larger shades could not. This is the main reason why the NASA Standard Star Tracker, the 8×8 degree, $M_V = +6$, BBRC CT-401 tracker could not be used for AMPS. Its aperture diameter of 3.5 inches translated into an excessive shade requirement under the AMPS/SIPS operating conditions.

The HEAO-B tracker has been selected for the AMPS application because it will be readily available and it meets the AMPS requirements. Using the star-position-dependent field of view calibration, as used

on HEAO-B, the tracker accuracy is specified as 1.81 arc-sec (1σ) for 0.32 second of data. This error budget is broken down further into 0.75 arc-sec of systematic or calibrated uncertainties and 1.06 arc-sec of noise equivalent angle.

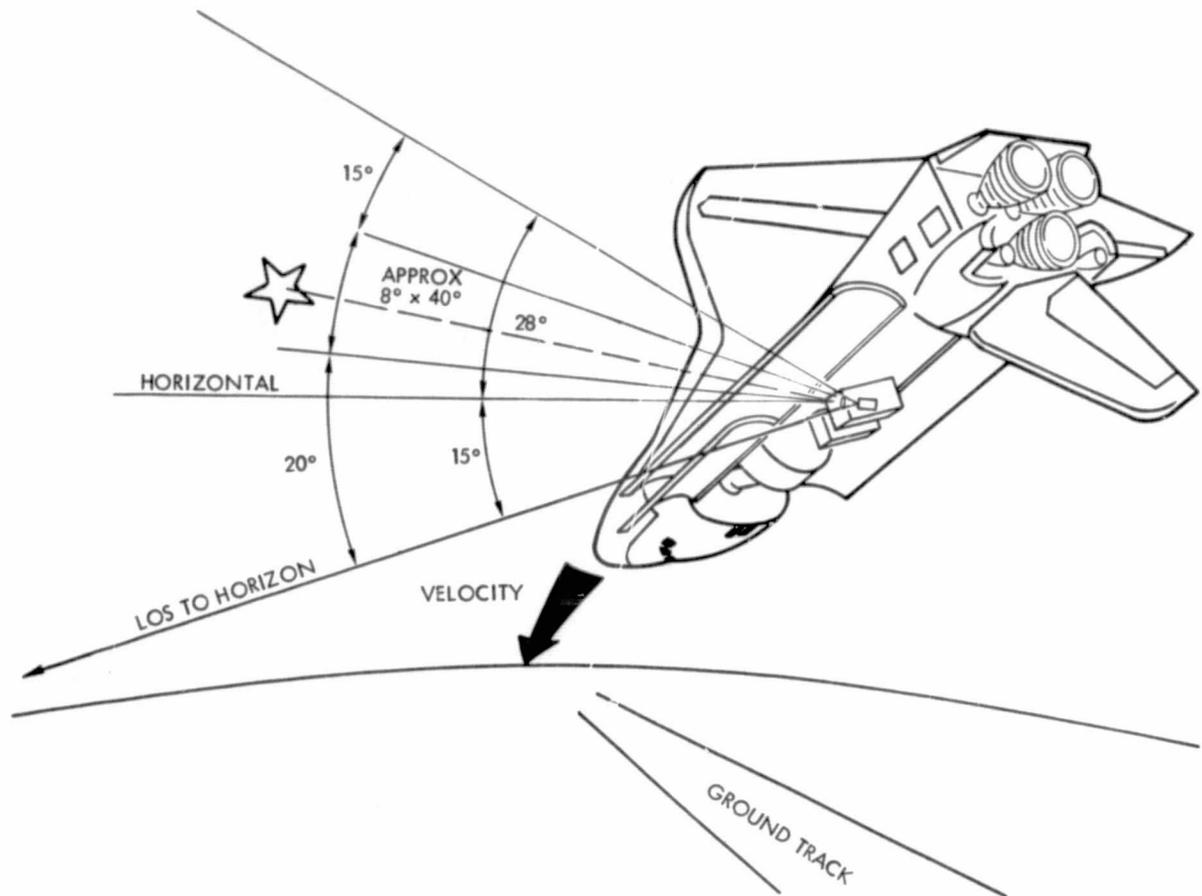


Figure 4-2. SIPS Star Tracker Field of View of Celestial Sphere

Even though the Honeywell/HEAO-B tracker has been preliminarily baselined for the AMPS/SIPS star tracker, the new development of all solid state, staring trackers using charge coupled devices (CCD's) cannot be ignored. The Jet Propulsion Laboratory of Pasadena, California, has recently developed and tested such a tracker with very promising results. It is expected that by the time AMPS/SIPS goes into the hardware and integration phase, the development status and commercial availability of CCD trackers will have substantially advanced and serious consideration should then be given to using such a tracker for AMPS/SIPS.

Since in the worst case, the window on the celestial sphere available to the star tracker is limited to a region comprising only 8×40 degrees = 320 square degrees (see Figure 4-2), star availability must be examined. Figure 4-3 plots the square-field-of-view width required on the celestial sphere versus star visual magnitude to assure, with 95 percent confidence, the presence of at least one star of the indicated brightness or brighter. For example, if one can detect stars up to $M_V = +6$, then with 95 per cent confidence one can find a usable star within any 25 square degrees of the celestial sphere. The actual worst case is also plotted using Yale star catalog data. For $M_V = +6$, the plot indicates that one can find no 10×10 degree area on the entire celestial sphere where there is not at least one 6th order

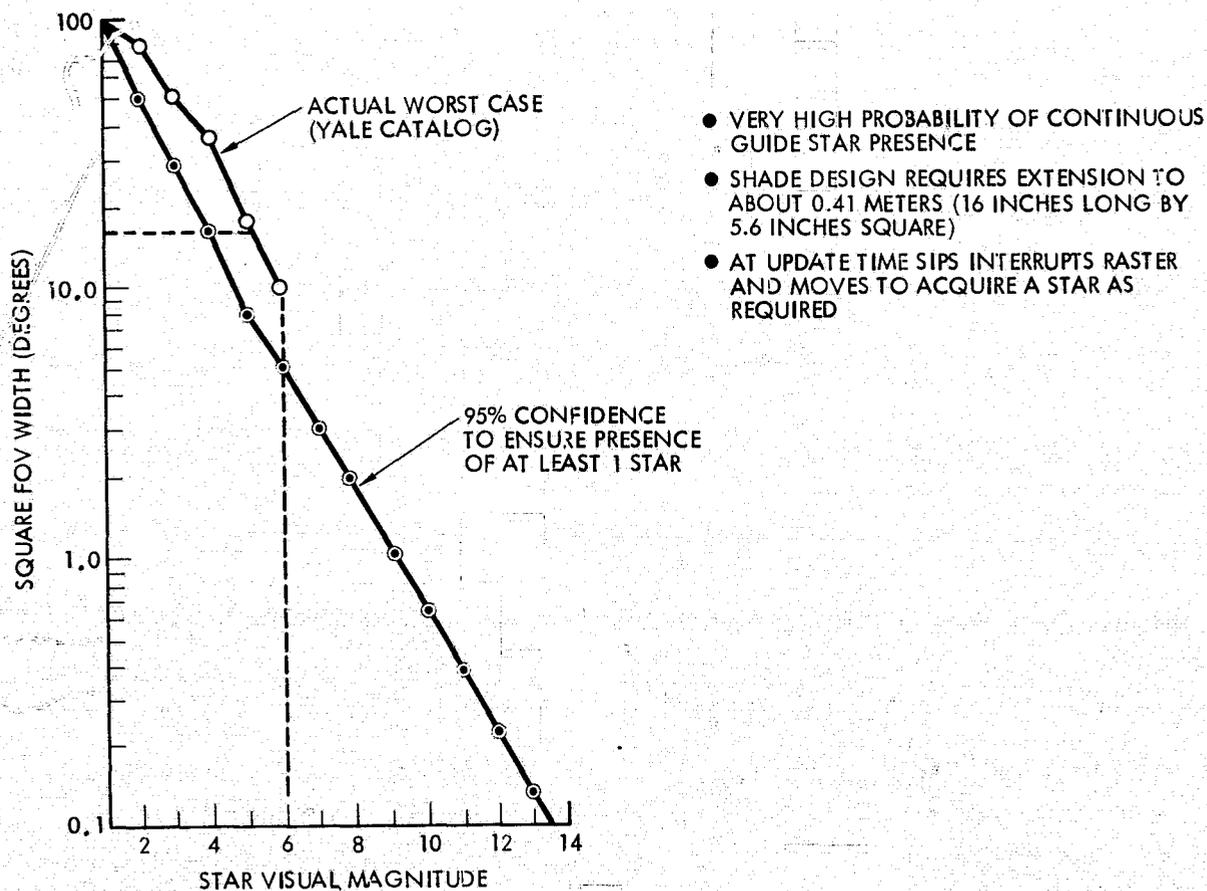


Figure 4-3. Guide Star Availability

magnitude star or brighter. Guide star availability within the 320 square degree window is, therefore, assured. In fact, with 95 per cent confidence, the guide star will be a 4th order magnitude star or brighter. At update time, the SIPS interrupts the raster scan and moves, if required, to acquire a star with the 2 x 2 degree FOV tracker within the available 8 x 40 degree area of the celestial sphere.

Figure 4-4 provides a functional block diagram of the attitude reference system. The gyro data processing consists mainly of correcting for the gyro biases and scale factor errors. Inertial rate is derived from the gyro pulse counts per sample, and the inertial attitude of the SIPS gimbaled platform (canister) is then computed using a closed form quaternion algorithm

$$\hat{q}_k = e^{(1/2)\hat{\Omega}_k T} \hat{q}_{k-1}$$

where

\hat{q} = quaternion attitude estimate

$\hat{\Omega}$ = 4 x 4 skew symmetric matrix containing the gyro sensed three-axis rates

T = gyro sample period

The matrix exponential $\exp (1/2 \hat{\Omega}_k T)$ can be computed in closed form and is given by

$$e^{(1/2)\hat{\Omega}T} = (\cos bT) I + \frac{\sin bT}{b} (1/2)\hat{\Omega}$$

where

$$b = (1/2) [\omega_x^2 + \omega_y^2 + \omega_z^2]^{1/2}$$

and I is the identity matrix. The direction cosine matrix A_{pI} of the gimbaled platform relative to inertial space (ECI) is then computed from the quaternion. The block diagram shows, thereafter, mainly direction cosine computations and this is generally a good way to proceed since it is easy to visualize and the availability of direction cosine matrices is often a requirement. However, a tradeoff should eventually be conducted between the use of quaternion algebra and direction cosine matrix manipulations to determine which approach results in a smaller software requirement.

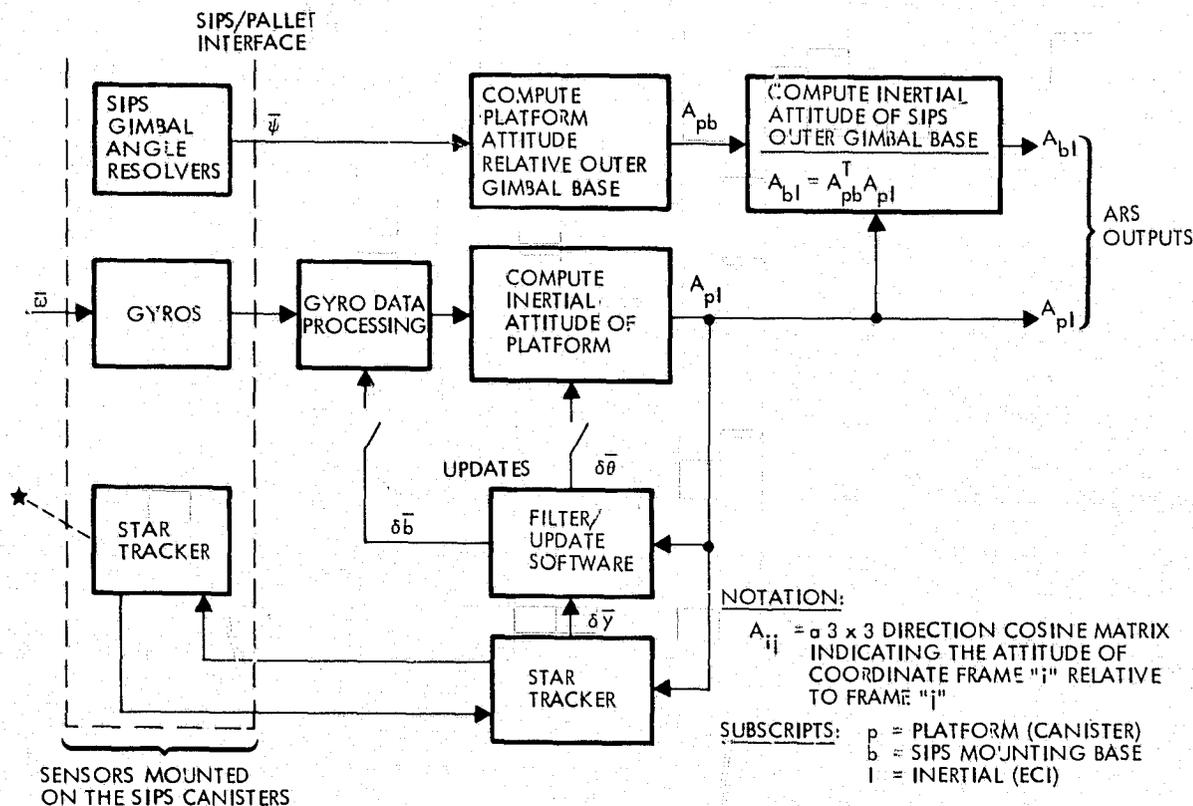


Figure 4-4. Inertial Attitude Reference System Functional Block Diagram

The star tracker data is processed by an extended Kalman filtering algorithm and optimal attitude and gyro bias updates are computed. The frequency of the required star updates depends on the quality of the system components and the attitude determination accuracy desired. As shall be shown shortly, an update interval of about 23 minutes is sufficient for AMPS/SIPS. Using the SIPS gimbal angle resolvers, the inertial attitude of the Orbiter pallet at the SIPS outer gimbal base is also computed by the attitude reference system.

4.1.2.2 Pointing Control System Design

A functional block diagram for the SIPS gimbal servos is shown in Figure 4-5. There are two modes, one to acquire the target and one for tracking and holding. When a new target is acquired, the SIPS gimbal angles are commanded and controlled relative to the gimbal base and the resolvers are the position error sensors. When the SIPS is pointing at or tracking a target, the pointing error is directly computed from the inertial attitude reference and resolver inaccuracies do not affect

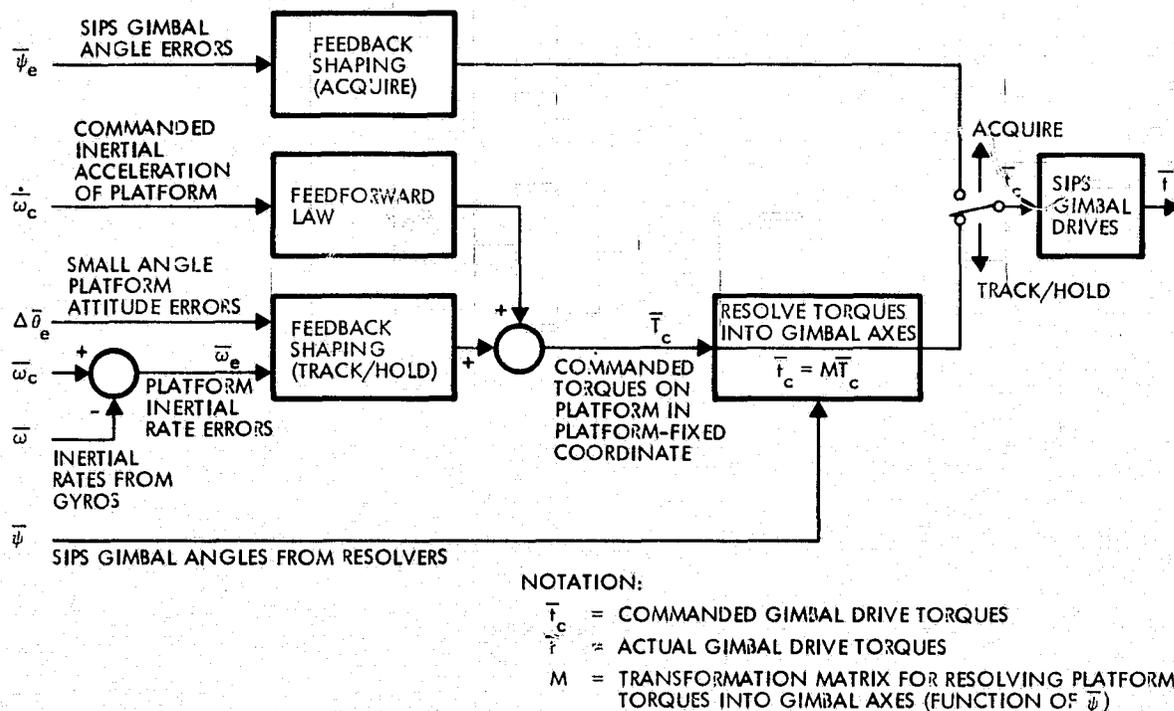


Figure 4-5. SIPS Servo Control Law Shaping Block Diagram

fine pointing. The control laws used are position plus integral, plus rate feedback with an additional lag network to stabilize high frequency gimbal compliance poles. An important feature of the control system is the angular acceleration feedforward which improves pointing performance and permits a reduction in the controller bandwidth, thus making it less sensitive to sensor noise. The feedforward control law is particularly important for raster scanning and tracking of earth fixed targets.

4.1.2.3 Optical Attitude Transfer/Alignment

Attitude transfer/alignment between pallet mounted instruments is accomplished by optical means using the attitude and position error sensor (APES) shown in Figure 4-6. The figure illustrates the attitude alignment measurement between the vector magnetometer and the electron beam accelerator which are located on different pallets. The alignment system contains three light sources, a solid-state detector and a convex mirror. Light source Number 1 is a focussed light source illuminating the convex mirror and light sources Numbers 2 and 3 can be thought of as point light sources, implemented by LED's.

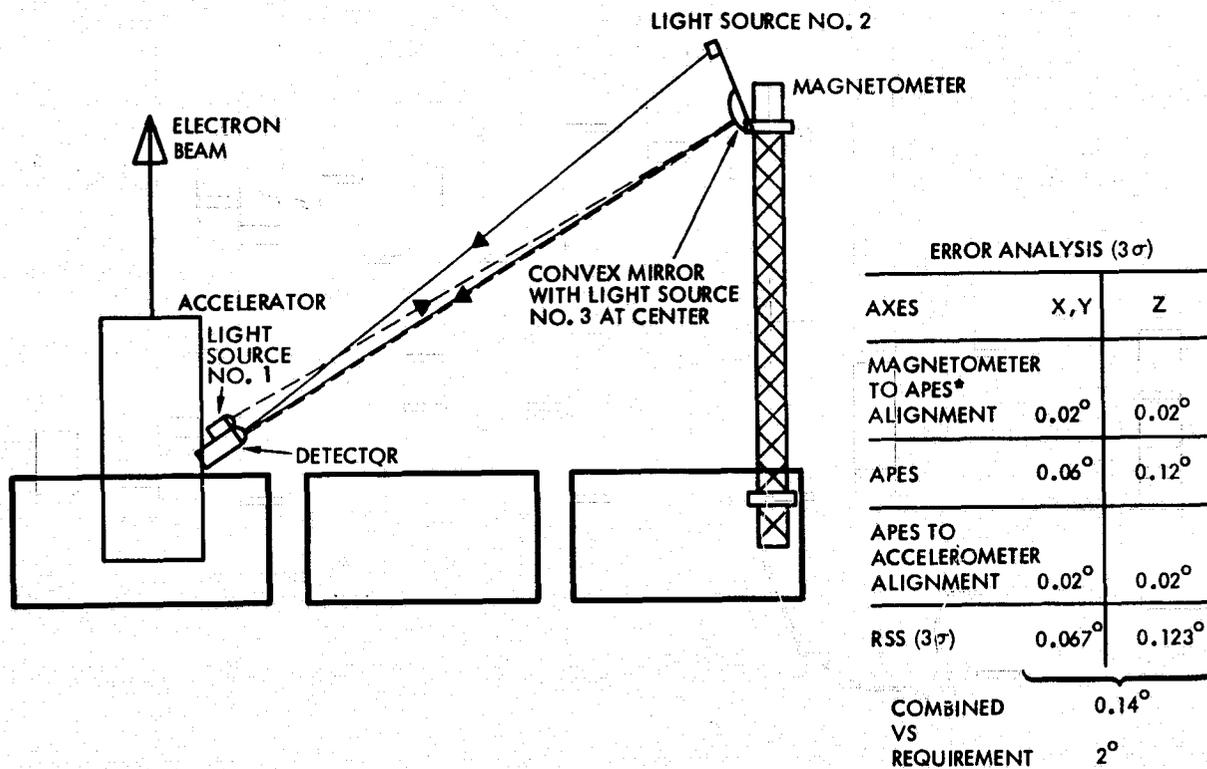


Figure 4-6. Attitude and Position Error Sensor (APES)

The mirror is convex in order to permit the magnetometer boom to move up to 10 degrees and still maintain the reflected image of light source Number 1 within the FOV of the detector. The position of the reflected image of light source Number 1 within the FOV of the detector indicates the bending motion of the boom transverse to the path of the light beam, while the image locations of light sources Numbers 1 and 2 together determine the twist motion. Light source Number 3 is used in conjunction with light source Number 2 to determine translational motion of the tip of the boom. The distance between the two light sources, as seen by the detector, decreases as the magnetometer recedes from the detector and increases as it comes closer. An error analysis of the APES is also given in Figure 4-6 which demonstrates that an alignment knowledge within 2 degrees can be easily provided by this system.

4.1.3 Pointing and Control Subsystem Performance

Pointing and control subsystem performance has been established by covariance analysis of the attitude reference system and analog

simulation of the SIPS gimbal control system. Overall performance is summarized in Table 4-2 which clearly demonstrates that the system meets and exceeds the AMPS/SIPS performance requirements. Figure 4-7 illustrates how the overall pointing accuracy is made up of attitude determination and alignment errors, and short term pointing stability errors which are mainly caused by the gimbal controller dynamic response to the disturbances acting on the system.

Table 4-2. Pointing and Control Subsystem Performance Summary

	EXPERIMENT REQUIREMENTS (2σ)	SIPS* CAPABILITY (2σ)
<ul style="list-style-type: none"> • ATTITUDE DETERMINATION • MISALIGNMENTS 		18.0 SEC ** 3.3 SEC
LONG TERM ERROR <ul style="list-style-type: none"> • SHORT TERM STABILITY 	20 SEC 20 SEC	18.3 SEC (RSS) 0.5 SEC *
OVERALL ACCURACY	120 SEC	18.8 SEC (SUM)

*SIPS ACCURACY INCLUDES DEGRADATION DUE TO CENTER OF MASS OFFSET

**STAR UPDATES EVERY 23 MINUTES

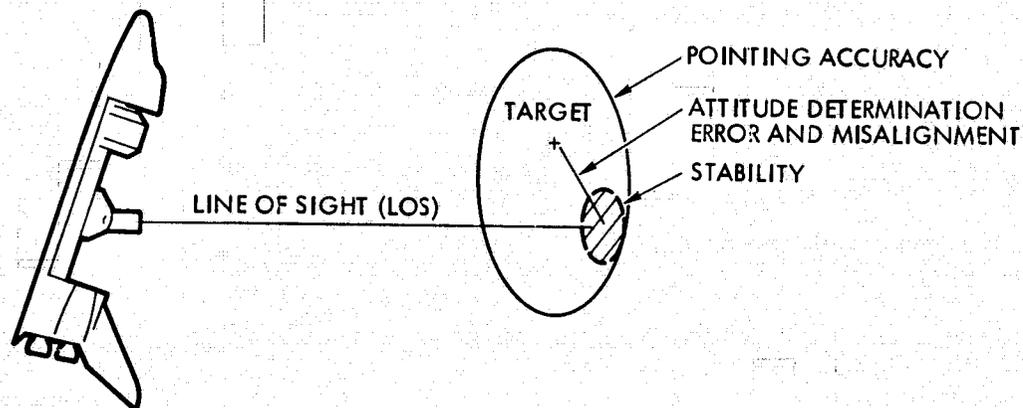


Figure 4-7. Pointing Error Definitions

4.1.3.1 Attitude Reference System Performance

The results of the attitude reference system covariance analysis are shown in Figure 4-8. Note that the covariance analysis provides 1σ performance levels, and that the entries in Table 4-2 are 2σ , i.e., 95 per cent confidence level numbers.

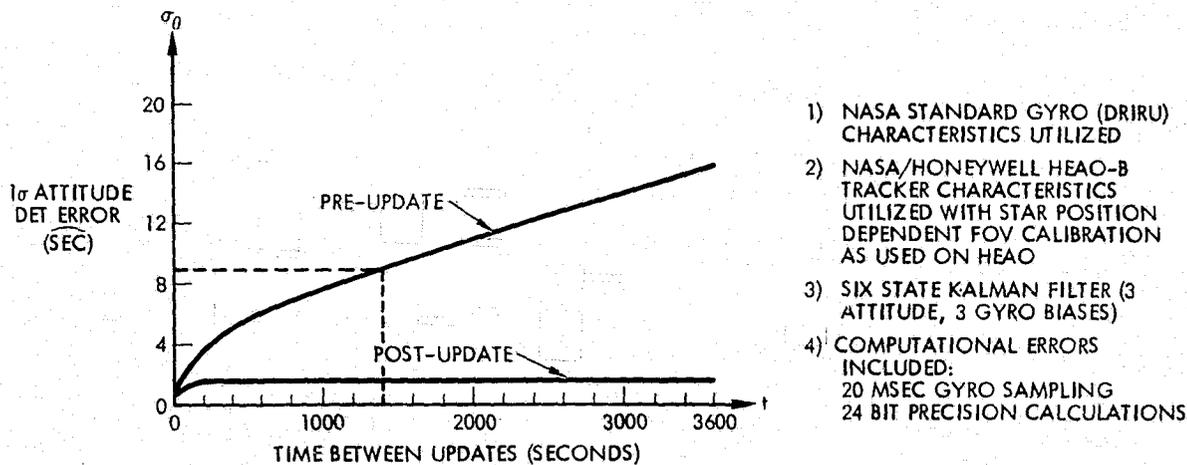


Figure 4-8. Attitude Reference System Covariance Analysis

As expected, the post-update attitude determination error reflects the star tracker accuracy of 1.81 arc-sec (1σ). Only for very short update intervals is the low tracker noise of 1 arc-sec further attenuated by the filter. The consistent 0.75 arc-sec (1σ) bias error cannot be improved upon no matter how frequent star updates are processed. The preupdate error mainly reflects the gyro random drift, computational errors and how well the gyro biases were calibrated [to about 0.0013 deg/hr (1σ)]. As can be seen, in order to meet the inertial attitude determination accuracy of 20 arc-sec (2σ) star updates are required about every 23 minutes.

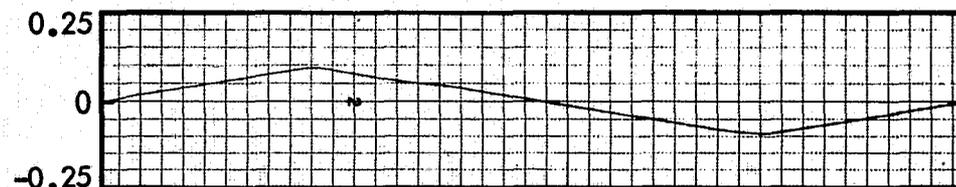
4.1.3.2 Pointing Control System Performance

Analog simulation has been used to establish pointing performance. A representative run is shown in Figure 4-9 for pointing of the cryogenic instrument. Effects of orbiter limit cycle motion, center of mass offset, man motion and hinge friction disturbances are included in the simulation. The 7 cm center of mass offset from the gimbal hinge point is the dominant error source at Orbiter limit cycle reversal; the hinge friction, also excited at Orbiter limit cycle reversal, has the smallest effect on pointing accuracy. The center of mass offset of the cryogenic instrument is caused by depletion of the cryogen and the offset is at its maximum value of 7 cm at the beginning of life and again at the end of life; the instrument has been packaged into the SIPS canister such that it is balanced when exactly half the

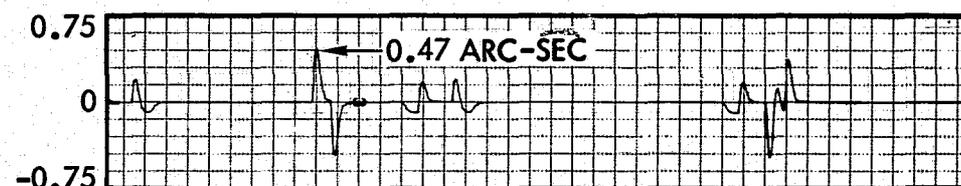
cryogen has been expended. As can be seen on Figure 4-9, the peak pointing error is 0.47 arc-sec and it occurs at Orbiter limit cycle reversal. The 0.5 arc-sec short term pointing stability given in Table 4-2 corresponds to this pointing error. The error could be reduced further, if required, by mounting an accelerometer to the SIPS canister and deriving feedforward signals from it which would compensate up to 80 per cent any pointing errors caused by payload center of mass offsets from the hinge point. For the present AMPS/SIPS application this is not necessary, however, since the short term pointing stability requirements are not only met but are exceeded by more than an order of magnitude.

- 1) CENTER OF MASS OFFSET OF 7 CM IS DOMINANT ERROR SOURCE
- 2) MAN MOTION DISTURBANCES INCLUDED
- 3) NOMINAL FRICTION DISTURBANCE EXCITED BY ORBITER LIMIT CYCLE REVERSAL
- 4) POINTING CONTROL SYSTEM CONTROLS TO WITHIN 0.47 ARC-SEC
- 5) DATA
 MASS = 618 KG
 INERTIA = 427 KG-M²
 CONT. NAT. FREQ. = 10 RAD/SEC

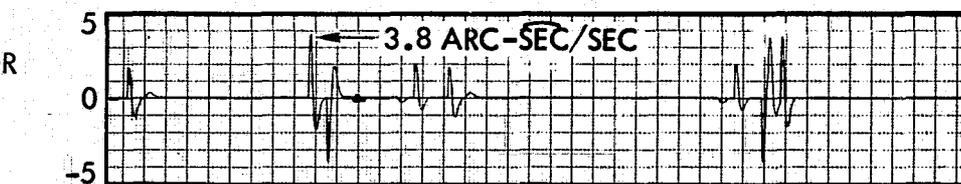
ORBITER

 ϕ_B (DEG)

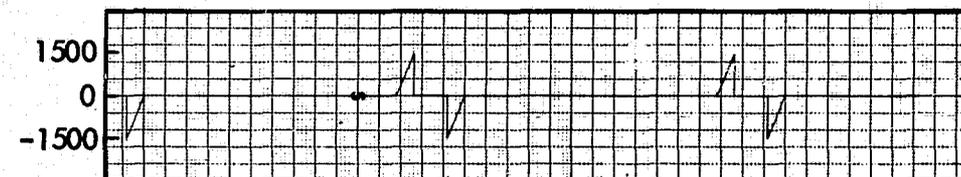
PAYLOAD ERROR

 ϕ_p (ARC-SEC)

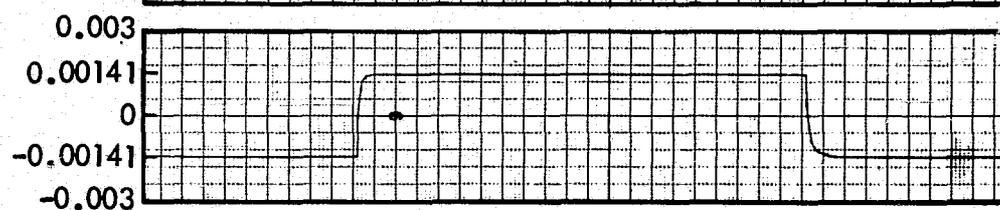
PAYLOAD RATE ERROR

 $\dot{\phi}_p$ (ARC-SEC/SEC)

CREW MOTION

 T_{CR} (NM)

HINGE FRICTION

 T_f (NM)

0

20

40

SEC

Figure 4-9. SIPS Controller Stability for Cryogenic Limb Scanner by Analog Simulation

4.2 STRUCTURES AND DYNAMICS

The structural support for AMPS equipment will follow conventional aerospace structure design and fabrication practice using aluminum material. Support structure commonality will be incorporated as much as possible to permit its reuse on subsequent flights. Trade studies have shown that this approach yields the lowest cost system with no impact on payload accommodation.

Trade study results also have shown that the selection of high factors of safety may reduce the overall cost of the structure by reducing structural testing requirements. Similarly, an evaluation of the environments has revealed that by designing the support structure to 35 Hz or higher, the requirements for extensive loads analyses can be minimized.

The acoustic environment in the payload compartment, as presently understood, can be accommodated using standard design and test matrices for electronic components and/or lightweight large surface type structural elements.

The deployable booms, required by several experiments, are available and many proven concepts have been developed. Candidate deployment devices have been identified and their feasibility established for the selected lengths and other deployment requirements.

4.2.1 Requirements

The major requirements imposed on the structural and mechanical design of an instrument support structure are that it: 1) meets the mechanical design capability of the Orbiter and the Spacelab (these are defined in JSC-07700, Space Shuttle Systems Accommodations, Reference 1, the ESA/Spacelab Payload Accommodation Handbook, Reference 2, and the Space Transportation System (STS) Safety Policy and Requirements, Reference 3); 2) minimizes the Orbiter and/or instrument performance constraints due to adverse dynamic effects, and 3) utilizes techniques which result in a low cost/high reliability system.

4.2.2 Design Criteria

The major parameter which influences the structural design at this phase of the program is the factor of safety. The factor of safety used in the subsequent design investigations was selected so as to reduce the

testing requirements and, correspondingly, the total cost of a qualified flight structure. The chosen factors are identical to those utilized on HEAO-A spacecraft design which implemented a similar design philosophy. These are: Yield Load = 2.0 x Limit Load and Ultimate Load = 3.0 x Limit Load.

4.2.3 AMPS Support Structure

The support structure for the instruments to be accommodated on Flights 1 and 2 can be readily designed using conventional aluminum aerospace construction. These would include aluminum honeycomb platforms and panels, tube truss-type, and sheet-stringer structures, as is shown in Figures 4-10 and 4-11. By the judicious design of platforms and panels, it is possible that these structural members can be used on different flights for supporting other instruments. An example is the support structure for the Environmental Sensor Package (ESP) and/or the support structure for the gas release canisters.

4.2.3.1 Typical Pallet Support Reactions

The design safety factors described above were used in conjunction with the flight load factors of Reference 1 to obtain the design load factors in Figure 4-12. The design factors for flight conditions are larger than those associated with crash, leading to the critical design condition being the flight and not the crash loads. The consequences of this was investigated for the lidar support to the pallet and the results of the analysis are shown in Figure 4-13. The lidar is one of the heaviest (approximately 779 kg) instruments of the Flight 1 and Flight 2 payloads. Two-pallet tiedown capabilities were compared with the reactive forces for the crash condition, and it will be noted that in all cases where there are known capabilities the loads are less than the capability of the pallet. The analysis was based on the conservative assumption that all the crash load factors are applied simultaneously. Referring to Figure 4-12, and evaluating the design loads for a given event, it will be recognized that the reactive forces may be larger than those in Figure 4-13, but they still are substantially below the capability of the tiedown points. It is concluded that for the functional assemblies investigation, the pallet reactions will be below the capability of the tiedown points. Further, for the functional assemblies investigated, the pallet reactions will be

4-19

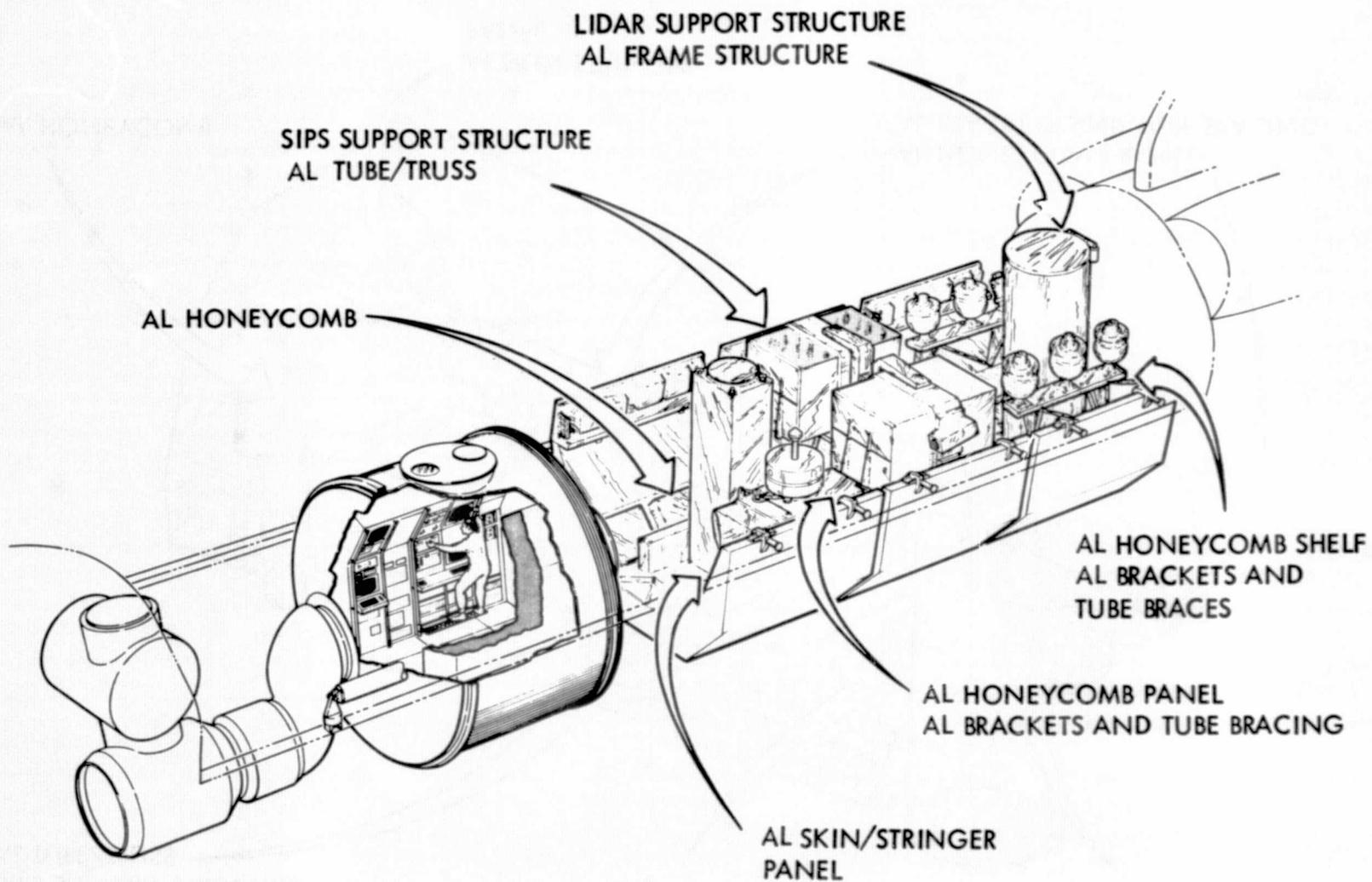


Figure 4-10. AMPS Flight 1 Support Structure

4-20

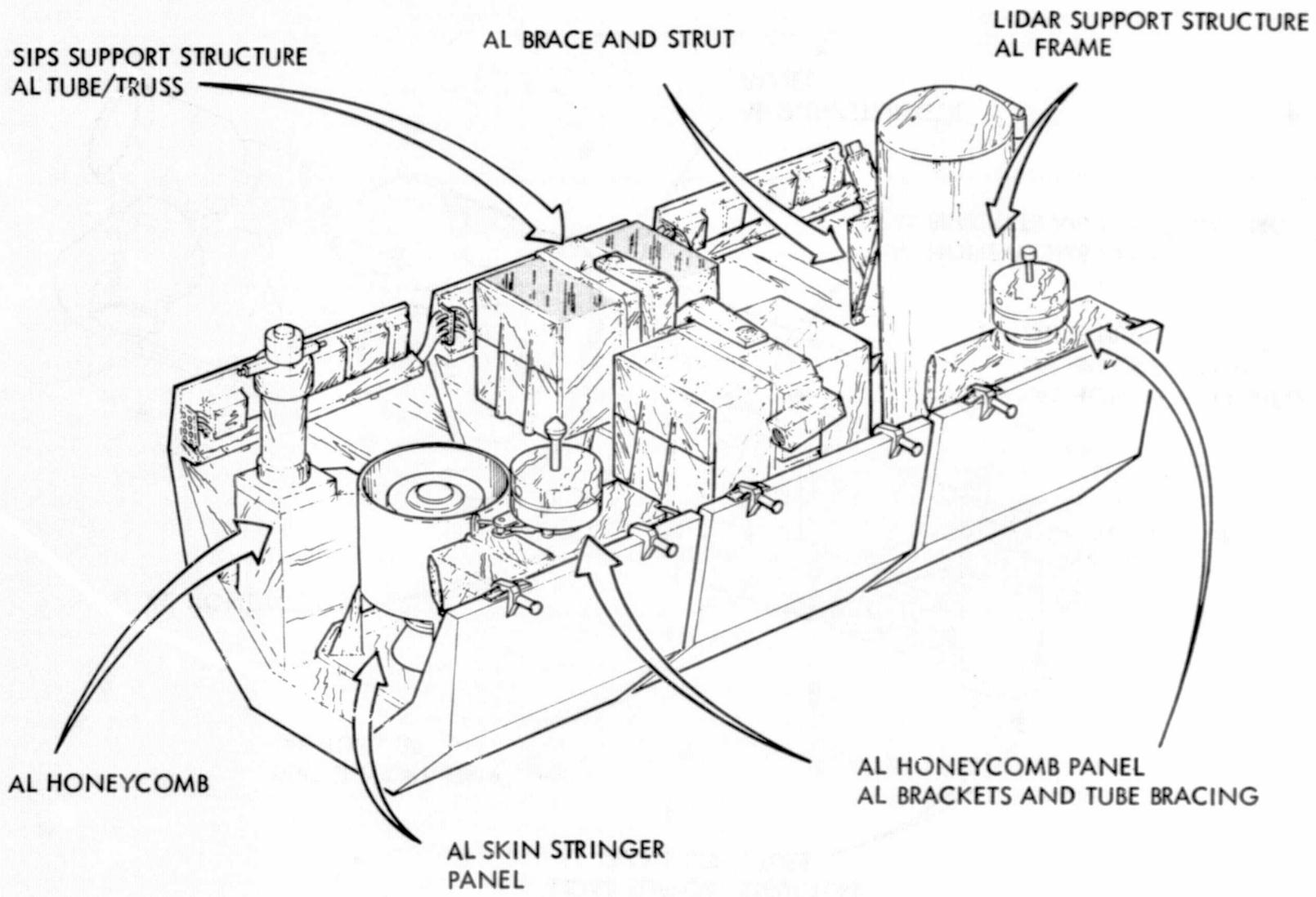
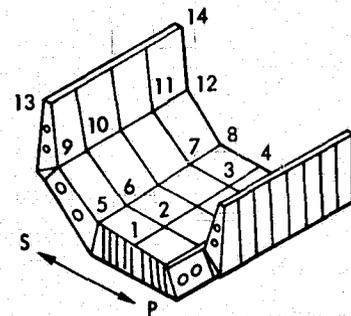


Figure 4-11. AMPS Flight 2 Support Structure

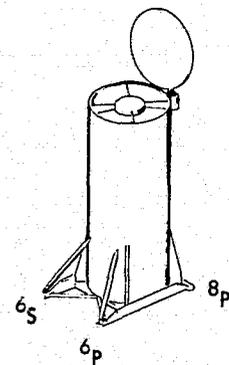
LOAD DIREC- TION	EVENT	BOOST - MAX L. F.						ENTRY AND DESCENT						CRASH ULT
		LIFTOFF		STACK		ORBITER		PITCHUP		YAW		LANDING		
		YIELD	ULT	YIELD	ULT	YIELD	ULT	YIELD	ULT	YIELD	ULT	YIELD	ULT	
X		-0.2	-0.3	-5.4	-8.1	-5.4	-0.1	2.12	3.18	+1.5	+2.25	+2.0	3.0	+9.0
		-5.8	-8.7	-6.6	-9.9	-6.6	-9.9	-0.04	-0.06	+1.5	+2.25	-1.6	-2.4	-1.5
Y		±2.0	±3.0	±0.4	±0.06	±0.4	±0.6	0	0	±2.50	±3.75	±1.0	±1.5	±1.5
Z		±3.0	±4.5	-0.6	-0.9	-1.5	-2.25	+5.0	+7.5	+2.0	+3.0	+5.6	+8.4	+4.5
								-2.0	-3.0		+4.4	+6.6	-2.0	

YIELD ACCELERATION LOAD = 2.0 X 
 ULTIMATE ACCELERATION LOAD = 3.0 X 
 = JSC 07200, VOL. XIV, REV. D TRANSLATIONAL ACCELERATIONS
 FLIGHT EVENT LOADS ARE COMBINED IN X, Y, Z; CRASH ARE APPLIED SINGULARLY

Figure 4-12. Preliminary Design Load Factors



PALLET HARDPOINT LOCATIONS



LIDAR

LIDAR
CRASH LOADING REACTIONS IN KNEWTONS

PALLET HARDPOINT	X		Y		X	
	REACTION	CAPABILITY	REACTION	CAPABILITY	REACTION	CAPABILITY
8	+17.2	+TBD	+2.9	+16.8	+19.3	+TBD
	- 2.9	-33.1	-2.9	- 4.5	-10.5	-33.2
6	+17.2	-TBD	+2.9	+22.4	+10.9	+33.2
	- 2.9	-62.1	-2.9	-25.3	-10.5	-100.0

Figure 4-13. Typical Pallet Support Reactions

below the capability of the tiedown points, and that a design philosophy where the "no test" factors of safety are used does not impose a penalty on the pallet design.

4.2.4 Design Environments

The significant environments for the mechanical design of AMPS payloads are provided in the "Space Shuttle System Payloads Accommodations," Reference 1. The critical flight events for the Shuttle payload are liftoff and landing, with maximum loads occurring at the first body bending frequency near 4 Hz for liftoff and 5.5 Hz for landing. Significant landing responses occur also at about 12 Hz, representative of the Orbiter wing torsion mode, and at 21 Hz, representative of a combined tail mode and the second bending mode of the main fuselage.

In the structural design of instrument support structures, a minimum 35 Hz design goal should be placed if they are rigidly attached to the pallets so that their fundamental frequencies fall outside the range of response associated with major flight events. This high minimum frequency reduces the amount of amplification imposed on a given component. Furthermore, the cost of the engineering evaluation could be reduced since a detailed dynamic analysis on the combined STS and payload system may not be required.

4.2.4.1 Acoustic and Vibration Environment

The acoustic levels within the payload bay and a typical component random vibration environment are shown in Figure 4-14. The acoustic levels are derived from Reference 1 by converting to octave band levels and adding 4 dB to establish the qualification environment. Both environments are typical of the present operational boosters and can be accommodated using current design practices.

4.2.4.2 Other Environments

In addition to the launch and landing accelerations and the acoustic and vibration environment, the AMPS experiments are subject to the full range of space environments including thermal, micro-meteoroid, aerodynamic drag at the lower altitudes, and gravity gradients. These are taken into consideration and are discussed where appropriate in the following sections.

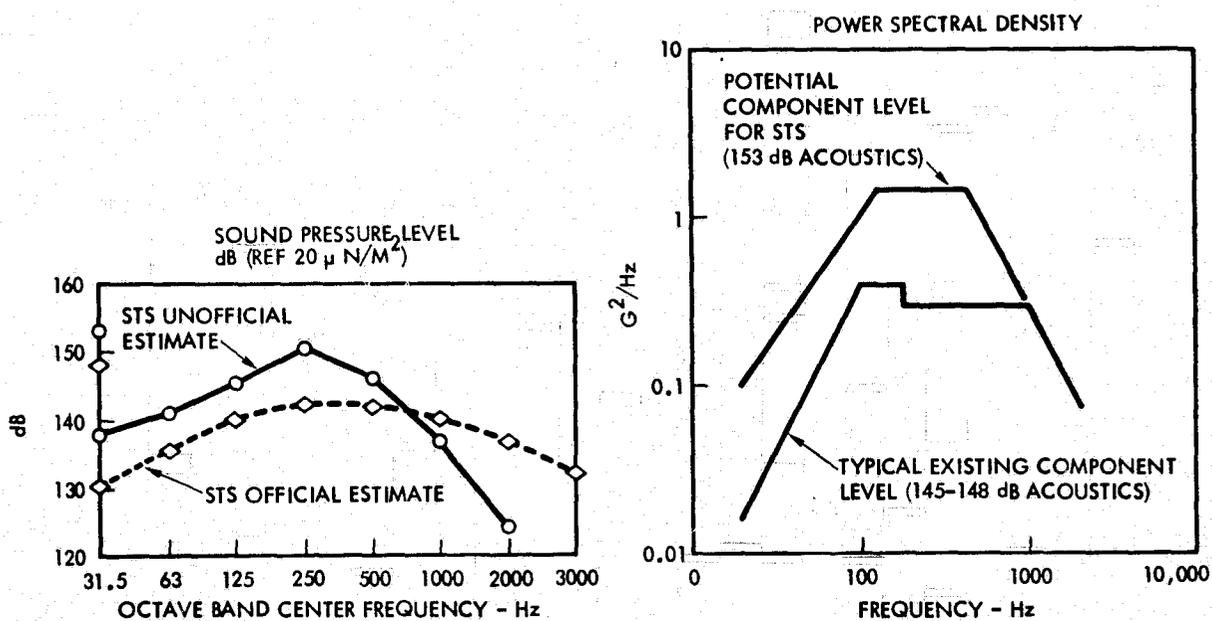


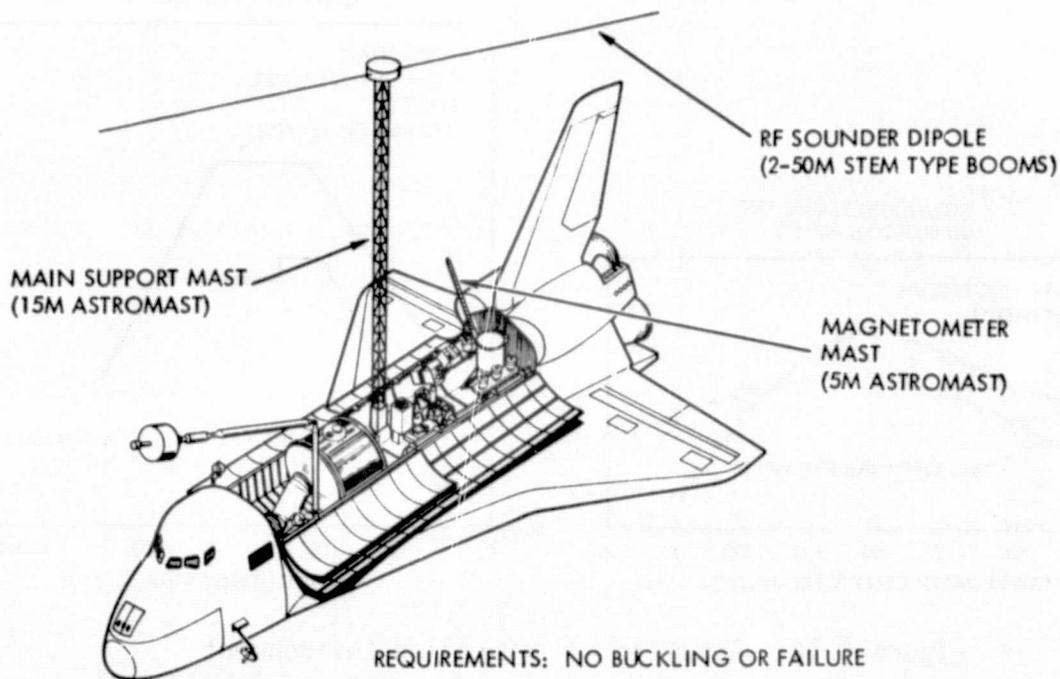
Figure 4-14. Acoustic and Vibration Environment

4.2.5 Deployable Appendages

A description of candidate deployable appendages with appropriate photographs is presented earlier in Section 3.2.

The deployable appendages vary for Flights 1 and 2. Flight 1 incorporates the magnetometer mast which is a 5-meter deployable and recoilable structure. The magnetometer housing is located at the end of the mast. Electrical cabling is attached to a longeron or a separate lanyard, which runs the length of the mast. Flight 2 incorporates the magnetometer mast and a main support mast. The main support mast is approximately 15 meters long and, when completely extended, two RF sounder dipoles are deployed. Each dipole is a 50-meter stem type boom. The baseline main mast is an Astromast using articulated steel longerons and diagonal steel cables. Figure 4-15 shows the deployed appendages for the Flight 2 AMPS.

The characteristics of each of these masts relative to potential suppliers, types of structure driving mechanism, weight and size, etc., are provided in Figures 4-16 and 4-17. The candidate mast and booms are all developed and available for use as an element of the AMPS payload. Additional development requirements are minimal and lie essentially in the area of the jettison or separation devices. These are not expected to present a major development problem.



REQUIREMENTS: NO BUCKLING OR FAILURE

MAGNETOMETER: LESS THAN 2° OF MISALIGNMENT
RELATIVE TO THE ACCELERATOR

RF SOUNDER: LESS THAN 10° OF BOOM BENDING

Figure 4-15. Flight 2 AMPS Deployable Appendages

SUPPLIER	ASTROMAST
STRUCTURE	FIBERGLASS LONGERON AND BATTEN FIBERGLASS OR PHOSPHOR BRONZE DIAGONALS
DRIVE MECHANISM	MOTOR DRIVEN LANYARD
CABLING	ATTACHED TO LONGERON OR LANYARD
THERMAL CONTROL	PRETWIST OF LONGERON
STORAGE LIFE	>1 YEAR
LIFE CYCLES	>600
STATUS	6-METER MAST ON USAF SATELLITE 12 METERS FOR 1977 MARINER-JUPITER-SATURN SATELLITE
WEIGHT	19 LB (BOOM, CANISTER, AND MOTOR)
SIZE	5 M x 22.9 CM DIAMETER
ADDITIONAL DEVELOPMENT	CAGING MECHANISM, SEPARATION/JETTISON DEVICE

Figure 4-16. Magnetometer Mast

	MAIN SUPPORT MAST	RF SOUNDER DIPOLE
SUPPLIER	ASTRO RESEARCH	ASTRO RESEARCH
STRUCTURE	ARTICULATED LONGERON DIAGONAL CABLES	BeCu INTERLOCKED STEM DEVICE
DRIVE MECHANISM	3 MOTOR DRIVES	MOTOR DRIVE
CABLING	SEPARATE REEL	NOT REQUIRED
STORAGE LIFE	LONG TERM	LONG TERM
LIFE CYCLES	>100	>50
STATUS	26-METER ENGINEERING MODEL DEVELOPED AND TESTED	MANY SIMILAR DESIGNS DEVELOPED AND FLOWN
WEIGHT	300 POUNDS	60 POUNDS (2 BOOMS + HOUSINGS + DEPLOYED MECHANISM)
SIZE	15 M x 50.8 CM DIAMETER	50 M x 3.4 CM DIAMETER
ADDITIONAL DEVELOPMENT	SYSTEM JETTISON DEVICE CABLE STOWAGE REEL MAST AZIMUTH DRIVE	SEPARATION/JETTISON DEVICE

Figure 4-17. RF Sounder Mast

The results of analyses investigating the combined effects of all significant environments on the deployed appendages is presented for the three extendible booms in Figure 4-18.

4.2.5.1 Boom Requirements

In addition to meeting the structure requirements and criteria stated in Sections 4.2.1 and 4.2.2, the booms must satisfy additional requirements which are unique to their application. These are: 1) they must not buckle or otherwise fail under indigenous or Orbiter-induced environments; 2) they must be capable of multiple extension and retractable cycles; 3) they must be capable of being jettisoned safely from the payload bay at any state of their extension/retraction cycle; 4) the magnetometer axes must be known to within 2 degrees of the accelerator axes; 5) the RF sounder appendages must not deflect more than 10 degrees, i.e., the angle between the tangent to root and line connecting root and appendage tip; and 6) they must use existing proven designs to minimize cost.

APPENDAGE \ DISTURBANCE	GRAVITY GRADIENT	SOLAR PRESSURE	DRAG ¹	THERMAL	ORBITER RCS ACCELERATION ²		MANUFACTURING	MAXIMUM TOTAL		ALLOWANCE
					PRIMARY	VERNIER		PRIMARY	VERNIER	
MAGNETOMETER MAST										
δ (IN.)	NEG	NEG	NEG	<0.7	0.6	0.01	<0.2	<1.5	<1.0	-
α (DEG)	NEG	NEG	NEG	<0.1	0.3	0.005	<0.1	<0.5	<0.3	1.0
M (LB-IN.)	NEG	NEG	NEG	-	130	2	-	130	2.0	179
MAIN SUPPORT MAST										
δ (IN.)	NEG	NEG	5.2 ³	-	8.2	0.3	3.8	17.2	9.3	-
α (DEG)	NEG	NEG	0.8	<2.8	1.2	0.04	0.4	5.2	4.1	-
M (LB-IN.)	NEG	NEG	120	-	21000	490	-	21120	610	34000
RF SOUNDER ANTENNA										
δ (FT)	0.08	NEG	2.4	-	-	9.7	<7.5	-	24.4	28.6
α (DEG) ⁴	NEG	NEG	0.8	-	-	3.4	<2.6	-	8.5 ⁵	10
M (LB-IN.)	NEG	NEG	5.0	-	-	20	-	-	25	633

¹200 KM ALTITUDE

²RIGID BODY ACCELERATIONS APPLIED IN WORST POSSIBLE MANNER TO MAGNETOMETER BOOM AND MAIN SUPPORT MAST; RF SOUNDER RESPONSE ASSUMES CHAIN OF 20 VERNIER RCS PULSES TUNED TO FIRST MODE BENDING PERIOD OF ANTENNA

³WITH BALLOON TEST BODY DEPLOYED

⁴ANGLE BETWEEN LINE CONNECTING ROOT AND TIP AND TANGENT LINE TO THE ROOT

⁵ROOT-SUM-SQUARE OF MAST AND RF SOUNDER SLOPE IS 5.5 DEGREES

Figure 4-18. Strength and Deflection Requirements of Booms and Masts

4.2.5.2 Magnetometer Mast

The environments of significance for the magnetometer mast are thermal distortion and Orbiter RCS accelerations. The thermal distortion is based on experience with the MJS mast and includes temperature ranges from -300°F to $+100^{\circ}\text{F}$. The RCS induced accelerations shown in Figure 4-19 (Reference 1) were applied to the magnetometer and support mast as if they were quasistatic. The vernier accelerations are produced by the RCS as it maintains the Orbiter within its limit cycle bandwidth. Primary accelerations result from the thrusters which can be used in a limit cycle mode or used to reorient the Orbiter at 0.25 to 1.0 degree per second. Accelerations acting on the magnetometer due to Orbiter rotational acceleration were computed assuming the system was separated 12 meters (40 feet) longitudinally from the Orbiter center of mass.

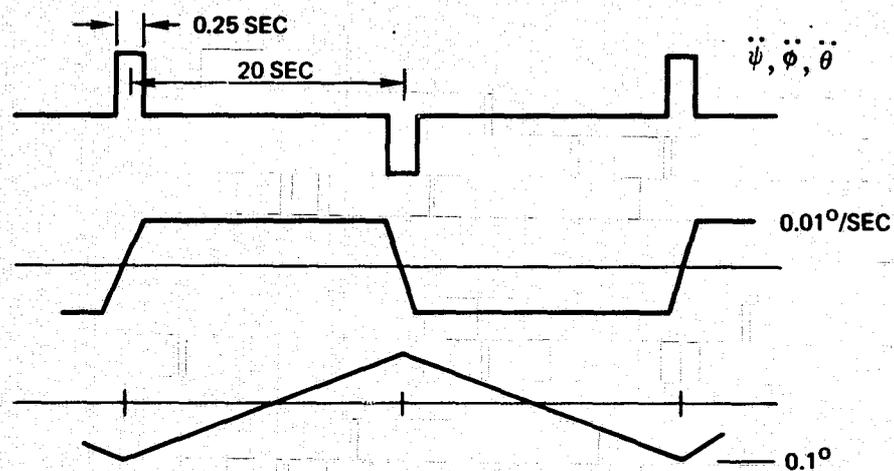
Displacements and loads were calculated for the case when all translational and rotational accelerations were applied simultaneously in the worst possible phase. Manufacturing tolerances are included in the total summation and represent an achievable quantity based upon demonstrated technology.

Alignment of the magnetometer system with respect to accelerator must consider not only the initial levels, which can be made quite small, but must include also the eventual distortion caused by thermal "hot-dogging" of the Orbiter, "settling" of the Orbiter/pallet structure as a result of boost loads, and errors introduced by the one-g effect during initial alignment on the ground. The magnetometer mast is located approximately two and one-half pallet lengths from the accelerator, so these concerns are important if accurate alignment is to be maintained. It is expected that the errors will be significant and an in-orbit pointing determination system will be employed.

Preliminary assessment summarized in Figure 4-17 indicates the selected magnetometer mast can withstand the in-orbit environments in any Orbiter attitude under both primary and vernier mode RCS action and meet the requirements.

DIRECTION	TRANSLATIONAL ACCELERATION FT/SEC ² (MPS ²)					ROTATIONAL ACCELERATION DEG/SEC ²			
	+X	-X	±Y	+Z	-Z	± $\ddot{\phi}$	+ $\ddot{\theta}$	- $\ddot{\theta}$	± $\ddot{\psi}$
PRIMARY THRUSTER	0.554 (0.169)	0.424 (0.129)	0.695 (0.212)	1.251 (0.381)	1.014 (0.309)	1.168	1.320	1.482	0.738
VERNIER THRUSTER	0 0	0 0	0.0070 (0.0021)	0 0	0.0080 (0.0024)	0.037	0.024	0.017	0.019

4-28



ASSUMED NOMINAL VERNIER DISTURBANCE

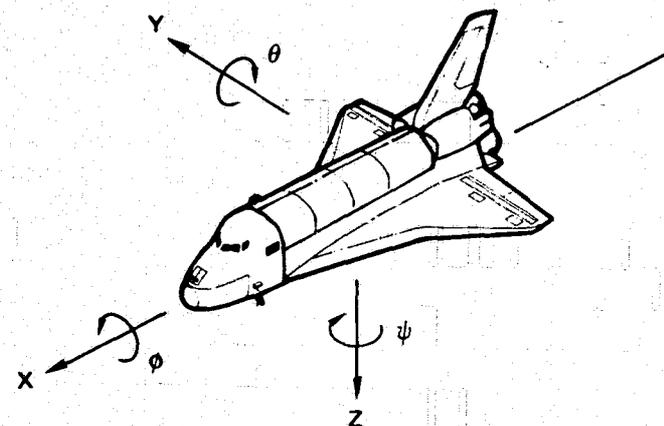


Figure 4-19. RF Sounder Antenna and Mast Loading Due to Orbiter RCS Action

4.2.5.3 Main Support Mast

The main mast deflections are influenced principally by drag forces. The drag induced deflections and loads were computed with the 10-meter diameter balloon wake experiment deployed. This experiment is not present when the RF sounder antennas are in operation. The thermal and manufacturing distortions are only estimates and require further study. Orbiter accelerations were applied to the main mast system as quasi-steady levels simultaneously acting in the worst manner. The possibility of dynamic amplification of the mast response to the RCS pulsing action is briefly explored in the next section.

Based upon preliminary evaluations summarized in Figure 4-18, the mast can sustain the loading imposed by vernier or primary accelerations in any Orbiter attitude. However, during the period of mast and RF sounder use, the Orbiter should be constrained to the vernier mode of limit cycle operation.

4.2.5.4 RF Sounder Antennas

Because of their long length and low stiffness, the RF sounder antennas are influenced more by the full range of in-orbit environments than the other appendages. Figure 4-20 displays the peak tip deflections for various standard sizes of overlapped STEM. The selected baseline antenna is an interlocked adaptation; the results shown in Figure 4-20 are representative. The diameter/stiffness requirements of the RF antenna are heavily influenced by drag considerations. Use of the antennas in a fully extended state at an orbital attitude of 200 km requires a STEM size of 3.4 cm (1.34 in.) to limit tip deflections to a reasonable magnitude. At 400 km the peak drag induced deflection is reduced by over a factor of 30. Figure 4-20 indicates that gravity gradient and solar pressures are not significant disturbances except for the smallest STEM sizes. Thermally induced distortions are sizable; however, the results are based on a perforated boom. Perforations of about 8 percent of the boom surface area, together with silver exterior and black interior surface coatings, can reduce these deflections below the shown values.

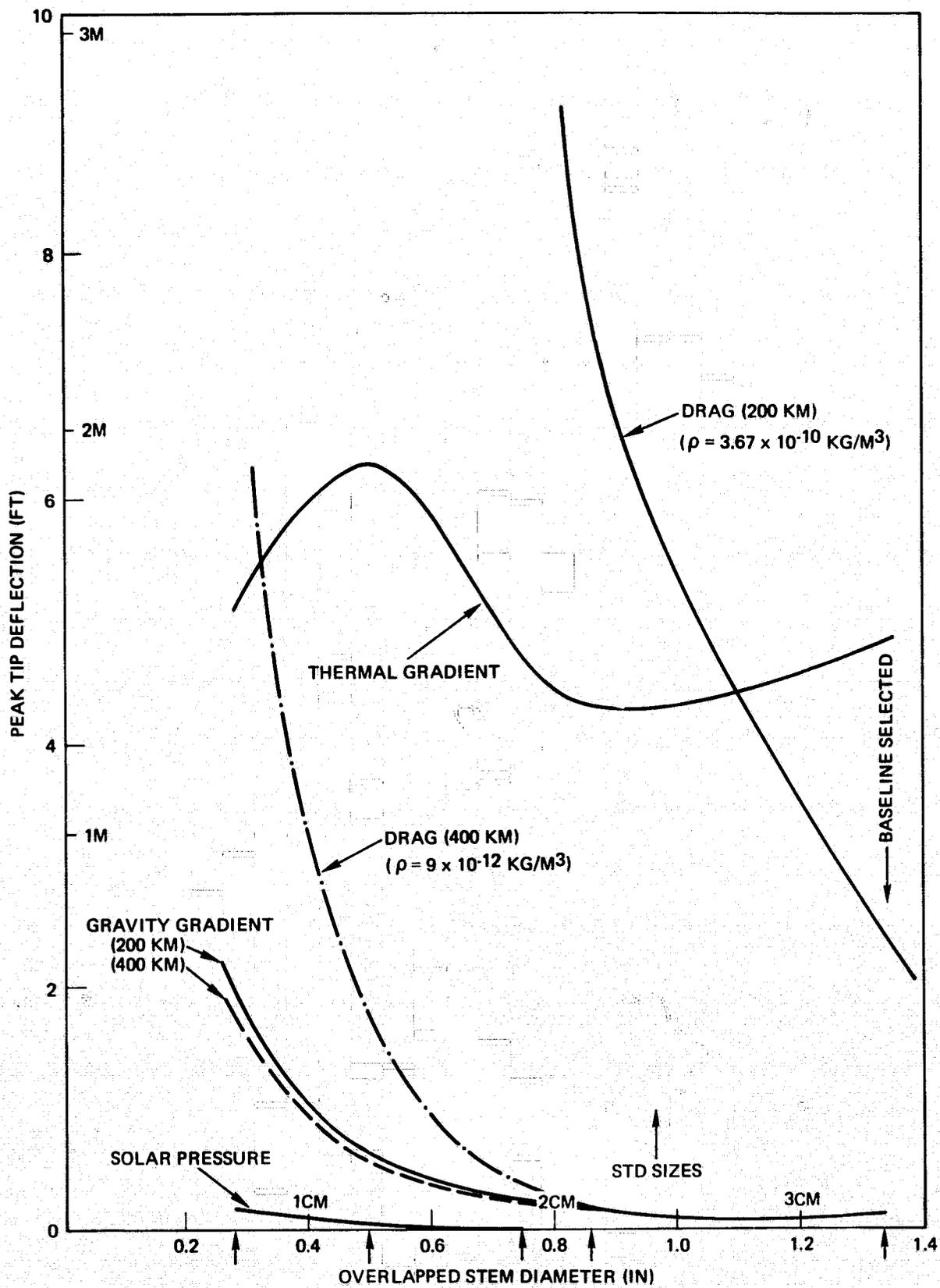


Figure 4-20. Tip Deflections for 50m (164 ft) Overlapped STEM at 200 and 400 km altitudes

Not included in Figure 4-20 are the distortions due to manufacturing and Orbiter accelerations. Available data indicate the radius of curvature due to manufacturing tolerances is greater than 549 meters (1800 feet). The distortions due to the Orbiter RCS action were computed by structurally modeling the fully deployed main support mast and RF sounder booms, determining their modal characteristics, and then calculating their response to the Orbiter pulse train accelerations.

In the analytical investigations, the mast was considered rigidly cantilevered at its base and the Orbiter accelerations were used as a base excitation. Figure 4-21 indicates the number of pulses that can be critically phased with the boom natural periods before the root bending moment exceeds the allowable. The RCS action is assumed to be impulsive. The impulses are assumed equal in magnitude, alternating positive and negative in sign, and spaced uniformly one-half cycle apart. The solutions apply also for impulses equal in magnitude and in sign, and spaced uniformly one cycle apart. The effect of damping was neglected in calculating the system response. The assumption of impulsive loading is reasonably valid up to the seventh mode where the natural vibration period is about ten times the assumed impulse width of 0.25 second. For higher modes, the pulse characteristics become important. The RF sounder deflections and loads shown in Figure 4-18 assume the vernier mode RCS action can be time phased to produce maximum response in the first and second bending modes for 20 pulses. This is equivalent to 16.7 minutes of continuous synchronized RCS action if phased for alternating impulses, or twice that for all positive or all negative impulses. Such continuous action is considered remote, although it must be a subject for additional study. The main support mast response (mode 7) can be amplified by the RCS pulse period of 20 seconds suggested by data in Reference 1, so actual critical phasing with the vernier pulse train is not likely. However, data in this reference indicates the time between pulses may be about 2 seconds if the primary RCS thrusters are used in a limit cycle mode. This is close to the fundamental bending period of the main mast. Although the mast can withstand the rigid body accelerations produced by the primary thrusters, they should not be used in a limit cycle mode when the mast or RF sounder appendages are deployed. Also, maneuvering of the Orbiter must

MODE	FREQUENCY Hz	PERIOD (SEC)	NUMBER OF PHASED IMPULSES REQUIRED TO EXCEED BOOM STRENGTH/CAPABILITY (1)			
			RF ANTENNA		MAIN SUPPORT MAST	
			VERNIER	PRIMARY	VERNIER	PRIMARY
1	0.0099	100	1540	40	(2)	(2)
2	0.0105	95	1360	30	(2)	1980
3	0.0615	16.3	(2)	310	(2)	(2)
4	0.0644	15.5	2360	53	(2)	960
5	0.170	5.9	(2)	1020	(2)	(2)
6	0.177	5.7	2940	83	(2)	680
7	0.327	3.1	740	18	302	7

(1) ACCELERATION IMPULSES FOR $\ddot{\psi}$, $\ddot{\phi}$ AND $\ddot{\gamma}$ ASSUMED PHASED TO PRODUCE MAXIMUM RESPONSE IN EACH MODE.

(2) > 10,000 PULSES

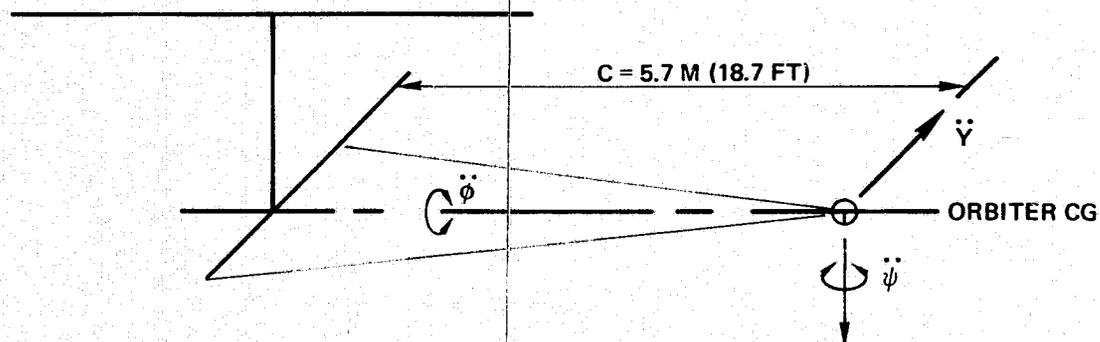


Figure 4-21. Orbiter RCS Induced Maximum Acceleration Levels

be limited while the RF sounders are extended. The 100-second fundamental bending period of these appendages will not permit a rapid translation or rotation of the main support body without causing the booms to buckle.

The effects of the sizable tip mass and flexibility of the main mast can become important to the stability of the Orbiter control system. Assessment of stability and calculation of appendage response feedback to the Orbiter must be accomplished with a model of the Orbiter, its control system characteristics, and flexible appendages but this is beyond the scope of the analysis presented here.

REFERENCES

1. JSC 07700, "Space Shuttle System Payload Accommodations," Vol. XIV, Revision D, Change 15.
2. ESA Ref. No. SLP/2104, "Spacelab Payload Accommodation Handbook," Preliminary Issue, May 1976.
3. "Safety Policy and Requirements for Payloads Using the Space Transportation System," Initial Issue, NASA Headquarters, Office of Space Flight, June 1976.

4.3 COMMUNICATION AND DATA HANDLING SUBSYSTEM

The AMPS Communication and Data Handling Subsystem (CDHS) provides to the payload all the services necessary for instrument command and control, data acquisition, data processing, data displaying, data storing, and data transmission to the ground. The AMPS CDHS configuration is a cost effective approach that makes maximum use of the Shuttle/Orbiter communications equipment and the Spacelab data management equipment. In most cases, the Spacelab Command and Data Management Subsystem (CDMS) interfaces directly with the AMPS instruments and provides the necessary command and control and data acquisition functions. One exception is certain critical caution and warning signals that are hardwired directly from the instruments to the Orbiter caution and warning system.

In the following subsections, the CDHS configuration is discussed. The data handling subsystem requirements and configuration are presented in Section 4.3.1. The communications system requirements of the AMPS payload and how the Shuttle/Orbiter communication subsystems are utilized in satisfying these requirements are presented in Section 4.3.2.

4.3.1 Data Handling Subsystem

A data handling subsystem configuration satisfying the AMPS instrument payload requirements is presented in the following subsections. This configuration requires no additional data handling equipment above that of the Spacelab Command and Data Management Subsystem (CDMS).

4.3.1.1 Instrument Data Requirements

The instrument data requirements derived in this study are based on the Instrument Functional Requirements Documents (IFRD's), Volume VI of this report, and on preliminary conceptual instrument designs performed during this study.

4.3.1.1.1 Command Requirements. A typical instrument data requirement table is presented in Table 4-3 for the lidar instrument. The table is separated into command and output data requirements. The command requirements are further subdivided to define functions, command positions, and command types. The command positions indicate the number of possible states a particular command function is required to accommodate. This ultimately determines the number of data bits required to define a particular command function.

The Spacelab RAU/AMPS instrument interface permits two types of commands, i.e., pulse on/off and PCM serial digital. In allocating command types for each command function, pulse on/off commands are reserved for certain critical functions such as controlling power and laser firing. The PCM serial digital commands are then used for less critical command functions that may have many states or that are essentially continuous over a range of values (see Table 4-3).

The above rationale is based on the fact that each Spacelab Experiment RAU has the capability for only 64 on/off commands. Some of the instrument command functions could have a large number of command states that would quickly use up all of the allocated 64 on/off commands per RAU. Consequently, where possible, PCM serial digital commands are used to handle the less-critical command functions. The PCM serial digital commands have the additional advantage that only one serial digital command line is required to handle all of the multi-state command functions, whereas the pulse on/off command requires a wire from the RAU to the associated AMPS instrument for each of the on/off commands. Unnecessary use of on/off commands would lead to a substantial increase in wiring between the RAU and the science instrument.

For PCM serial digital commands, a PCM command word is allocated for each command function for a given instrument. With a small command decoder within each instrument, all of the serial digital command functions can be processed efficiently over one serial command line. Another advantage is the flexibility permitted in this type of configuration. For instance during payload integration if command changes are required, only software modifications will be necessary with a minimum of hardware impact.

4.3.1.1.2 Output Data Requirements. Output data requirements are also defined in the sample instrument data requirement table shown as Table 4-3. The output data requirements are identified as to their function, type, word length, sampling rate, number of channels, and data rate. These requirements are derived mainly from the IFRD's and from preliminary conceptual instrument designs performed during this study.

Also included in the instrument data requirement table is a summary of the command and output data requirements. The summary is given in terms of the number and types of commands and output data rates.

4.3.1.2 Data Handling Subsystem Configuration

The AMPS science instrument data handling requirements for Flights 1 and 2 range from a few hundred bits per second to several megabits per second. However, no additional data handling support equipment is needed above what is provided by the Spacelab Command and Data Management System (CDMS) for AMPS Flights 1 and 2.

4.3.1.2.1 AMPS Flight 1 Configuration. The specific configuration of the Spacelab CDMS to accommodate the AMPS Flight 1 data handling requirements is shown in Figure 4-22. Orbiter/Spacelab data handling interfaces and the RAU placement necessary for command distribution and data collection on AMPS Flight 1 is also shown in Figure 4-22. Only representative subsystem RAU's are shown here, since detail data requirements on the subsystem RAU's will be defined during the next phase of the program.

Commands to the Spacelab CDMS can originate from three sources. These are from the ground via the Orbiter General Purpose Computer (GPC), preprogrammed onboard commands that are stored in the Spacelab computers, and adaptive commands that are interactively generated in real time using the Spacelab CRT keyboard. The ground commands are sent from the Orbiter GPC to the Spacelab I/O units via the GPC/MDM/I/O unit interface. All of these types of commands are distributed to the appropriate I/O unit, data bus, and, finally, to the proper RAU and instrument.

Low rate telemetry is acquired via the subsystem and experiment data bus with the RAU's interfacing directly with the instruments for data acquisition. The low rate science and engineering data processed by the computers can be displayed on CRT's and/or used by the computers for limit checks. The low rate data is also transmitted to the ground through the Orbiter payload data interleaver via the S-band or the Ku-band communication equipment.

Medium rate science data from the subsystem or experiment I/O units and high rate data from the electron accelerator and the cryogenic interferometer/spectrometer instruments are accepted by the high rate multiplexer (HRM) for transmission to the ground or are recorded for later

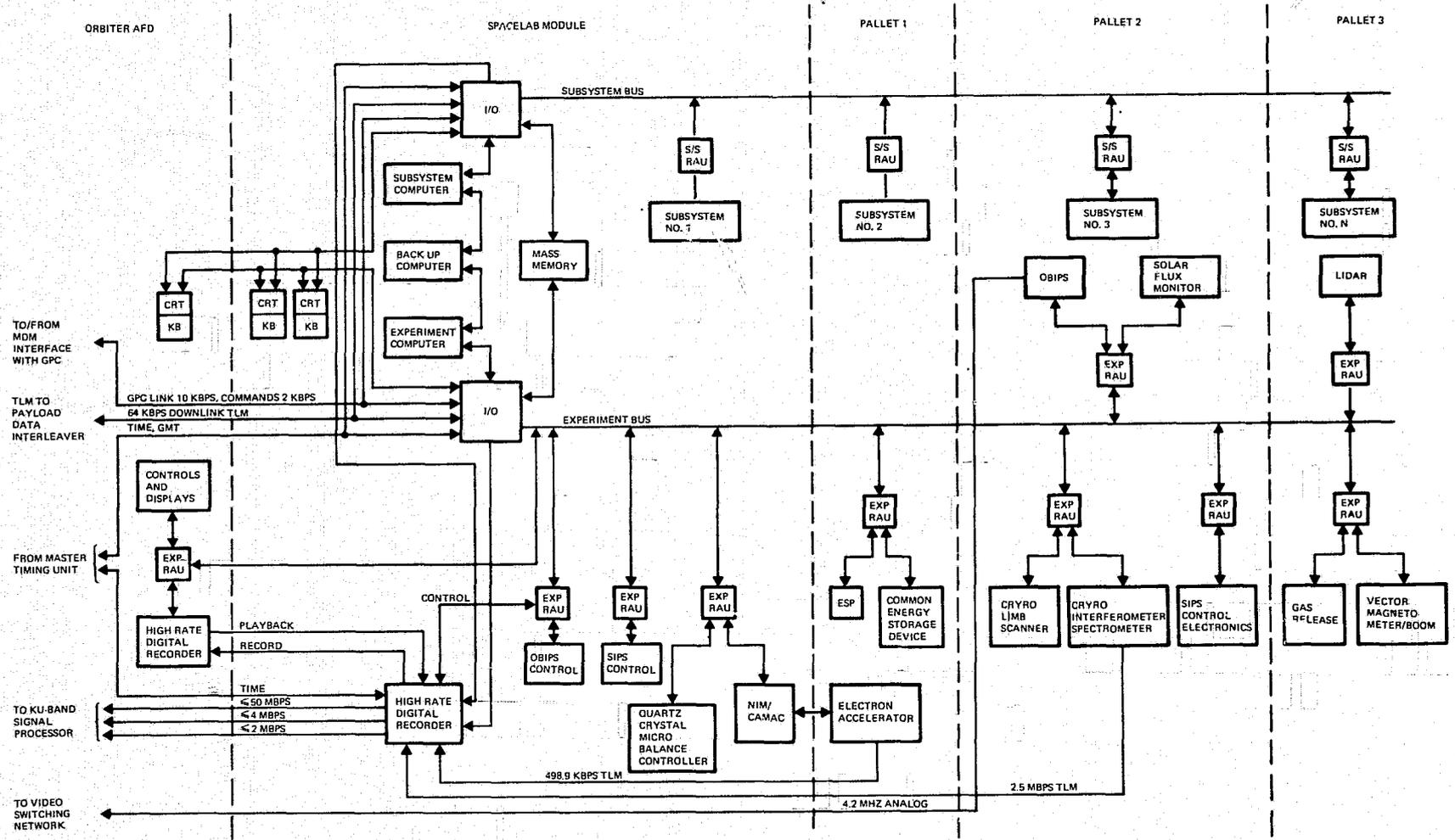


Figure 4-22. AMPS Flight 1 Data Handling Flight Subsystem Configuration

playback through the HRM. The HRM interfaces directly with the Ku-band signal processor which is part of the Ku-band communication equipment on the Orbiter.

An example of the detailed instrument command and data handling requirements for Pallets 1, 2, and 3 is shown in Figure 4-23. For each instrument, the number and types of commands and the science and status data requirements was identified. Also summarized in the figure is the RAU utilization in terms of serial PCM commands, on/off commands, serial telemetry, and flexible inputs. In all but one of the cases, the utilization of each experiment RAU is 50 percent or less for all types of inputs and outputs. Consequently the configurations shown here have substantial growth capability if the data requirements increase when the instruments become better defined.

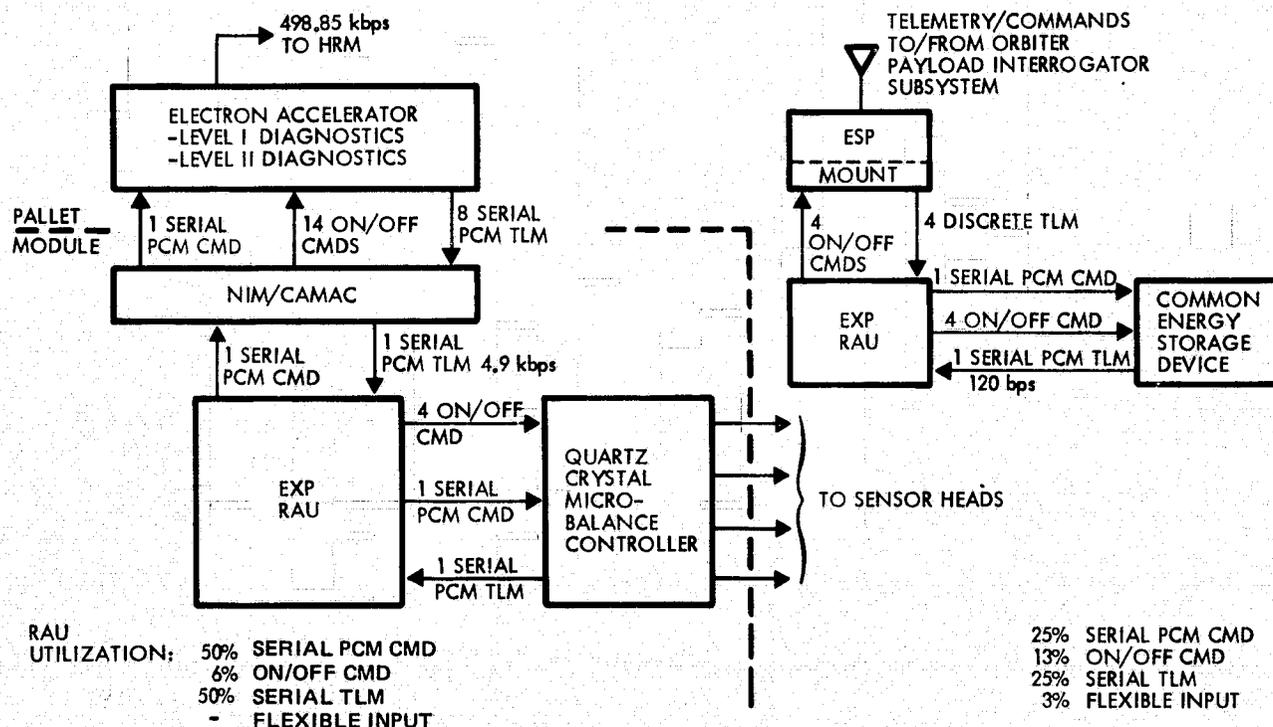


Figure 4-23. Flight 1 - Pallet 1 Data Requirements

4.3.1.2.2 AMPS Flight 2 Configuration. The AMPS Flight 2 Spacelab CDMS configuration for the Orbiter AFD and the Spacelab Module is similar to that shown in Figure 4-22 for Flight 1 and is not repeated here. However, different instruments used in this flight required minor modifications to the pallet data handling configuration.

An example of the detailed instrument command and data handling requirements for Pallets 1, 2, and 3 is shown in Figure 4-24. The RAU utilization for Flight 2 is slightly higher than Flight 1 because of a more complex payload on Flight 2. However, more than adequate margin exists for future growth capability if the data requirements increase when the instruments become better defined.

4.3.1.2.3 Data Processing. Some science instruments such as the lidar generate large blocks of science data. A mini-computer or processor can be teamed with the Spacelab computer to preprocess and condense the raw science data prior to entering the science data onto the Spacelab data bus. Without preprocessing, over 30×10^6 bits of data are generated each time the lidar is fired. With preprocessing, only 3.1×10^3 bits of data are presented to the Spacelab data bus.

Figure 4-25 shows some of the data handling configurations that utilize mini-computers for preprocessing. These techniques permit selective augmentation of the Spacelab CDMS and permit the flexibility necessary for the data handling subsystem to grow with instrument data requirements on the later flights.

Flight 1 data handling subsystem flexibility is shown in Figure 4-23. This configuration shows how the electron accelerator can be selectively augmented to permit additional data processing capability. The electron accelerator is augmented with NIM/CAMAC equipment, which contains a data processor within the instrument to reduce the data processing requirements on the Spacelab computer. These configurations are two of the alternative approaches used to demonstrate preprocessing capabilities that can be used to selectively augment the Spacelab CDMS as required.

4.3.1.2.4 Environmental Sensor Package Data Handling Configuration. The Environmental Sensor Package (ESP), described in Section 3.2.2, is a free flyer type of satellite launched from the Shuttle cargo bay and is part of

4-42

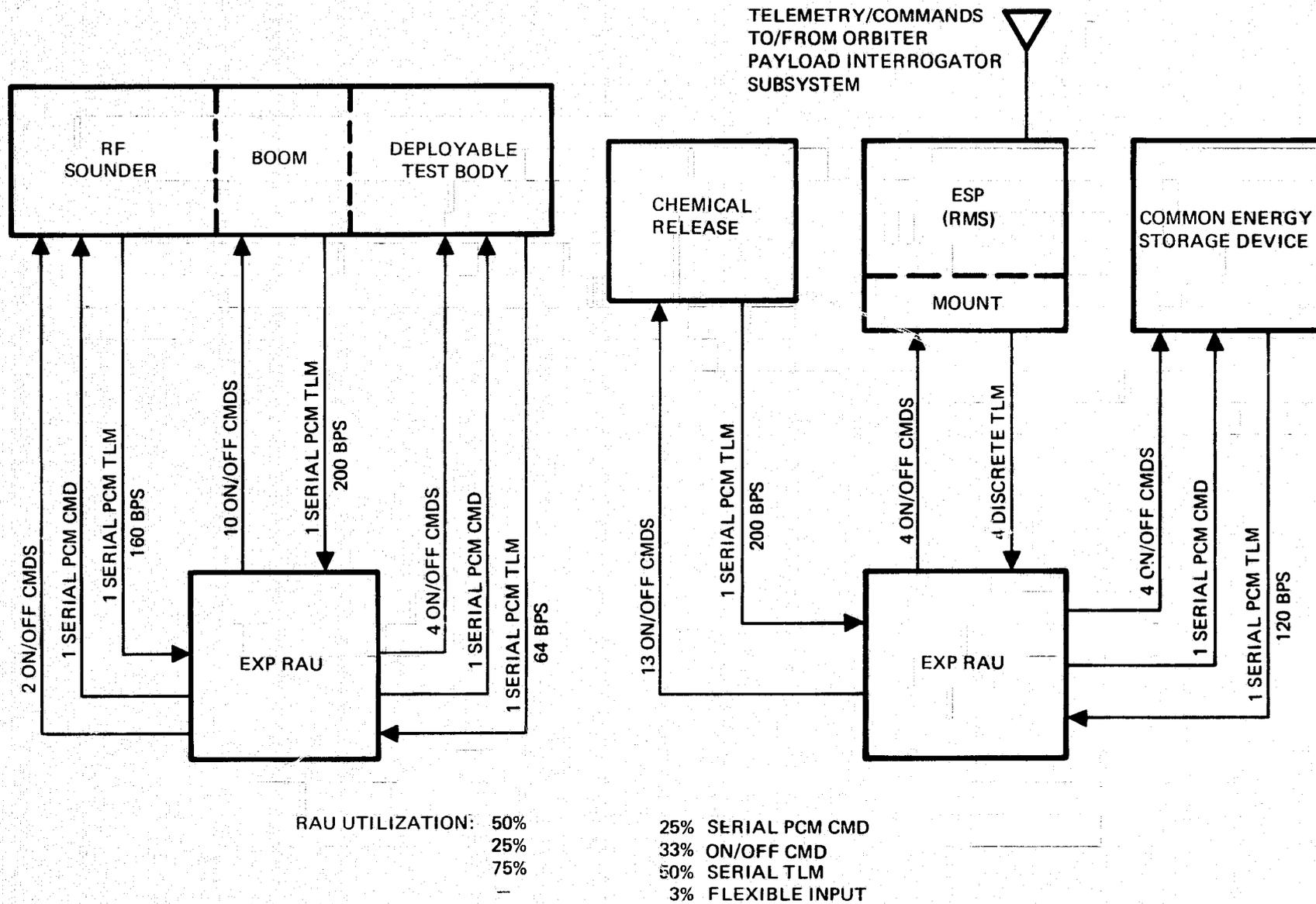


Figure 4-24. Flight 2 - Pallet 1 Data Requirements

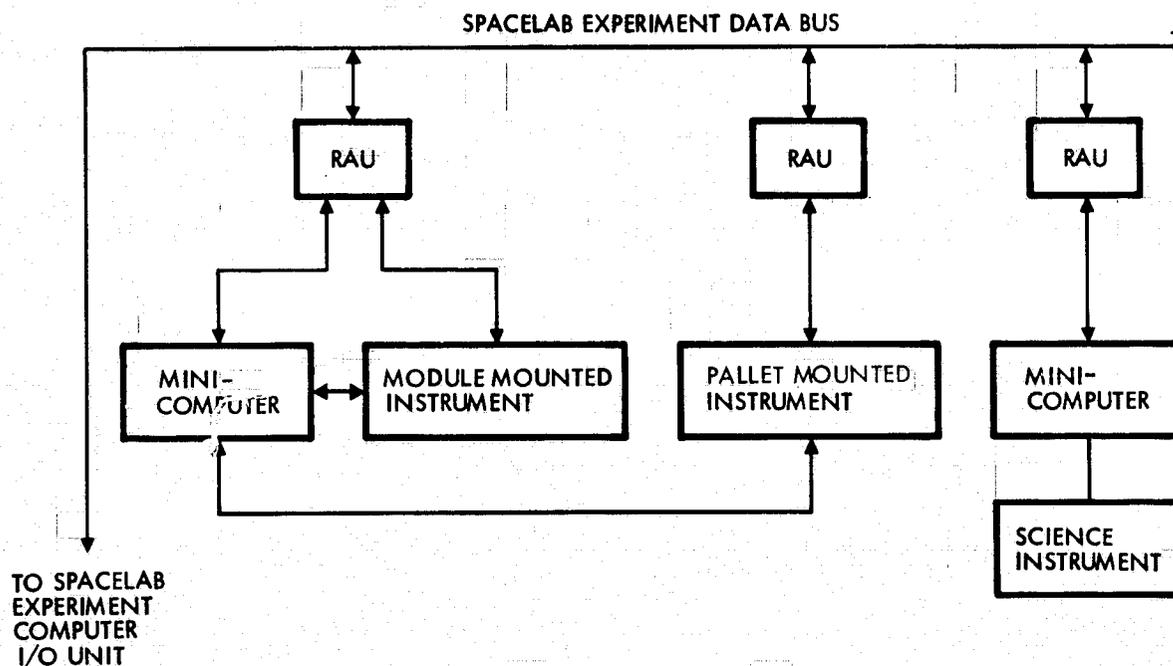


Figure 4-25. Typical Preprocessing Configurations for the Selective Augmentation of the Spacelab Command and Data Management Subsystem

the AMPS payload. The science instruments on board the ESP utilize the Orbiter Payload Interrogator Subsystem to provide the necessary command and telemetry links. Via this link, the Orbiter is capable of commanding at a 2 kbps rate and receiving telemetry data at 16 kbps.

The ESP's require essentially the same complement of science instruments for both Flights 1 and 2. The ESP will consist of seven science instruments. For Flights 1 and 2, all seven instruments are not required to be active during any one experiment. Therefore, the instruments can be grouped for operation in two different modes, i.e., Mode 1 and Mode 2 operation. The grouping of the instruments into two modes of operation permits the best utilization of the Orbiter payload interrogator data link and still remains within the total 16 kbps maximum data rate.

The active instruments for Mode 1 operation are:

- 1) Medium energy electron detector
- 2) DC electric field
- 3) E and B receivers
- 4) Vector magnetometer
- 5) Langmuir probe.

The active instruments for Mode 2 operation are:

- 1) Ion mass and distribution analyzer
- 2) Neutral mass spectrometer
- 3) Vector magnetometer
- 4) Langmuir probe.

Figure 4-26 shows the ESP data handling configuration that permits maximum science data return.

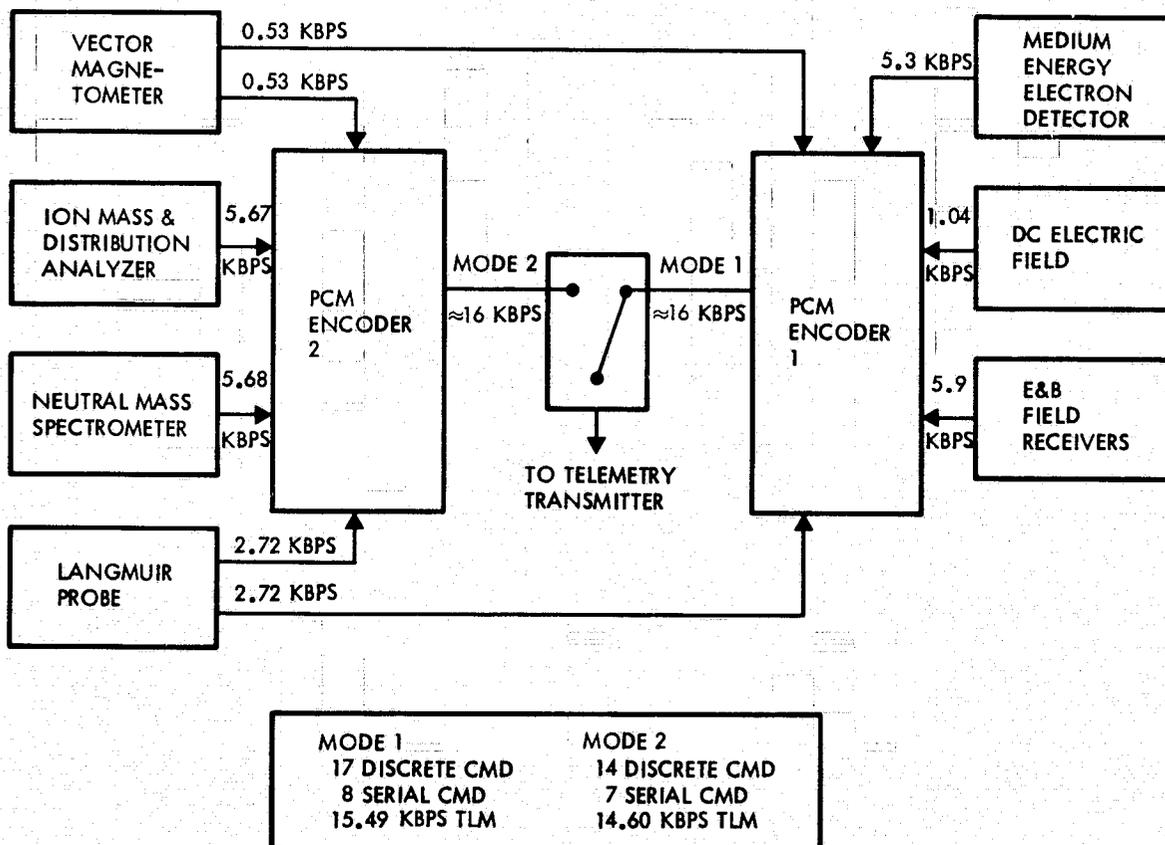


Figure 4-26. ESP Data Handling Configuration

4.3.1.2.5 Data Formats. Flight 1 consists of seven individual experiments and Flight 2 consists of 10 individual experiments. Flights 1 and 2 experiment/instrument complement matrices are shown in Section 2.5 of this volume. The synthesis of data formats that is compatible with the payload is highly dependent upon the mission operation and flight timeline that will be discussed in Section 5. Due to the large

number of instruments and experiments, a different format for each situation is undesirable. At the same time, one format capable of satisfying all the instruments would waste valuable transmission bandwidth. Therefore, a solution that is between these extremes is highly desirable.

Earlier, the data handling requirements on an instrument basis were presented. The data handling requirements on an experiment basis are presented for Flights 1 and 2 in Tables 4-4 and 4-5, respectively.

Examining these data handling requirements, both Flights 1 and 2 can be accommodated with two digital formats and one analog format. The data format requirements for Flights 1 and 2 are shown in Tables 4-6 and 4-7, respectively.

For Flight 1, two digital formats are required; i.e., 1A at 2.6 mbps and 1B at 536 kbps. Format 1A is capable of supporting any one of the high data rate experiments (Acoustic Gravity Wave, Gas Cloud Expansion, or the Minor Constituent) plus the low data rate Environmental Measurements and Solar Flux Radiation experiments simultaneously. Format 1B can support one of the medium data rate experiments (Electron Accelerator or Beam Plasma Interaction) plus the low data rate Environmental Measurements and Solar Flux Radiation experiments. Format 1C accommodates the video signal that is generated by the OBIPS in support of certain experiments shown in Table 4-6.

Flight 2 also requires two digital formats, i.e., 2A at 2.6 mbps and 2B at 49 kbps. Format 2A can support any one of the high data rate experiments (Minor Constituents, D-Region Composition, Anomalous Composition, or High Altitude Ozone) plus the low data rate Oxygen Variation and Solar Flux Radiation experiments simultaneously. Format 2B can support any two of the four medium data rate experiments (Conductivity Modification, HF Wave Particle Interaction, Long Delay Echoes, and Plasma Flow) simultaneously plus the low data Oxygen Variation and Solar Flux Radiation experiments. Format 2C is similarly used to accommodate the video signal that is generated by the OBIPS in support of certain experiments shown in Table 4-7.

Table 4-4. Data Handling Requirements for Flight 1

Experiment	Maximum Data Requirements		Comments
	Digital	Analog (MHz)	
Acoustic Gravity Wave	2.51 mbps	4.2	Analog is video signal generated by the optical band imager
Gas Cloud Expansion	2.51 mbps	4.2	
Electron Accelerator	516.0 kbps	4.2	
Beam Plasma Interactions	516.0 kbps	4.2	
Minor Constituents	2.56 mbps		Highest data rate
Environmental Measurements	16.7 kbps		
Solar Flux Radiation	3.2 kbps		Performed 30 minutes once per day

Table 4-5. Data Handling Requirements for Flight 2

Experiment	Maximum Data Requirements		Comments
	Digital	Analog (MHz)	
Conductivity Modification	17.4 kbps	4.2	Analog signal is video from optical band imager
HF/Wave Particle Interaction	17.3 kbps	-	
Long Delay Echoes	17.3 kbps	-	
Plasma Flow	17.2 kbps	4.2	
Minor Constituents	2.57 mbps	-	Highest data rate
D-Region Composition	2.57 mbps	-	
Oxygen Variation	10.2 kbps	-	
Anomalous Composition	2.57 mbps	4.2	
High Altitude Ozone	2.57 mbps	-	
Solar Flux Radiation	3.2 kbps	-	Performed 30 minutes once per day

Table 4-6. Flight 1 Experiment Data Format Requirements

Format No. Experiments	1A (2.6 mbps)	1B (536 kbps)	1C (4.2 MHz)
1) Acoustic Gravity Wave	x		x
2) Gas Cloud Expansion	x		x
3) Electron Accelerator		x	x
4) Beam Plasma Interaction		x	x
5) Minor Constituents	x		
6) Environmental Measurements	x	x	
7) Solar Flux Radiation	x	x	

Table 4-7. Flight 2 Experiment Data Format Requirements

Format No. Experiments	2A (2.6 mbps)	2B (49 kbps)	2C (4.2 MHz)
1) Conductivity Modification		x	x
2) HF Wave Particle Interaction		x	
3) Long Delay Echoes		x	
4) Plasma Flow		x	x
5) Minor Constituents	x		
6) D-Region Composition	x		
7) Oxygen Variation	x	x	
8) Anomalous Composition	x		x
9) High Altitude Ozone	x		
10) Solar Flux Radiation	x	x	

4.3.2 Communication Subsystem Description

The AMPS payload communication requirements are satisfied completely by the Orbiter S-band and Ku-band communication subsystems. Figure 4-27 shows an overview of the Orbiter communication links and capabilities to the ground.

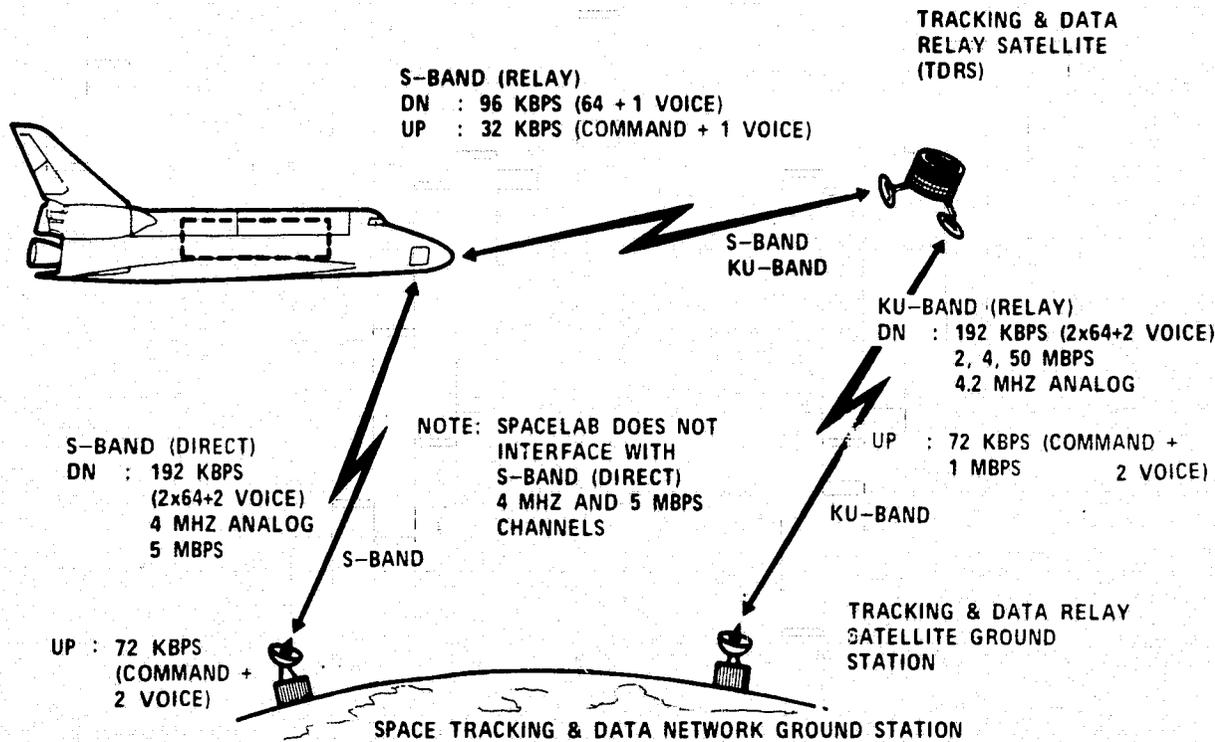


Figure 4-27. Orbiter Communication Links

4.3.2.1 Communication Subsystem Configuration

The transmission of the data generated by the AMPS payload is performed by the Orbiter avionics equipment. The Orbiter communication capabilities are discussed below in terms of the communication links necessary for the flow of data between the ground and the instrument payload.

4.3.2.2 Uplink/Forward Link Communications

The Orbiter communications equipment provides a 72 kbps uplink time division multiplexed (TDM) data channel containing command and voice data. This link is either via the STDN S-band or relayed via TDRSS S-band or

Ku-band (see Figure 4-27) forward link (ground/TDRS/Orbiter). The 72 kbps uplink data is composed of two voice channels at 32 kbps each and one command channel with a symbol rate of 8 kbps [2 kbps information rate which has been BCH (Bose-Chaudhuri-Hocquenghem) encoded].

The Ku-band forward link signal is routed to the network signal processor (NSP) via the Ku-band signal processor. If the uplink signal is from the S-band system, the same 72 kbps is provided to the NSP and processed in the same manner.

The NSP decodes the BCH encoded command data and routes it to the Orbiter general purpose computer (GPC) via the multiplexer/demultiplexer (MDM). From there the commands are sent to the Spacelab subsystem or experiment computer (via the MDM and the Spacelab I/O units) for execution. The digital voice channels are demodulated and converted into two analog voice signals by the NSP before routing to the audio central control unit for distribution to the Spacelab master intercom unit.

The Ku-band forward link also has the capability for a 1 mbps data channel. However, this channel will not interface with the Spacelab CDMS.

4.3.2.3 Downlink/Return Link Communications

The transmission of the data generated by the AMPS payload is separated into two categories:

- 1) Housekeeping and low rate payload data
- 2) Wideband payload data.

The processing and routing of these two data types by the Orbiter communications equipment are discussed below.

4.3.2.3.1 Housekeeping and Low Rate Payload Data. The Orbiter housekeeping and low rate data links to the ground is a 192 kbps data stream. The 192 kbps data stream is composed of the following:

- 1) Two voice channels, 32 kbps each
- 2) Orbiter telemetry data, 64 kbps
- 3) Spacelab data from the experiment and subsystem I/O unit outputs, 64 kbps nominal.

The composition of the data in the 192 kbps data stream is software controlled through the PCM master unit. The PCM master unit acquires the data from different sources (Orbiter GPC and payload data interleaver) in a demand response manner to provide the capability of a variable data channel allocation between the Orbiter and the payload data sources. This would mean that during periods of low activity on Orbiter subsystems some of its data channel capacity can be allocated to the payload subsystems to maintain a total data channel output of 128 kbps from the PCM master unit.

The PCM master unit provides the 128 kbps data stream to the NSP. Also provided to the NSP are two analog voice channels from the Audio Central Control Unit. The two-voice signals are digitized by the NSP into two 32 kbps digital signals. These digital streams are multiplexed into a single 192 kbps data stream for transmission to ground direct (STDN) or via TDRSS. When TDRSS coverage is not available, the 192 kbps telemetry data is recorded on the maintenance PCM recorder in the Orbiter for later playback when TDRSS coverage is available.

4.3.2.3.2 Wideband Payload Data. The AMPS wideband science data consists of the digital data output from the high rate multiplexer (HRM) and the 4.2 MHz video signals from the OBIPS instrument. The wideband data is transmitted to the ground only via the TDRSS Ku-band return link (Orbiter/TDRS/ground). When TDRSS coverage is not available, the Spacelab high rate digital recorder (HRDR) is used to record digital data during the TDRSS outage periods. Similarly, an analog/video tape recorder within the OBIPS instrument is used to record its wideband analog data for subsequent playback when the analog channel is not available for transmission to the ground.

The Ku-band signal processor is used to control the Ku-band return link operating mode. The return link data is either phase modulated (Mode 1) or frequency modulated (Mode 2). Latest changes in the Ku-band communication equipment permits both modes to have a three-channel capability, with a wideband channel (50 mbps or 4.2 MHz) and a subcarrier carrying two channels (≤ 2 mbps and 192 kbps) in an unbalanced QPSK format.

Mode 1 is strictly a digital link. The 2 mbps channel 1 can be used to dump Orbiter tape recorders such as the MSS tape recorder or interface with the 2 mbps channel from the Spacelab high rate multiplexer (HRM). The 50 mbps channel 2 is used to interface with the HRM. Channel 3 is used to transmit the housekeeping and low rate payload data at 192 kbps.

Mode 2 is a frequency modulated link whose channels 1 and 3 contain the same type of data as in Mode 1. Channel 2 can be selected to transmit either 4 mbps digital data from the HRM or the 4.2 MHz analog signal from the OBIPS instrument via the Orbiter video switching network.

4.3.2.3.3 Environmental Sensor Package Communication Link. The ESP will utilize the Orbiter Payload Interrogator Subsystem to provide the necessary command and telemetry links. The command link (Orbiter to ESP) provides an information rate of 2 kbps (8 kbps symbol rate). These commands can originate from the ground, the Orbiter GPC, or the Spacelab computer via the Spacelab I/O unit/MDM interface.

The telemetry link is capable of accommodating a 16 kbps data rate maximum. The ESP telemetry is received by the Payload Interrogator Subsystem and routed to the payload signal processor. The payload signal processor detects and bit synchronizes the telemetry stream and provides data and clock signals to the payload data interleaver for transmission to the ground via the S-band or Ku-band communication equipment.

The science data from the ESP is complementary to that generated by the pallet mounted instruments; i.e., the ESP data is used in real time or stored on board the Spacelab CDMS for delayed use in processing and analyzing data from pallet-mounted instruments. In this study it had been assumed that capability to transfer ESP data from the payload signal processor (or some other equivalent interface unit) to the Spacelab CDMS exists. However, current Orbiter avionics equipment and software requirements do not permit a direct real-time transfer of the 16 kbps ESP data from the Orbiter to the Spacelab CDMS. This interface needs to be further assessed in future studies.

The ESP RF link performances via the Orbiter Payload Interrogator Subsystem are given in Tables 4-8 and 4-9. Commands can be received by the ESP at a range of 100 km with command margin of 10.6 dB after

Table 4-8. Payload Interrogator Orbiter to ESP Command Link Performance

Parameter	Nominal Value	Adverse Tolerance	Notes
1) Modulation format	PCM/PSK/PM		
2) Frequency (MHz)	2115.6	0.0	
3) Transmit power (dBm)	37.0	0.5	5 watts nominal
4) Transmit circuit loss (dB)	6.6	0.5	Rockwell estimate
5) Transmit antenna gain (dBi)	3.0	0.0	Minimum for 100-deg cone off the top of orbiter
6) Orbiter EIRP (dBm)	33.4	0.7	RSS tolerance
7) Space loss (dB)	139.0	0.0	Maximum range = 100 km
8) ESP antenna gain (dBi)	3.0	0.0	Toroidal beam antenna; minimum gain ± 9 deg from beam peak
9) ESP receive circuit loss (dB)	2.0	0.5	Assumed tolerance
10) Total received power at receiver input (dBm)	-104.6	0.9	RSS tolerance
11) ESP system noise temperature referred to receiver input (°K)	2900.0	0.0	Assumes antenna temperature = 290°K, circuit loss = 2 dB, and receiver noise figure = 10 dB
12) Spacecraft noise spectral density referred to receiver input (dBm/Hz)	-154.0	0.0	
13) Total received power to noise spectral density (dB-Hz)	59.4	0.9	RSS tolerance
<u>Carrier Performance</u>			
14) Carrier modulation loss (dB)	2.9	0.6	Assumes command data is phase modulated onto carrier at 1.1 rad ± 10 percent
15) Received carrier to noise spectral density (dB-Hz)	56.5	1.1	RSS tolerance
16) Carrier loop noise bandwidth (dB-Hz)	24.8	0.0	$2 B_{LO} = 300$ Hz
17) Carrier-to-noise ratio (dB)	31.7	1.1	RSS tolerance
18) Carrier loop threshold (dB)	7.0	2.0	Assumed tolerance
19) Carrier performance margin (dB)	24.7	2.3	RSS tolerance
20) Carrier performance margin less adverse tolerance (dB)	22.4	-	
<u>Command Performance</u>			
21) Data modulation loss (dB)	3.5	0.7	See Note 14)
22) Received signal-to-noise spectral density ratio (dB-Hz)	55.9	1.1	RSS tolerance
23) Command data rate (dB-Hz)	33.0	0.0	2 kbps
24) Received energy per bit to noise spectral density ratio (dB)	22.9	1.1	RSS tolerance
25) Required theoretical energy per bit to noise spectral density ratio (dB)	9.6	0.0	For 10^{-5} BER
26) Degradation due to nonoptimum detection (dB)	1.5	0.5	Due to hardware implementation
27) Command performance margin (dB)	11.8	1.2	RSS tolerance
28) Command performance margin less adverse tolerance (dB)	10.6	-	

Table 4-9. Payload Interrogator ESP to Orbiter Telemetry Link Performance

Parameter	Nominal Value	Adverse Tolerance	Notes
1) Modulation format	PCM/PM		
2) Frequency (MHz)	2297.5	0.0	
3) ESP transmitter power (dBm)	36.0	0.0	4-watt minimum
4) Transmit circuit loss (dB)	2.0	0.5	Assumed tolerance
5) Transmit antenna gain (dBi)	3.0	0.0	Toroidal beam antenna; minimum gain ± 9 deg from beam peak
6) ESP EIRP (dBm)	37.0	0.5	RSS tolerance
7) Space loss (dB)	139.7	0.0	Maximum range = 100 km
8) Orbiter antenna gain (dBi)	3.0	0.0	Minimum for 100-deg cone off the top of orbiter
9) Orbiter receiver circuit loss (dB)	6.6	0.5	Assumed tolerance
10) Total received power at receiver input (dBm)	-106.3	0.7	RSS tolerance
11) Orbiter system noise temperature referred to receiver input ($^{\circ}$ K)	1829.8	0.0	Assumes antenna temperature = 290° K, circuit loss = 6.6 dB, and payload interrogator receiver noise figure = 8.0 dB
12) Orbiter noise spectral density referred to receiver input (dBm/Hz)	-166.0	0.0	
13) Total received power to noise spectral density (dB-Hz)	59.7	0.7	RSS tolerance
<u>Carrier Performance</u>			
14) Carrier modulation loss (dB)	2.9	0.6	Telemetry data directly phase modulates carrier at 1.1 rad ± 10 percent
15) Received carrier to noise spectral density (dB-Hz)	56.8	0.9	RSS tolerance
16) Carrier loop noise bandwidth (dB-Hz)	24.8	0.0	$2 B_{LO} = 300$ Hz
17) Carrier to noise ratio (dB)	32.0	0.9	RSS tolerance
18) Carrier loop threshold (dB)	7.0	2.0	Assumed tolerance
19) Carrier performance margin (dB)	25.0	2.2	RSS tolerance
20) Carrier performance margin less adverse tolerance (dB)	22.8	-	
<u>Telemetry Performance</u>			
21) Data modulation loss (dB)	3.5	0.7	See Note 14)
22) Received signal-to-noise spectral density ratio (dB-Hz)	56.2	1.0	RSS tolerance
23) Telemetry data rate	42.0	0.0	16 kbps
24) Received energy per bit to noise spectral density ratio (dB)	14.2	1.0	RSS tolerance
25) Required theoretical energy per bit to noise spectral density ratio (dB)	9.6	0.0	For 10^{-5} BER
26) Degradation due to nonoptimum detection (dB)	1.5	0.5	Due to hardware implementation
27) Telemetry performance margin (dB)	3.1	1.1	RSS tolerance
28) Telemetry performance less adverse tolerance (dB)	2.0	-	

adverse tolerance. The telemetry link from the ESP to the Orbiter requires a 4-watt S-band transmitter to provide a nominal telemetry performance margin of 3.1 dB (2.0 dB margin after adverse tolerances) at a maximum range of 100 km.

4.4 CONTROLS AND DISPLAYS

4.4.1 Introduction

A primary objective of this study was to define the controls and displays (C&D) required to conduct AMPS Flight 1 and 2 experiments from the pressurized module of the Spacelab and/or aft flight deck (AFD) of the Orbiter. Secondary objectives were to define the experiment or flight support equipment (FSE) C&D applicable to subsequent AMPS flights or to other Labcraft payloads.

4.4.2 General Description of Control and Display Development Process

The experiment and FSE C&D were developed through a comprehensive analysis and evaluation process. During the conceptual phase of the study, the effort concentrated upon definition of requirements. In the preliminary design phase, the activities were directed toward evaluation of configuration alternatives, procedural and command data requirements, and development of instrument and FSE dedicated C&D panel designs.

4.4.2.1 Control and Display Requirements Definition

The objective of this requirements analysis effort was to identify the instrument and FSE C&D which would enable the payload specialist to function in his role as a scientist, observer, and decision maker to enhance the quality of scientific data acquired, rather than as a "procedural switch flipper." The definition of these C&D requirements was accomplished through the following analytical activities:

- a) Instrument/FSE functional analysis
- b) Experiment operations sequence analysis
- c) Instrument/FSE function allocation analysis
- c) Instrument/FSE payload control and display synthesis.

4.4.2.1.1 Instrument/FSE Functional Analysis. In order to define experiment operations and related C&D requirements, it was essential to first define operating functions of each instrument and the FSE as they would be employed in performing the various experiments in the Spacelab environment. This was accomplished in steps. First, the purpose of each experiment was described. Second, the instruments and FSE required were identified. Next, the constraints on operations imposed by the experiment

were defined. Fourth, the on-orbit operations requirements and assumptions were stated. Finally, using the respective IFRD's, an operations functional analysis was performed to define the operations control and data display capability required for the instrument/FSE complement associated with a specific experiment. Figure 4-28 illustrates the type of documentation developed for each experiment, with the functional analysis documented in the form of functional flow diagrams (Figure 4-29).

4.4.2.1.2 Experiment Operations Sequence Analysis. The operations requirements derived from the instrument/FSE functional analysis, on-orbit requirements, and experiment operations/display requirements were then combined into an experiment operations sequence timeline. An example of this type of documentation is shown in Figure 4-30.

4.4.2.1.3 Instrument/FSE Function Allocation Analysis. The C&D requirements for each of the instruments and FSE used in an experiment were extracted from the above documentation and tabulated. A preliminary determination was made of which control functions could be preprogrammed for automatic operation, and which should be manually controlled either through the CDMS keyboard or a dedicated control. These operational control functions were then compared with the Spacelab Payload Accommodations Handbook to determine which functions could be accommodated by Spacelab mission independent or dependent equipment. Figure 4-31 is typical of the tabulations performed for each experiment.

A preliminary assignment of the control function was then made according to the following criteria and which would achieve the objective stated in paragraph 4.4.2.1.

- a) Automatic functions
 - Activation/calibration/shutdown sequences
 - Nominal operations
 - Status monitoring and update
 - Data acquisition, processing and routing
 - Display generation
 - Safety monitoring.

MISSION 1

Experiment A-1(a), Acoustic Gravity Wave

Experiment A-2(a), Gas Cloud Expansion Dynamics

Experiment Purpose - Experiments A-1 and A-2 are combined to study gas cloud dynamics (expansion temperature and braking) from the ground station at Arecibo, $\lambda 18.3^{\circ}W$, and the Orbiter.

Instruments

- 1) Chemical Gas Release (I-21) - Up to seven, 50 to 100 Kg, canister releases. One release per pass. Chemical proposed is Xe at 300 atmospheres.
- 2) OBIPS (II-3) - Visual only for 1st flight. Used to observe gas cloud expansion.
- 3) NIR Spectrometer/Photometer (II-9) Detection and measurement of g . cloud emissions.

Constraints

- 1) Gas release at 200 Km altitude
- 2) Shuttle at 200 Km altitude
- 3) Observation time 10 seconds
- 4) All gas releases as near as possible to Arecibo

Operations Requirements and Assumptions

- 1) Canister release is forward along orbit path approximately two orbits before Arecibo pass. Canister orbit is ~ 203 Km x 200 Km. Timing and orbit such that Shuttle is leading canister by 20 Km and at the same altitude upon gas release.
 - a) Assume ground processing of Spacecraft position and release time calculations.
 - b) Assume fixed ejection impulse and automatic release.
 - c) Assume canister tracking by ground radar.
- 2) Canisters provided with self-contained timer
- 3) Arming and firing method is self-contained
- 4) OBIPS and NIR mounted on SIPS Pointing and tracking will be automatic. Automatic start of sensors.
- 5) No overboard dump of H_2O immediately prior to or during experiment. All optical surfaces will be protected from contamination.
- 6) No direct outgassing or venting onto optical surfaces: Minimum outgassing and venting during operation.

Figure 4-28. Example of Experiment Documentation

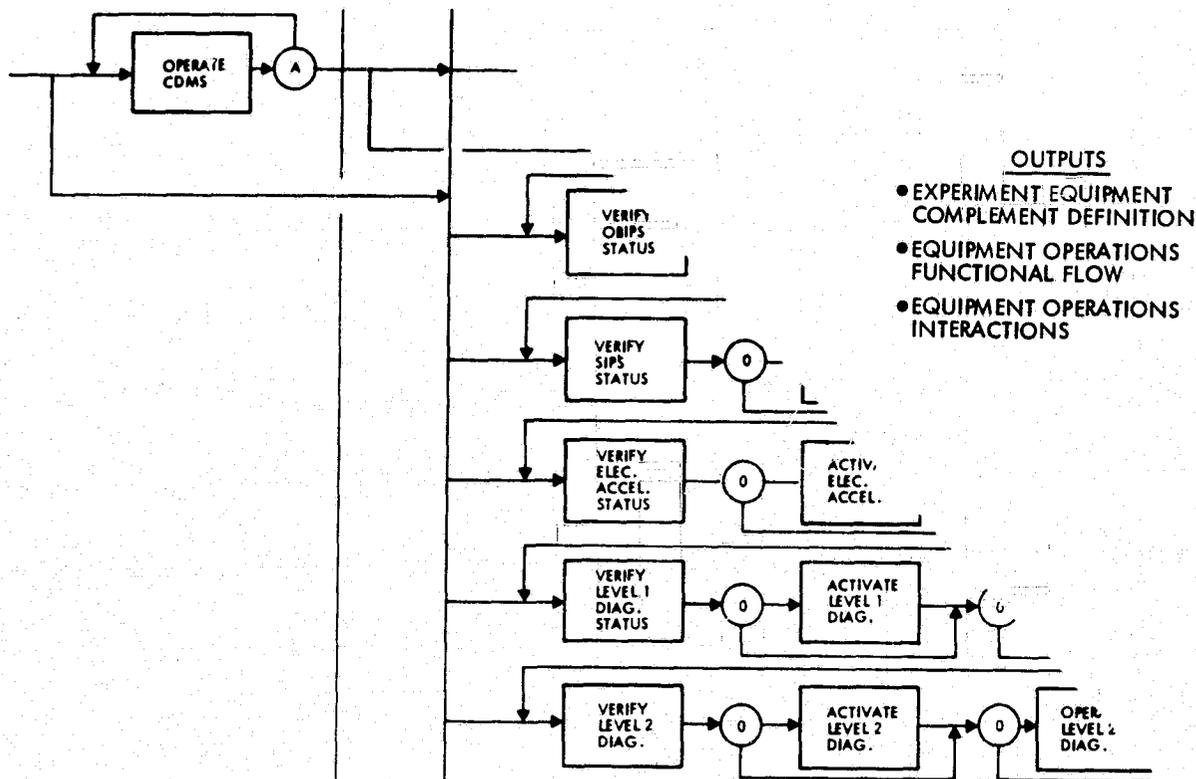


Figure 4-29. Instrument/FSE Operation Analysis Example

TIME	SHUTTLE OPS	INSTRUMENT/FSE OPS	DATA ANALYSIS OPS
	MANEUVER FOR Z AXIS PARALLEL TO MAG FIELD - TRANSMIT S BAND COMMANDS TO ESP	PERFORM CATHODE BURN IN - INCREASE HEATER VOLTAGE (STEPED INCREASE) - CALL UP CATHODE CURRENT DISPLAY - REPEAT TIMED SEQUENCES	
		INITIATE A-3 OPERATIONS - INITIATE AUTO-OPS SEQUENCE ESP 1, OBIPS - INITIATE ELECTRON ACCEL AUTO SEQUENCE	

Figure 4-30. Operations Sequence Diagram

Mission 1, Rev 2 Experiments A-1(a), A-2(a)

Controls Requirements

OBIPS

S/L Dedicated

- TV Monitor Controls (M)
- Video Switching (M)
- Video Recording (M)
- Telemetry Output (A)
- Video Playback (M)

Note: M = manual
A = automatic
However, manual may be a dedicated control or keyboard input.

Experiment Dedicated

(Pointing - see SIPS)

Optical Elements Control M/A

- a) FOV 16⁰ to 1⁰
- b) Interference filters
- c) Light Shield
- d) Lens Cover

Scan Mode Select M/A

- a) Standard
- b) Integrated
- c) Special

Subsystem Select M/A

- a) No. 1
- b) No. 2
- c) No. 3

TV Camera Control M

- a) No. 1
- b) No. 2
- c) No. 3

Photometer Gains M/A

- a) No. 1
- b) No. 2
- c) No. 3

TV Gains M/A

- a) No. 1
- b) No. 2
- c) No. 3

Voltage (Power ON/OFF)

Calibration 11

Figure 4-31. Function Allocation Analysis Worksheet

b) Manual functions - CDMS keyboard

- Mode selection
- Sequence control
- Parameter change
- Display callup
- Command activation.

c) Manual functions - dedicated control

- Target of opportunity painting
- Real time modifications
- Field-of-view adjustments
- Safety enable, jettison control.

4.4.2.1.4 Instrument/FSE Payload Control and Display Synthesis. The final C&D requirements definition activity was then directed toward consolidating the C&D requirements for each instrument and related FSE as the equipment is applied in each experiment operation. Included in this integrated analysis were the requirements derived for later AMPS and Labcraft payloads which employed the same instruments or those with expanded capability. The consolidated C&D requirements along with their preliminary assignment to automatic or manual keyboard/dedicated control were tabulated for each item (e.g., OBIPS, UV-VIS-NIR Spectrometer, SIPS, etc.).

4.4.2.2 Preliminary Control and Display Design Process

The information developed in the conceptual design phase was used in the selection of the type of controls and displays required by the Payload Specialist to operate and control the instruments and interpret the data in relation to experiment performance and operations timeline. The following criteria were used in selecting the specific controls, displays, and display formats:

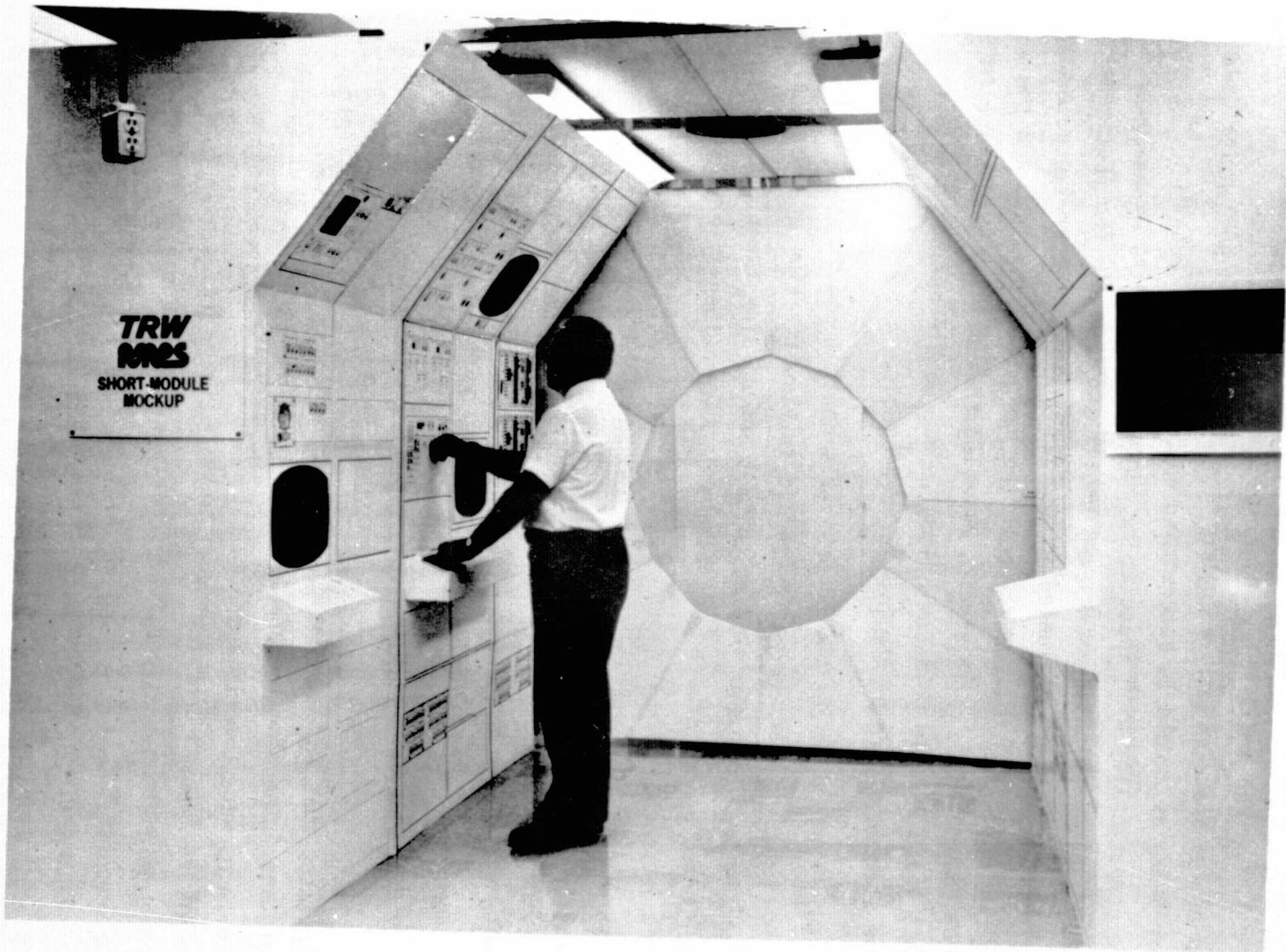
- a) Accommodate safety requirements
- b) Compatible with Spacelab resources
- c) Meet science requirements
- d) Meet payload specialist requirements, capabilities and training
- e) Make effective use of Spacelab provided C&D capability
- f) Accommodate future/multiple payload requirements

- g) Minimize cost
- h) Comply with human engineering requirements.

4.4.2.2.1 Control and Display Selection and Layout Studies. A full scale mockup of the Spacelab module control and display racks was constructed (Figure 4-32) as an aid in determining an optimum configuration. To ensure compatibility with Spacelab resources the existing panels were indicated. The dedicated experiment panels indicated in the requirements definitions study were evaluated in various positions to arrive at a configuration which was optimum for both the payload specialists and the science requirements. Our studies revealed that all C&D panels for AMPS Flights 1 and 2 could be accommodated in one double and one single rack on the starboard side of the pressurized module.

Full scale drawings of the Orbiter's control and display panels were available during this evaluation to aid in visualization of the total flight crew activity. Specific mission control and display functions which should be performed at the orbiter aft flight deck were identified. Some of these functions, such as the SIPS and OBIPS C&D, were placed there for operational reasons, others, such as the instrument safing panels, were dictated by safety considerations. The basic safety approach was to include interlocks on all mission functions when the inadvertent operations would jeopardize the flight or damage AMPS instruments. The capability to configure the payload safe return from the aft flight deck after the emergency evacuation of the Spacelab was also included.

An operating ground rule was established early in our study which stipulated that any Orbiter AFD interfaces with the experiment C&D were inviolate. Therefore, emphasis was placed upon assuring that such hardware interfaces were totally compatible and that no experiment software requirements were imposed upon the Orbiter GPU. Our analyses have shown that the AMPS experiment operations requirements at the AFD can be accommodated with the Spacelab DDU and keyboard at panel L-11 and a minimum of mission unique or backup (safety) C&D. The latter are readily implemented into the CORE-AFD equipment concept.



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Figure 4-32. A Soft Mockup of the Core Segment

Evaluations of C&D selection and layout were performed using scientists specializing in the scientific fields associated with the various experiments and instruments, because their skills, knowledge, and requirements should closely approximate those of the AMPS payload specialists.

The control and display requirements defined as manual were, where practical, assigned to the experiment computer and DDU/keyboard. Exceptions to the assignment of functions to the CDMS are areas where a spurious signal would adversely affect safety, where current status of experiment instrument status must be continuously displayed, and where the operator's observations must be fed back into the system in real time. A representative example of these types of instrument/FSE dedicated panels, based on those manual functions which evolved during this study, is shown in Figure 4-33. The control and/or display requirements of proposed future AMPS missions were considered in the panel design and space allocated in the proper areas to enable inclusion of these expanded requirements. Workstation layouts for both the pressurized module and AFD experiment C&D equipment are provided in Section 3 of this report.

In addition to the development of preliminary designs and layouts of the experiment/FSE dedicated panels, those manual functions which could readily be incorporated and accommodated by the Spacelab mission-dependent equipment were assigned to the experiment CDMS DDU/keyboard. The C&D designers and scientists worked with the software developers to achieve functional descriptions of the operation of the instrument which satisfied the scientific requirements and would also enable the flight crew to control and monitor the instruments through the CDMS.

4.4.2.2.2 Procedures, Commands and Status Data Display and Control Concept Developments. Analyses was also performed of the amount and types of procedural, command, command status and reference data required by the payload specialist (PS) to perform, or to direct his performance, of experiment operations in a timely and effective manner. Figure 4-34 identifies the types of data and the recommended display or control device required to provide the PS with the data he needs at the time and in the form he needs it.

Typical experiment operations procedural/command sequence logic and command status display approaches are shown in Figures 4-35 and 4-36, respectively. Figure 4-35 illustrates a functional description of the

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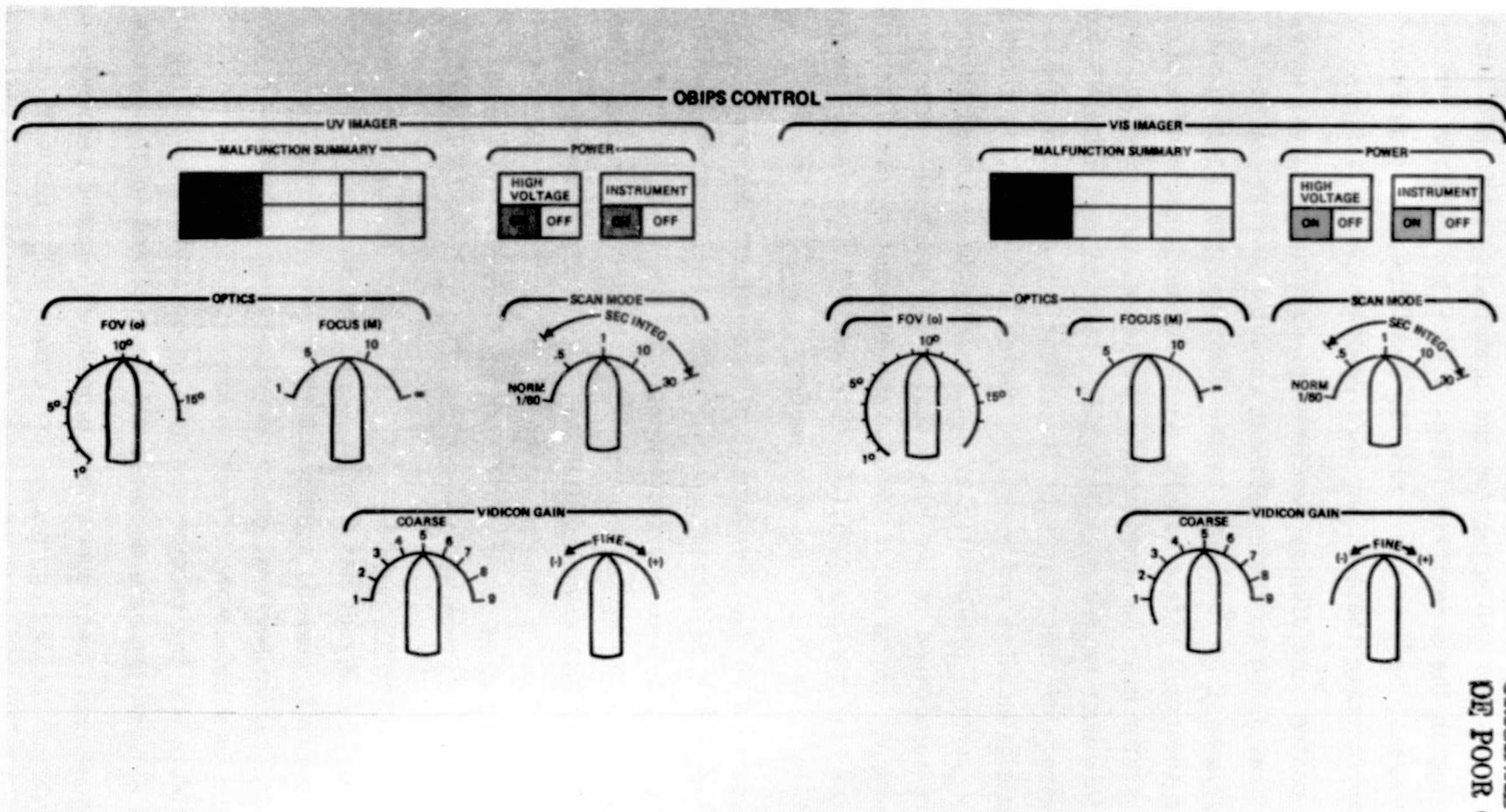


Figure 4-33. OBIPS Control Panel

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- 1) CDMS/CRT (AN, VECTORS, GRAPHICS)
 COMMAND SCENARIO AND STATUS, EXPERIMENT
 STATUS, SCIENCE DATA, LEVEL 3 AUTOMATIC ALARM
 LINE, KEYBOARD REQUEST DISPLAY
- 2) CDMS/KEYBOARD
 EXPERIMENT PROGRAM CALL-UP, MODE SELECTION,
 COMMAND SEQUENCE INITIATION AND CONTROL,
 PROGRAM/PARAMETER CHANGE, DISPLAY CALL-UP,
 BULK DATA DISPLAY SELECTION
- 3) TV MONITOR - DATA RETRIEVAL UNIT
 SCHEMATICS AND DRAWINGS, EQUIPMENT OPERATIONS,
 STANDARD (UNCHANGING) PROCEDURES, CCTV
- 4) FLIGHT DATA FILE
 PROCEDURES, PARAMETERS

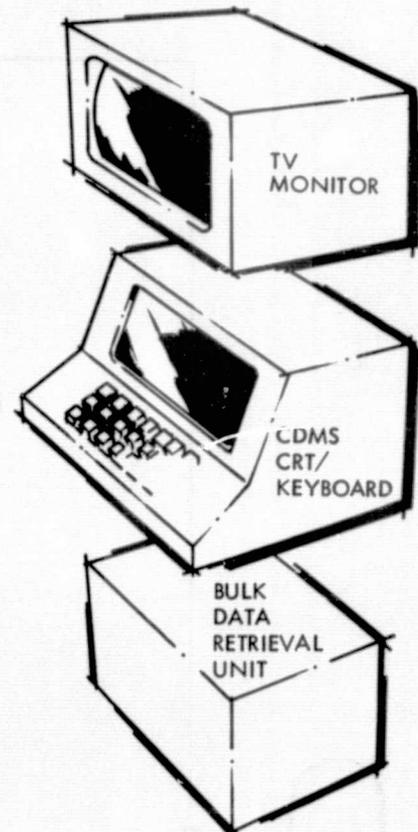


Figure 4-34. Recommended Display and Control Devices

operation of the lidar trace of acoustical gravity waves in the sodium layer. Figure 4-36 is a typical DDU/CRT display for the setup of the lidar receiver in the logic structure. Operator inputs into the sequence are performed via keyboard entry or dedicated control panel.

The bulk data retrieval unit of the ultra-fiche type, Norden-United Aircraft, would display the types of data indicated on the CCTV. It is suggested that this unit be designated MMSE and tied into the CDMS-DDU.

The flight data file (FDF) is standard equipment. Considerable discussion has centered around the need for a hard copy device which could be used to update the FDF. Analyses have indicated that such a capability would be desirable, but not mandatory for AMPS operations. It is recommended that such a device could serve both Orbiter and Spacelab operations and should be developed as standard Orbiter AVE.

THIS FLOWCHART SHOWS THE BREAK-DOWN OF THE LIDAR TRACE OF ACOUSTICAL GRAVITY WAVES IN SODIUM LAYER EXPERIMENT INTO ITS MAJOR FUNCTIONAL COMPONENTS.

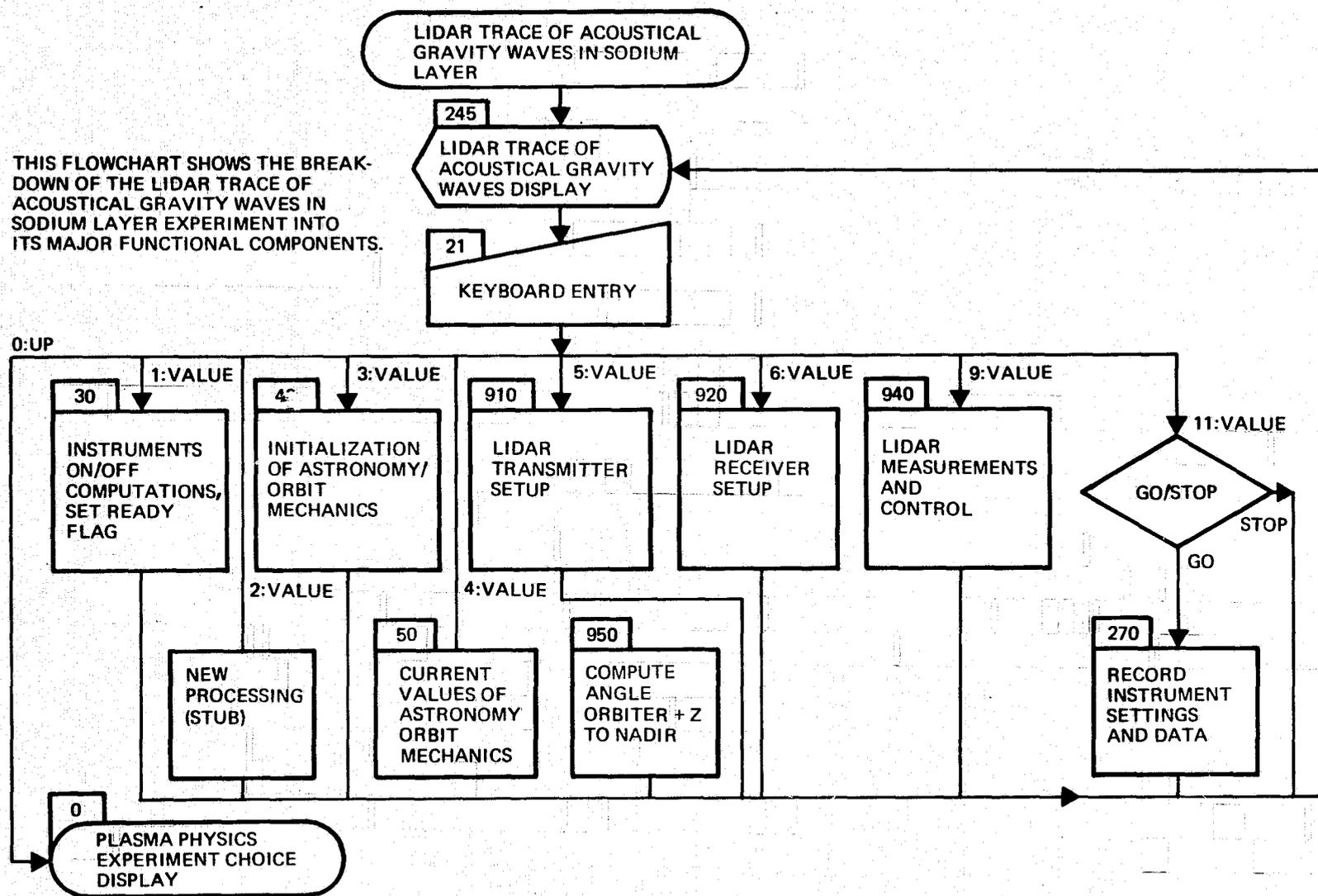


Figure 4-35. Lidar Command Sequence Logic

GMT DD HH MM SS

MET DD HH MM SS

TITLE: RECEIVER SET-UP/LIDAR NA WAVE EXPT

1. LIDAR RECEIVER	<u>ON</u>	ON/OFF	<input checked="" type="checkbox"/> READY	READY
2. SET APERTURE (0 TO 550)	<u>247</u>	XXX	MILLIMETERS	
3. SET FIELD STOP	<u>41.2</u>	XX.X	MILLIRADIANS	
4. NA D-LINE FILTER	<u>OUT</u>	IN/OUT OF BEAM		
5. FABRY-PEROT MOTOR	<u>OFF</u>	ON/OFF		
6. FABRY-PEROT BANDPASS	<u>0</u>	X, XXX	NANOMETERS	
7. DUST COVER	<u>ON</u>	ON/OFF		
8. RECEIVER WAVELENGTH	<u>0</u>	XXX.XX	NANOMETERS	
9. DETECTOR HI-VOLTAGE	<u>0</u>	XXXX	VOLTS	

RECEIVER READY FOR MEASUREMENTS

NO

INPUT/STATUS 4:1N/WAITING FOR INPUT

E ERROR MSG

Figure 4-36. Lidar Command Status Display

4.4.3 Automated versus Manual Instrument Operation

Specific evaluations were made relative to defining those instrument/experiments which require minimum human operator intervention and those where manned operation is essential to experiment performance or to maximize scientific data acquisition. In general, the operation of the atmospheric survey instruments (e.g., spectrometers, interferometers, limb scanner, solar flux monitor) and ESP instruments require little human interaction. Table 4-12 provides a summary of the manual versus automatic aspects of the AMPS Flights 1 and 2 equipment.

Table 4-10. Automated versus Manual Operating Requirements Summary

Instrument/FSE	Function			Notes
	Calibrate	Operate	Control	
Chemical gas release	N/A	A (timing)	M (enable)	
Electrical accelerator (NIM-CAMAC)	A/M	A (timing)	M (adj-enable)	Protective cover
- Level I diag.	N/A	A	A	Integral with electronic acceleration
- Level II diag.	A	A	A	Integral with electronic acceleration
Vector magnetometer*	A	A	A	Boom mounted
Solar flux monitor*	A	A	A	SIPS mounted
OBIPS	A	A	M/A	SIPS mounted
LIDAR*	M/A	A	A/M	Protective cover
Cryo limb scan*	A	A	A	SIPS mounted
Cryo IR interferometer*	A	A	A	SIPS mounted
NIR spectrometer*	A	A	A	SIPS mounted
UV-VIS-NIR spectrometer*	A	A	A	SIPS mounted
ESP's*	A	A	A	RMS/free flyer
Test body	N/A	N/A	M (infl-rel)	Boom mounted
RF sounder	A	A	M/A	Antenna/boom mounted
SIPS	N/A	A/M	A/M	Pointing platform
RMS		M/A	M/A	ESP positioning
Booms	N/A	A	M (enable)	

*Minimum local operator requirements.

4.5 THERMAL CONTROL

This section summarizes the results of thermal analyses performed on the AMPS payload bay and both attached and detached payloads.

4.5.1 Introduction and Summary

The thermal control subsystem encompasses those elements which control and maintain the temperature of the AMPS scientific payload and support equipment. An overall view of the scientific payload with the thermal control system in place is shown in Figure 4-37. Trade studies were conducted early in the phase B study on the thermal insulation technique most suitable for AMPS. As a result of these studies, it was decided to utilize multi-layer insulation blankets with silvered Teflon exterior surfaces to surround the instruments to minimize temperature transients and heat losses. Electric heaters are utilized for standby temperature control and reduction of instrument gradients. Maximum use is made of Spacelab-provided multilayer insulation blankets for the pallet areas, avionics air cooling for module mounted equipment, and the active pallet freon cooling loop. Additionally, the SIPS standard heat pipe canister is used for pointed instrument thermal control. With the thermal control system presented in this section, all instrument temperature requirements are met.

One of the major elements of the thermal control subsystem is the special canister required to support the cryogenically cooled limb scanner. This instrument presents a major design challenge in the areas of cryogenic fluid selection, instrument design, and cryogenic fluid handling techniques to meet the safety requirements of the Space Transportation System. Extensive trades were accomplished to define an optimum system. Baseline to our system is the selection of solidified hydrogen for detector cooling and gaseous nitrogen/solidified nitrogen for optics cooling. The hydrogen and nitrogen are kept solidified prior to launch by a service dewar of liquid helium mounted on the pallet. Provisions also exist for a safe return of any unused hydrogen in an abort from orbit mode.

4.5.2 Thermal Design Process

The basic thermal design approach taken in the study was to develop and utilize a nodal network computer model of the payload bay and the attached scientific instruments and to accomplish the thermal design of the detached

payloads (such as the ESP's and magnetometer package) through a combination of computer and hand analyses as described in the following paragraphs.

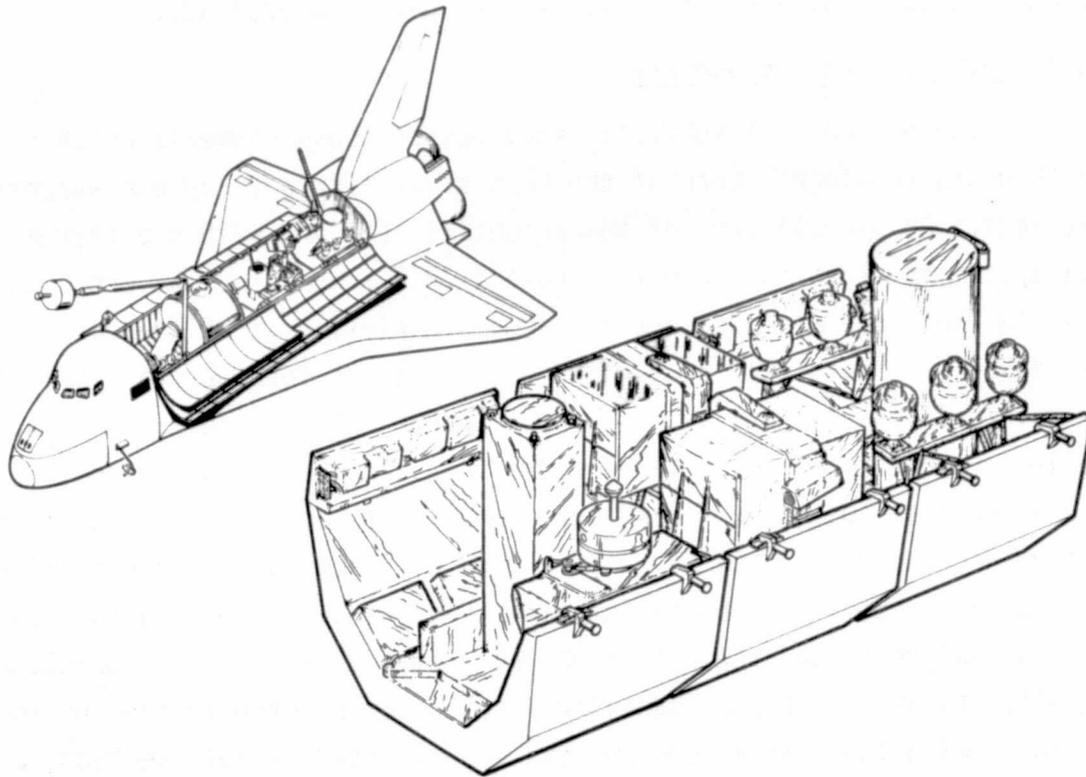


Figure 4-37. AMPS Thermal Control System Flight 1

4.5.2.1 Computer Thermal Model

The basic computer nodal model consisted of approximately 50 nodes. These nodal descriptions were input to LOHARP. A check on this input is obtained from an ancillary graphics program, PICTURE, which produces a line drawing of the model. LOHARP can simulate whole orbits and determine earth and sun heat input rates including the effects of multiple reflections and surface shadowing.

In order to accomplish this analysis, worst case hot and cold orbits was defined. These cases are shown in Figure 4-38. Data from LOHARP program was then utilized in the TAP program. TAP is three dimensional heat transfer program that utilizes explicit techniques to calculate steady state and transient temperatures.

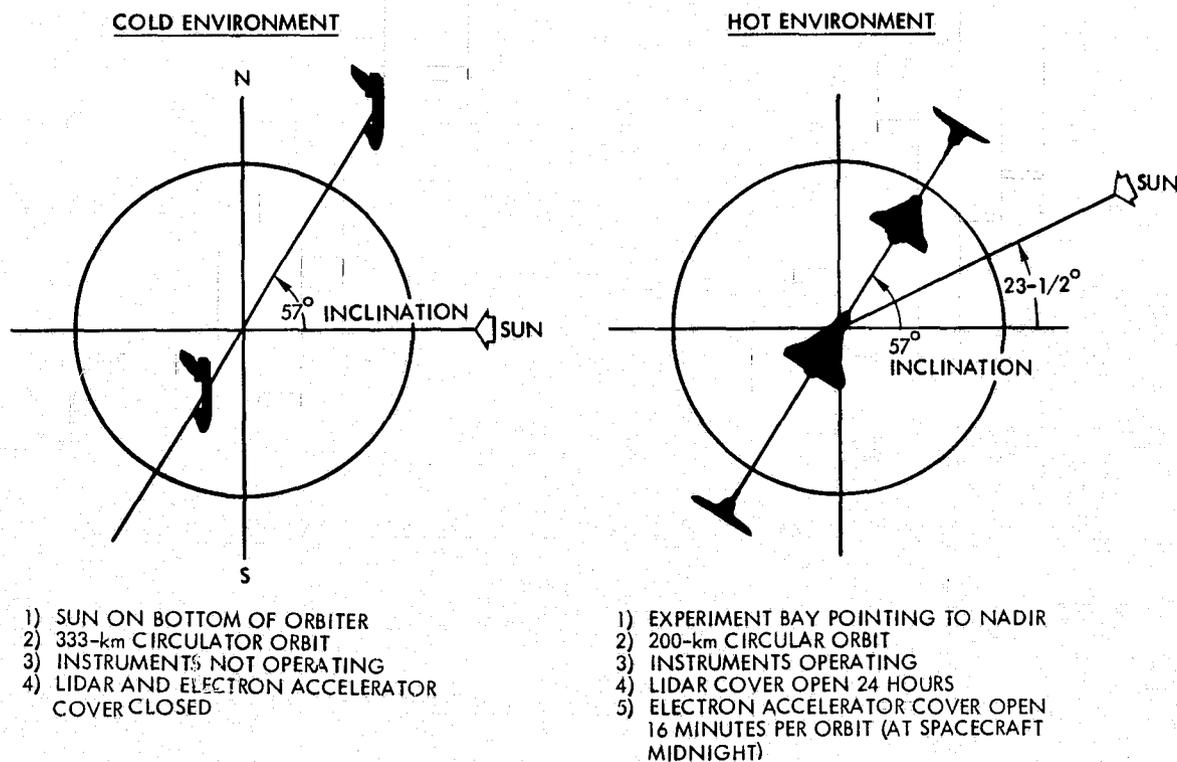


Figure 4-38. Thermal Control System Design Conditions

4.5.2.2 Thermal Analysis Results

The thermal analysis results are shown in Table 4-11. In every case it can be seen that the temperature requirements have been met. As pointed out earlier, heaters are utilized in some cases to maintain the lower temperature limit and the power for this technique is summarized in Table 4-12. The power required is well within the capability of the Spacelab supplied power and has no impact on the AMPS mission.

The configuration of the Spacelab provided active cooling loop and the instruments serviced by it are shown in Figure 4-39. The heat dissipated into the active Spacelab cooling loops was determined from the thermal analysis and is shown in Table 4-13. As expected, this value is well below the electrical power consumption due to radiation cooling from all the detached payloads, SIPS, etc.

Table 4-11. AMPS Instrument Temperature Requirements and Thermal Analysis Results

INSTRUMENT	TEMPERATURE LIMITS			PREDICTED TEMPERATURES	
	DEG C		GRADIENTS	DEG C	
	MIN	MAX		MIN	MAX
MODULE-MOUNTED INSTRUMENTS					
CONTROLS AND DISPLAYS	0	50		35	39
CRT DISPLAYS	0	50		35	39
NIM	-10	60		35	39
CAMAC	-10	60		35	39
PALLET-MOUNTED INSTRUMENTS					
COMMON ENERGY STORAGE	-20	50		20	22
ELECTRON ACCELERATOR	-30	50	±5°C, ±9°F	20	24
LIDAR					
● LASER AND DYE RESERVOIR	NA	64		20	36
● PHOTOMULTIPLIER TUBE	-23	0		PROVIDED BY SUPPLIER	
● TELESCOPE, OPTICS & ELECTRONICS	-30	50	±5°C, ±9°F	20	28
● PHOTON COUNTER DETECTOR	7	37		20	36
● SOLID STATE DETECTOR	NA	-196		PROVIDED BY SUPPLIER	
SIPS					
OBIPS					
● IMAGER (STORAGE)	-30	50		19	22
● IMAGER (OPERATING)	5	35		19	22
● OTHER OPTICS	-30	50		19	22
SOLAR FLUX MONITOR	-30	50		19	22
CRYO LIMB SCANNER					
● DETECTOR	NA	-265		-	-265
● OPTICS	NA	-183		-	-183
ESP INSTRUMENTS					
MEDIUM ENERGY ELECTRON DETECTOR	0	45		14	38
DC ELECTRIC FIELD VECTOR MAGNETOMETERS	0	45		14	38
E AND B RECEIVERS	0	45		14	38
LANGMUIR PROBE	0	45		14	38
ION MASS AND DISTRIBUTION ANALYZER	0	45		14	38
MAST-MOUNTED INSTRUMENT					
VECTOR MAGNETOMETER	-30	50		-20	20
GAS RELEASE CANISTER	10	38		18	21

Table 4-12. Instrument Standby Power Requirements

Instrument	Maximum Standby Heat Requirement (watts)
Electron Accelerator	62
SIPS System	--
Vector Magnetometer	4
LIDAR	82
Gas Release Canister	20 Each
ESP	23* Each
Common Energy Storage	9

*0.36 M² White Strip

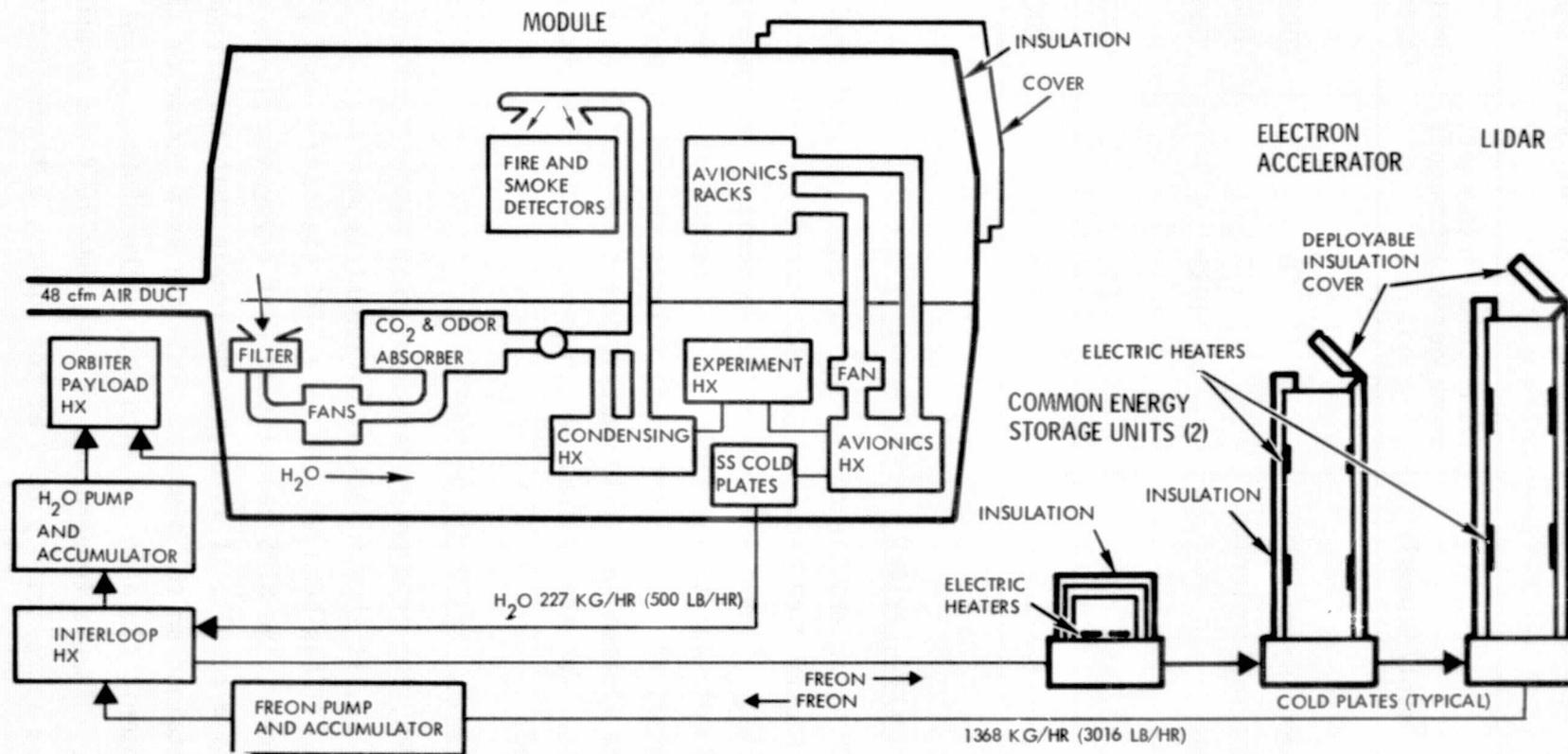


Figure 4-39. Spacelab Active Control System

Table 4-13. Heat Dissipated Into the Active Cooling Loop

Instrument/Assembly	Peak Cold Plate/Heat Exchanger Requirements (watts)	Orbiter Cooling Capacity (watts)
Pallet		
Common Energy Units	98	2000 (2 Plates)
Electron Accelerator	116	1000 (1 Plate)
LIDAR	1991	2000 (2 Plates)
Module		
Avonics HX	920	4510
Maximum, instantaneous	2151	5200

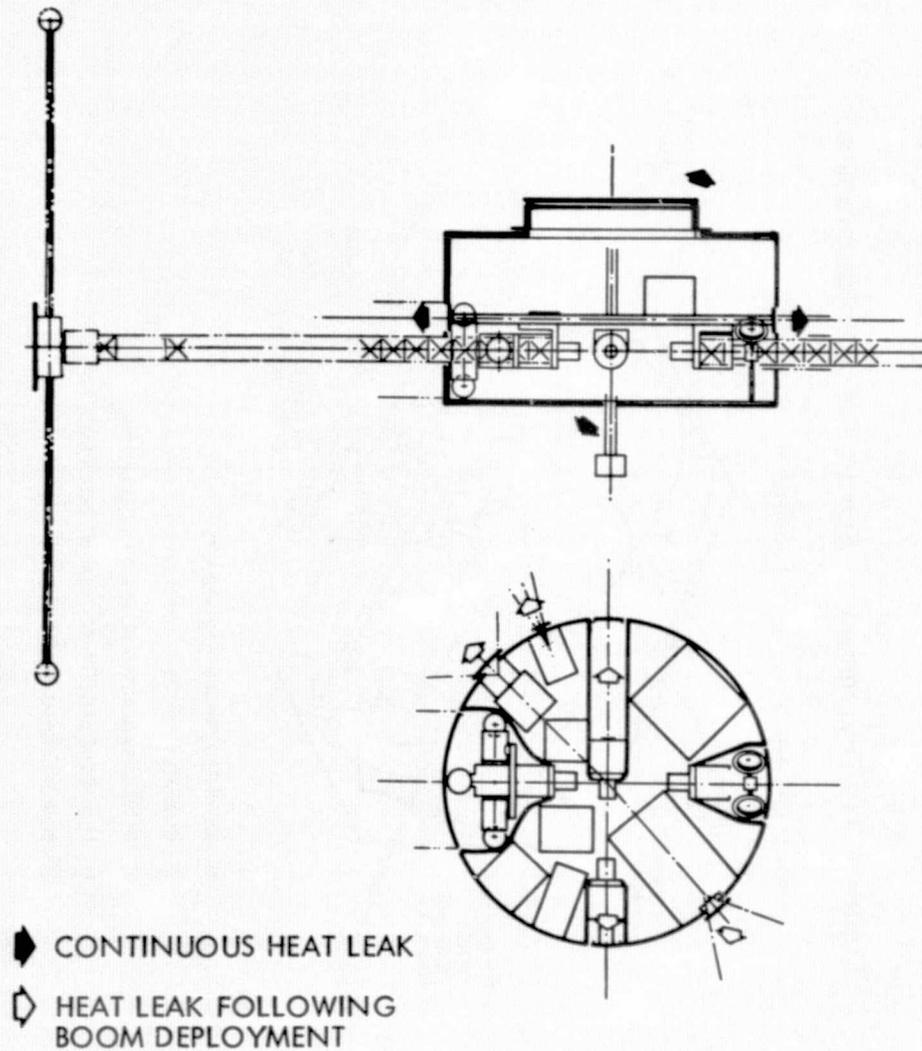
4.5.2.3 Detached Payload Thermal Analysis

The ESP is an example of the thermal analysis performed for detached payloads. Figure 4-40 shows an ESP and identifies some of the unit's thermal properties. Figure 4-41 shows the temperature swings expected for both the hot and cold worst case orbits as a function of the radiant area of the circumferential stripe. Figure 4-41 relates the heater power required to maintain the minimum satellite temperature of 0°C again as a function of radiant area. For the nominal design point, only approximately 18 watts are required.

4.5.3 Cryogenic Instrument Accommodation

For AMPS, the cryogenic instrument must raster scan above the earth's horizon. Also, it should look at different segments of the horizon. For this reason, it has been mounted on a SIPS standard gimbal platform (Figure 4-42). However, the SIPS cannot transport cryogenic fluids across its four gimbals. As a result, it is highly desirable that the cryogenic fluids be totally contained on the gimbals. This places a burden on the efficiency of the cryogenic fluid in order to obtain the minimum weight and volume for the system. Since a temperature of less than 10⁰K was required at the detector and approximately 80⁰K at the optics, a tradeoff study determined

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- 1) WHITE PAINT CIRCUMFERENTIAL RADIATOR
- 2) MULTILAYER INSULATION ENCLOSURE
- 3) MULTILAYER INSULATION INSIDE COMPARTMENT AROUND BOOM SUPPORT
- 4) MULTILAYER INSULATION BLANKETS AT END OF BOOMS
- 5) MULTILAYER INSULATION CAPS OVER INSTRUMENT VIEW PORTS
- 6) FIBERGLASS CYLINDER
- 7) COMPONENTS AND PLATFORM PAINTED BLACK
- 8) THERMAL INTERFACE MATERIAL BETWEEN PLATFORM AND HIGH POWER DISSIPATING COMPONENTS
- 9) ELECTRIC STRIP HEATERS

Figure 4-40. Environmental Sensor Package Thermal Control Features

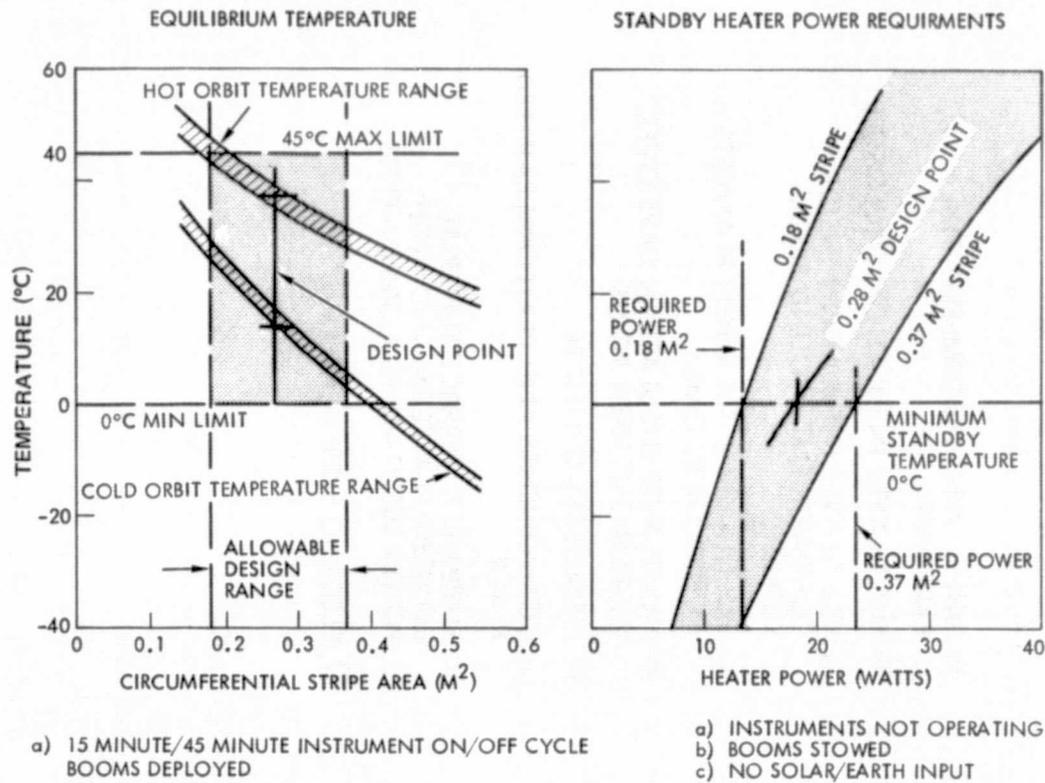
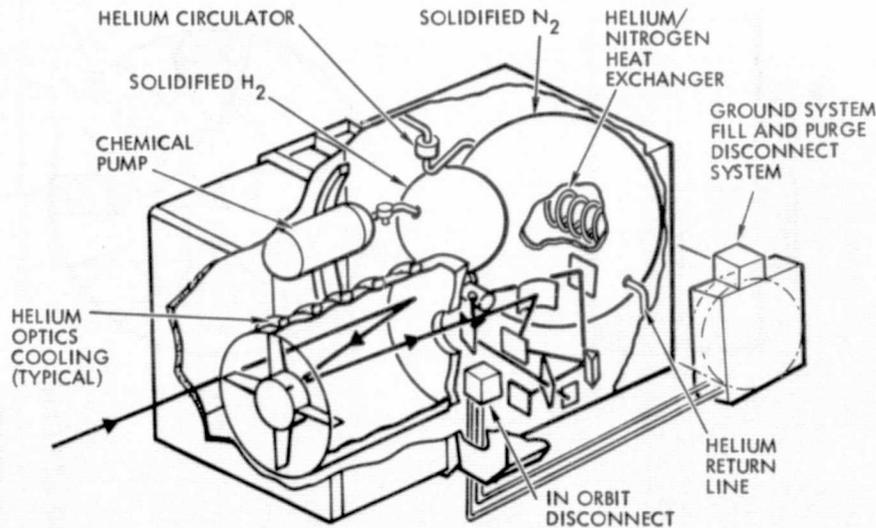


Figure 4-41. Environmental Sensor Package Temperature



- 1) MOUNTED ON A STANDARD SIPS
- 2) COMMON OPTICS COOLED TO 80°K BY GHe/SOLID N₂ SYSTEM
- 3) DETECTORS COOLED TO 8°K BY SOLIDIFIED H₂
- 4) LHe DEWAR UTILIZED TO PREVENT PRELAUNCH GH₂ BOILOFF
- 5) BOILOFF FORMS PURGE GAS

Figure 4-42. Cryogenic Instrument

that an open loop cooler utilizing solid hydrogen for cooling the detector and a system using gaseous helium and solidified nitrogen for cooling the optics was the most suitable. The gaseous helium was circulated around the optics and then re-cooled by a heat exchanger located within the solidified nitrogen bottle. However, since hydrogen in the wrong concentration with oxygen is an inflammable mixture, precautions must be taken to prevent the presence of hydrogen while in the atmosphere. This is accomplished prior to launch by the use of a service dewar of liquid helium mounted on the pallet. The system is sized to give a 24 hour launch pad contingency capability. After SIPS deployment this connection is severed. Under normal circumstances the hydrogen would be fully depleted prior to reentry. To cover an abort from orbit condition, heaters are provided to heat the solidified hydrogen container to rapidly boil off the cryogen. Additionally, prior to reentry, the system is sealed off and a vent is opened to a chemical pump. This system is often referred to as a getter system and it is merely an evacuated bottle containing calcium powder pressed on stainless steel sieves. This system is sized for a 3-hour standby period under a worst case condition before purge gas is supplied.

4.6 ELECTRICAL SUBSYSTEM

Power for the electrical power and distribution subsystem is provided from a dedicated power source in the Orbiter through the Spacelab power distribution system. However, electrical power and energy requirements for instruments and support equipment must be established, power and signals on the pallets must be distributed, and interface requirements with the Spacelab or Orbiter must be satisfied. Electrical power and distribution subsystem functions include:

- Establish power and energy requirements
- Distribute primary power
- Distribute secondary power
- Distribute command and signal
- Distribute caution and warning
- Perform fault isolation

No changes are required to the existing Spacelab/Orbiter system. A review of the electrical power and energy requirements for Flights 1 and 2 revealed that power and energy values are within the capabilities provided. A comparison of the capabilities and requirements is shown in Table 4-14.

Table 4-14. Power Requirements for AMPS Flights 1 and 2 Are Within the Spacelab Capability*

	Capability	Requirement	
		Flight No. 1	Flight No. 2
Sustaining (kW)	3.4	3.0	3.2
Peak (kW)	7.4 15 min max/3 hr	3.2	3.5 10 min/10-15 hr
Energy (kW-hr)	369	194	261

* ESA SLP-2104 Spacelab Payload Accommodation Handbook, May 1976.

The sustaining power available for the short module and three pallet segment Spacelab configuration as reported in the Spacelab System PDR-A is 3.4 kW with a peak of 7.4 kW. The total energy available is 369 kW-hours. The sustaining power requirement for Flights 1 and 2 in Table 4-14 are maximum sustaining power but are not required over the total mission. A large margin of energy is available for both flights.

4.6.1 Requirements

Each of the experiment, instrument, and subsystem support requirements for Flights 1 and 2 were reviewed in detail. The power and energy requirements for both flights were determined for each instrument, each combination of instruments for each experiment and the subsystem support equipment for each experiment for the periods in the mission timeline. These requirements and the one piece of flight support equipment that must be added, the common energy storage system, are discussed in the following paragraphs.

4.6.1.1 Flight 1

The power and energy requirements for Flight 1 experiments and support equipment are shown in Table 4-15. Because the Small Instrument Pointing System (SIPS) is used with the environmental measurements and solar flux experiments, it is used almost continuously. The NIM/CAMAC supports the electron accelerator for the vehicle neutralization beam and beam plasma interactions experiments. The control and display requirements are also shown. As a result of a thermal analysis (as reported in Section 4.5), the thermal requirements were also included. A value of 20 percent of all other loads was added to account for wire losses, growth, and contingencies. The power and energy data presented in Table 4-15 were used to construct the payload power and energy profile in Figure 4-43. The solid line is the power profile and the dashed line is the energy profile. The average power requirement is approximately 800 watts, except for the 10-minute peaks during the first 120 hours of the mission and the longer duration peaks after 120 hours. The peaks shown in Figure 4-43 occur during the vehicle neutralization beam, beam plasma interactions, and minor constituents experiments. The first two experiments require the electron accelerator and the last experiment the lidar. The electron accelerator is operated for five periods. In each period, the accelerator is operated eight

times at about one-orbit intervals. Each of the narrow peaks represent one of the eight operations of about 10 minutes duration. During the 10 minutes, the accelerator is fired 10 times at 1-minute intervals. Each firing of the accelerator requires a 1-minute period to charge the capacitors of the common energy storage system. The first charge requires more power than subsequent charges resulting in an initial peak that is higher. The wider peaks that occur near the end of the mission represent the operation of the lidar.

Table 4-15. Electrical Power and Energy Summary (Flight 1)

Experiment/Experiment Support	Power (watts)		Energy (kilowatt-hours)
	Operating	Average	
Acoustic Gravity Wave	172	1.9	0.3
Gas Cloud Expansion Dynamics	172	1.9	0.3
Vehicle Neutralization Beam	1128	45.5	7.1
Beam Plasma Interactions	1128	18.0	2.8
Minor Constituents	2185	376.3	58.7
Environment Measurements/Solar Flux	26	9.0	1.4
SIPS	208	213.5	33.3
NIM/CAMAC	761	48.7	7.6
RMS, Control and Display	160	135.9	21.2
Thermal	-	183.3	28.6
Wire Loss, Growth and Contingency	-	207.0	32.3
Totals	-	1241.0	193.6

4.6.1.2 Flight 2

The power and energy requirements for Flight 2 for each experiment and for subsystem support equipment are shown in Table 4-16. The resultant payload power and energy profiles are shown in Figure 4-44. In this figure, the solid line is the power profile and the dashed line is the energy profile. The power requirements for Flight 2 are less than 800 watts for most of the mission. The higher power requirements beginning at about 85 hours are a result of the atmospheric instrument experiments. The

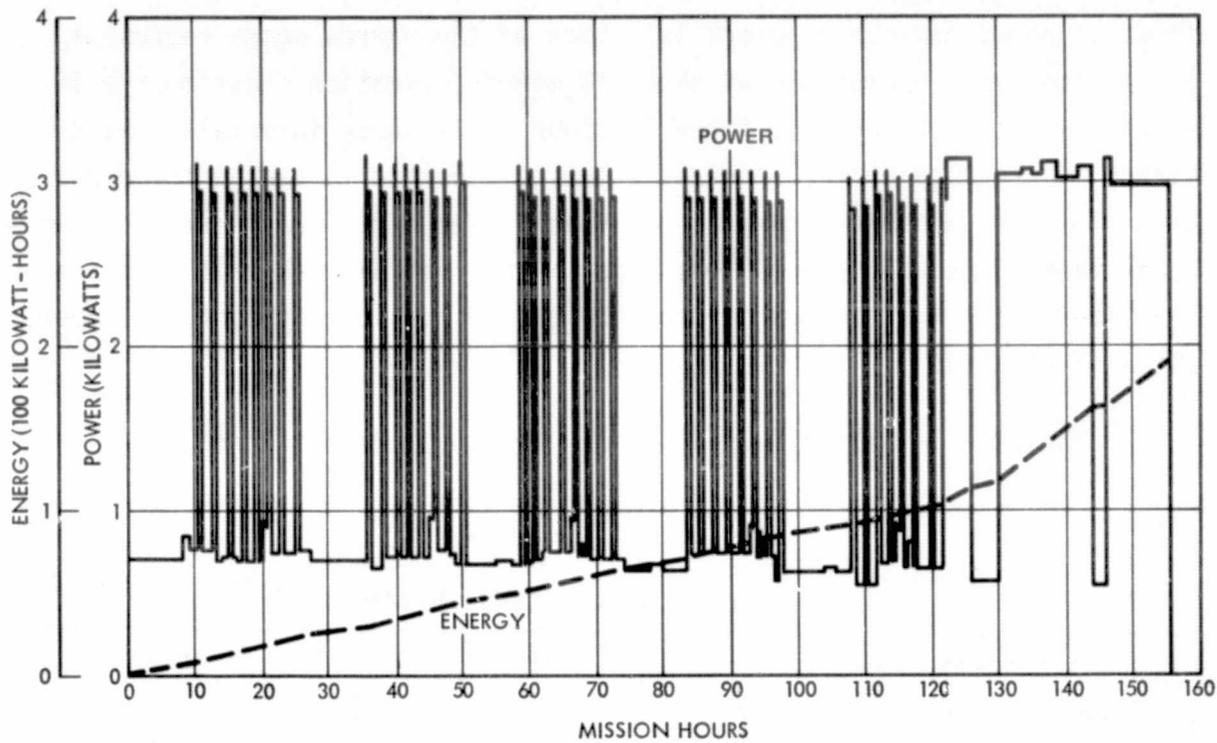


Figure 4-43. Payload Power and Energy Profile (Flight 1)

Table 4-16. Electrical Power and Energy Summary (Flight 2)

Experiment/Experiment Support	Power (watts)		Energy (kilowatt-hours)
	Operating	Average	
Conductivity Modification	133	7.8	1.2
Wave Particle Interaction	277	46.8	7.2
Long Delay Echoes	222	17.3	2.7
Plasma Flow	67	12.2	1.9
Atmospheric Measurements	2441	991.6	152.7
SIPS	208	91.8	14.1
Control, Display, CRT	220	176.6	27.2
Thermal	-	68.2	10.5
Wire Loss, Growth, Contingency	-	282.5	43.5
Totals	-	1694.8	261.0

mission timeline has identified the time period from about 85 hours to near end of mission as "targets of opportunity" for these experiments. All instruments required to perform any of the atmosphere's experiments must be operating during this time period.

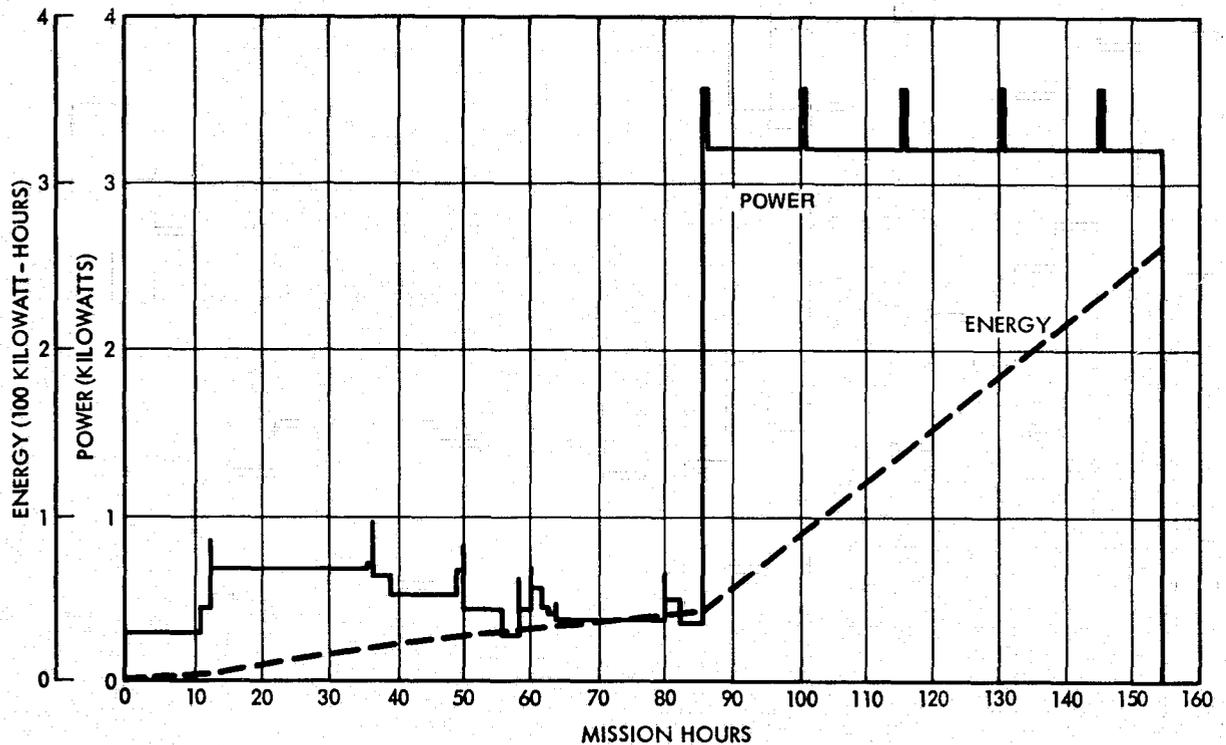


Figure 4-44. Payload Power and Energy Profile (Flight 2)

4.6.1.3 Common Energy Storage System

A common energy storage system must be added to the Spacelab electrical subsystem. This system stores and delivers high energy and high voltage pulses to the electron accelerator. An investigation of devices to satisfy the high energy and voltage requirement was conducted. These devices included flywheels, pumped pressure systems, batteries, capacitors, and inductors. Only capacitors and inductors could meet the high-power and high-voltage output requirements. Capacitors were selected because of low cost and available hardware. The common energy storage system is modular and can be increased in size to provide the increased energy requirements for later flights. The common energy storage system, as shown in Figure 4-45, consists of a capacitor bank and a low-power processor. The low-power processor accepts a 28-volt input and delivers an output to the

capacitors of 450 volts maximum. A capacitor bank is made up of 124 standard electrolyte capacitors of 1600 μf each. The output of a single device is 20k joules. Fuse and diode protection is provided for each capacitor as shown in the figure. In addition, each capacitor bank is pressurized to one atmosphere with gaseous nitrogen; temperature and pressure sensors are provided. Although not shown, a resistor load dumping capability is also provided.

- 1) CONSISTS OF A LOW POWER PROCESSOR AND UP TO SIX CAPACITOR BANK UNITS
- 2) LOW POWER PROCESSOR
 - a) 28V \pm 4V 500V MAX INPUT OUTPUT
 - b) CURRENT SETABLE LIMITING
- 3) CAPACITOR BANK 20K JOULES EACH (TWO BANKS UTILIZED ON FLIGHT 1)
 - a) 124 MALLORY LES 21823-A ELECTROLYTIC CAPACITORS
 - b) 1600 μf EACH
 - c) PRESSURIZED TO 1 ATMOSPHERE GN_2
 - d) TEMPERATURE AND PRESSURE SENSORS
 - e) FUSE AND DIODE FAULT PROTECTION
 - f) OPERATING TEMPERATURE RANGE 0° TO 65°C
 - g) LOAD DUMPING CAPABILITY PROVIDED

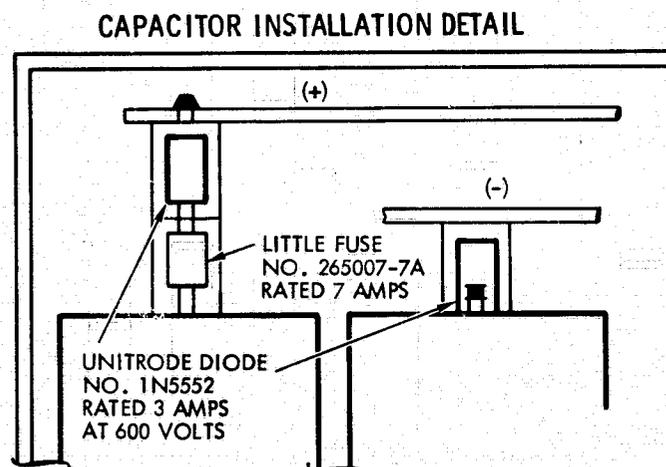
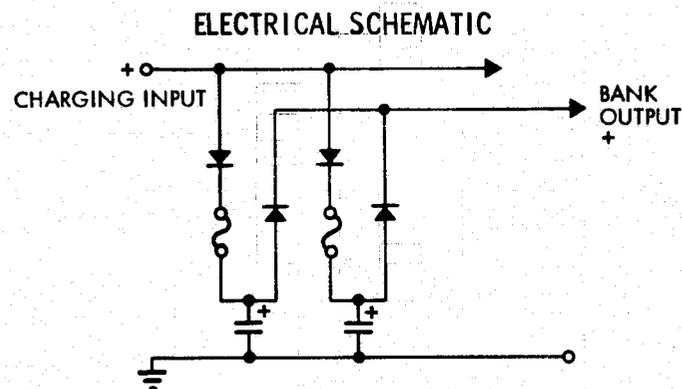


Figure 4-45. Common Energy Storage System Design

4.6.2 Electrical Power Distribution

Electrical layouts were developed for all of the instruments and supporting equipment on Flight 1, including cabling for the modules, each of the three pallets, and the SIPS. An example of the detail that was developed is shown in Figure 4-46. The provided items include the remote acquisition units (RAU's), the electrical power distribution box (EPDB) and the power bus, interconnect station, and essential power bus cabling. The

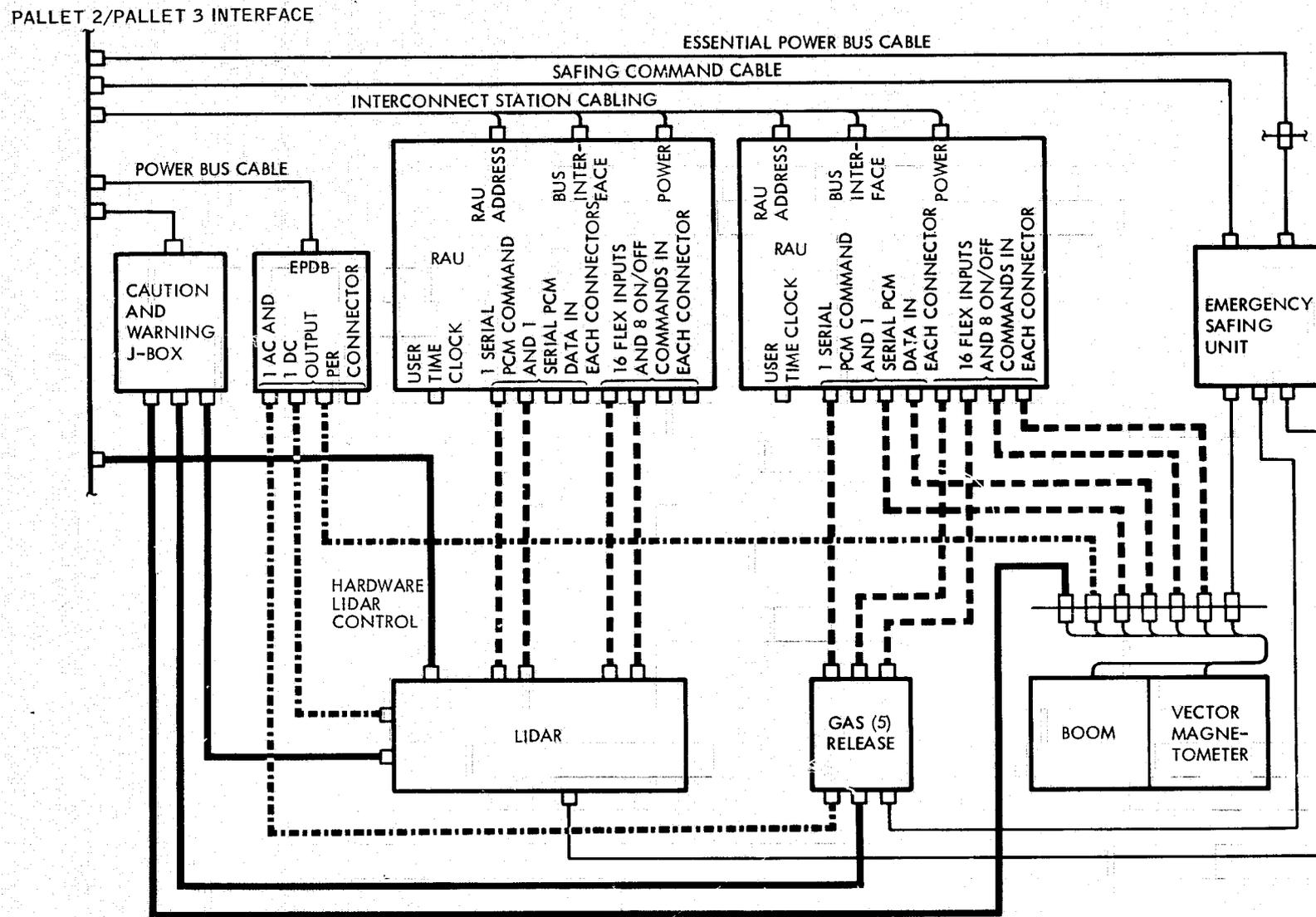


Figure 4-46. Flight 1, Pallet 3 Electrical Cable Schematic Showing the Implementation of Standardized Cables

AMPS payload provides these items with a caution and warning J-box; an emergency safing unit; the instruments shown; and the caution and warning, power, signal and essential/emergency cabling. In the event of power failure, the emergency safing unit, which is modular, provides safing functions. Figure 4-46 assumes the implementation of standardized cables.

The use of standardized cabling has been studied and is recommended for AMPS and all Labcraft payloads. However, as presently configured, the RAU connectors make it difficult to distribute services to more than one user. This is shown in Figure 4-47 by the presently configured small RAU, which is made up of four slices with two connectors each. With this configuration, a slice can only support one instrument. If one SER PCM CMD and one SER PCM data are available at each of the first four connectors and 16 flex inputs and eight on/off commands are available at each of the last four connectors, each connector can support one instrument, which can be supported by each slice. The capability can be increased by one of the two alternatives that are listed in Figure 4-47. The use of an interface bracket appears more likely than a design change by ESA. The interface bracket alternative is shown at the center of Figure 4-47. The outputs from the small RAU are mixed as shown to achieve the increased capability. Once the interface bracket is made and installed, it can be used on all Labcraft flights.

The cables required for Flight 1 were counted as summarized in Table 4-17. In accordance with standard design practices, the use of various connector types and sizes, keying and color coding is recommended to eliminate the possibility of interchanging cables.

OBJECTIVE: TO ESTABLISH AN INSTRUMENT TO RAU INTERFACE THAT ALLOWS THE USE OF STANDARD CABLES (WHICH VARY ONLY IN LENGTH) FOR ALL INSTRUMENTS

PROBLEM:

AS PRESENTLY CONFIGURED, THE RAU CONNECTORS DO NOT ALLOW AN EASY DISTRIBUTION OF SERVICES TO MORE THAN ONE USER.

ALTERNATES

1. PERSUADE ESA TO DISTRIBUTE THE RAU INPUT AND OUTPUT FUNCTIONS AMONG THE CONNECTORS
2. ACCOMPLISH FOR LABCRAFT THROUGH USE OF AN INTERFACE BRACKET

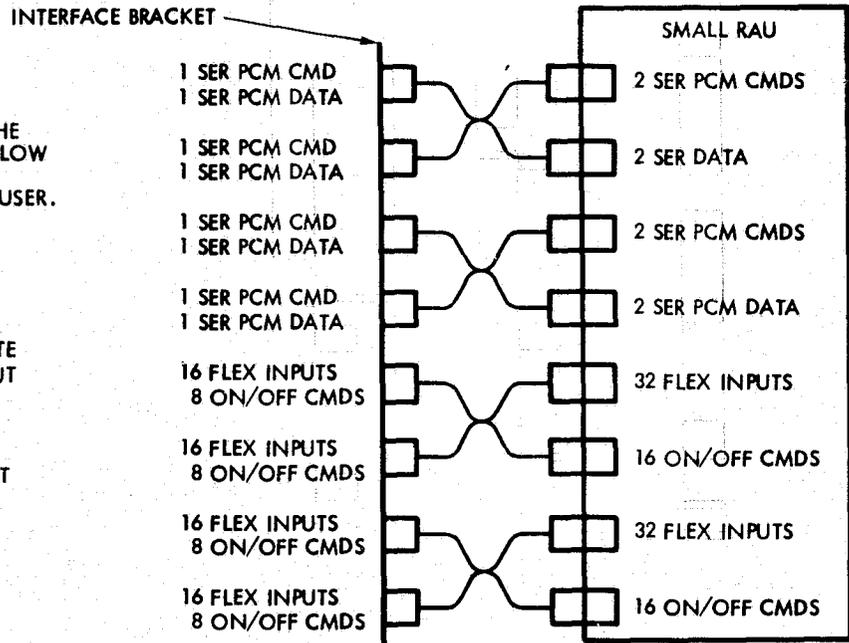


Figure 4-47. Standard Cable Implementation

Table 4-17. AMPS Flight 1 Cable Count

	Cables
AFD	3
Module	44
Pallet 1	29
Pallet 2	30
SIPS Canisters	28
Pallet 3	25
Total Cables Required	159

5. MISSION ANALYSIS

This section discusses the effects of experiments on the mission profiles for AMPS Flights 1 and 2 within the Orbiter capability.

5.1 AMPS FLIGHT ONE MISSION ANALYSIS

The three AMPS experiments that most strongly affect the mission profile for Flight 1 are:

- The acoustic gravity wave and gas cloud expansion dynamics experiments using the Xenon gas release system which require repeated night overflight of the Thomson Scatter Facility at Arecibo, Puerto Rico at altitudes from 150 to 300 km
- The determination of atmospheric constituents by measuring IR and UV absorption using a lidar system which requires altitudes of from 150 to 300 km
- The detection of atmospheric minor constituents using an IR interferometer and IR radiometer.

Other desirable, although not mandatory, orbit characteristics include orbit inclination as high as possible and as much northern hemisphere dark time as possible. Finally, the mission profile selected must be within the basic Orbiter OMS ΔV capability (no OMS kits). Phasing orbit inclination and altitude are chosen to produce an initial Arecibo overflight after allowing adequate time for experiment setup. In addition, the Orbiter must be in the final orbit at least 1.5 hours before the Arecibo overflight to allow for the required Xenon gas canister separation from the Orbiter. Subsequent Arecibo overflights are obtained by adjusting the orbit period to 16 revolutions per sidereal day and compensating for nodal regression by the appropriate period bias. The local time of day at the first Arecibo overflight is determined by the launch time of day from the Eastern Test Range as discussed later.

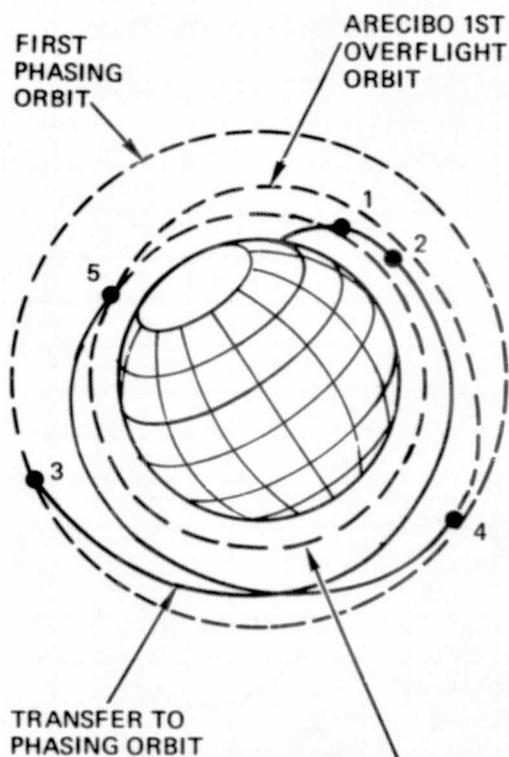
For the two experiments related to measuring atmospheric constituents a low altitude circular orbit is desirable. There are no other particular phasing constraints.

A mission profile has been selected which meets all of the above experimental objectives and is within the basic OMS and RCS impulse capabilities. Although it is by no means unique nor necessarily optimum, it has two interesting features:

- 1) Six Arecibo overflight opportunities are available for Xenon gas release experiments
- 2) The repeating ground track orbit is near circular, is low enough in altitude and provides extensive time to conduct atmospheric constituent measurements over an interval of five days.

The orbit maneuver sequence, shown in Figures 5-1, -2 and -3 begins with a northeasterly Shuttle launch from the Eastern Test Range at about 10 P.M. into a 104 by 302 km orbit inclined at 52.9 degrees. At first apogee, a trim maneuver of +64 m/sec tangentially raises apogee to the first phasing orbit altitude. Approximately 43 minutes later at next apogee, the OMS adds +6.4 m/sec tangentially to establish the near circular initial phasing orbit. The Orbiter remains in the initial phasing orbit for about 9 hr 55 min until on the fifth revolution at a latitude of 18.3 degrees south the OMS adds a 39 m/sec retro ΔV transferring to the second phasing orbit with perigee at 195 km altitude at the latitude of Arecibo. The first Arecibo experiment pass occurs 11 hr 08 min later (22 hr 33 min GET). The Orbiter passes directly over the tracking facility at about 12 P.M. local time in a southwest to northeast direction, remaining above the local horizon for about 6 minutes. At the next perigee approximately 1 hr 33 min later, an OMS - 34 m/sec retro ΔV reduces the orbit period to 88.586 minutes, causing the second Arecibo pass to occur approximately 22 hr 09 min later at about 11:38 P.M. on the second day.

At the first perigee following each Arecibo experiment pass, the orbit period is trimmed by increasing apogee altitude slightly. This accounts for expected atmospheric drag effects during the time interval until the next Arecibo pass, 15 revolutions later. Strictly speaking, the Arecibo second overflight orbit does not have a repeating ground track and must be trimmed slightly to produce the third Arecibo overflight. To obtain a repeating ground track orbit, the period would be trimmed to 16 revolutions/day at the perigee following ΔV_3 and then the phasing orbit altitude and



EVENT	TIME (GET) HR:MIN	ΔV M/SEC	ORBITAL CHARACTERISTICS		INC DEG
			H _p (KM)	H _A (KM)	
1 ET SEPARATION THROUGH INSERTION	0:11	70	104	302	52.9
2 TRANSFER TO PHASING ORBIT ALTITUDE	0:46	64	307	330	52.9
3 CIRCULARIZE AT PHASING ORBIT ALTITUDE	1:29	64	320	340	52.9
4 LOWER PERIGEE FOR 1ST ARECIBO OVERFLIGHT	11:24	-39	195	329	52.9
5 TRIM PERIOD FOR 2ND - 6TH ARECIBO OVERFLIGHTS	22:33	-34	187	222	52.9
6 DEORBIT	166:15	-85	-	-	-

REPEATING GROUND TRACK ORBIT FOR 2ND - 6TH ARECIBO
OVERFLIGHTS AND MINOR CONSTITUENTS EXPERIMENTS

Figure 5-1. An Orbit Sequence has been Chosen Which Meets All Scientific Objectives Including Six Acoustic Gravity Wave Experiment Opportunities

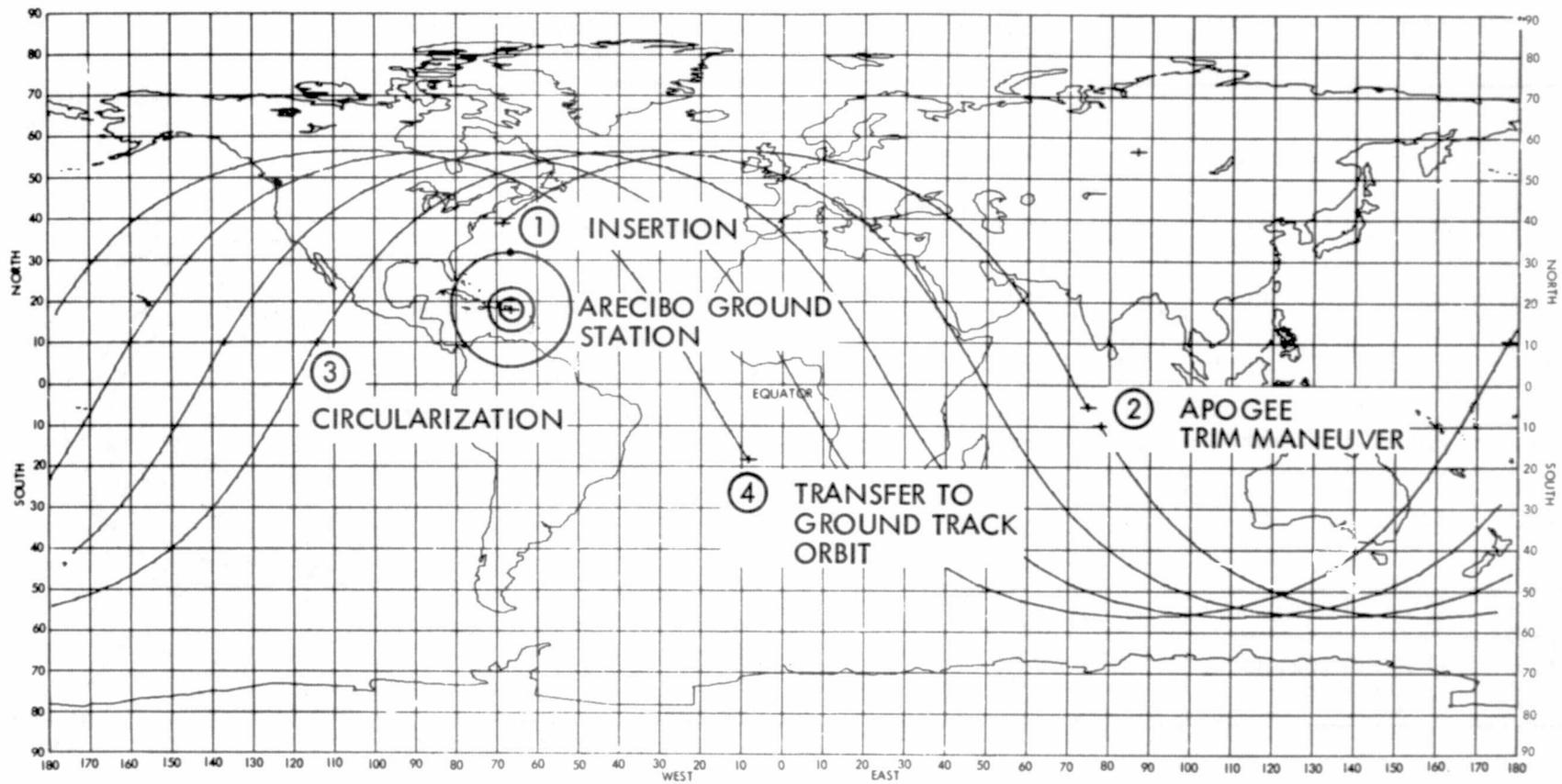
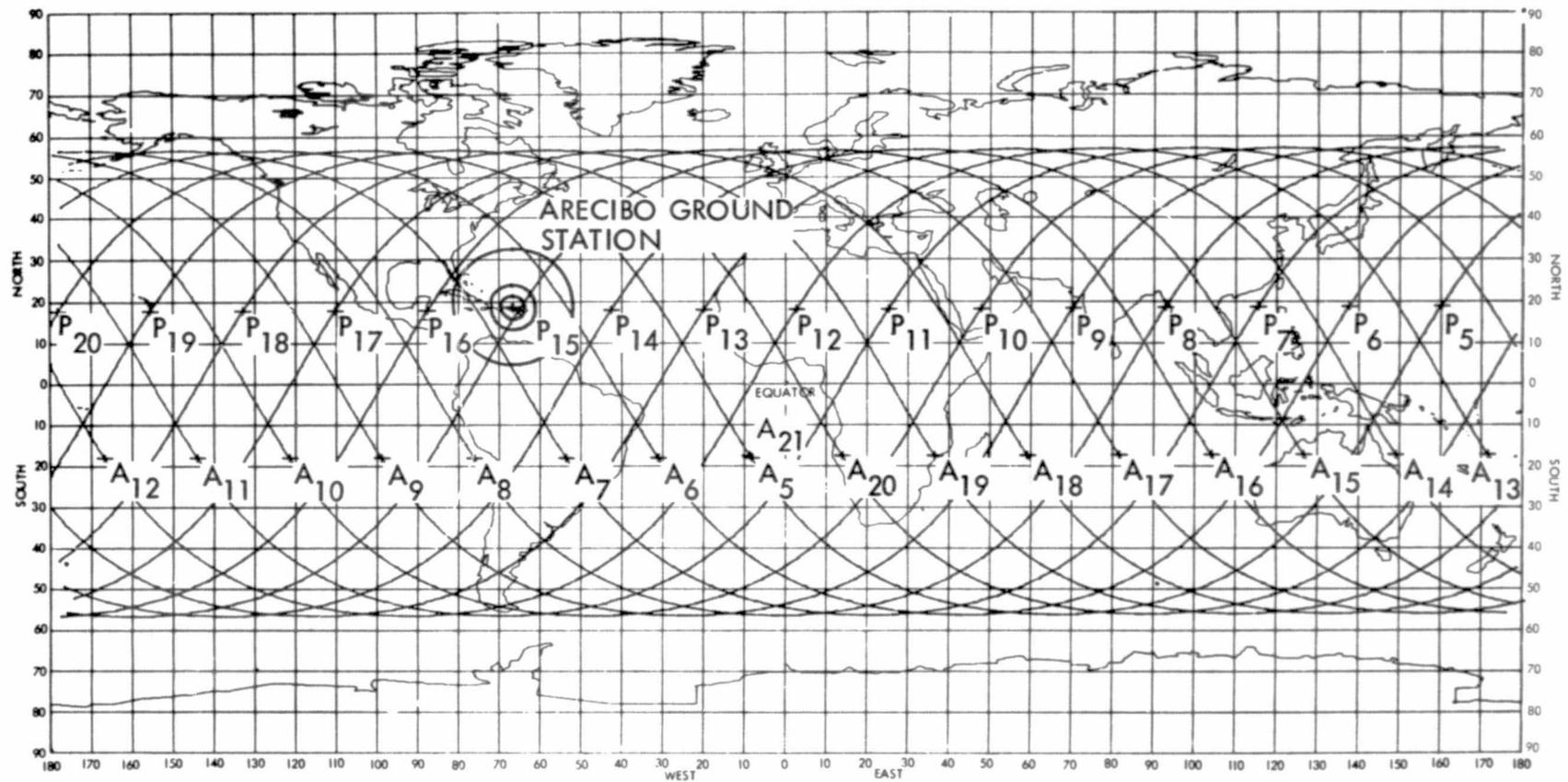


Figure 5-2. Flight 1 Ground Tracks During Initial Maneuvers



ARCIBO PASSES OCCUR ON ORBITS 15, 31, 47, 63, 79, 95

Figure 5-3. Ground Tracks Repeat Over Arcibo Every 24 Hours for 5 Days

inclination would be adjusted to obtain the desired geographic node. Mean solar time at the first Arecibo experimental pass is determined by the ETR launch time of day. As illustrated in Figure 5-4, an ETR launch time of about 10 P.M. results in a midnight Arecibo local time first pass. The combined effects of orbit plane regression and earth motion about the sun will cause the local time of Arecibo flyover to occur approximately 22 minutes earlier each pass.

Atmospheric drag effects on the initial and final phasing orbits and the repeating ground track orbit were determined. The Orbiter flight was simulated in each of the three principal attitudes over a time interval using the NASA LIFTIM program. The results are presented (Table 5-1) in terms of daily changes in period, apogee and perigee radius, and semimajor axis. The gradual decrease in orbit period as drag contracts the orbit semimajor axis causes the ground track to drift eastward at an increasing rate. This must be compensated for by raising apogee altitude at regular intervals. Since the spacecraft remains in the initial phasing orbit for only 9 hr 55 min, the period change is no greater than 0.0049 minute even in the worst cast attitude and does not require compensation. Similarly, during the 11 hr 09 min the Orbiter is in the final phasing orbit, the period change will be no greater than 0.0691 minutes. This could be compensated for by slightly decreasing ΔV_3 which will result in a slightly higher initial perigee altitude.

In the repeating ground track orbit, drag effects must be compensated for on a regular basis to maintain the required ground track. Premature reentry is also a factor since the lifetime of this orbit can be as short as 2 or 3 days if the worst case attitude is assumed continuously. The total drag compensation ΔV assuming the worst case attitude for the entire flight is approximately 61 m/sec. If the minimum drag, head-on, attitude is continuously flown, drag compensation ΔV is reduced by about an order of magnitude. The above results are based on a nominal solar activity model and do not include the effect of worst case increases in atmospheric density due to possible variations in solar activity.

C-3

5-7

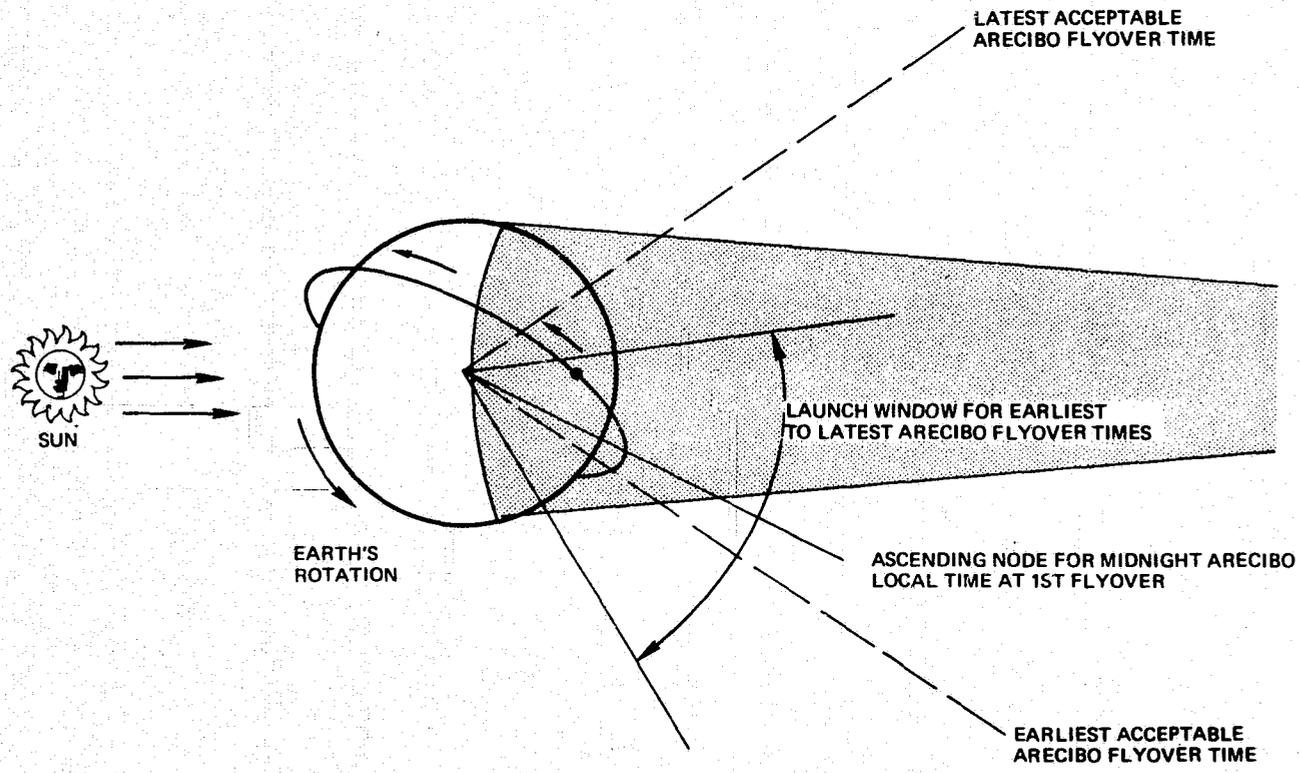


Figure 5-4. Flight 1 Launch Window Geometry Provides Midnight Local Time at First Arecibo Flyover

Table 5-1. Atmospheric Drag Effects on AMPS Flight One Orbits

Orbit	ΔP	ΔRP	ΔRA	ΔA	Drag Makeup ΔV	
	(min/day)	(km/deg)	(km/day)	(km/day)	m/sec/rev	m/sec/deg
Initial Phasing						
Nose forward	-0.0013	- 0.05	- 0.08	- 0.07	0.003	0.0043
Sideways (yaw 90 ⁰)	-0.0060	- 0.23	- 0.36	- 0.29	0.012	0.189
Nose down (pitch 90 ⁰)	-0.0118	- 0.46	- 0.70	- 0.58	0.024	0.375
Final Phasing						
Nose forward	-0.0165	- 0.22	- 1.41	- 0.82	0.034	0.527
Sideways (yaw 90 ⁰)	-0.0749	- 1.01	- 6.38	- 3.69	0.149	2.387
Nose down (pitch 90 ⁰)	-0.1507	- 2.20	-12.81	- 7.44	0.302	4.804
Repeating Ground Track						
Nose forward	-0.0335	- 1.22	- 1.91	- 1.51	0.061	0.972
Sideways (yaw 90 ⁰)	-0.1460	- 5.98	- 9.09	- 7.22	0.293	4.673
Nose down (pitch 90 ⁰)	-0.3231	-12.04	-20.00	-16.02	0.646	10.345

5.2 AMPS FLIGHT 2 MISSION ANALYSIS

The Flight 2 experiments that affect the choice of mission profile are:

- Low altitude conductivity modification which involves a barium canister release and detonation at approximately 200 km altitude over a cooperative high latitude tracking site at twilight
- Long delay echo observation using ESP-borne equipment at altitudes of at least 300 km over the mid- and equatorial latitudes
- Plasma flow experiments over low latitudes both day and night with no altitude restriction
- Detection and measurement of atmospheric minor constituents using a lidar system
- D-region changes and high latitude ozone detection experiments requiring high latitudes
- Oxygen variation measurements requiring both high latitudes and darkness.

The latitude of the available candidate tracking sites to support the low altitude conductivity modification experiment, such as Fort Churchill or Saskatoon, limits orbit inclination to no less than the latitude of the southern most site, which is Saskatoon at 52 degrees north. However, since some of the other experiments require latitudes as far north as possible at an inclination of 57 degrees, the limiting value from ETR, was chosen. Because this is a one time only experiment phasing is not critical, as on Flight 1. Also, because the tracking sites are located so far north, the Orbiter is visible to them several revolutions in a row as is shown later.

Twilight conditions at the tracking site at flyover are obtained by launching at the appropriate time of day. However, it is not possible to have both twilight and darkness at high latitudes on the same orbit over a six-day interval, so a compromise must be made.

A candidate mission profile has been developed for Flight 2 that satisfies the experiment constraints and is well within the OMS and RCS propellant budgets. The mission begins with a northeasterly launch

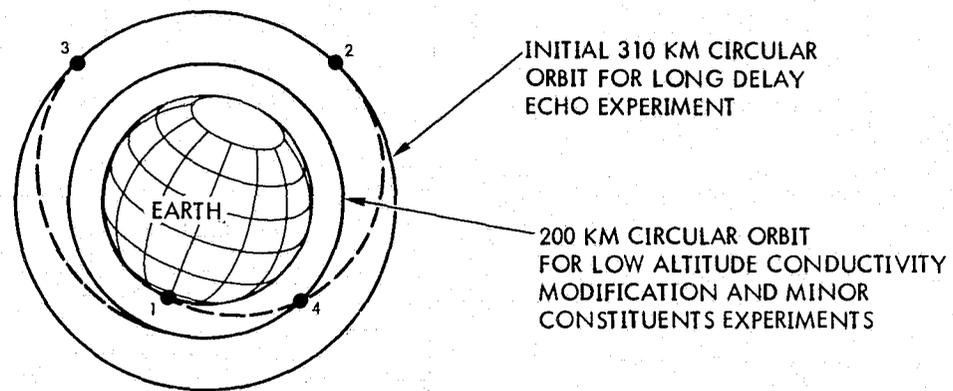
from ETR into a 115 by 310 km orbit inclined at 57 degrees. At first apogee the OMS adds 57 m/sec tangentially to circularize the orbit at 310 km. The Orbiter remains in this orbit for about 53 hours performing both the long delayed echos and a part of the D-region composition experiments. At approximately 53 hr 45 min after liftoff, a 32 m/sec OMS retro ΔV initiates a Hohmann transfer down to 200 km. Forty-five minutes later, a second retro ΔV of 33 m/sec circularizes the orbit at 200 km for the low altitude conductivity plasma flow, minor constituents, D-region changes, and oxygen variation experiments. This sequence is summarized in Figures 5-5, 5-6 and 5-7.

Figures 5-8 and -9 illustrate the ground track geometry for the Fort Churchill overflight. Revolutions 18, 19, and 20 all provide long tracking passes in excess of 6 minutes and are all equally well suited for the conductivity modification experiment. As mentioned earlier, phasing is not critical to obtain adequate Fort Churchill tracking coverage as indicated by the relative insensitivity of tracking time to tracking pass location over revolutions 18, 19 and 20. If, for example, atmospheric drag is not compensated for quite correctly, and the ground track shifts slightly to the east, the tracking geometry and tracking times will still be adequate.

Atmospheric drag effects will require correction by periodic RCS period trim maneuvers totaling less than 1 m/sec while in the 310 km orbit and 50 m/sec while in the 200 km orbit. This appears to be well within the Orbiter's RCS ΔV capability.

The requirement for Fort Churchill flyover at twilight drives the selection of launch time. Two daily launch windows are available. The first occurs during the early afternoon resulting in Fort Churchill flyover at dusk twilight. The second window occurs shortly before midnight and results in a dawn twilight flyover as shown in Figure 5-10.

Fort Churchill flyover at the dark limit of the dawn twilight region is preferred because the orbit plane regression motion will cause the flyover local times in the northern hemisphere to become progressively later as the mission continues, thereby improving darkness conditions for the other experiments. Conducting this mission at or near the winter solstice will provide the best darkness conditions at high northern latitudes.



EVENT	TIME HR:MIN	ΔV MPS	ORBITAL CHARACTERISTICS (KM)		INCLINATION (DEG)
			hp	ha	
1 ET SEPARATION THRU INSERTION	0:11	70	115	310	57
2 CIRCULARIZATION	0:43	57	310	310	57
3 TRANSFER	53:45	32	200	310	57
4 CIRCULARIZATION	54:30	33	200	200	57
5 DEORBIT	166:00	85	-	-	-

Figure 5-5. A Profile for Flight 2 has been Selected Which Meets All Scientific Objectives

- ① SHUTTLE INITIAL ORBIT INJECTION T = 0 HR 11 MIN
- ② OMS V_1 ; CIRCULARIZE IN 310 KM ORBIT T = 0 HR 46 MIN
- Ⓐ END OF 10TH REV IN 310 KM CIRCULAR ORBIT T = 15 HR 53 MIN

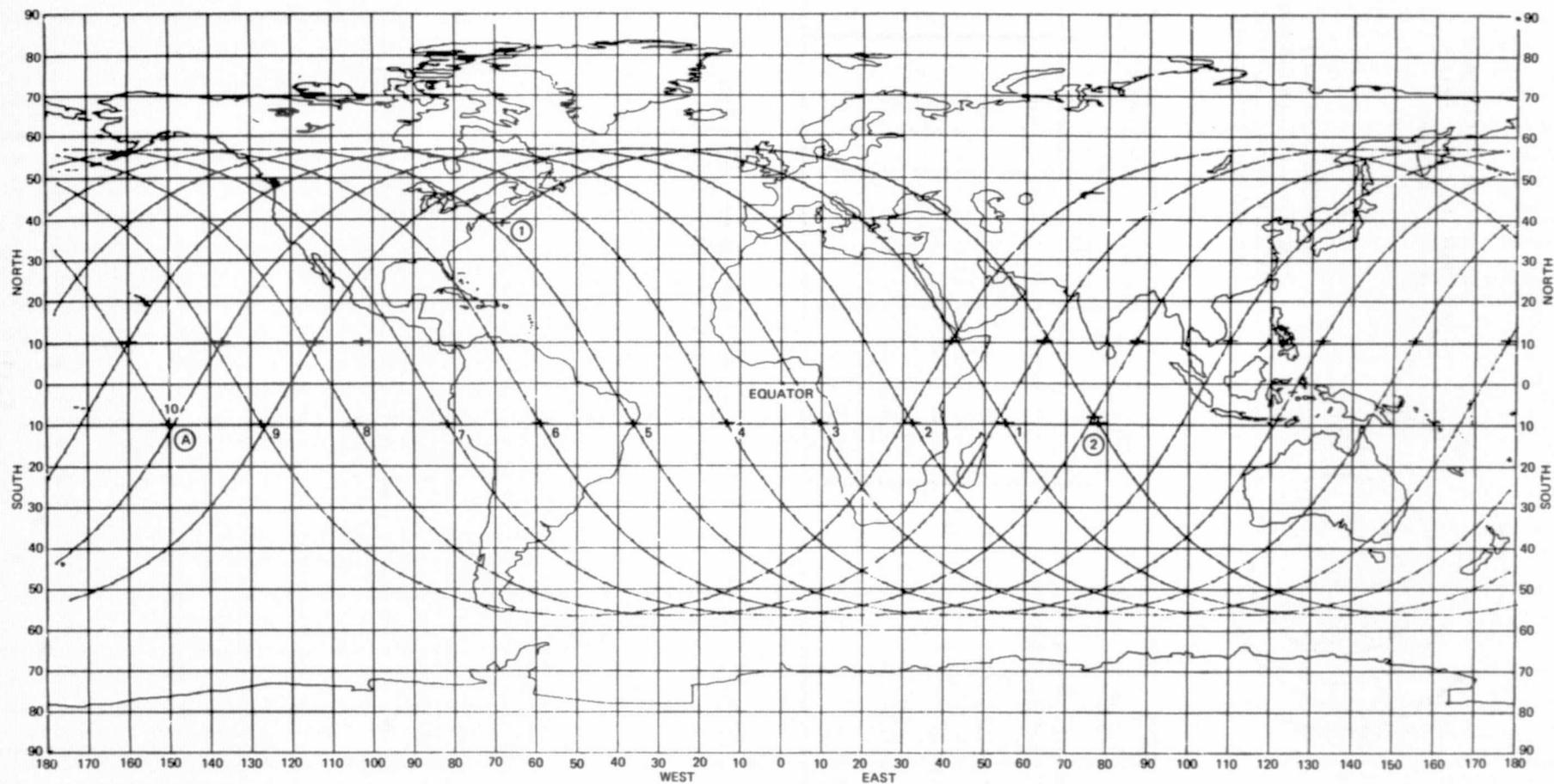


Figure 5-6. Flight 2, Shuttle Injection and First 10 Revolutions in 310 km Circular Orbit

REVS 10-20

T = 15 HR 53 MIN TO 31 HR 00 MIN

(A)

(B)

5-13

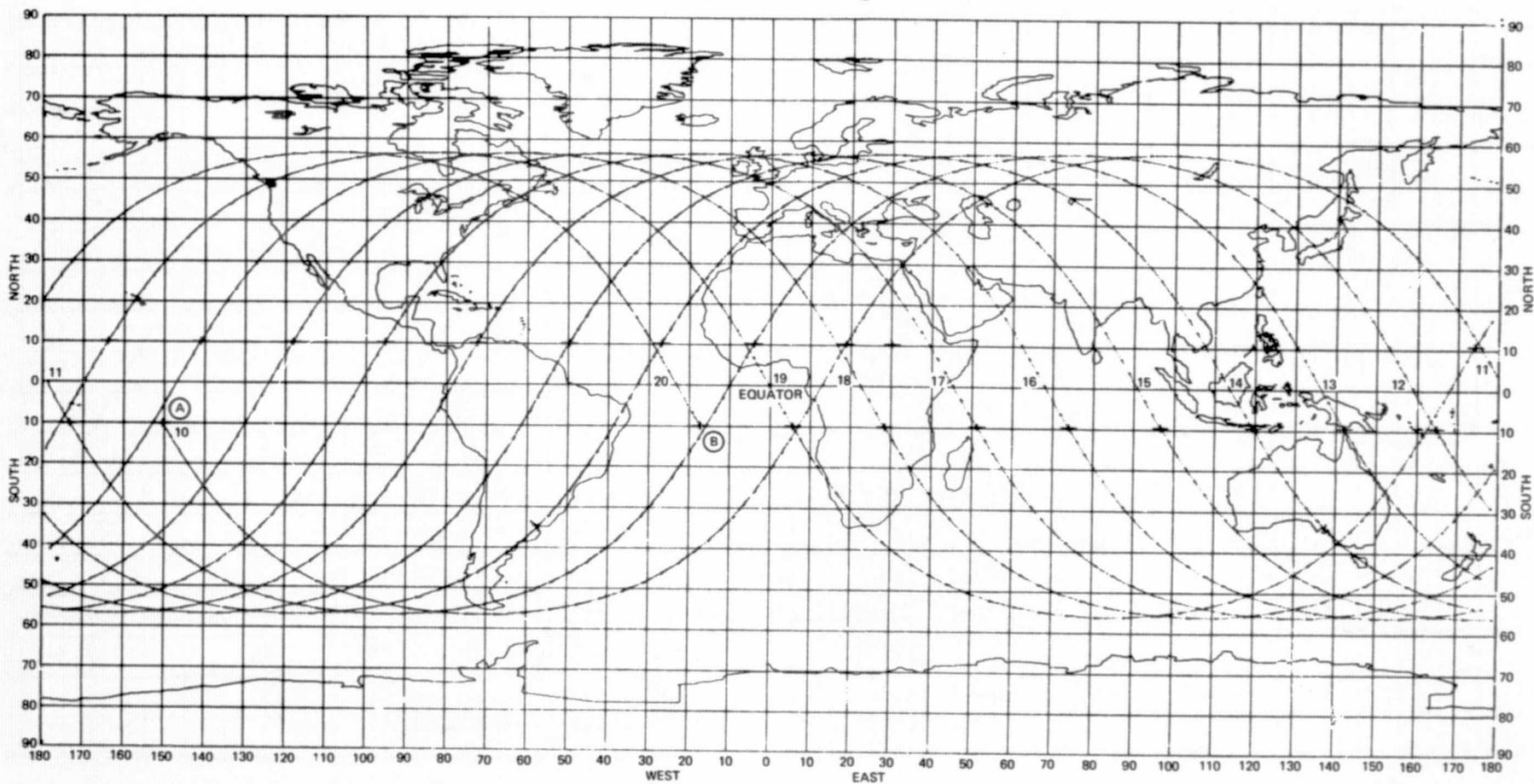


Figure 5-7. Flight 2, 310 km Circular Orbit

- ③ RETRO V_1 LOWER PERIGEE TO 200 KM 31 HR 00 MIN
- ④ RETRO V_2 LOWER APOGEE TO 200 KM, CIRCULARIZE 31 HR 45 MIN

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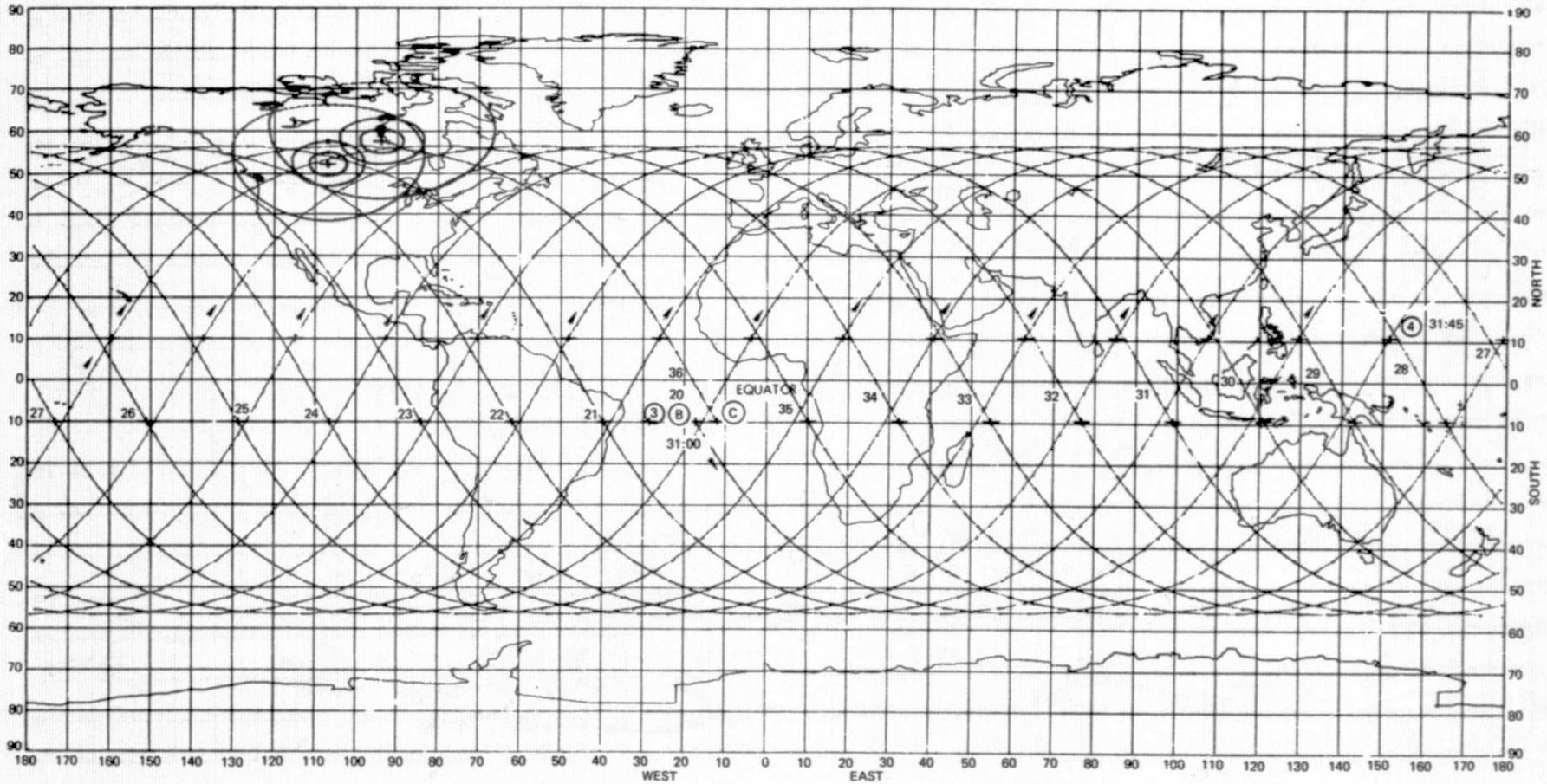


Figure 5-8. Flight 2, 200 km Circular Orbit
From (B) 31 hr 0 min to (C) 54 hr 35 min

5-15

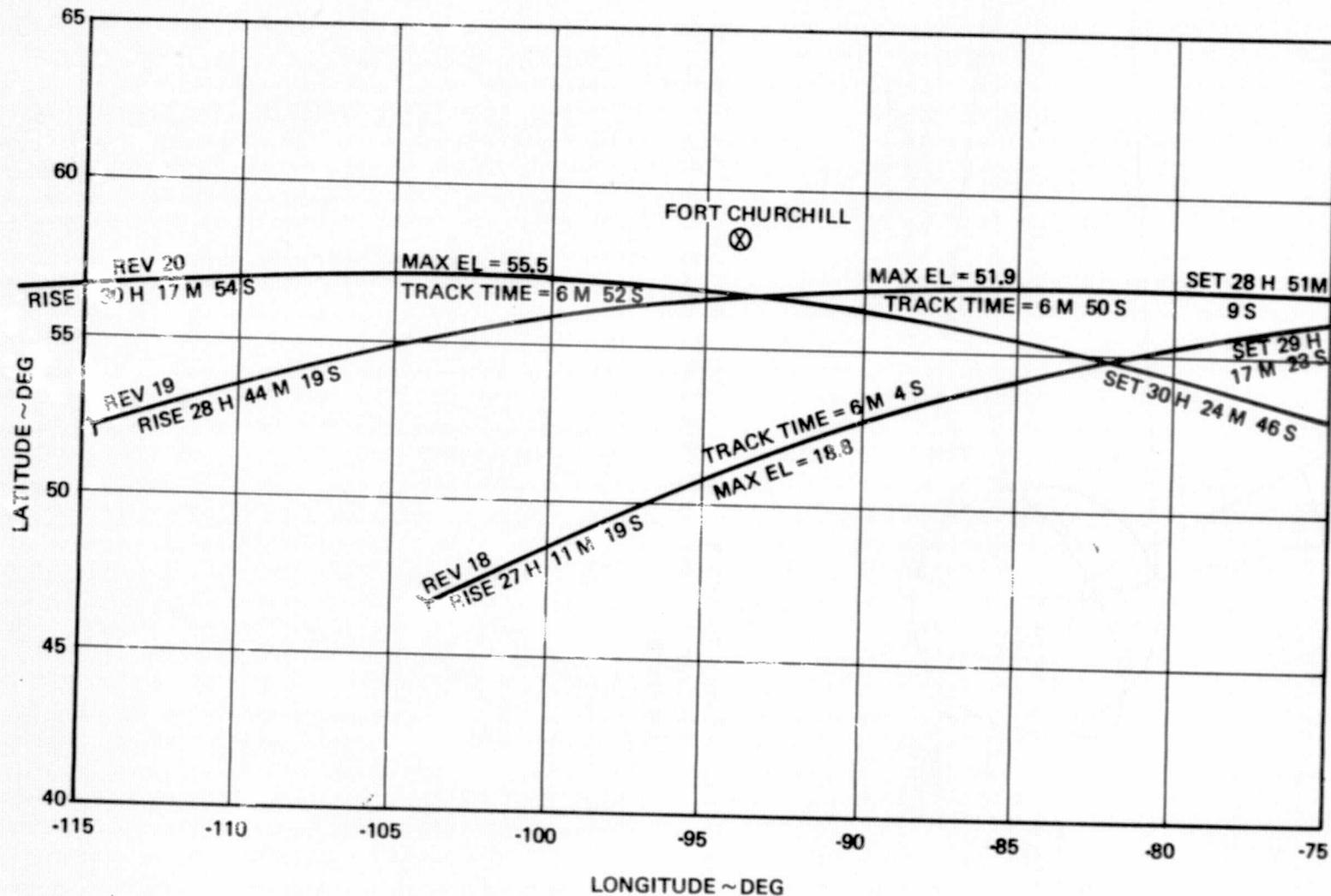


Figure 5-9. Fort Churchill Tracking Geometry for Conductivity Modification Experiment

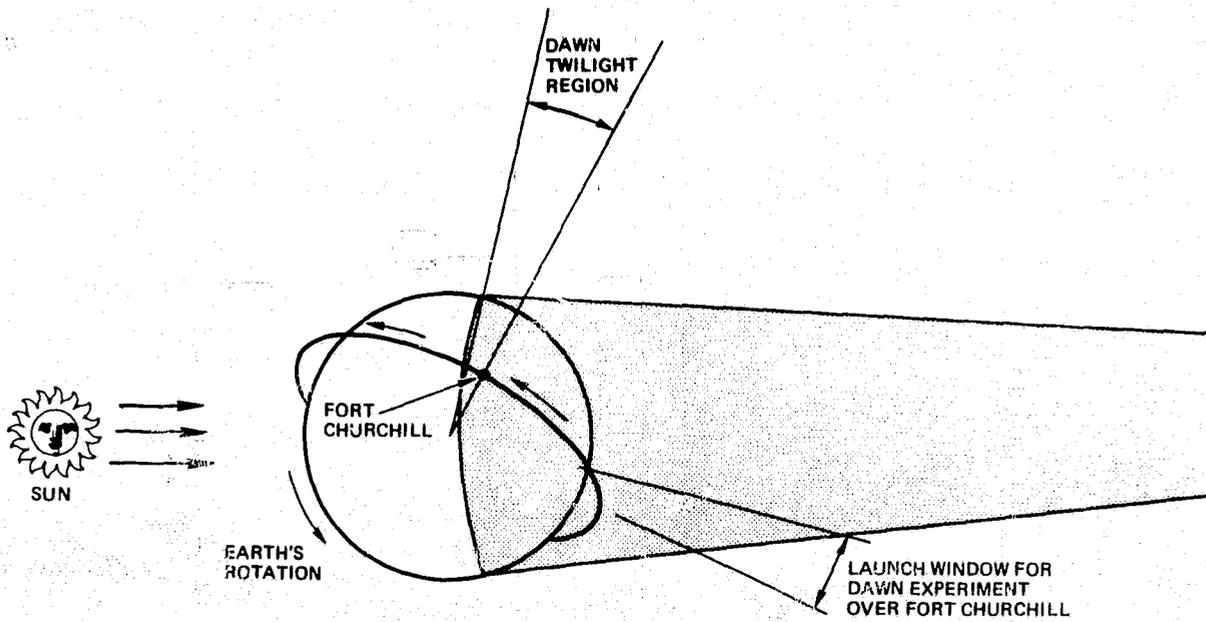


Figure 5-10. Flight 2 Launch Time Selected to Provide Fort Churchill Flyover at Dawn Twilight for Conductivity Modification Experiment

6. OPERATIONS ANALYSIS

This sections summarizes the AMPS operations analyses requirements for flight planning, payload flight support and flight crew operations.

6.1 FLIGHT PLANNING

The objective of the flight planning requirements analysis was to establish the products and documentation required by the STS planning activity and to search for approaches which would minimize planning costs. AMPS flight planning includes:

- Selection of orbits, trajectories, and schedules
- Development of operational procedures and timelines
- Planning of resource utilization
- Development of attitude and pointing profiles
- Preparation of a flight data file.

6.1.1 Flight Planning Products

An overview of the products and interfaces of the planning function is shown in Figure 6-1. As can be seen from the figure, a detailed understanding of user requirements, STS capabilities, AMPS payload design and capability, and the STS planning sequence is required to accomplish these functions. The output of the flight planning and analytical integration activities is combined to produce the Mission Support Requirements Document (MSRD). This document defines the STS resources required to support each flight. A preliminary version of this document has been developed (Vol II), the outline is shown in Figure 6-2. The next document is the AMPS Detail Flight Plan. This document provides the information required by the STS planning function to prepare the Integrated Flight Plan. The AMPS Detail Flight Plan provides the AMPS experiment related:

- Flight requirements
- Orbit selection
- Experiment timeline
- Attitude and pointing requirements

- Consumables/resource utilization
- Experiment procedures.

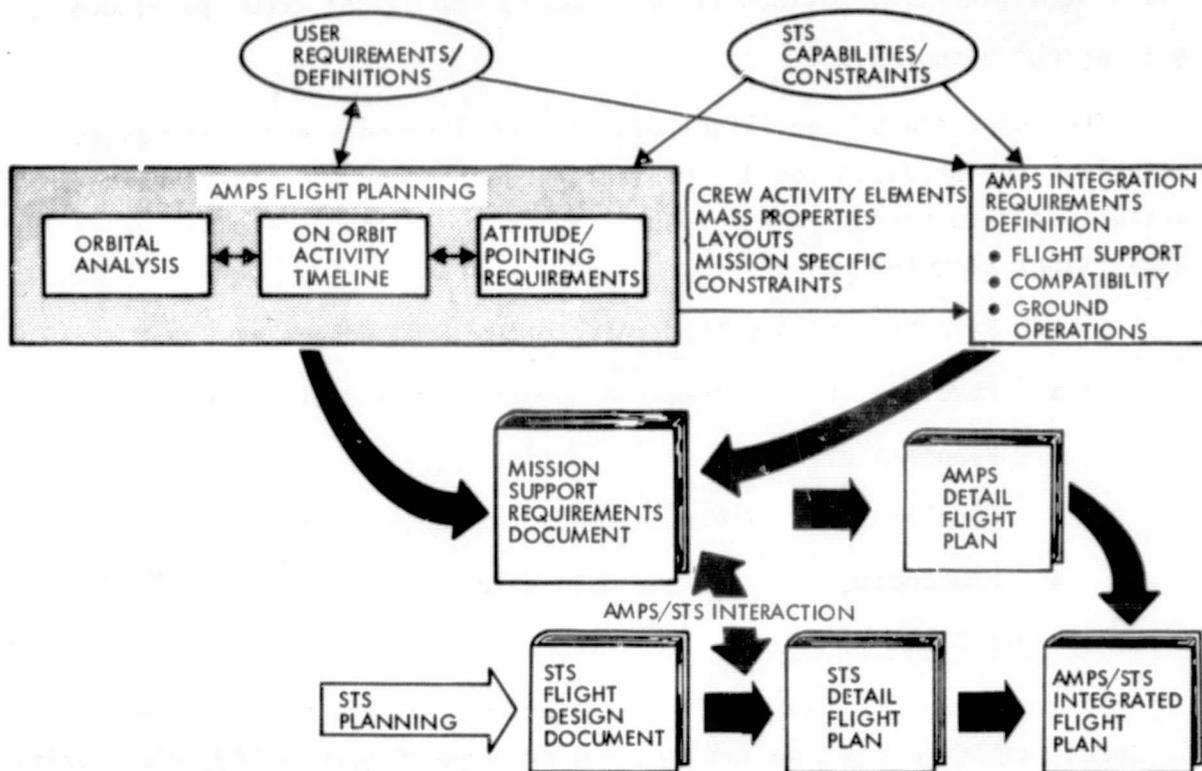


Figure 6-1. Flight Planning Products and Interfaces

These data, when combined with the detailed instrument procedures and characteristics, are included in the Flight Data File (FDF). The FDF is the data carried on board the Spacelab for reference during the flight. Portions of the FDF are incorporated in the Spacelab experiment computer and the bulk data file. The mass memory of the experiment computer stores nominal operations and parameters for display as an aid during experiment operations. These data are also included in the book part of the FDF along with the event timeline and more detailed procedures. These data are changeable in flight. Data which do not change in flight are placed in the Bulk Data File (BDF). This device establishes permanent records such as schematics and CDMS procedures. The experiment computer searches the BDF for display on the video system.

<ul style="list-style-type: none"> 1. INTRODUCTION <ul style="list-style-type: none"> 1.1 PURPOSE 1.2 FLIGHT OBJECTIVES 1.3 FLIGHT REQUIREMENTS 1.4 SUCCESS CRITERIA 1.5 APPLICABLE DOCUMENTS 2. EXPERIMENT OPERATIONS <ul style="list-style-type: none"> 2.1 EXPERIMENT SEQUENCE DEFINITIONS 2.2 EXPERIMENT OPERATION/EQUIPMENT MATRICES 2.3 EXPERIMENT OPERATIONAL CONSTRAINTS 2.4 INSTRUMENT OPERATIONAL CONSTRAINTS 3. PAYLOAD CONFIGURATION <ul style="list-style-type: none"> 3.1 PAYLOAD MANIFEST AND PHYSICAL PROPERTIES 3.2 LAYOUT 3.3 INSTRUMENT DESCRIPTIONS 3.4 SUPPORT EQUIPMENT DESCRIPTIONS 3.5 COMMON PAYLOAD SUPPORT EQUIPMENT 3.6 SPACELAB MISSION-DEPENDENT EQUIPMENT 3.7 MULTIMISSION SUPPORT EQUIPMENT 3.8 SOFTWARE 3.9 FLIGHT DATA FILE 3.10 MISCELLANEOUS EQUIPMENT 4. PAYLOAD FLIGHT REQUIREMENTS <ul style="list-style-type: none"> 4.1 FLIGHT PLAN <ul style="list-style-type: none"> 4.1.1 ORBIT CHARACTERISTICS 4.1.2 FLIGHT TIMELINE 4.2 SPACELAB/ORBITER RESOURCE REQUIREMENTS <ul style="list-style-type: none"> 4.2.1 ELECTRICAL POWER 4.2.2 THERMAL CONTROL 4.2.3 DATA HANDLING 4.2.4 SOFTWARE 4.2.5 POINTING 4.2.6 STOWAGE 4.2.7 OMS/RCS PROPELLANTS 4.2.8 CONTROLS AND DISPLAYS 4.3 SPACELAB/ORBITER INTERFACE REQUIREMENTS <ul style="list-style-type: none"> 4.3.1 CONTAMINATION 4.3.2 ELECTROMAGNETIC COMPATIBILITY 4.3.3 SAFETY 4.4 DATA AND COMMUNICATION REQUIREMENT <ul style="list-style-type: none"> 4.4.1 REAL-TIME DOWNLINK DATA 4.4.2 UPLINK COMMANDS 4.4.3 STORED AND DUMPED DATA 4.4.4 VOICE AND TV 4.4.5 TDRSS REQUIREMENTS 4.4.6 DOMSAT REQUIREMENTS 4.4.7 ON-BOARD DATA PROCESSING 4.4.8 POST-FLIGHT DATA PROCESSING 4.4.9 DATA DISTRIBUTION 	<ul style="list-style-type: none"> 5. PAYLOAD GROUND REQUIREMENTS <ul style="list-style-type: none"> 5.1 SPACELAB ELEMENT STAGING REQUIREMENTS FOR PAYLOAD INTEGRATION CENTER <ul style="list-style-type: none"> 5.1.1 MISSION-DEPENDENT EQUIPMENT 5.1.2 MISSION-INDEPENDENT EQUIPMENT 5.1.3 MULTIMISSION SUPPORT EQUIPMENT 5.1.4 COMMON PAYLOAD SUPPORT EQUIPMENT 5.2 PAYLOAD INTEGRATION/GROUND OPERATIONS <ul style="list-style-type: none"> 5.2.1 PAYLOAD INTEGRATION 5.2.2 GROUND OPERATIONS 5.3 GSE REQUIREMENTS <ul style="list-style-type: none"> 5.3.1 PAYLOAD INTEGRATION CENTER 5.3.2 LAUNCH SITE 5.4 FACILITIES AND SERVICES <ul style="list-style-type: none"> 5.5.1 STORAGE 5.5.2 SERVICES 5.5.3 SPECIAL FACILITIES/EQUIPMENT 5.5 LOGISTICS <ul style="list-style-type: none"> 5.5.1 PAYLOAD INTEGRATION CENTER 5.5.2 LAUNCH SITE 6. PAYLOAD FLIGHT SUPPORT REQUIREMENTS <ul style="list-style-type: none"> 6.1 POCC REQUIREMENTS <ul style="list-style-type: none"> 6.1.1 GROUND COMMANDS 6.1.2 GROUND DISPLAY AND CONTROL 6.1.3 GROUND DATA PROCESSING 6.2 POCC SUPPORT OPERATIONS FOR PAYLOAD INTEGRATION 6.3 POCC PERSONNEL SIMULATIONS TRAINING REQUIREMENTS 7. FLIGHT CREW REQUIREMENTS <ul style="list-style-type: none"> 7.1 CREW SIZE 7.2 PAYLOAD SPECIALIST QUALIFICATIONS 7.3 TRAINING REQUIREMENTS 7.4 TRAINING EQUIPMENT REQUIREMENTS 7.5 SPACELAB/ORBITER TRAINING REQUIREMENTS
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Figure 6-2. Mission Support Requirements Document

In-flight changes to the FDF are made through uplink computer loads, text and graphics uplink, and manual entry. The text and graphic uplink provide a permanent record of procedure and mass memory changes. This mode of operation assures that new procedures are followed by other crew members, who also conduct the experiments.

6.1.2 Planning Approaches That Reduce Costs

Lower costs for payload flight planning can be achieved by careful attention to three major factors. These factors have been identified as important for reducing manpower per flight. In the material that follows, each factor is analyzed to determine its contribution to cost-savings and implementation methods for achieving these lower costs. The factors are:

- 1) Minimize contingency/malfunction planning
- 2) Minimize flight planning iterations
- 3) Maximize common use of manpower.

6.1.2.1 Minimize Contingency/Malfunction Planning

The likely payload contingencies, their causes and remedial actions, have been identified and summarized in Table 6-1. It is important to note that all elements of the flight (Orbiter, Spacelab, experiment equipment, procedures, and crew timelines) will be developed to minimize the occurrence of malfunctions or contingencies; accordingly, we expect malfunctions and contingencies to decrease as the STS and payload technology mature. The need for payload contingency planning will correspondingly decrease.

The probable actions, shown in Table 6-1, are all within the capabilities of the POCC and its supporting complement of Principal Investigators (PI), experiment engineers, and flight planners. The resources of the MCC will provide comparable support for work-around procedures for orbit insertion errors and Spacelab subsystem malfunctions. From Skylab, experience shows that the flight crew is capable of implementing corrective actions for payload malfunctions and contingencies.

6.1.2.2 Minimize Experiment Planning Iterations

Manpower and computer hours for payload flight planning are directly related to the number of times the flight plan is updated. It is recommended that a new plan, or an update of an existing plan, be implemented only at the following times:

- When a flight plan is needed to support experiment equipment design specifications, or to assemble requirements for flight support from the STS, the launch site, communications networks and other support agencies
- When hardware test data become available for integrating into detailed timelines, procedures, consumables, and pointing analysis. As a subset, refinement of a detailed flight plan may be necessary on the basis of simulation of experiment operations and training exercises.

Table 6-1. Most Contingencies are Solved by Change to the Timeline

Experiment Contingency Factor	Probable Cause	Probable Action	General Characteristics
Experiment Equipment Performance	Equipment breakdown	<ul style="list-style-type: none"> ● Fault isolation ● Work around 	<ul style="list-style-type: none"> ● Large number of variations ● Largely unpredictable ● Generally only causes loss of experiment time ● Most likely action is a change in the timeline
Orbital deviations	Variation in launch time, insertion orbit, etc.	<ul style="list-style-type: none"> ● Revised timeline 	
Natural phenomena	Occurrence of flares, weather, etc.	<ul style="list-style-type: none"> ● Revised timeline 	
Human factors	Variation of crew performance in zero-environment	<ul style="list-style-type: none"> ● Revision of experiment performance time ● Revised timeline 	
Scientific data	Scientific phenomena not as expected	<ul style="list-style-type: none"> ● Reconfiguration of equipment 	

The advantage of limited iterations is in a reduction of costs (manpower and computers) from the costs of continuous flight planning during the preflight periods.

For Spacelab payloads, Figure 6-3 shows the minimum flight-plan iteration requirement together with their intended purpose. Flight Plan A defines total flight characteristics and constraints that must be considered in design and test of the experiment equipment. Flight Plan B is a detailed

plan that includes experiment timelines, procedures, consumables, and pointing, all of which should be compatible with the actual flight hardware. Flight Plan C is an update that considers the impact of simulations and integrated crew training; this flight plan becomes part of the Flight Data File.

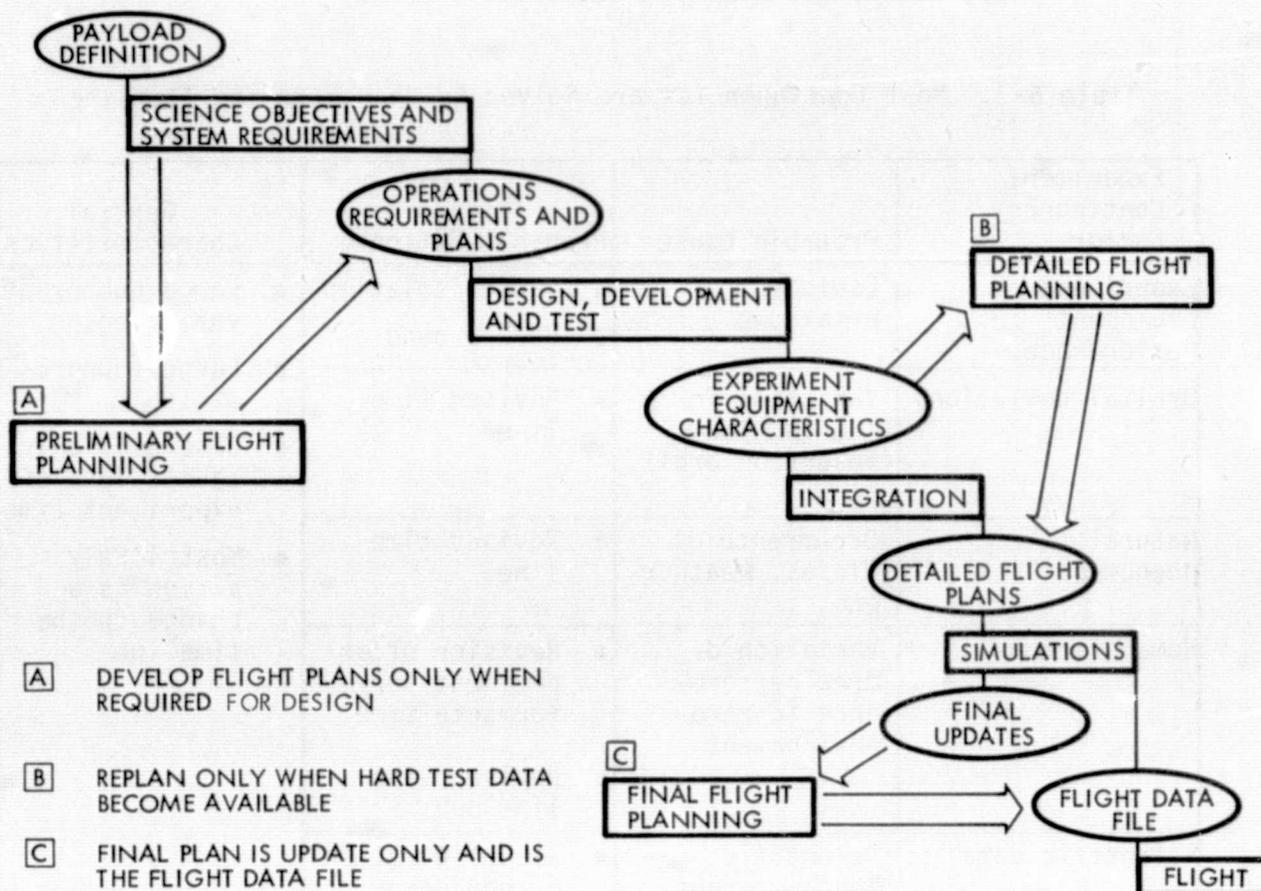


Figure 6-3. Experiment Flight Planning Iterations

Manpower estimates were developed for the three iteration planning schemes. The estimates for manpower were provided by TRW people who supported the flight planning for Skylab, Apollo, and ASTP. A definition of the planning functions and a relative manpower profile is shown in Figure 6-4 for the three iterations to the flight plan and for a limited degree of flight plan maintenance. The values are consistent with the plan described in Figure 6-3.

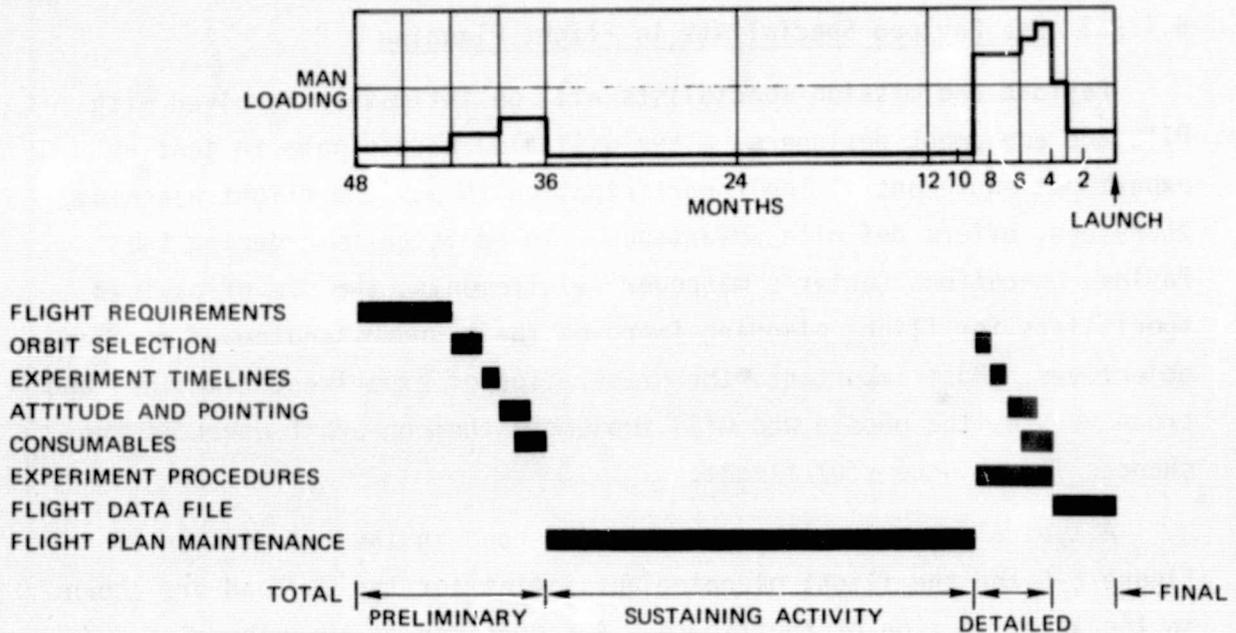


Figure 6-4. Nominal Payload, Man-Month Estimates for Flight Planning

Flight plan maintenance is shown for a period that is typical of the manufacture and test of new experiment equipment. For reflights, this period would be shorter because this equipment would require only refurbishment or minor modifications. Flight plan maintenance would be reduced accordingly.

AMPS instruments and flight support equipment are expected to be used on additional flights with only slight modification between flights. Thus, large portions of the previous flight plan can be reused, resulting in lower manpower requirements for planning. For example, procedures for operating the equipment will change only slightly and much of the Flight Data File can be used again. Also, sustaining activity is lower than for the first flight of the payload because the equipment-procurement cycle is greatly reduced in scope and time.

6.1.2.3 Use Payload Specialists in Flight Planning

Payload and mission specialists will be intimately involved with PI's and equipment designers. They will also participate in testing experiment equipment. Their participation in payload flight planning, therefore, offers definite advantages. In addition to reducing the Payload Operations Center's manpower requirements, the use of payload specialists for flight planning improves their understanding of mission objectives. Most important, the preparation of experiment timelines and procedures by the people who will implement them on-orbit enhances the chances for a successful flight.

A typical crew training schedule is shown in the top section of Figure 6-5 and the flight planning activities for the payload are shown in the lower section of the figure. A comparison of the schedules and the activities being performed indicates that it is both possible and desirable to use payload specialists to perform significant portions of the payload flight plan. For example, block B (procedural training on experiments) and block 5 (experiment procedures) occur in parallel and should really be performed together (i.e., experiment procedures must be written in order to do the procedural training, and their use in training shows what changes are needed to make them realistic. Also, orbit selection, experiment timelines, attitude and pointing, and consumables analyses can impact the procedures for operating the experiments and should be considered during procedural training on experiments; accordingly, blocks 1, 2, 3, and 4 are deemed logical activities for the payload and mission specialists during their training on experiment procedures.

Analysis of the training load for the payload and mission specialists indicates that time is available for payload flight planning up to 4 months prior to launch. The likelihood that backup crew members will be assigned and trained increases the amount of specialist time that can be applied to the flight planning activity.

TRAINING SEQUENCE

- (A) SCREENING, ORIENTATION AND SYSTEM FAMILIARIZATION
- (B) PROCEDURAL TRAINING ON EXPERIMENTS
- (C) EXPERIMENT/SPACELAB INTERFACE TRAINING
- (D) COMBINED EXPERIMENTS TRAINING
- (E) HABITABILITY AND SAFETY TRAINING
- (F) INTEGRATED OPERATIONS TRAINING
- (G) STS/SPACELAB SYSTEMS O&M TRAINING

PLANNING SEQUENCE

- (1) ORBIT SELECTION
- (2) EXPERIMENT TIMELINES
- (3) ATTITUDE AND POINTING
- (4) CONSUMABLES
- (5) EXPERIMENT PROCEDURES
- (6) FLIGHT DATA FILE

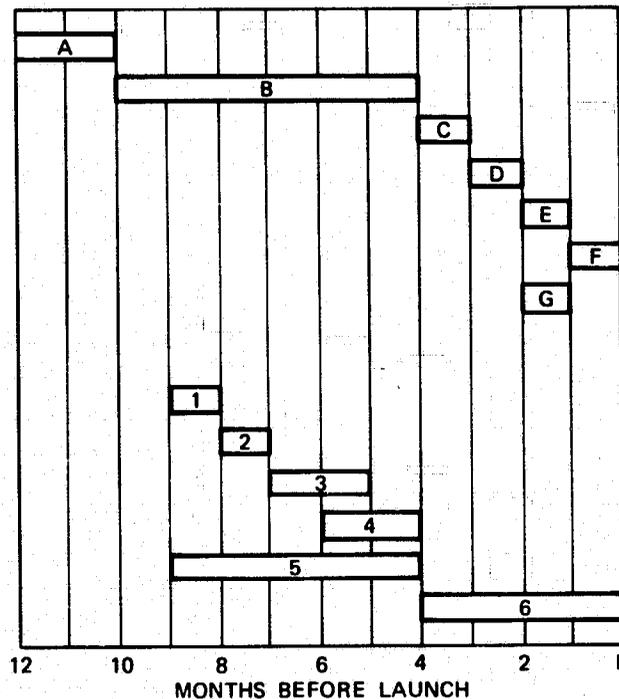


Figure 6-5. The Payload Flight Crew Can Participate in Flight Planning

6.1.2.4 Summary and Recommendations

The manpower reduction factors, the rationale for their selection and the advantages are summarized in Table 6-2. Based on the analysis performed during the AMPS study, it is recommended that GSFC review the scope and timing required for flight planning and consider the combination of some aspects of flight planning and flight crew training.

6.2 PAYLOAD FLIGHT SUPPORT

The AMPS flight system and Payload Operations Control Center are designed to complement the National Space Transportation System command, control, and communications network and maximize the scientific return for each flight. The flight system concept includes the proper mix of automated and manual controls to enhance the crew's ability to make decisions and adjustments during the experiments.

Table 6-2. Evaluation of Manpower Reduction Factors

Cost Manpower Reduction Factors	Rationale	Advantages
Minimize contingency/malfunction planning	<ul style="list-style-type: none"> ● Most contingencies solved by changes to timeline ● Science/equipment must be evaluated in real time 	<ul style="list-style-type: none"> ● Reduces total manpower
Minimize flight planning iterations	<ul style="list-style-type: none"> ● Develop flight plans only when required to support payload operations planning or design ● Replan only when hard test data become available 	<ul style="list-style-type: none"> ● Reduces total manpower ● Allows use of planners for other systems analysis activities during hardware development ● Reduces computer usage
Maximize common use of manpower	<ul style="list-style-type: none"> ● Crew training and flight planning are closely related ● Payload flight crew can participate in flight planning ● Training and planning sequences are synchronized 	<ul style="list-style-type: none"> ● Reduces lead center manpower requirements ● Maximizes use of highly qualified people ● Enhances continuity from flight planning through operations

The Payload Operations Control Center (POCC) provides support such as activity replanning, payload hardware performance assessment, fault isolation, and the assessment of scientific data. The combination of these design features allows the scientific flight crew to concentrate on scientific observations and permits on-the-spot decision making.

AMPS studies indicate that onboard scientific crew members can contribute significantly to the investigative process. Payload specialists, coordinating with ground investigators, can make decisions on power levels, frequencies, and wavelengths, based on realtime analysis of data. They can make judgments on the appropriate conditions for conducting experiments and determine what previously unexpected effects might be interfering with an experiment.

An overview of the approach is shown in Figure 6-6. Real-time interaction between the inflight payload specialists and ground-based scientists is the key to AMPS success. The flight and ground crews each have significant roles. The flight crew concentrates on instrument operation, science observation, "on the spot" decisions, and implementation of ground developed adjustments to plans and procedures. The POCC team provides data verification, Principal Investigator observation, activity scheduling and replanning, instrument performance evaluation, and fault isolation and the development of "work around" plans and procedures.

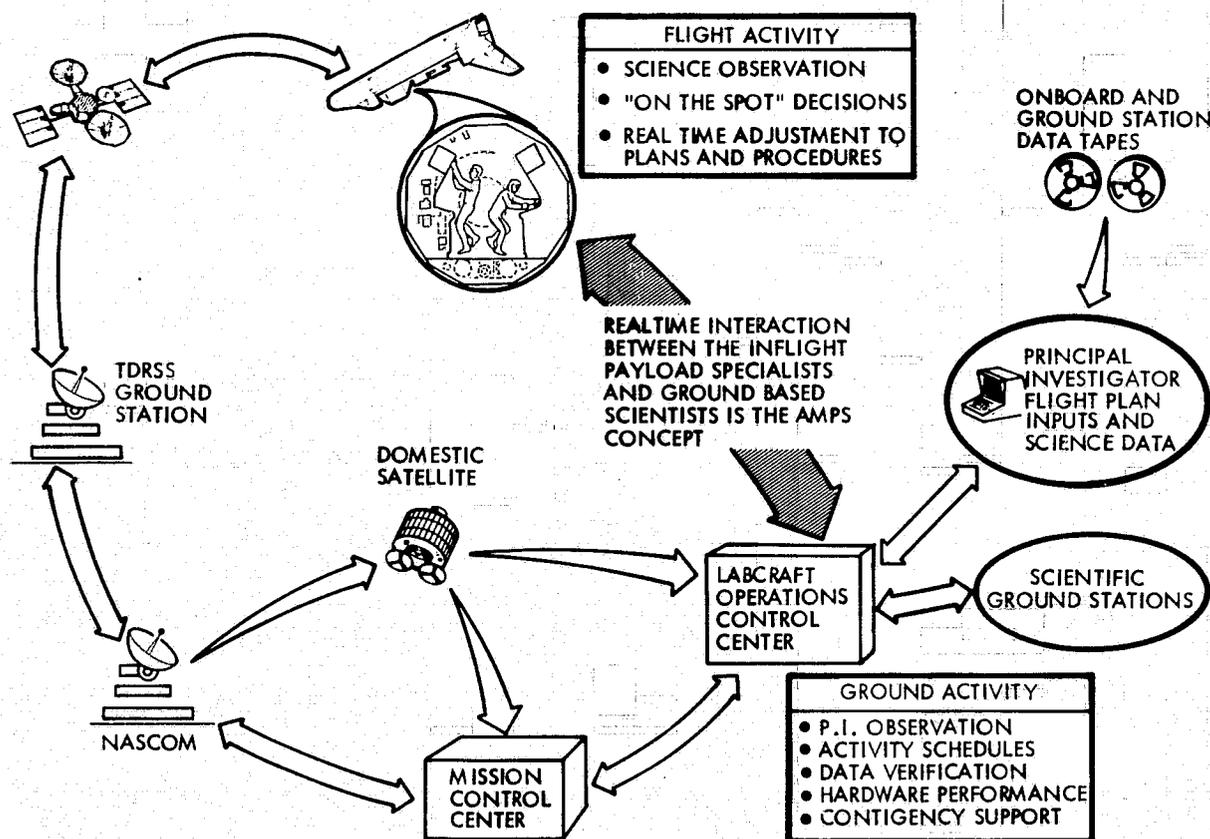


Figure 6-6. AMPS Flight Operations Concept Maximizes Payload Specialist Science Investigation Time

The Principal Investigators not present in the POCC have remote access to the data for review and evaluation. This takes advantage of the GSFC role as the Spacelab data reduction facility for non-time critical data. All data required for experiment assessment is transmitted through the

Orbiter Ku-band system, the TDRS system, and domestic satellite to the POCC. All data is time tagged and includes state vector and instrument housekeeping, as well as scientific information.

Support of the AMPS mission requires ground facilities other than the POCC, including:

- Mission Control Center (MCC)
- Subsatellite Control Center (SCC)
- Science Observation Stations.

The MCC is responsible for integrated flight planning and command verification and will provide Spacelab and Orbiter systems status and Orbiter state vector. There is no need to duplicate the processing and formatting of these data in the POCC. The SCC provides subsatellite maneuver and attitude control, and trajectory determination for those AMPS flights which will use free flyers. The science observation stations are required to perform special data acquisition and evaluation for specific experiments. Use of these stations is generally based on continued experimentation from balloon, sounding rocket, or other space programs.

The following paragraphs present the results of studies to establish the preliminary characteristics of the AMPS POCC.

6.2.1 Requirements Definition

An analysis was made of the experiments of the first and second AMPS flights, as they are defined in the IFRD's. This analysis established the ground support requirements for each of the planned experiments. A summary of the requirements is shown in Figure 6-7.

6.2.1.1 Payload Operations Control Center Design

Based on the requirements established, a number of information displays and communication situations were postulated. These flight planning aids were developed so that a necessary and complete set of aids could be defined for each experiment.

The information displays are broken down into:

- Those that are of a dynamic nature and so require computer assistance in their formulation either for formatting of data or computation of data products

EXPERIMENT	MANEUVERS	POINTING	TIME DEPENDENCY	POSITION RELATIONSHIP	VOICE	COMMAND	DOWNLINK DATA		
							TV MHZ	RATE MBPS	TOTAL MB
ACOUSTIC GRAVITY WAVE AND GAS CLOUD EXPANSION	CANISTER JETTISON			RELEASE OVER GROUND STATION	✓		4.2	2.51	7,500
ELECTRON ACCELERATOR		Z II B X II V	DARKNESS		✓		4.2	0.52	28,100
BEAM PLASMA INTERACTIONS		Z II B X II V	AFTER ELECTRON ACCELERATOR		✓		4.2	0.52	9,300
MINOR CONSTITUENTS		Z NADIR Y POP			✓			2.6	221,000
OZONE DENSITY		Z NADIR			✓			2.6	157,000
ENVIRONMENTAL MEASUREMENTS					✓			0.02	450
SOLAR RADIATION		Z-LOCAL VERTICAL Y-POP						0.003	40
LOW ALTITUDE CONDUCTIVITY	CANISTER JETTISON	OBIPS TRACKING	TWILIGHT	RELEASE OVER GROUND STATION	✓	✓	4.2	0.02	10
HF/WAVE PARTICLE INTERACTION	ORBITER/ESP POSITION	Y-POP Z-NADIR			✓			0.02	1,680
LONG DELAYED ECHOS	ORBITER/ESP POSITION	Y-POP Z-NADIR		EQUATORIAL LATITUDES	✓			0.02	1,000
PLASMA FLOW		Y II V X-POP		LOW LATITUDES	✓		4.2	0.02	1,920
HIGH LATITUDE OZONE ANOMALOUS COMPOSITION D-REGION COMPOSITION		EARTH TARGETS	NATURAL ACTIVITY		✓			2.6	610,600
OXYGEN VARIATION		EARTH TARGETS			✓			0.01	2,400

Figure 6-7. Ground Support Requirements for AMPS Flights 1 and 2

- Those that become fixed when the actual orbit has been achieved, such as ground track, or those that are supplied by external agencies, such as weather prediction.

Three levels of autonomy were defined in order to develop equipment requirements for the Payload Operations Control Center (POCC), as they relate to the amount of planning and operational autonomy allowed the crew.

- 1) Assistance Only. Full autonomy is allowed the crew except that the POCC must be ready to assist in diagnosis of malfunctions and in recommending remedial measures either through repair or through changes in procedures and plans.
- 2) Minimum Command. This level provides the minimum amount of equipment necessary for the POCC to command instruments when the crew is not available. It also allows the POCC to develop daily activity plans for recommendation to the crew.
- 3) Full Control. This level provides adequate equipment for the POCC to do all the planning and instrument commanding. It does not provide for a console dedicated to each instrument in those cases where all instruments will not be operated simultaneously.

The ground equipment and manpower required to operate the POCC is a function of the level of flight crew autonomy. An assessment of the displays, command positions and communication functions required to support AMP's for each of the POCC alternatives is shown in Figure 6-8.

FUNCTION	CONTROL CENTER CAPABILITY		
	ASSISTANCE ONLY	MINIMUM COMMAND	FULL CONTROL
DISPLAYS			
DYNAMIC DISPLAYS	4	8	15
DAILY DISPLAYS	3	3	3
COMMAND POSITIONS	0	1	6
COMMUNICATIONS			
SPACE/GROUND	X	X	X
MISSION CONTROL	X	X	X
GROUND OBSERVERS	X	X	X

Figure 6-8. Information Display and Communication Alternatives

The functional classes of these displays include:

- Investigation status
- Planning aids
- Instrument status
- Consumables status
- Scientific data
- Instrument configuration
- Data status.

Our approach to flight operations assigns a significant role to both the flight and ground crews. The "minimum command" POCC configuration best fits this approach. The number of displays is adequate to support the anticipated data review and monitor and evaluation functions planned. A single command station is planned to format computer loads for the experiment computer.

The POCC presents sufficient data to payload personnel so they can assist in optimizing the scientific observations. Except for commanding instruments, most of the decisions in the POCC have a time scale on the order of hours as contrasted with the short time scale of safety-related decisions. This aspect, by relieving much POCC equipment from real-time operation, simplifies the computational and display components.

The POCC design provides for payload data to be routed directly to the POCC. Any payload data that is interleaved with Orbiter instrumentation data is stored in the MCC data base and is accessible by the POCC computer; the POCC can directly access Orbiter and Spacelab systems data and trajectory information in MCC format. Based on these assumptions and the equipment requirements, a generalized POCC design was developed. The functions and interfaces of the POCC are shown in Figure 6-9.

The POCC provides for historical storage of all payload data, formatting commands to the payload, voice communication with the Spacelab, and display of Spacelab TV. The number of consoles and other peripherals to be used can be adjusted, over a reasonable range, as demanded by the particular flight.

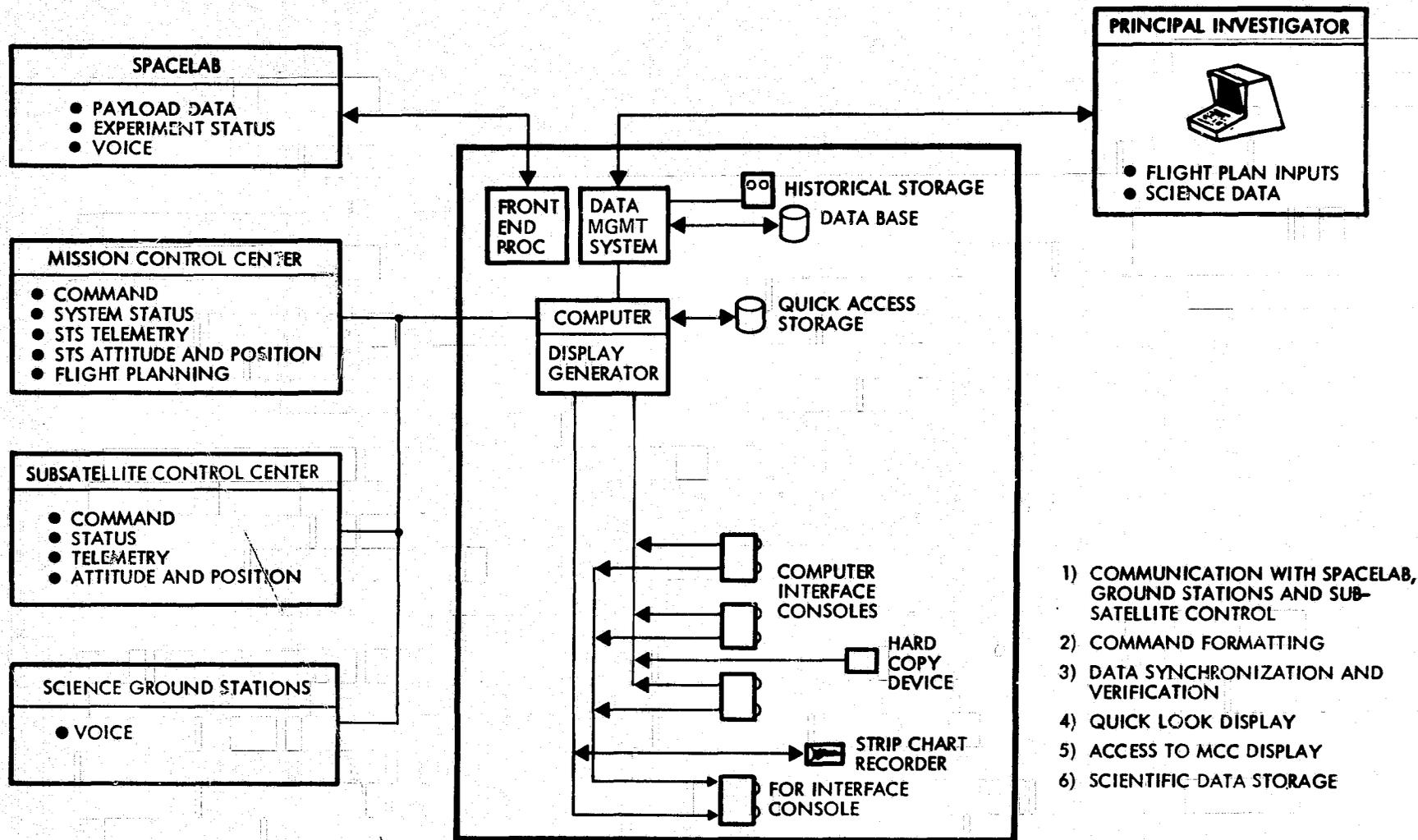


Figure 6-9. Labcraft Operations and Control Center Functions and Interfaces

The POCC design includes a front-end processing system to perform science data synchronization and verification; data management system to control data base access; data storage; and computers to control operations, perform calculations, and develop displays and consoles which provide the interface between the ground crew and each of the other elements of the flight operation. The consoles will be selected to interface with the POCC computer and display generator. It would be advantageous if they had similar characteristics to those in the MCC so that all consoles could access Orbiter and Spacelab data. If this is not practical, a two MCC consoles should be provided.

Figure 6-10 is a summary of the POCC equipment functions and capability required to support the AMPS mission. The maximum downlink data rate is defined by operation of the OBIPS instrumentation on Flight 2. The data storage estimates are defined to provide 24 hours of data for quick look display. Preliminary estimates indicate that approximately 10 percent of the science data stream is required for quick-look analysis. The tape storage is sized to accommodate all science and housekeeping data. Something similar to a PDP 11/70 or an Eclipse 200 should meet the computer requirements.

EQUIPMENT	FUNCTION	AMPS CAPABILITY REQUIRED
FRONT END PROCESSOR	BIT SYNC, DECOMMUTATE, POSITION & TIME CORRELATE, ROUTE TO DATA STORAGE	1) 2.6 MBPS
DATA STORAGE	HOLD DATA FOR ACCESS BY COMPUTER SYSTEM AND REMOTES	1) QUICK ACCESS(DISK) ~ 20 MBYTES 2) TAPE STORAGE ~ 80,000 MBYTES
COMPUTER SYSTEM	ACCESS DATA FROM DATA STORAGE & MCC, DEVELOP DISPLAYS, GENERATE COMMANDS, PERFORM SCIENTIFIC CALCULATIONS, INTERRUPT/PRIORITIZE	1) INTERROGATED BY ~ 10 PERIPHERALS
CONSOLES	REQUEST AND DISPLAY DATA, ASSEMBLE AND TRANSFER COMMANDS	1) GRAPHICS 2) REFRESHING 3) PARTIAL UPDATE 4) SYMBOL GENERATOR

Figure 6-10. AMPS Operations and Control Center Equipment Requirements

The study was performed only to the depth that would develop a general understanding of the equipment needed to perform the functions. Actual design of a POCC and sizing of the components requires an in-depth analysis of the nature of the data and its flow rates.

6.3 FLIGHT CREW OPERATIONS

The objective of this study has been to make a preliminary evaluation of the size of the AMPS payload specialist team, their qualifications, and the plan and equipment necessary to train them.

6.3.1 Crew Size

The number of payload specialists necessary to support experiment operations is dependent on three major factors:

- 1) Actual time available to support experiments
- 2) Number of simultaneous experiments
- 3) Number of crewmen required for each experiment.

A baseline payload specialist day has been developed. This model is a combination of the results of Concept Verification Test (CVT) simulations at MSFC and life sciences lab work at JSC. The AMPS baseline days allow 12 hours of experiment operation per day, as shown in Figure 6-11.

Simultaneous operation of the AMPS experiments is severely constrained due to:

- Unique Orbiter altitudes
- Common use of equipment
- Sequence and timing conflicts.

The flight timelines, shown in Section 5, show that each experiment is conducted independently. The exception is the environmental measurements experiment which is conducted in conjunction with the accelerator experiments. This is possible because of the common use of the RMS and ESP and the long periods of set-up time between accelerator firings.

ACTIVITY	HOURS:MINUTES
POST SLEEP ACTIVITIES	:30
EAT PERIODS (0:30 EACH)	1:30
PLANNING/REVIEW OF NEXT DAY'S ACTIVITIES	:45
DAILY STATUS	:15
PRESLEEP ACTIVITIES	1:00
SLEEP	8:00
TOTAL	12:00
TOTAL AVAILABLE EXPERIMENTATION TIME	12:00

Figure 6-11. AMPS Baseline Payload Specialist Day Allows 12-Hours of Experiment Operation

The experiment task analysis results indicate that one payload specialist per 16-hour shift is sufficient to support scientific activities. There are short periods of time, when two payload specialists are required at the same time. These times are accommodated by use of the mission specialist/pilot and by providing 8 hours per day overlap in the payload specialist shifts.

The results of the analysis show that the commander, pilot, mission specialist and two payload specialists are sufficient to perform the AMPS baseline missions.

6.3.2 Payload Specialist Qualifications

The payload specialist is responsible for the attainment of experiment objectives. In this role he is responsible for the management of the payload operations, proficient in experiment performance, and knowledgeable in the operation of the caution and warning systems, hatches, tunnel, and life support systems. In addition to these general characteristics, the payload specialists for each AMPS flight require specific scientific training and experience. For AMPS flight 1, each payload specialist should

be an experimental physicist, one with a background in accelerators and the other in atmospheric physics. On flight 2, the accelerator specialist should be replaced by an experimental ionospheric physicist.

6.3.3 AMPS Payload Specialist Training Analysis

In order to define cost effective approaches to AMPS flight crew training, it was first necessary to define the training requirements. A systems analysis approach was used to define these requirements. This systems approach consisted of the following steps. First, an analysis of the AMPS experiments in order to define the function and operation of the equipment. Next, an analysis was performed to identify the following:

- Manned operations and interactions with the equipment
- Types of training equipment required to develop an understanding of the operations and interactions.

Recommendations as to the types of equipment necessary to support the training of the flight crews have been developed based on these requirements, schedule and cost.

6.3.3.1 Preliminary Training Plan

An overview of AMPS payload specialist training is shown in Figure 6-12; the schedule of this activity is shown in Figure 6-13. The letter identifiers indicate the relationship between the equipment and the schedule. The following steps make up the training process:

- Certification for flight
- Procedural training on experiments
- Experiment/Spacelab interface training
- Combined experiments training
- Habitability and safety training
- Integrated operations training.

The first milestone in the schedule is the certification of the payload specialist for flight. This is accomplished by exposure to zero-g in aircraft flight, medical examination and possibly exposure to the launch and landing environments using a centrifuge. This certification is performed by NASA/JSC and could be accomplished up to 10 months before launch.

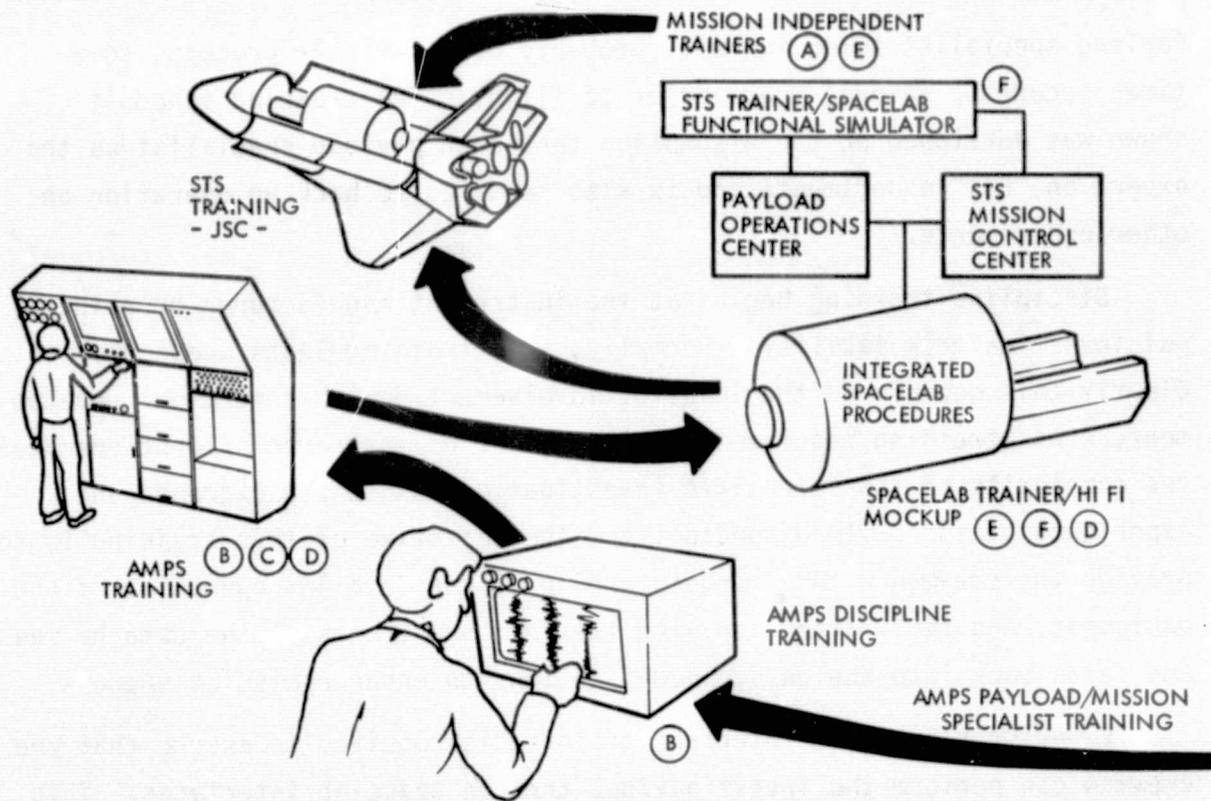
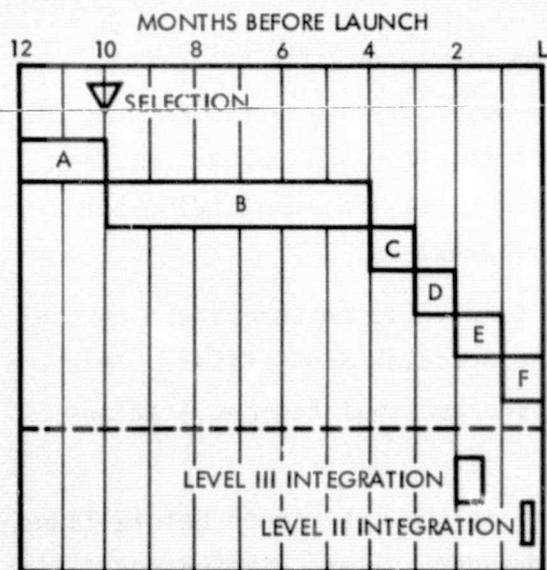


Figure 6-12. AMPS Payload Specialist Training Interfaces



* PART TIME ACTIVITY

- (A) SCREENING, ORIENTATION AND SYSTEM FAMILIARIZATION
 - WATER IMMERSION FACILITY
 - ZERO-G AIRCRAFT
- (B) PROCEDURAL TRAINING ON EXPERIMENTS
 - CLASS ROOM
 - EXPERIMENT HARDWARE
- * (C) EXPERIMENT/SPACELAB INTERFACE TRAINING
 - CLASS ROOM
 - PART TASK TRAINERS
- * (D) COMBINED EXPERIMENTS TRAINING
 - CLASS ROOM
 - ONE-G TRAINER
- * (E) HABITABILITY AND SAFETY TRAINING
 - WATER IMMERSION
 - ONE-G TRAINER
- * (F) INTEGRATED OPERATIONS TRAINING
 - ONE-G TRAINER

Figure 6-13. Payload Specialist Training Sequence

Payload specialist selection will probably be a variable process, sometimes occurring several years prior to flight. The training schedule shown was developed on the assumption that each payload specialist is the expert on "his" experiments and is also required to back up operation on other experiments.

Discipline training begins at the instrument manufacturer or principal investigator's facility. Normally, this training lasts about 6 months. Clearly this depends on the level of involvement during instrument development, cross-training requirements, the number of instruments to be operated, the complexity of the scientific investigation, and the background and experience of the payload specialist. The objective of this training is to provide the crewman with an understanding of the detailed operation of the equipment, and familiarization with the form and trends of the data he sees and feeds back into the design modifications to enhance mission success.

Experiment/Spacelab interface training is required to assure that the crewman can perform the investigations through Spacelab interfaces. This is particularly critical for CDMS operations. At this time, the payload specialist learns the relative position of the various controls, displays, and switches within the pressurized module.

Next the crewmen participates in combined experiment exercises. This training helps to build the teamwork necessary for efficient use of on-orbit experiment time. The crewmen work together following a mission timeline learning how they can support each other and identifying and possibly resolving procedural and timeline problems.

The last phases of the training are to develop an understanding of the habitability and safety systems on the Spacelab and Orbiter. This involves the preparation of food, use of the personal hygiene stations, and safety drills and procedures.

Integrated operations training occurs in the last month before launch. At this time, the entire flight crew (commander, pilot, mission specialist and payload specialist) practices the coordination of Orbiter maneuvers, RMS operations and instrument operations. This training also includes ground crews from the MCC and POCC and probably includes at least one mission simulation.

6.3.3.2 Training Equipment Requirements

The equipment required to accomplish the AMPS training plan has been selected to maximize the multiple use of AMPS or other existing equipment. A summary of the factors and applications of the selected equipment is shown in Figure 6-14. The training program can be accomplished using the following equipment:

- Instrument development breadboards
- Combined AMPS work station mockup and payload simulator
- AMPS integration facility
- Spacelab/Orbiter trainer.

TRAINING EQUIPMENT	TRAINING APPLICATIONS	NEGATIVE FACTORS
INSTRUMENT DEVELOPMENT BREADBOARDS AND SIMULATIONS	<ul style="list-style-type: none"> ● INSTRUMENT PROCEDURES ● DATA TRENDS 	<ul style="list-style-type: none"> ● NOT "FLIGHT LIKE" ● LIMITED AVAILABILITY
WORK STATION MOCKUP + PAYLOAD SIMULATOR	<ul style="list-style-type: none"> ● EXPERIMENT PROCEDURES ● EXPERIMENT/SPACELAB INTERFACES ● INTEGRATED EXPERIMENT OPERATIONS ● SPACELAB FAMILIARIZATION 	<ul style="list-style-type: none"> ● LIMITED AVAILABILITY PRIOR TO LEVEL IV INTEGRATION
AMPS INTEGRATION FACILITY	<ul style="list-style-type: none"> ● PROCEDURES REFRESHER ● MISSION SIMULATION 	<ul style="list-style-type: none"> ● LIMITED INSTRUMENT OPERATION ● LIMITED AVAILABILITY ● INTERFERENCE WITH TESTING
SPACELAB/ORBITER TRAINER (JSC)	<ul style="list-style-type: none"> ● CDMS PROCEDURES ● SPACELAB FAMILIARIZATION ● INTEGRATED FLIGHT OPERATIONS 	<ul style="list-style-type: none"> ● LIMITED AVAILABILITY ● NO EXPERIMENT SIMULATION

Figure 6-14. AMPS Training Equipment Has Been Selected To Maximize Multiple Use of Hardware

This equipment should prove very satisfactory for training, however its use is very sensitive to flight rate. In each case, the equipment is also used for other functions which would compete with training for usage. This scenario is based on a maximum of two flights per year. Other flight rates and mission models require a reassessment.

7. INTEGRATION AND TEST

This section discusses AMPS payload integration, test, ground support and operations requirements.

7.1 TEST REQUIREMENTS

The following paragraphs describe the AMPS verification test program developed during the study, including principal verification techniques.

7.1.1 General

The AMPS system verification test program consists of the development, qualification, acceptance, payload integration, and ground operations test phases. In this program, it is assumed that flight certified scientific instruments have been received and only verification of the development and manufacture of contractor-built flight support equipment is required. System level tests are performed during the payload integration and ground operations test phases.

The AMPS verification program addresses three distinct levels of hardware buildup:

- 1) Flight support equipment (FSE)
- 2) Functional assemblies
- 3) Integrated pallet, rack, and payload systems.

A functional assembly is defined as an instrument integrated into a flight support equipment element, e.g., a Fabry Perot instrument installed within a SIPS canister. The AMPS verification program uses the protoflight concept, i.e., the AMPS flight hardware also serves as a prototype. Therefore, this hardware is subjected to environmental tests and analyses using both qualification and acceptance criteria.

Verification of the quality of workmanship of all FSE is demonstrated by environmental testing during the development test program. These tests also assure that the FSE is qualified and accepted to the operational environmental levels. Functional assemblies are qualified and accepted, in some cases by testing and in others by analysis. The system level operational environmental verification is accomplished by analysis using the lower level hardware environmental test results.

During the study, a verification plan was developed. It was structured around the three hardware categories (FSE, functional assemblies and systems). The manufacturing and quality control programs support the test program. Documentation, control, and scheduling are tailored for fabrication and quality assurance activities. These, however, are also adequate for use in the verification activities.

Verification at the flight support equipment level is detailed and comprehensive to assure successful verification at the higher and more critical functional assembly or system levels. Flight support equipment tests are to standards established for AMPS. These standards are established taking into consideration the reusable capability of the hardware. The flight support equipment testing includes a series of environmental and functional tests. Early development tests on breadboards are conducted if such tests are determined to be critical in providing confidence in design and determining design margins. Qualification or qualification-acceptance tests are conducted to demonstrate that the flight support equipment can survive and/or operate in all the expected environments of handling, shipping, storage, launch, and orbital operations. Where instrument performance influences flight support equipment tests, mockups or nonflight versions are mounted. Acceptance tests on the flight hardware demonstrate that the design is the same as the qualified design and that workmanship was performed to the required standards.

Functional assembly verification is performed primarily by analysis. Testing is performed only if required. The need for testing is determined on a case-by-case basis. In those cases where tests are performed, the tests are designed to demonstrate that the assemblies function properly before, during, and after exposure to all the expected environments. These tests are supported by analysis and by data which were obtained at the lower flight support equipment level.

System verification begins after the flight support equipment and functional assemblies are integrated into the Spacelab hardware elements. The system verification program is, in essence, an in-process qualification and acceptance program. Environmental verification is performed by analysis.

Several criteria were used in the development of the system verification program. The most important of these are:

- 1) The program should permit end-to-end system verification
- 2) The operational sequences expected in flight should be simulated
- 3) Baseline data for evaluating functional performance should be included
- 4) Scheduling and sequencing of tasks should be planned to permit fault isolation and correction
- 5) The integration of the AMPS hardware should be accompanied by suitable and sufficient test data to verify satisfactory performance at each critical step prior to continuation.

7.1.2 Principal Verification Techniques

During the study, a verification program was developed which was designed to achieve maximum flight readiness. The program includes active verification planning and control, which is exercised through a continuous review and evaluation process. The program includes close coordination with the customer so that verification status, participation, and support by the customer can be scheduled.

Principal Investigators are involved throughout the complete verification program to support the instrument and flight support equipment integration and test. The documentation used in the program assures timely planning, control, and verification status visibility.

The basic approach to assuring mission success is to perform extensive verifications, primarily by analysis for the piece parts and materials that are described in the reliability program plan and by test at the flight support equipment level. At this level, a full series of environmental exposures is conducted, as necessary, to assure flight performance. At the system level, the verification test program could require that built-up pallet assemblies be subjected to an acoustic test in order to verify workmanship performance.

A detailed check of the functional performance, before each environmental test, provides the baseline data for comparison with a similar

post-test functional check. Operational monitoring during environmental exposure is designed to reveal problems which might not be evident until exposure to the actual environment simulated during the test.

The verification program reduces testing cost with a minimum risk to accomplishing the experiment. Program economies are realized by reducing environmental facility expenditures and test crews, and shortening schedules.

The risk in this approach is that difficulties can appear during the higher level of testing which could affect schedule performance. The potential on-orbit instrument and flight support equipment risk is reduced, however, by the testing accomplished at the lower levels, which consists of both functional and rigorous environment testing. In this way, the problem with the individual flight support equipment or functional assembly is discovered early enough to be resolved without impact to the progress of the other items. Each item is fully characterized so that its interface is well understood before installation at the higher system level. While the rationale is to reduce the number of functional assembly tests, there are special tests that have to be performed at the functional assembly level.

7.2 PAYLOAD INTEGRATION

Payload integration, as defined within the context of this report, consists of instrument certification, pre-Level IV, and Level IV activities. This definition is depicted in Figure 7-1. The instrument certification is primarily a government responsibility and, during that activity, it is verified that the instruments are ready for flight. The pre-Level IV activities consist of the preparation of the instruments, racks, and pallets for integration. The Level IV activity is the integration of instruments, flight support equipment and/or functional assemblies onto racks and pallets.

A functional level flow of the AMPS payload integration operation is shown in Figures 7-2 and 7-3. The pre-Level IV activities begin when the certified instruments, FSE, and the staged Spacelab elements are received at the Payload Integration Center (PIC) and are moved to the assembly area. The operation includes all of the functions performed and identified in Figures 7-2 and 7-3 and continue until instruments, cables, and FSE are actually installed on pallets and racks.

In order to determine the requirements for the AMPS integration and ground operations, a flow similar to that shown in Figures 7-2 and 7-3 was analyzed. Each function was expanded in detail to activity levels and procedural levels so that the functions, activities performed, and procedures required could be examined. This approach to the flow analysis is shown in Figure 7-4. The kinds of requirements derived from the flow analysis are also shown in that figure. Another product of the flow analysis, also shown in the figure, is a definition of the documentation required to integrate the system; e.g., system level, subsystem level, and interface checkout procedures.

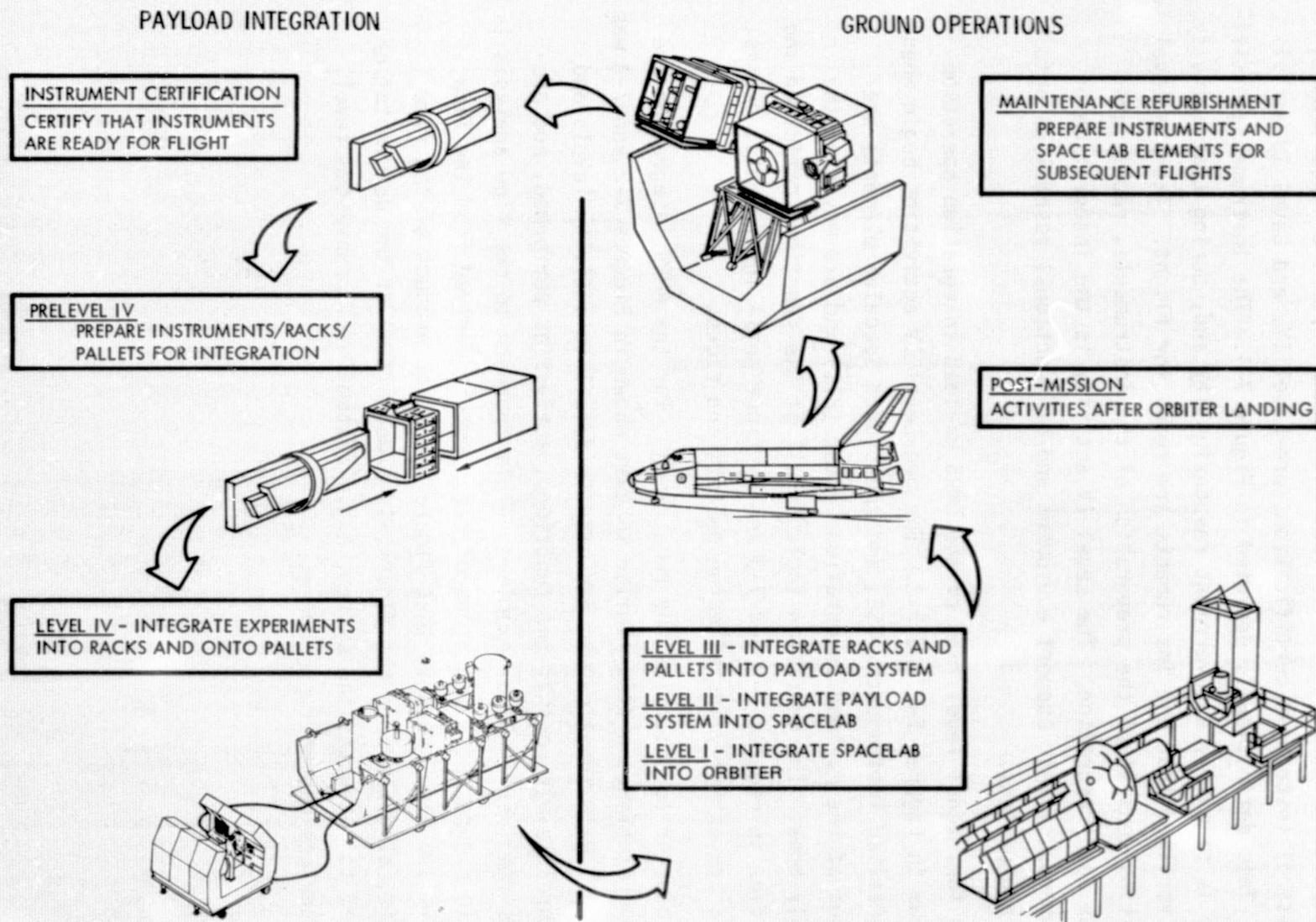


Figure 7-1. Payload Integration/Ground Operations Definitions

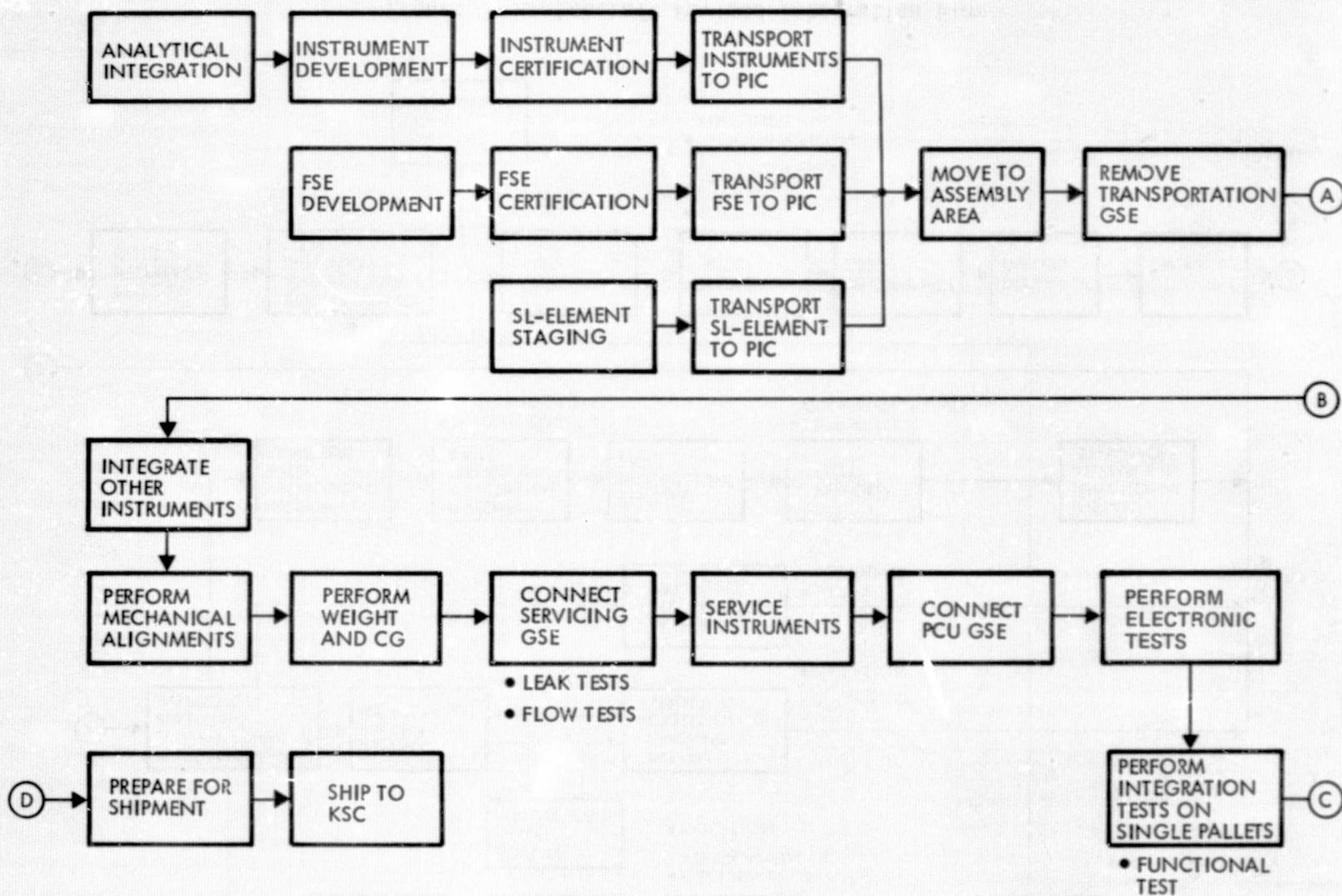


Figure 7-2. Labcraft Payload Integration Flow

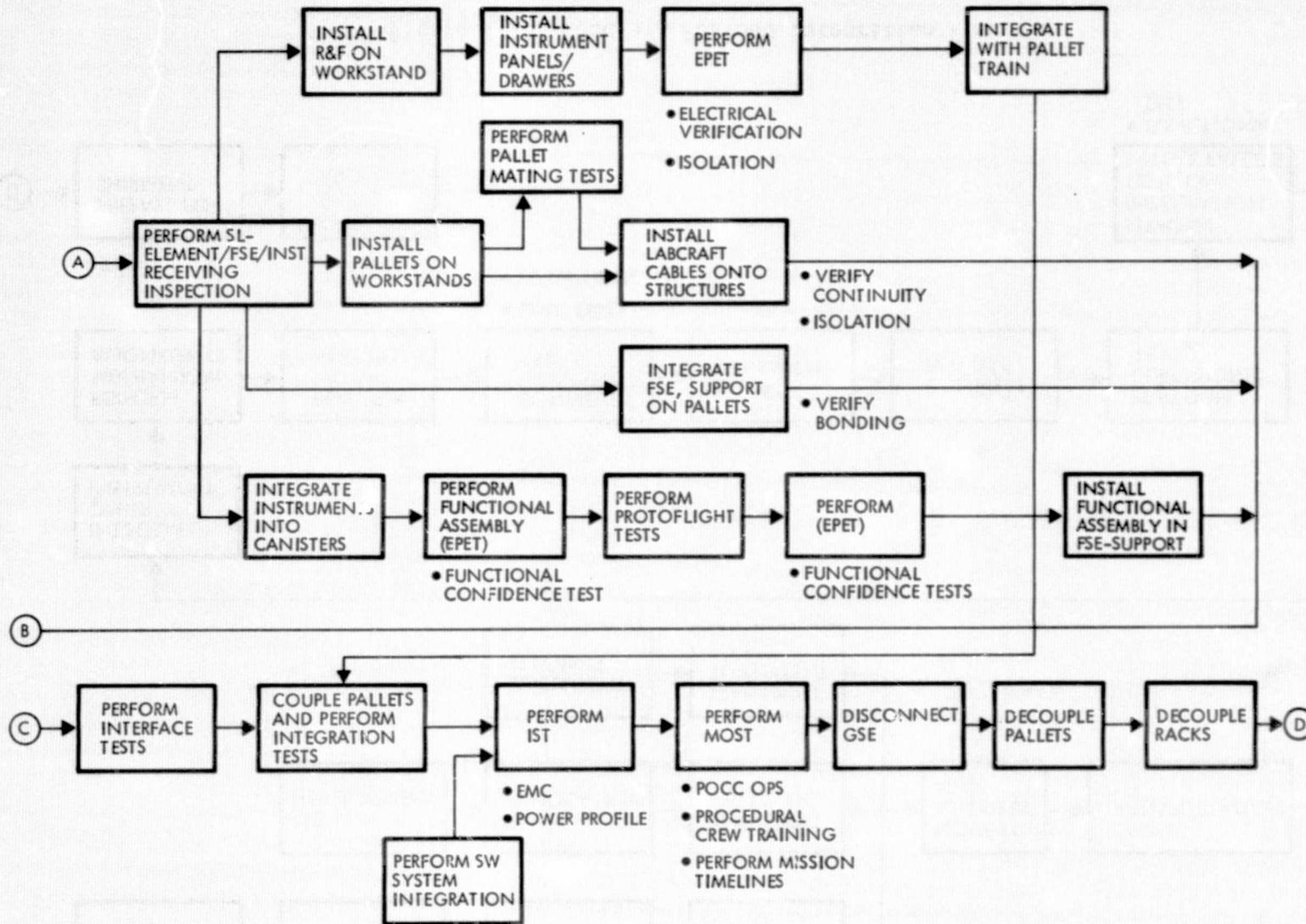


Figure 7-3. Labcraft Payload Integration Flow

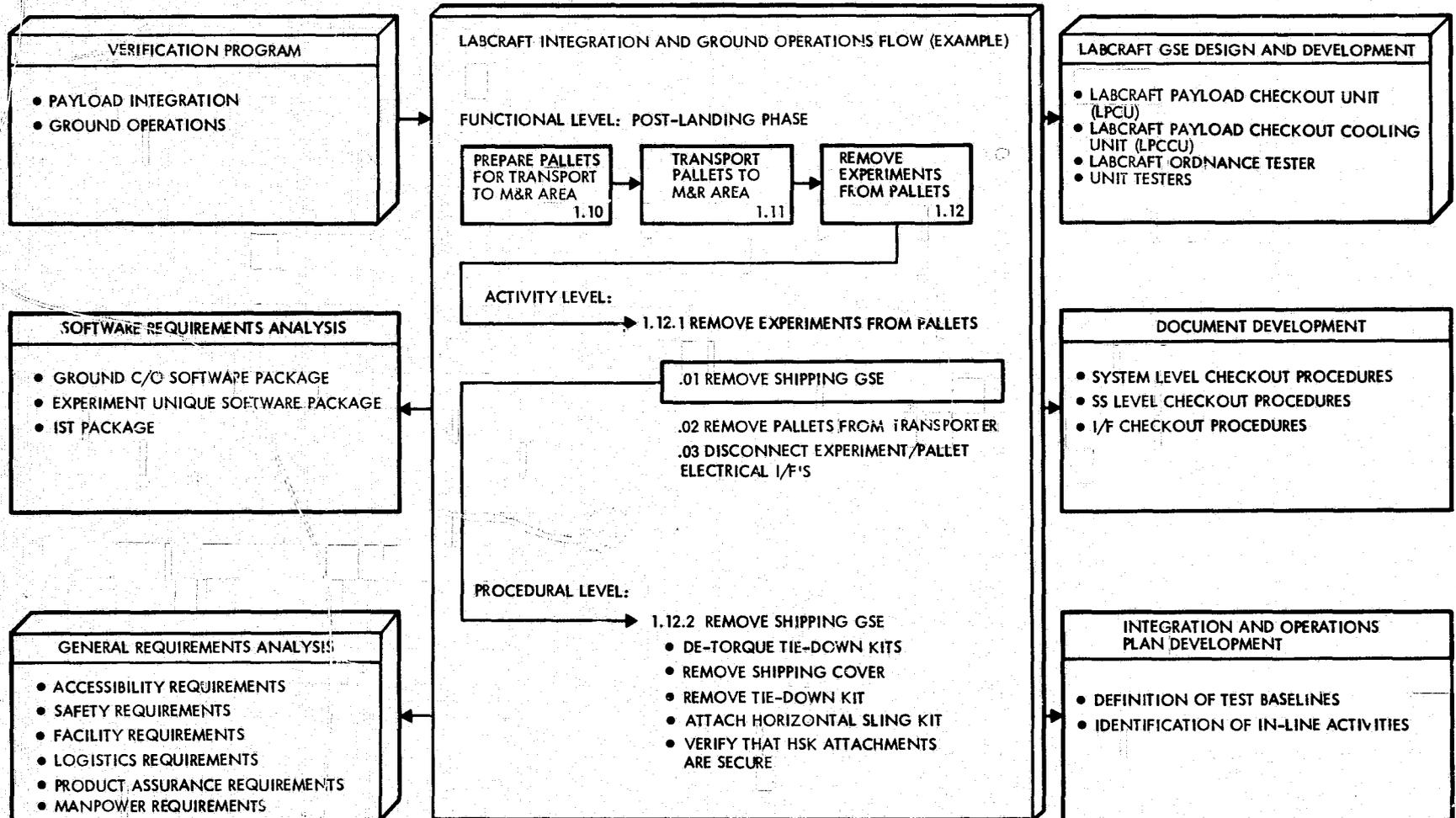


Figure 7-4. Payload Integration/Ground Operations Flow Analysis Approach

7.3 GROUND OPERATIONS

Ground operations, in the context of this report, addresses all of the functions which are performed upon the receipt of the payload at KSC. These include the Levels-III, -II, and -I integration; post-mission activities and maintenance and refurbishment of Spacelab elements for subsequent flights (see Figure 7-1). During these activities, support to KSC is provided by the payload lead center(s) and its payload integration contractors.

Ground operations support primarily consists of providing technical assistance during the activities conducted at the Operations and Checkout Building after receipt of the payload. This includes performing post-shipment tests and instrument checkout and calibration. Support continues throughout the activities performed at the Orbiter Processing Facility, the Vertical Assembly Building, and the pad, and includes the tasks performed during post-landing, maintenance, and refurbishment (Figure 7-5).

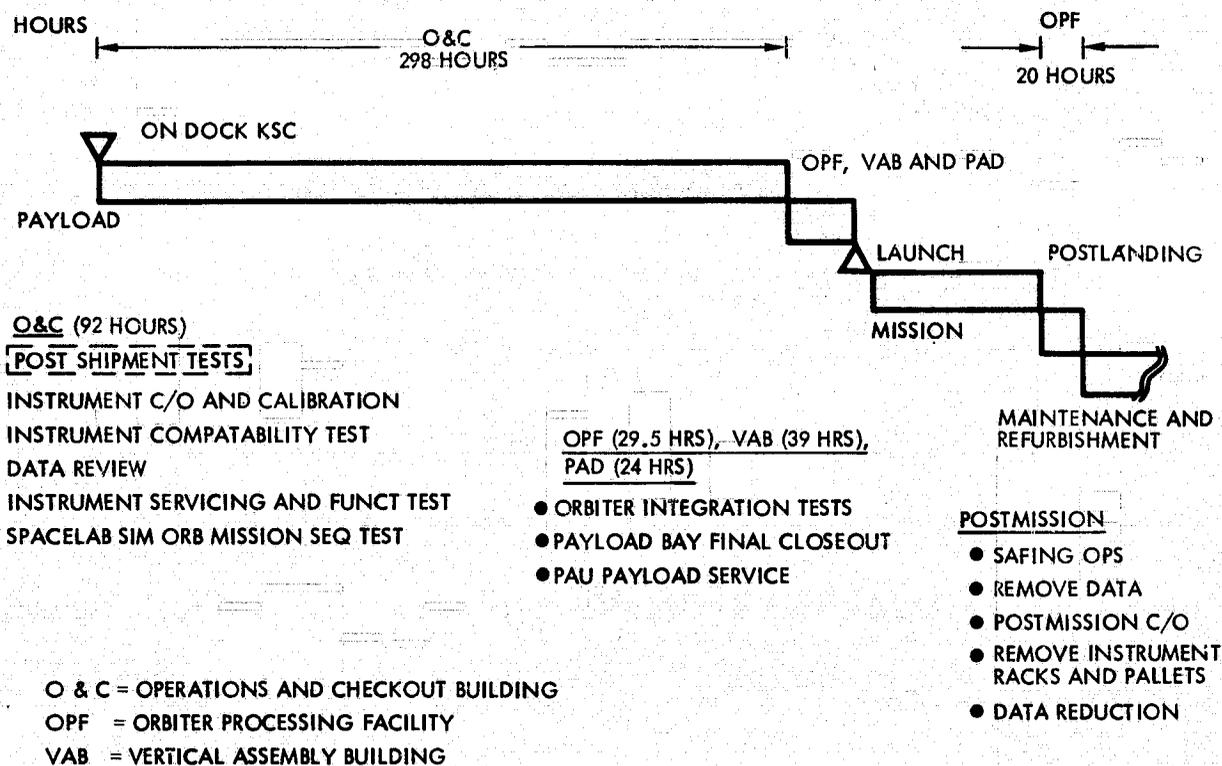


Figure 7-5. Ground Operations Support

Typical ground operations support activities include:

- 1) Preparing test summary sheets
- 2) Supporting KSC in test readiness and status meetings and providing overall payload checkout coordination
- 3) Performing data analysis and reduction
- 4) Participating in payload scheduling and coordinating test-day activities
- 5) Providing inputs to the payload checkout procedures and STS integrated procedures
- 6) Assisting laboratory personnel in checkout operations
- 7) Participating in payload systems and integrated tests
- 8) Directing and performing troubleshooting, including disposition of failed items and recommended design fixes
- 9) Conducting pre-test and post-test briefings for each test activity.

7.4 GROUND SUPPORT EQUIPMENT (GSE)

The GSE to be developed for AMPS payload integration during pre-Level IV and Level IV at the Payload Integration Center (PIC) were identified during the study. It was assumed that the Spacelab Program would provide the GSE for Levels-III, -II and -I integration, as well as the transportation equipment for the Spacelab elements during Level IV. The remaining required GSE would then be developed by the Level IV payload integration contractor.

The mechanical ground support equipment (MGSE), which should be developed, includes hardware for flight support equipment (FSE) development testing, handling, transportation, servicing, and storage. Specific identification of the MGSE is listed with the key requirements in Table 7-1.

The key requirements for the EGSE are shown in Table 7-2. Conceptual designs have been prepared during the Phase B AMPS Study for each of these hardware items in order to demonstrate that all requirements can be met and to provide a firm basis for defining costs. The designs are within the state of the art, can be fabricated using proven techniques, and require a minimum of test and verification.

Table 7-1. Key MGSE Requirements

Hardware	Requirements/Functions
Pallet/Rack Hard Mockup	<ul style="list-style-type: none"> ● Available on a schedule that supports the fabrication, test and operational checkout schedule of the flight support equipment ● Designs that allow for a high-fidelity reproduction of the mechanical features of the pallets and racks ● Flexibility to accommodate various instrument configurations
Handling Dolly	<ul style="list-style-type: none"> ● Provide in-plant mobility for a single pallet segment and a rack and floor structure ● A rail design capable of interfacing with the rails of the checkout stands
Pallet/Rack Checkout Stands (5-pallet stand)	<ul style="list-style-type: none"> ● Provide a dedicated working area for pallet and rack integration ● Provide accessibility to the pallets and racks ● Design which incorporates the Spacelab rail "roll in/out" concept
Optical Alignment Kit	<ul style="list-style-type: none"> ● Provide the capability to perform assembly stand, instrument, pallet mating operation alignment
Thermal Blanket Handling Kit	<ul style="list-style-type: none"> ● Provide the capability to support and protect the thermal blankets associated with the instruments and pallets during handling and hoisting
Rack and Floor Braces Kit	<ul style="list-style-type: none"> ● Provide the required structural support to the racks while on the floor segments during assembly and integration
Labcraft Payload Checkout Cooling Unit	<ul style="list-style-type: none"> ● Provide cooled, dehumidified air to the rack mounted experiments ● Perform the functions of draining, purging, evacuating, and circulating freon through the pallet cold plates

Table 7-1. Key MGSE Requirements (Continued)

Hardware	Requirements/Functions
Vertical Sling Kit	<ul style="list-style-type: none"> ● Provide the capability to hoist and rotate the pallets and racks from a vertical to a horizontal position
Pallet Segment Floor Covers 15/Kit	<ul style="list-style-type: none"> ● Provide protection to the pallet floor during integration operations
Rack and Floor Installation and Removal Kit	<ul style="list-style-type: none"> ● Provide support for any rack and floor assembly combination when on the assembly stand
Pallet Segment Support	<ul style="list-style-type: none"> ● Support the pallets empty or full during all integration and transportation activities
Pallet Mate/ Demate Kit	<ul style="list-style-type: none"> ● Provide the capability to deflect the top of a pallet segment in order to facilitate the alignment of the pallet attach fittings
X-Ray Unit	<ul style="list-style-type: none"> ● Perform the functions of determining internal malfunction, stress or corrosion in small connectors or small metallic components

Table 7-2. Key EGSE Requirements

Hardware	Requirements/Functions
<p>Labcraft Payload Checkout Unit (LPCU)</p>	<ul style="list-style-type: none"> ● Provide, monitor distribute and control the power required for the individual or integrated experiment testing ● Provide control, display and real time monitoring of sequenced events ● Process and display in "real time" emergency caution and warning signals ● Provide capability to detect out-of-tolerance conditions and to perform fault isolation to the experiment level ● Provide recording capability for wideband analog and digital experiment data ● Provide off-line data processing capabilities for processing recorded data (except wideband) ● Provide and record data ● Provide automatic control of the experiment coolant and heating and monitor in real time ● Provide timing signals (GMT and MET) and sync signals for experiment events ● Provide interface capability for experiment-unique GSE ● Provide interface capability for the Payload Operations Control Center
<p>Ordnance Tester</p>	<ul style="list-style-type: none"> ● Utilized for Level IV integration, and in support of KSC operations ● Verify ordnance firing commands and circuitry ● Verify "no voltage" status of ordnance circuits prior to the installation of live ordnances
<p>Aft End Cone Adapter</p>	<ul style="list-style-type: none"> ● Provide the aft end cone to pallet electrical and fluid physical interface bulkhead

Table 7-2. Key EGSE Requirements (Continued)

Hardware	Requirements/Functions
Electrical Harness	<ul style="list-style-type: none">● Provide electrical harnesses from the PCU to the pallet elements and/or pallet train
Continuity Tester	<ul style="list-style-type: none">● Perform continuity tests on Labcraft electrical cables
Grounding/Bonding Tester	<ul style="list-style-type: none">● Verify the bonding compatibility of the equipment after installation
EMC Test Equipment	<ul style="list-style-type: none">● Verify electromagnetic compatibility between instruments

8. LOGISTICS

During the AMPS study we examined an approach to satisfying logistics requirements for the AMPS program. The integrated logistics program must be consistent with the on-going logistics support effort initiated by NASA and currently supported by ESA/ERNO and other Governmental organizations and contributors. The requirements for those operations are described in the Level II Spacelab Operations Requirements, Volume IV, Logistics Operations, SL-OP-033, MSFC. Logistics support for the AMPS Project will contribute to the success of the project:

- By early analysis of emerging designs, eliminating unnecessary support cost or potential causes of delay
- By guaranteeing the availability of the proper resources at the proper time to support both routine and contingency activities.

Logistics engineering includes activities occurring during the design and planning phases to assure a cost effective program. Principal tasks reviewed during the study included the following:

- Performance of optimum repair level analysis (ORLA)
- Performance of maintainability evaluations and establishment of maintainability design criteria for flight and ground hardware
- Evaluation of selected hardware for maintenance characteristics and establishment of operational maintenance concepts
- Establishment of overall spares control and supply plan
- Provision of on-going coordination and support associated with PIC and KSC activities including facilities activation, operations and planning
- Assistance to GSFC in planning transportation for GFE deliveries to the PIC and shipment of flight and support hardware to KSC, as well as return shipments from KSC.

8.1 OPTIMUM REPAIR LEVEL ANALYSIS (ORLA)

As part of the logistics engineering program and in accordance with the requirements of SL-OP-33, an Optimum Repair Level Analysis (ORLA) is required for Spacelab hardware. The ORLA is aimed at determining for each

equipment item: (1) whether it should be designated as "repairable" or "throwaway" and (2) if repairable, where it should be repaired. Standardized ORLA methods defined in AFLCM/AFSCM 800-4, "Optimum Repair Level Analyses—ORLA," and KSC K-SM 12.7, "Logistics Engineering Analyses Plan," is used. Since the ORLA is a costly formal process we do not recommend that it be performed on the AMPS FSE and GSE, but rather that repair level decisions be based on less formal analysis using the results of the maintainability and maintenance engineering analysis (See Sections 8.2 and 8.3.)

8.2 MAINTAINABILITY

The maintainability of an item should be considered in optimizing the AMPS system design. A maintainability program should be established for AMPS to ensure minimum support costs through inclusion of maintenance features in the AMPS flight and ground support equipment. This results in a simplified maintenance concept and a reduction in potential support problems. The maintainability program should accomplish the following tasks:

- Develop detailed system maintenance concepts and major logistics policies
- Establish maintainability design goals and criteria. Maintainability is measured in terms of mean-time-to-repair (MTTR). According to SL-OP-33 the MTTR goal for a Spacelab subsystem should be 1 hour. A similar standard should be applicable to AMPS subsystems
- Control and monitor maintainability process.

Accessibility and interchangeability are critical criteria to be considered in the equipment design to assure that the maintenance concept of remove-and-replace is achievable. The design concept for all equipment should allow maximum access to units or components through all the integration levels. The use of off-the-shelf hardware is a particular constraint on maintainability, but it can result in a more cost-effective program. Off-the-shelf hardware selection should be evaluated in terms of impact on the AMPS maintenance concept and AMPS LCC (life cycle cost).

8.3 SPARES AND SUPPLY SUPPORT

A centralized spares provisioning system should be established and implemented for the purpose of selecting, quantifying, cataloging, documenting, and furnishing spares and repair parts for the AMPS program.

Spare part quantity recommendations should be determined by utilizing maintainability and failure analysis data, total quantity used per subsystem, number of units to be supported, support period, and required turnaround time on repairable items (see Figure 8-1).

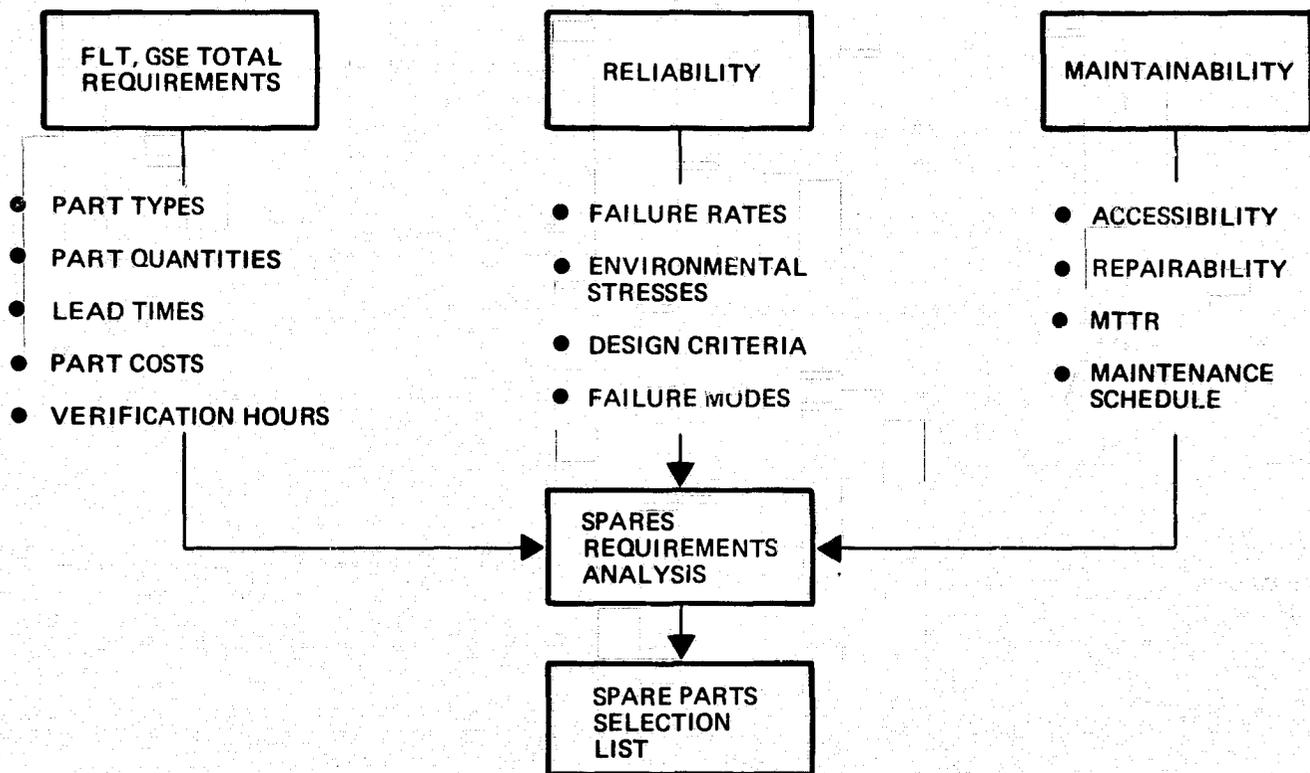


Figure 8-1. Spares Analyses

8.4 STORAGE

The primary storage of AMPS project spare parts will be at the PIC. Spares are also required at the depot maintenance facility. Storage is also required for launch support spare parts at KSC. All items received at the stores should be inventoried and issued with integration logistics stores controls.

Inventory control at PIC and KSC integration stores should conform to procedures utilizing the integration planning and logistics hardware control system. Controls for stores should follow standard procedures using bonded stores. The travel history of all spares flow should be recorded on a routing and operations traveller. A receipt and withdrawal record will establish a permanent store record of the hardware.

A controlled access list should be maintained in the stores showing project personnel authorized to withdraw hardware for preplanned use. Use should be specified on the routing and operations traveller for the unit by the person withdrawing the spare part.

8.5 HANDLING AND TRANSPORTATION

Packaging and handling, as part of the logistics support plan, should ensure maintenance of the reliability of program hardware consistent with the environmental and cleanliness requirements. The packaging function must ensure that adequate protection is provided for the following:

- Raw materials, supplies, components, and assemblies purchased from outside vendors
- Program hardware during in-plant manufacture, assembly, test, and storage
- Items shipped from suppliers, the PIC or KSC
- Experiment hardware during shipment to and from suppliers, PIC and KSC.

Packaging of electro-explosive devices should conform to the packaging and marking requirements of AFM 71-4, or T. C. George's Tariff 27, or CAB 82 regulations, as applicable to the mode of transportation. Transportation of radioactive materials will conform to the Department of Transportation document 49 CFR Part 100-199, "Transportation for Handling of Radioactive Sources."

Transportation of Spacelab and AMPS elements to and from the PIC will be the responsibility of the AMPS project. Transportation and handling equipment for Spacelab elements will be supplied by Spacelab.

In general, the transportation plan should ensure safe movement and timely delivery of the hardware by the most economical mode of transportation consistent with the fragility of the item and the program schedule. This will be accomplished in compliance with NHB 6000.1B "Packaging, Handling and Transportation Requirements."

Loaded pallets and racks should be transported from the PIC to KSC by air using a C-5A. Shipment by air transport is recommended because of the following advantages:

- a) It requires a minimum schedule time and provides maximum schedule flexibility
- b) It exposes the equipment to a more benign environment for a shorter period of time than road transport
- c) It exposes the equipment to fewer hazards and provides maximum physical security.

8.6 FACILITIES

Two major facilities will be required to support the integration operations of AMPS. These are the Payload Integration Center (PIC) and the Operation and Checkout (O&C) facilities. The support required for the facilities should consider the space and personnel, as well as the environment required.

8.6.1 The Payload Integration Center

The payload integration and test center (PIC) is the principal integration center for AMPS instruments and flight support equipment. Facilities must be available at the PIC to permit the following functions to be performed:

- a) Certification of functional assemblies
- b) Pre-level IV integration
- c) Level IV integration
- d) Maintenance and refurbishment on selected instruments.

Figure 8-2 shows a concept of the AMPS PIC facility at which all the functions described above can be performed. Ample room is provided for experimenter laboratories, general offices and storage areas.

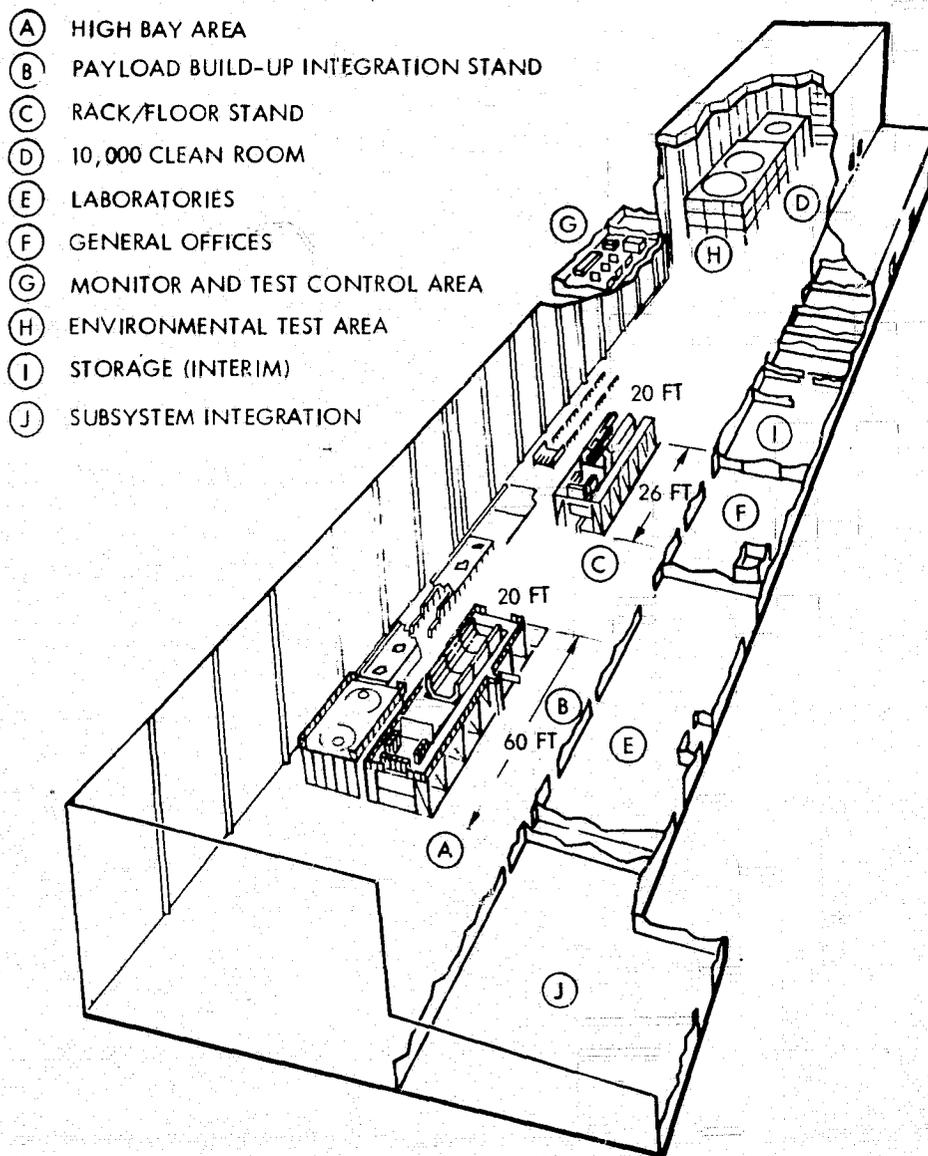


Figure 8-2. AMPS PIC Facility

8.6.2 Facilities at the Launch Site

Facilities will be required at the launch site to perform the following AMPS off-line functions:

- a) Post shipment tests
- b) Interface tests
- c) Integration and checkout of late arrival instruments

A preliminary evaluation of presently available facilities shows that these functions can readily be performed at KSC using the KSC Operations and Checkout (O&C) facility (Figure 8-3). Sufficient area (approximately 70 ft x 30 ft) would be required in that facility to accommodate the AMPS payload build-up integration stand and the payload checkout unit (PCU).

Coordination between other contractors and agencies scheduled to use this facility is required to ensure a noninterference operation and smooth transition between operations.

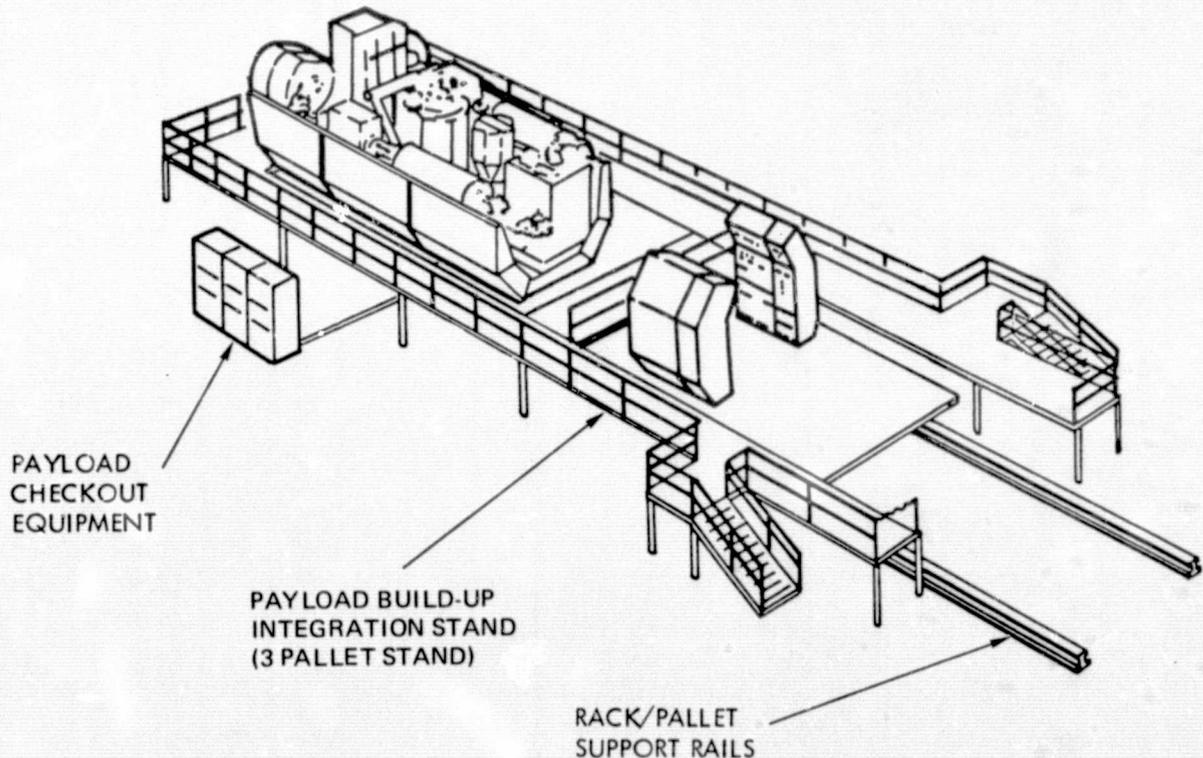


Figure 8-3. Off-Line KSC Facility (O&C)

9. DESIGN ANALYSIS AND MASS PROPERTIES

9.1 MASS PROPERTIES

9.1.1 Payload Weight

The AMPS payload weight is constrained by two overlapping sets of requirements, one imposed by the Spacelab Payload Accommodation Handbook, SLP 2104, and the other by the Space Shuttle System Payload Accommodations, JSC 07700, Volume XIV. The requirements specified in the latter document cover the entire Orbiter payload including the Spacelab module, pallets, and mission independent equipment. The former document specifies the allowable mass of the payload in the module, on the pallets, and on the aft flight deck. These requirements are summarized in Table 9-1 for a short module-three pallet configuration.

Table 9-1. Payload Mass Requirements

Source of Mass Limit	Mass Limit (kg)		Included in Payload Allowance
	Launch	Landing	
JSC-07700 Vol. XIV Rev. D	29,484	14,515	<ol style="list-style-type: none"> 1) All Spacelab equipment in the cargo bay and on the Orbiter aft flight deck. 2) All payload unique hardware in the cargo bay and on the Orbiter aft flight deck. 3) Any of the following optional items: <ol style="list-style-type: none"> a) Docking module b) Tunnel adapter kit c) ATCS radiator panel kit (provided as part of Spacelab) d) Second manipulator arm e) Crew in excess of four <ol style="list-style-type: none"> (1) Crew mass (2) Expendables (3) Seat and restraints f) OMS ΔV kit (above 1,000 ft/sec) g) EPS energy kit (above 890 kWh) h) RCS propellant (above baseline) i) Personal rescue enclosure j) EVA gear (above two men) k) MSS PCM recorder kit l) Additional payload attach fitting (above baseline) m) Airlock repressurization 4) All payload cabling

Table 9-1. Payload Mass Requirements (Continued)

Source of Mass Limit	Mass Limit (kg)		Included in Payload Allowance
	Launch	Landing	
NASA/ESA SLP-2104 PDR-B 1976		6222	<ol style="list-style-type: none"> 1) All payload unique hardware in the cargo bay and on the Orbiter aft flight deck. 2) Any optional equipment noted in paragraph 3. above except item (c) 3) All payload cabling <ol style="list-style-type: none"> a) From payload to Spacelab b) Between payload elements c) From payload to service panels 4) Any Spacelab mission dependent equipment such as: <ol style="list-style-type: none"> a) Airlock b) High quality window c) Film vault d) High rate multiplexer e) High data rate recorder f) Equipment racks g) High vacuum venting facility h) Experiment computer i) experiment I/O unit j) Stowage containers k) Experiment inverter l) Pallet thermal cover m) Data display unit and symbol generator n) Subsystem units (above baseline) <ol style="list-style-type: none"> (1) RAU (2) Coldplate (3) Switching panel (4) Intercom station o) Program reserve, not released

The AMPS flights 1 and 2 payloads are for 7-day missions, using one additional payload specialist and requiring no additional energy kits, RCS propellants, or OMS propellants. The mass properties estimates for Flight 1 is given in Table 9-2. The payload margins are adequate for this early phase of the AMPS program. In Table 9-3 the masses of the mission dependent equipment for Flights 1 and 2 are itemized.

In Table 9-4, the Flight 2 mass properties are estimated. This payload also is well within allowable margins.

Table 9-2. AMPS Flight 1 Payload Mass Summary

	Launch Mass (kg)	Landing Mass (JSC-07700) (kg)	Landing Mass (SLP-2104) (kg)
<u>Pallet Segment No. 1</u>			
ESP Subsystem Structure	18	18	18
ESP Cradle Assy.	11	11	11
ESP	171	171	171
Accelerator System	328	324	324
Common Energy Storage Subs.	210	210	210
Cables, J-boxes & Attachment Hrdwr.	<u>45</u>	<u>45</u>	<u>45</u>
Pallet Seg. No. 1 Summary	783	779	779
<u>Pallet Segment No. 2</u>			
SIPS Subsystem	527	527	527
CRYO Canister No. 1	72	72	72
Roll Gimbal Assy.	46	46	46
Instrument Support Spar	55	55	55
Cryo Cooled Common Optics & Cryo System	375	286	286
Cryo Cooled Limb Scanner Spectrometer	20	20	20
Std. Canister No. 2	119	119	119
Instr. Support Structure	29	29	29
OBIPS	93	93	93
Solar Flux Monitor	50	50	50
Cables, J-boxes & Attachment Hrdwr.	40	40	40
Pallet Seg. No. 2 Summary	<u>1446</u>	<u>1357</u>	<u>1357</u>
<u>Pallet Segment No. 3</u>			
Lidar System	779	779	779
Port Side Gas Rel. Supp. Str.	20	20	20
Sphere Release Assy.	29	29	29
Sphere Assy. (3)	324	--	--
Starboard Side Gas Rel. Supp. Str.	20	20	20

Table 9-2. AMPS Flight 1 Payload Mass Summary (Continued)

	Launch Mass (kg)	Landing Mass (JSC-07700) (kg)	Landing Mass (SLP-2104) (kg)
Sphere Release Assy.	19	19	19
Sphere Assy. (2)	216	--	--
Vector Mag. Mast Adapter	7	7	7
Truss Type Deploy. Mast	9	9	9
Vector Magnetometer	2	2	2
Quartz Crystal Microbal.	7	7	7
Cables, J-boxes & Attachment Hrdwr.	<u>45</u>	<u>45</u>	<u>45</u>
Pallet Seg. No. 3 Summary	1477	937	937
<u>Module</u>			
Safing Cmd. J-Box	10	10	10
C&W J-Box	10	10	10
Nim-Camac	80	80	80
QCMB Controller	20	20	20
Lidar Controller	20	20	20
OBIPS Controller	25	25	25
SIPS Controller	30	30	30
Bulk Data Retrieval Unit	22	22	22
Analog/Video Recorder	23	23	23
Cables & Attachment Hrdwr.	<u>50</u>	<u>50</u>	<u>50</u>
Module Summary	290	290	290
<u>Orbiter AFD</u>			
Safing Cmd. J-Box	15	15	15
C&W J-Box	15	15	15
Safing D&C	30	30	30
Cables & Attachment Hrdwr.	<u>20</u>	<u>20</u>	<u>20</u>
OAFD Summary	80	80	80
Payload Specialist	210	210	210
Mission Dependent Equipment	736	736	736
Baseline Spacelab Equipment	<u>7562</u>	<u>7162</u>	<u>----</u>
Total Payload Mass	12583	11550	4388
Allowable	29484	14515	6222
Margin	16799	2965	1834

Table 9-3. Mission Dependent Spacelab Equipment

Item Description	Unit Weight	Flight 1		Flight 2	
		Quantity	Weight (kg)	Quantity	Weight (kg)
Single Rack	36.1	2	72	2	72
Double Rack	57.3	2	115	2	115
Pallet Hardpoints	1.0	38	38	33	33
Cold Plate	6.3	5	31	2	13
Cold Plate Standoff	0.9	30	27	12	11
Thermal Capacitor	12.5	0	0	0	0
Freon Line System		1	50*	1	50*
Pallet Thermal Cover	19.8	3	59	3	59
Expt Switch Panel	3.7	1	4	1	4
Power Harness		1	50*	1	50*
Expt. Computer	30.4	1	30	1	30
Expt. I/O Unit	29.5	1	30	1	30
Expt. Rau	8.9	10	89	10	89
Keyboard	4.5	3	14	3	14
CRT Display	24.25	3	73	3	73
High Rate Digital Recorder	43.00	1	43	1	43
High Rate Multiplexer	11.0	1	11	1	11
			736		697

* Estimated.

Table 9-4. AMPS Flight 2 Payload Mass Summary

	Launch Mass (kg)	Landing Mass (JSC-07700) (kg)	Landing Mass (SLP-2104) (kg)
<u>Pallet Segment No. 1</u>			
RF Sounder	218	218	218
Barium Release & Cradle	595	100	100
Test Body	20	-0-	-0-
ESP No. 1	171	171	171
ESP Subsys. Structure	18	18	18
ESP Cradle Ass'y	75	75	75
15 Meter Mast	<u>45</u>	<u>45</u>	<u>45</u>
Pallet Segment No. 1 Summary	1153	638	638
<u>Pallet Segment No. 2</u>			
SIPS Subsystem	527	527	527
Cryo Canister No. 1	72	72	72
Roll Gimbal Ass'y	46	46	46
Inst. Support Spar	55	55	55
Cryo Cooled Common Optics & Cryo System	375	286	286
Cryo Cooled Limb Scanner	20	20	20
Cryo Cooled Interferometer Spectrometer	20	20	20
Std. Canister No. 2	119	119	119
Inst. Support Structure	29	29	29
OBIPS	93	93	93
Solar Flux Monitor	50	50	50
UV-Vis Spectrometer	65	65	65
Cables, J-Boxes & Attach. Hrdwr.	<u>40</u>	<u>40</u>	<u>40</u>
Pallet Segment No. 2 Summary	1511	1422	1422
<u>Pallet Segment No. 3</u>			
Lidar System	779	779	779
Vector Magnetometer			
Mast Adapter	7	7	7
Truss Type Deploy. Mast	9	9	9
Vector Magnetometer	2	2	2

Table 9-4. AMPS Flight 2 Payload Mass Summary (Continued)

	Launch Mass (kg)	Landing Mass (JSC-07700) (kg)	Landing Mass (SLP-2104) (kg)
Quartz Crystal Micro.	7	7	7
ESP No. 2	171	-0-	-0-
ESP Subsystem Structure	18	18	18
ESP Cradle Assy	11	11	11
Cables, J-Boxes & Attach. Hrdwr.	<u>45</u>	<u>45</u>	<u>45</u>
Pallet Segment No. 3 Summary	1049	879	879
<u>Module</u>			
Safing CMD J-Box	10	10	10
C & W J-Box	10	10	10
QCMB Controller	20	20	20
Lidar Controller	20	20	20
OBIPS Controller	25	25	25
SIPS Controller	30	30	30
RF Sounder Control	20	20	20
Pulse Generator	10	10	10
Analog/Video Recorder	23	23	23
Bulk/Data Retrieval Unit	22	22	22
Cables and Attachment Hrdwr.	<u>50</u>	<u>50</u>	<u>50</u>
Module Total	240	240	240
<u>Orbiter AFD</u>			
Safing CMD. J-Box	15	15	15
C & W J-Box	15	15	15
Safing D & C	30	30	30
Cables and Attach. Hrdwr.	<u>20</u>	<u>20</u>	<u>20</u>
	80	80	80
Payload Specialist	210	210	210
Mission Dep. Equip	697	697	697
Baseline Spacelab Equipment	<u>7562</u>	<u>7162</u>	<u>-----</u>
Total Payload Mass	12502	11328	4166
Allowable	<u>29484</u>	<u>14515</u>	<u>6222</u>
Margin	16982	3187	2056

9.1.2 Payload Centers of Gravity

The centers of gravity of the Flights 1 and 2 payloads were computed and compared with the requirements of SLP/2104 (PDR-B 1976) Space Lab Payload Accommodation Handbook. It is well known that the critical cg constraint is along the longitudinal or X-axis. An important consideration during the layout of each payload is the placement of heavier equipment toward the rear of the payload bay. Both Flight 1 and Flight 2 payloads, as can be seen on Figures 9-1 and 9-2, have centers of gravity within the allowable envelopes specified in SLP/2104. The x-axis locations are very close to and within the allowable boundary. In the event of increased mass in the forward parts of the payload it may be necessary to relocate payload elements or to add more payload or ballast on the aft pallets. Note that there is ample weight margin for this.

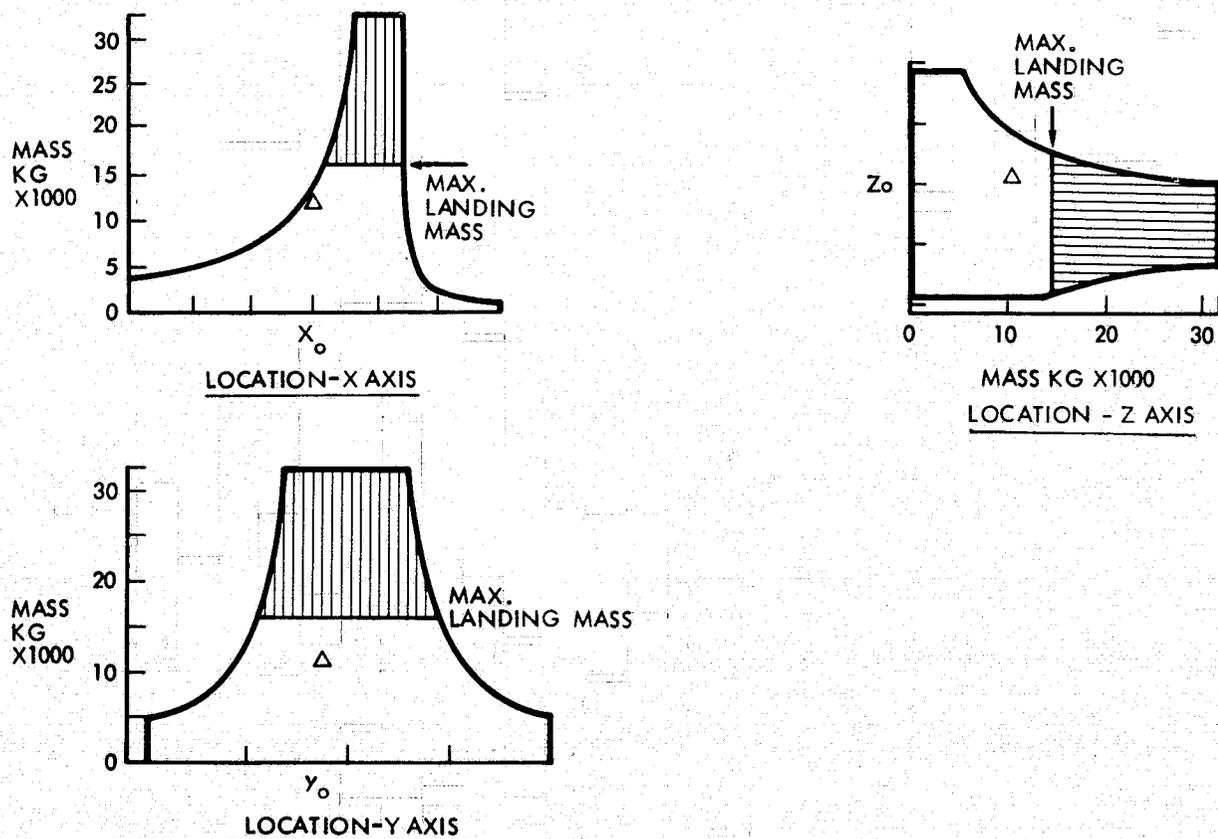


Figure 9-1. Flight 1 Payload Center of Gravity Estimates

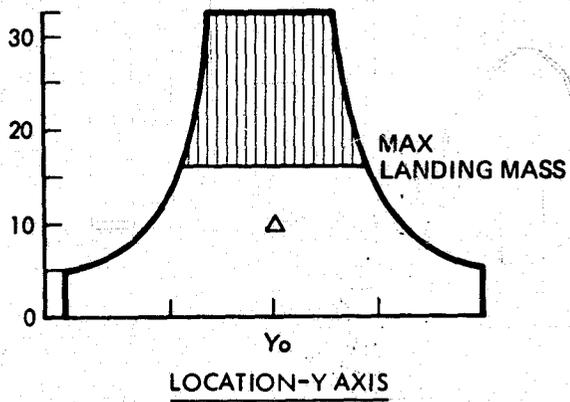
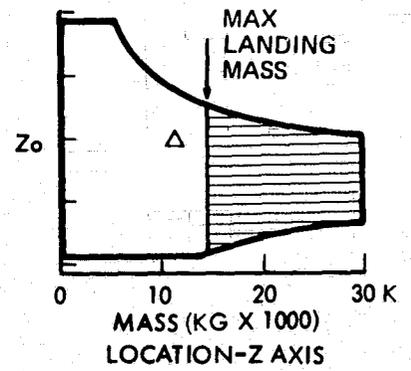
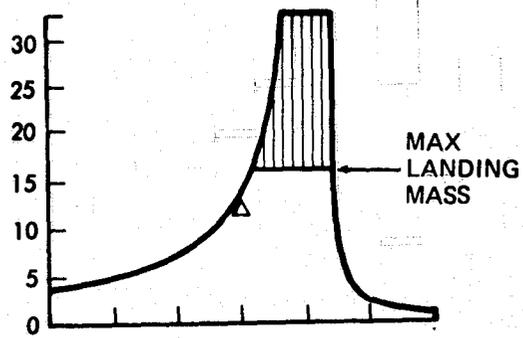


Figure 9-2. Flight 2 Payload Center of Gravity Estimates

9.2 CONTAMINATION ANALYSIS

The AMPS payloads consist of several scientific instruments and sensors which are sensitive to gaseous, molecular deposition, and particulate contamination. During the AMPS Phase B Study, the following tasks were performed (Reference 1):

- a) Determine the contamination susceptibility of the AMPS instrument complement. Many of the instruments were in the development stage during this study; therefore, only relative ratings may be obtainable at this time and quantitative data must be supplied by the principle investigation at a future date.
- b) Relate instrument susceptibility to estimates of Orbiter/Spacelab contamination environments. The basis for the contamination estimates was the computer study Reference 2 sponsored by MSFC.
- c) Define measures that should be taken to minimize the effects of Orbiter/Spacelab contamination.

9.2.1 Maximum Allowable Levels of Contamination

While the maximum allowable levels of contamination may vary with the specific AMPS instrument or sensor, a general requirement had to be established as a design parameter for the STS (Space Transportation System). NASA established a Contamination Requirements Definition Group (CRDG) to determine these levels. The group's recommendations for orbital operations are published in Reference (4) and are summarized below. These values should be used as program goals.

- a) Molecular Column Density
(along any line-of-sight within 60 degrees of +Z)
 - 1) 10^{11} molecules/cm² for H₂O + CO₂
 - 2) 10^{13} molecules/cm² for O₂ + N₂
 - 3) 10^{10} molecules/cm² for any other species
- b) Particulate Background
 - 1) Less than one discernable particle/orbit in a 1.5×10^{-5} steradian field of view along any line-of-sight within 60° of +Z.
 - 2) No discernable particles for 90% of the operational period.

Note: A discernable particle is a particle larger than 5 micrometers diameter within a range of approximately 5 kilometers.

c) Background Radiation

Total ultraviolet and visible radiation background levels from spacecraft induced and molecular scattering and emission are defined as a function of wavelength in Reference 1.

d) Molecular Deposition

The cumulative deposition after 30 days in orbit at the prime measurement point (PMP), X = 1107, Y = D, Z = 507, and perpendicular to the Z axis is shown below:

Temperature, Acceptance Angle, and Altitude	Accumulated Deposition
300 ⁰ K 2π steradians any altitude	10 ⁻⁵ gm/cm ²
300 ⁰ K 0.1 steradians 400 km	10 ⁻⁷ gm/cm ²
20 ⁰ K 0.1 steradian 400 km	10 ⁻⁵ gm/cm ²

e) Deposited Particulates

Maximum levels for particulate deposition were not established.

In a computer study for MSFC, estimated contaminant levels from the major Shuttle Orbiter sources, Reference 2, were as follows (measured at the PMP):

a) Molecular Column Density

The column density for water vapor (polar molecules) must be less than 10¹² molecules/cm².

b) Particulate Background

Less than one particle per orbit larger than 5 micrometers in the field-of-view of an instrument.

c) Background Radiation

Background brightness from scattering or emission less than 20th magnitude/sec² in the UV range.

d) Return Flux

Return flux of less than 10^{12} molecules/cm²/sec.

e) Molecular Deposition

Control to 1 percent absorption of UV, visible, and IR radiation by condensibles on optical surfaces.

f) Deposited Particulates

Maximum levels for particulate deposition were not established.

A comparison of the two sets of values shows the latter set to be somewhat less stringent than those established by CRDG although both sets represent extremely tight cleanliness requirements. It should be noted that the return flux level of 10^{12} molecules/cm²/sec is not compatible with the CRDG requirement of a molecular deposition of 10^{-5} or 10^{-7} gm/cm² after 30 day exposure unless an extremely small sticking coefficient is assumed. For instance, if we assume a molecular weight of 20 and specific gravity of 1.0 for the deposited molecules and a sticking coefficient of 1.0, then the molecular deposition after 30 days is approximately 7.3×10^{-5} gm/cm² (6900Å).

9.2.2 Sources of Contamination

The sources of molecular and particulate contamination are the surface exterior to the payload bay, the interior of the payload bay, or the AMPS structure and instruments. The contamination due to external sources has been analyzed and the results published in Reference 2. The main external sources considered in the study were outgassing from thermal control paints, adhesives and other polymeric materials, offgassing from external surfaces, leakage of cabin atmosphere, exhaust from the supplemental evaporator, and exhaust from the vernier reaction control system (RCS). The results of the study for various lines-of-sight (LOS) are summarized in Table 9-5. The LOS are defined as:

LOS 1A: Parallel to +Z
LOS 2A: 50° off +Z towards +Y
LOS 3A: 25° off +Z towards ±
LOS 4A: 50° off +Z towards ±Y and 45° toward +X (aft)
LOS 5A: 50° off +Z toward -X (fwd)
LOS 6A: 50° off +Z toward +X (aft)

Note: All LOS originate at X = 1107.

Table 9-5. Predicted Contamination Levels from Payload Bay External Sources²

	(mol/cm ²) NCD	Return Flux (0.19 steradian FOV) (mol/cm ² /sec)		
		200 km	435 km	700 km
LOS 1A				
Outgassing ⁽¹⁾	2.0×10^{11}	0	9.1×10^{10}	3.4×10^9
Offgassing ⁽²⁾	2.1×10^{12}	2.0×10^{12}	4.0×10^{10}	1.3×10^9
Cabin leakage ⁽³⁾	2.2×10^{13}	1.9×10^{13}	3.7×10^{11}	1.2×10^{10}
Supplemental evaporator ⁽⁴⁾⁽⁵⁾	1.7×10^{14}	1.6×10^{14}	2.9×10^{12}	1.1×10^{11}
Reaction control systems	4.4×10^{14}	3.9×10^{14}	7.6×10^{12}	2.7×10^{11}
LOS 2A				
Outgassing	1.4×10^{11}	0	6.7×10^{10}	2.4×10^9
Offgassing	1.6×10^{12}	1.4×10^{12}	2.8×10^{10}	9.3×10^8
Cabin leakage	2.2×10^{13}	2.0×10^{13}	3.7×10^{11}	1.2×10^{10}
Supplemental evaporator	3.3×10^{14}	2.8×10^{14}	5.3×10^{12}	1.9×10^{11}
RCS	8.3×10^{14}	7.3×10^{14}	1.4×10^{13}	6.8×10^{11}
LOS 3A				
Outgassing	1.7×10^{11}	0	7.9×10^{10}	2.9×10^9
Offgassing	1.8×10^{12}	1.7×10^{12}	3.3×10^{10}	1.1×10^9
Cabin leakage	2.3×10^{13}	2.1×10^{13}	3.9×10^{11}	1.3×10^{10}
Supplemental evaporator	2.4×10^{14}	2.1×10^{14}	4.3×10^{12}	1.4×10^{11}
RCS	8.3×10^{14}	7.3×10^{14}	1.4×10^{13}	4.9×10^{11}
LOS 4A				
Outgassing	1.7×10^{11}	0	7.9×10^{10}	3.0×10^9
Offgassing	2.0×10^{12}	1.8×10^{12}	3.3×10^{10}	1.2×10^9
Cabin leakage	1.9×10^{13}	1.7×10^{13}	3.2×10^{11}	1.1×10^{10}
Supplemental evaporator	3.8×10^{14}	3.3×10^{14}	6.3×10^{12}	2.2×10^{11}
RCS	1.4×10^{15}	1.3×10^{15}	2.4×10^{13}	8.5×10^{11}

Table 9-5. Predicted Contamination Levels from Payload Bay External Sources² (Continued)

LOS 5A				
Outgassing	1.5×10^{11}	0	7.3×10^{10}	2.7×10^9
Offgassing	1.7×10^{12}	1.6×10^{12}	3.0×10^{10}	1.0×10^9
Cabin leakage	3.5×10^{13}	3.1×10^{13}	5.6×10^{11}	2.0×10^{10}
Supplemental evaporator	4.1×10^{13}	7.3×10^{13}	1.4×10^{12}	5.0×10^{10}
RCS	1.8×10^{14}	1.6×10^{14}	3.2×10^{12}	1.1×10^{11}
LOS 6A				
Outgassing	2.4×10^{11}	0	1.1×10^{11}	4.1×10^9
Offgassing	2.7×10^{12}	2.3×10^{12}	4.7×10^{10}	1.6×10^9
Cabin leakage	1.9×10^{13}	1.7×10^{13}	3.2×10^{11}	1.1×10^{10}
Supplemental evaporator	1.6×10^{14}	1.4×10^{14}	2.7×10^{12}	9.2×10^{10}
RCS	7.8×10^{14}	7.1×10^{14}	1.3×10^{13}	4.7×10^{11}

- (1) Assumed outgassing rate of 5×10^{10} g/cm²/sec at 100°C
- (2) At 10 hour mark in on-orbit vacuum exposure
- (3) Polar modules (H₂O + CO₂) are approximately 2 percent of total shown
- (4) Both evaporators on
- (5) Study shows that relocation of the evaporators can reduce N to 10^{12} mol/cm² and R to 10^{12} mol/cm²/sec. This relocation has been implemented.

No study has been performed to date to assess the contamination due to the interior of the bay or the AMPS payload.

Since the study was performed, several changes have been made in the Orbiter design that will effect the numbers in Table 9-5. RTV 568 has replaced the higher outgassing RTV 560 as the adhesive bonding of the tiles near the payload bay. Large areas of ceramic tiles have been replaced by Nomex with an epoxy thermal control paint. The supplemental evaporators have been relocated and on-board water storage tanks added to

reduce the contamination from this source. The overall impact of these changes has not been determined as of this date.

Of the types of contamination sources considered, outgassing and cabin leakage are steady state sources, while offgassing is short term and evaporator and RCS exhaust is intermittent. The values for offgassing shown in Table 9-5 are taken at the 10 hour vacuum exposure mark. These values will be reduced by a factor of about 2.5 at the 24-hour mark and will fall to zero after about 48 hours. The assessment is conservative and does not take into account the short RCS duty cycles and the interactions of the wing reflected RCS exhaust with the atmosphere or the RCS plume. Therefore, the calculated return flux from this source may be very high with respect to the actual flux, especially at the lower altitudes. The actual contamination levels which should be used for design of AMPS contamination control equipment are those to be measured by the proposed contamination monitoring package.

9.2.3 AMPS Equipment Sensitivity

While the maximum allowable levels of contamination have not been defined for individual instruments and sensors in the AMPS payload, it is known that some are very sensitive to contamination and must be protected during on-orbit operation. The relative sensitivities to contamination of equipment on AMPS Flights 1 and 2, and subsequent are shown in Table 9-6.

Most of the instruments and sensors listed as "very sensitive" have optical surfaces which can be degraded by deposited molecular films or deposited particulates. The maximum thickness of molecular film that an optical surface can tolerate is a function of the contaminant species, the wave length of the radiation to be transmitted or reflected, and the effect of the incoming radiation on the contaminant film. For instance, it is known that far UV radiation can cause a deposited molecular film to polymerize and drastically change the optical and thermo-physical properties of a surface. A maximum contaminant thickness of 100\AA has been used on several programs as the maximum acceptable level for critical optical surfaces. This value was used for the HEAO contamination assessment (Reference 5). A 100\AA thick molecular film buildup is equivalent to

Table 9-6. Instrument Relative Contamination Sensitivity

Very Sensitive	Medium Sensitivity	Low Sensitivity
Optical band imaging photometer system (OBIPS)	Ion mass spectrometer and distribution analyzer	Faraday cups
IR interferometer	Thermal control surfaces	Retarding potential analyzer (RPA)
Electron accelerator	Electrostatic analyzer	Langmuir probe
IR radiometer		RF sounder
Lidar		Magnetometer
Solar flux monitor		
Neutral mass spectrometer		
UV-Vis-IR spectrometer		
Optical sensors		
Ion accelerator		

a deposition of 10^{-6} gm/cm² (assuming a specific gravity of 1.0) and a return flux of approximately 1.4×10^{10} molecules/cm²/sec (assuming a molecular weight of 20 and a sticking coefficient of 1.0). It is apparent from Table 9-4 that the predicted return flux for altitudes of 435 km and under exceeds this value for nearly all LOS, even if we assume that contributions from the evaporators and RCS can be avoided by proper mission planning and that all offgassing has ceased.

The IR interferometer, IR radiometer, and UV-Vis-IR spectrometer are sensitive to contaminant atmosphere IR radiation. The maximum molecular column densities defined by CRDG and the study are exceeded for most LOS. This is especially true during evaporator venting and RCS firing periods. The evaporator venting is especially serious since polar molecules such as H₂O emit IR radiation in the spectrum of interest. The optics in the IR interferometer and IR radiometer are especially sensitive to return flux since liquid nitrogen and solid hydrogen cooling are employed. The resulting low temperature of the optical surfaces will cause the sticking coefficient to be 1.0 and all impinging contaminants will condense on the optics. Therefore, it will be necessary to protect these surfaces by active or passive means.

The electron accelerator and the neutral mass spectrometer have electron beam sources which are thin filaments which are heated to a high temperature. If the source is activated at a pressure greater than 10^{-4} torr, the filament will burn out causing loss of the instrument. Adequate venting area and proper timing of the experiments must be employed for these units.

Maximum allowable deposited particulate levels have not been defined nor has any assessment been made to predict actual levels. In addition, while a goal has been established for the number of free flying particulates in an instrument field of view per orbit, no assessment has been made of expected levels. The particulate problem can be extremely serious unless stringent cleanliness control is employed for the AMPS equipment and the interior of the payload bay. An easily cleanable payload bay liner would be extremely desirable and would result in a greatly reduced particulate contamination level. It should be noted that even under a clean room environment appreciable contaminants can accumulate on exposed surfaces. For instance, a 30-day exposure of a horizontal surface in a class 100,000 (FED-STD-209) clean area can result in about 24 percent area coverage with particles 50 microns or less in diameter. These particles are too small to be seen by the unaided eye. Vibration or shock loading can jar the particles loose from the surfaces and they can then deposit on critical surfaces or penetrate the field of view of an instrument. It is strongly recommended that a final cleaning, just before launch, be performed on the AMPS and payload bay area (or liner). Solvent wiping with a non-shredding cloth or towel is the best approach. All instrument covers should remain in place until the last possible moment prior to launch. Controllable covers should only be opened after all surface-released particles have been swept away or have settled. If the contamination monitoring package is accepted and flown on an early flight, then the resulting data can be used to determine what special cleaning and protective measures should be used.

9.2.4 Methods of Controlling Contamination

The approach to AMPS contamination control should be to protect the AMPS payload from existing external and payload bay contamination and to design AMPS in such a manner as to prevent self contamination. The following are steps which can be taken to assure a clean AMPS design:

- 1) Select materials which are low outgassers, resistant to shedding, flaking, corrosion, or other types of particle generation. The maximum outgassing rate for materials selected for AMPS tested according to NASA specification JSC SPR-0022 should be 1 percent TWL (total weight less) and 0.1 percent VCM (Vacuum condensible matter).
- 2) Where necessary, special processing techniques should be employed to minimize outgassing and particulate contamination. These would include postcure baking of polymeric materials (paints, adhesives, etc.), special cleaning and handling methods, black light inspection for detection of hydrocarbon contaminants, and special clean area fabrication and assembly.
- 3) Critical AMPS surfaces should be located out of direct line-of-sight view of major contaminant sources such as organic paints and adhesives and lubricants (hinge assemblies, motors, etc.). Where necessary, shields must be provided to protect critical surfaces. It is probable that a shield must be provided to intercept contaminants resulting from outgassing from the vertical stabilizer. Shields should be constructed of inorganic material and have inorganic coatings (such as anodize) to prevent outgassing. If organic material must be used due to special requirements (thermophysical properties, etc.), then special processing should be employed to minimize outgassing.
- 4) Covers should be used to protect all critical surfaces during fabrication, assembly, integration, test, prelaunch, launch, and return modes. The covers should remain in place until the instrument is ready to begin operation or as late into the operation as is possible. Covers should be in place during periods of high particulate release or generation such as RCS firings or payload bay door opening and closing. If possible, the covers should be constructed of inorganic materials with inorganic coatings. Ideally, the covers should be opened and closed on command. For some instruments, such as the cryogenically cooled IR interferometer and IR radiometer assembly, controllable covers should be a requirement.

- 5) Mission time-lining should be established with contamination control as a major parameter. Experiments should not be operated during periods of maximum contaminant generation. For instance, adequate time should be allowed for peak off-gassing to occur (approximately 24 hours) and for particulates to settle or be swept away after the launch, maneuvering, and payload bay door opening operations prior to experiment activation. Contaminant peaks associated with critical STS orientations with respect to the sun should also be established. The proposed on-board monitoring system will provide hard data for establishing AMPS mission time-lining.
- 6) Cold traps or fingers may be required to protect critical optical surfaces from return flux. These traps would be located to intercept molecular contaminant. The IR radiometer and IR spectrometer assembly may require such traps.
- 7) A positive purge of nitrogen gas from the LN₂ cryogenic tanks should be used on the IR radiometer and IR spectrometer assembly. This purge would prevent contaminants from entering into the telescope housing containing the cooled optics. This approach has been suggested for the IR Telescope Facility which is being designed by NASA/AMES (Reference 6) for the Space Shuttle.
- 8) The CRDG recommended the use of a liner in the payload bay to protect contaminant sensitive payloads from particulate and molecular contaminants from the bilge area (Reference 4). This type of liner would be highly desirable for the AMPS payload. The liner should be designed for easy cleaning, constructed of low-outgassing material, and be inspectable by black light and normal light visual techniques.
- 9) It has been recommended that the payload bay doors be closed during the RCS firing (Reference 2). However, it is not at all clear whether closing and reopening the doors would help or worsen the situation. Certainly, particulates would be emitted from exposed surfaces during the door openings and closings and these may pose serious contamination problems unless the individual critical surfaces have covers which open and close on command. In addition, the RCS engine duty cycles are so short (bursts of the order of 40 milliseconds) that the resulting column densities and return fluxes will only represent momentary spikes in the steady state levels and may not significantly effect the operation of the AMPS instruments. Therefore, it is recommended that the payload bay doors remain open during the AMPS mission.

While most of the above steps can be implemented with minimum cost and design impact to the AMPS and Space Shuttle systems, some, such as the use of controllable covers, cold traps, and a payload bay liner, can result in significant cost increases or complexity in the design of the instruments. Therefore, these more cost-critical steps should only be implemented after hard data has been accumulated and a contamination problem has been identified. This data could best be obtained from the proposed contamination monitoring package on an early STS flight. It is also desirable that the AMPS payload contain contamination monitoring instruments, such as quartz crystal microbalances, which can indicate contaminant flux peaks or critical deposition thicknesses. These monitors can be used as a warning system and the experiments could be shut down temporarily and controllable covers (if present) would be closed.

References

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5. L. A. Rosales, "Assessment of the Contamination of Critical Surfaces During Launch and Orbit Operation for the HEAO-A Observatory, 26000-200-007, TRW Systems, 1 May 1975.
6. F. C. Witteborn and L. S. Young, "A Cooled Infrared Telescope for the Space Shuttle - The Spacelab Infrared Telescope Facility (SIRTF)," AIAA Paper No. 76-174, AIAA 14th Aerospace Sciences Meeting, Washington, D.C., January 26-28, 1976.

9.3 ELECTROMAGNETIC COMPATIBILITY

This section summarizes the results of our electromagnetic compatibility analysis effort during the AMPS Phase B study.

9.3.1 General Discussion

Within the context of the AMPS Program, the control of the electromagnetic compatibility (EMC) and interference (EMI) characteristics of the AMPS payload configurations can be loosely divided into two separate but complimentary activities. Arbitrarily labeling one of these activities as "Engineering EMC/EMI Control" to denote those aspects of the control effort, which can and should be addressed using standard EMC/EMI engineering practices, allows the other portion to then be labeled "Sciences EMC/EMI Control." This latter category would relate to the study and resolution of those problems associated with the unique aspects of the scientific instruments which cannot be realistically resolved by implementing the more common interference containment techniques, such as case and cable shielding, brute force line filtering, circuit grounding and/or isolation. Applying such measures directly to the instrument sensors and intentional field emitters would seriously degrade their capabilities or even negate the mission objectives.

The Sciences EMC/EMI Control effort will require extensive trade-off studies leading to practical engineering solutions to the problems generated by the needs or desires of the scientific community. These solutions may take the form of designing special labyrinth shields, operational timing of experiments, locating instrument clusters on extendable booms or on throw-away or recoverable free-flyers, or by negotiating with the scientific community to scale down those requirements which result in extraordinarily difficult and costly designs and approaches. One of the primary early efforts in an AMPS follow-on program must be the translation of the scientific community's EMI requirements into standard engineering terminology. The existing IFRD's are laced with qualitative statements such as:

- "Susceptible . . . at relatively high levels."
- ". . . could be severe source of EMI to other instruments."
- "None."
- "Moderately susceptible to EMI noise."

Where quantitative limits are included in the IFRD's, they are neither standardized, nor are they readily translatable into engineering terminology.

In this section we will define the Engineering EMC/EMI Control requirements which should be imposed at the unit, subsystem, and payload level on the electrical/electronic equipments which directly or indirectly interface with the Orbiter/Spacelab systems. Sections 9.4 and 9.5 give examples of the proposed accommodation methods which can be used to protect scientific instrument sensors from Orbiter/Spacelab generated fields and, conversely, limit the effects of payload emitters upon the parent space vehicle system.

9.3.2 Engineering EMC/EMI Requirements

The basic EMC/EMI design requirements for Orbiter equipments and subsystems and for the Spacelab modules are defined in total or in part in the following documents:

- NASA/JSC SL-E-0001, "MIL-E-6051D Space Shuttle Amendment, Electromagnetic Compatibility, Systems"
- NASA/JSC SL-E-0002, "MIL-STD-461A Space Shuttle Amendment, Electromagnetic Interference Requirements, Equipment"
- NASA/JSC 07700, VOL XIV, "Space Shuttle System Payload Accommodations"
- ESTEC/MSFC SLP/2104, "Spacelab Payload Accommodation Handbook."

It is recommended that the conducted and radiated interference limits and the conducted susceptibility criteria of these documents, and in particular SL-E-0002, be imposed at the AMPS Payload interfaces with Orbiter/Spacelab. This will ensure the continuity of design and functional compatibility in the integrated system. The radiated susceptibility criteria of SL-E-0002, however, should be increased by a factor of at least 20 dB for pallet-mounted equipment which do not have the benefit of the inherent shielding afforded by the skins of the Orbiter and Spacelab modules. It is also essential that primary power distribution within the payload be electrically isolated from structures in order to maintain the single-point ground system of Spacelab.

9.3.3 EMC/EMI Design Implementation

The existing EMC/EMI data base for the AMPS program is wholly insufficient to allow any comprehensive analysis of the probable EMC/EMI characteristics of the eventual system, nor can specific recommendations be formulated from the current data. The following, however, is a synopsis of the type of control measures which would be applicable to the electrical/electronic units and subsystems:

- Equipment housings electrically bonded per MIL-B-5087, Class R
- Housings designed as RF tight enclosures
- In-line, passive EMI filters at all power interfaces
- Fully shielded electrical cable harnesses
- Primary power returns electrically isolated from structure, secondary power, and signal returns
- No intentional power or signal return currents in structural members
- Single-point grounding for all circuits except coaxial RF circuits
- All input/output circuitry protected against indirect lightning strike and spacecraft differential charging/discharge effects.

Implementation of these general measures, accompanied by test verification at the unit and subsystem level, should ensure functional compatibility between the payload equipment and the Orbiter/Spacelab modules. As further definitive data become available during the design and development phases, they can be factored into a computer-assisted compatibility analysis program which can identify pertinent problem areas and lead to specific recommendations for correction. A chart depicting the data flow during these analytical studies is shown on Figure 9-3. The computer program proposed is that currently being used by ESA/ERNO on Spacelab.

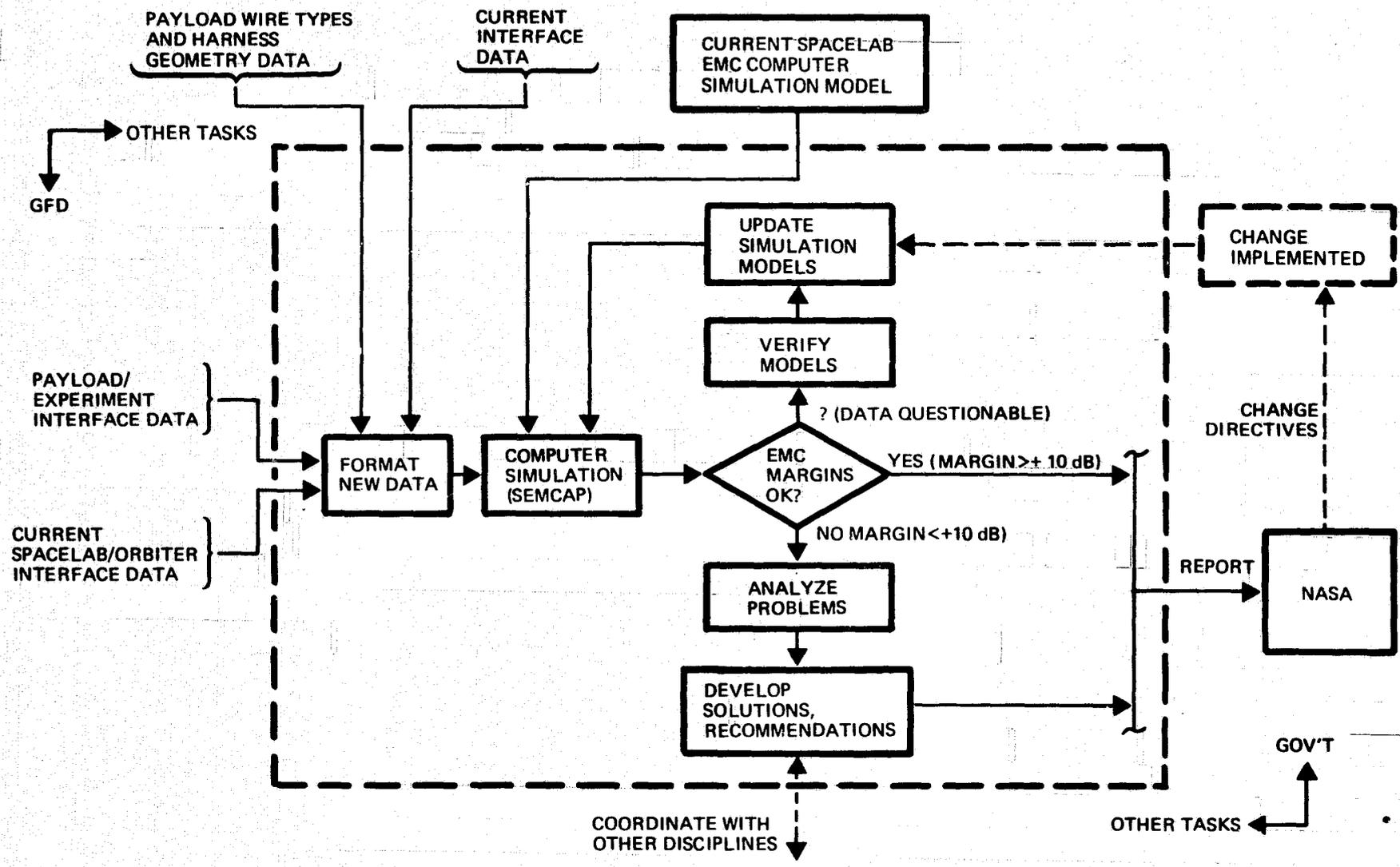


Figure 9-3. Payload/Spacelab Electromagnetic Compatibility (EMC) Analysis

9.4 MAGNETIC INTERFERENCE

In this section, magnetics is defined to cover the 0 to 20 Hz frequency range to distinguish it from the standard EMC regime which starts at 20 Hz. The usual EMC analytical and measurement techniques are not directly applicable to magnetics.

9.4.1 Magnetic Requirements

Magnetic interference considerations for AMPS payload design arise from the compatibility requirements for the scientific instruments and their associated control and data processing and display equipment. The latter category includes electron beam devices such as tape recorders, relays, motors, and magnetic core storage. Generally, these are sensitive to magnetic fields greater than one gauss. However, at close distances within the Spacelab module and on the pallet, it is entirely possible that such fields and even larger magnitudes could be encountered. Contributions from Spacelab and the Orbiter as well as the payload must be considered.

The scientific instruments, particularly in the AMPS payload complement, impose very severe requirements on the acceptable magnetic environment. Again, two categories of requirements exist, those for the pointing of instruments (accelerators, antennas) relative to the local geomagnetic field, and those for magnetic field sensors. The various requirements are summarized in Table 9-4.

9.4.2 Sources of Magnetic Interference

Magnetic fields may be created by electrical currents or may be due to magnetic materials. Fields due to currents are called "strays" in spacecraft technology and are defined by Amperes' law:

$$d\vec{h} = \vec{i} \cdot \frac{d\vec{l} \times \vec{r}}{r^2}$$

Fields due to magnetic materials may be permanent or induced. Permanent or remanent magnetization is due to the hysteretic property of magnetic materials in which magnetization remains after an inducing field is applied and then removed. If the retained magnetization is high, the material is "hard." It is "soft" if the remanence is low and easily changed or removed by small fields (1 to 10 gauss). Induced magnetic moments, in the

Table 9-4. Magnetic Environment Requirements for AMPS Equipment

Functions/Devices	Sensors	Required Thresholds	Frequency Range
Scientific Measurements	Alkali vapor,	0.01 nT	0.01-20 Hz (ULF)
Geomagnetic Field, ULF, Ionospheric Currents	Fluxgate, and Search coil magnetometers	10 nT	0-10 Hz (dc)
Geomagnetic Attitude Determination	Fluxgate magnetometers	100 nT	0-10 Hz
Electron Beam Devices	CRT's, etc.	1 gauss	0-100 MHz
Magnetic Devices	Tape recorders, etc.	10 gauss	0-20 MHz

linear regime, exist only as long as inducing fields are present. The magnetization depends strongly on the size and geometric shape of the magnetic body as well as its orientation relative to the inducing field. For most magnetic materials with permeability, μ , much greater than unity, induced fields are relatively independent of the actual magnitude of μ .

Figure 9-4 compares the magnetic moment per unit weight of various spacecraft, the NASA Convair 990 aircraft and the estimated value for Orbiter. The very low values were achieved for the Pioneer spacecraft as a result of a magnetic cleanliness control program instituted in response to contractual specifications. The value for the Orbiter was an early estimation supplied by NASA/JSC. It corresponds to a field generated by taking the total Orbiter power of 16.1 kW, assuming that this power is all supplied at 28 volts dc and 575 amperes, and assuming that the current flows in a 1.1 by 20 meters rectangular loop (22 m² area) around the Orbiter bay.

9.4.3 Magnetic Modeling

An extremely useful technique in achieving payload/spacecraft magnetic compatibility is that of magnetic modeling. The model is started very early with whatever data is available and is updated continuously as

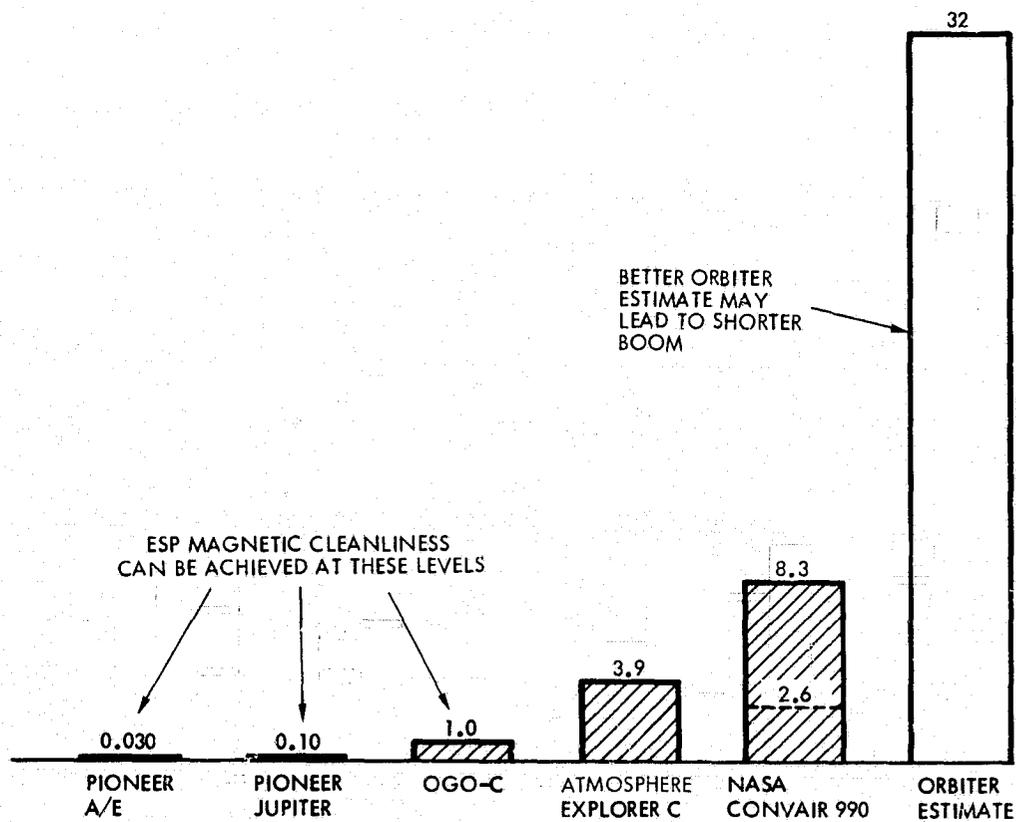


Figure 9-4. Per Unit Weight Magnetic Moments (Pole-cm/kg) of Different Spacecraft and Aircraft Show a Wide Variability

additional information is generated. By comparing the model with the various requirements, problem areas and nonproblem areas are defined and cost-effective tradeoffs in design may be made on an overall systems basis without costly over design. Information on areas where better data must be obtained either by test or by analysis is also provided by comparison of the model with requirements. The number and scope of costly tests are minimized in that emphasis is placed on the verification of the model only in critical areas.

In the model, all significant sources are identified and characterized as to location and falloff with distance (dipole or otherwise) in the vector sense. In contrast to higher portions of the frequency spectrum, magnetic modeling of individual sources is a more valid approach in that shielding of magnetic fields, unless intentionally introduced, is generally ineffective over large distances. Contributions from the Orbiter and Spacelab may be included directly in the magnetic model for AMPS. Frequency and time-lining parameters of ac sources are also included.

9.4.4 Achievement of Magnetic Compatibility for AMPS

In the case of the AMPS equipment, there are two regimes of accuracy requirements: below and above one gauss. For the latter at Orbiter altitudes, induced fields are generally not of concern because the geomagnetic inducing field is about 0.5 gauss. Thus, the compatibility of the high field category of equipment is easily taken care of as long as the possible areas of interference are identified. In the area of high sensitivity dc and ULF scientific measurements, the high cost impact of long booms or magnetic cleanliness control of the Orbiter is best avoided by emplacing the sensors on throw-away or recoverable subsatellites. In fact, some prospective experiments require such remote platforms with these sensors on board. A magnetic cleanliness control program will be required for the subsatellite but will be of reasonable cost if properly implemented.

For geomagnetic attitude/pointing purposes, it has been determined that computations using available mathematical models of the geomagnetic field are not sufficiently accurate. The worst case errors could be as large as 7.5 degrees and accuracies of ± 2 degrees or better are required. Although estimated Orbiter fields 7.3 meters away from its long axis (5-meter boom) are in the order of 1000 nT, we have determined that such a boom length will permit magnetic orientation measurements to be made to a 100 nT accuracy which is equivalent to ± 2 degrees if the invariant part of the Orbiter field is calibrated out by a one-orbit comparison of the magnetometer output with the geomagnetic field model. The factors considered in the analysis are the induced magnetization and eddy current changes due to variable Orbiter attitudes with respect to the ambient geomagnetic field and the field change due to a change in the power loading of the Orbiter. The results are shown in Table 9-5.

As indicated in Figure 9-4, the Orbiter estimated fields are too large. It could be possible to decrease the boom length when better data become available.

Table 9-5. Estimated Variable Fields of the Orbiter

Source Variable Moment	Estimated Moment	Estimated Field at 5 M
Induced Magnetization	2125 G-cm ³	3.4 nT
Eddy Currents	204 G-cm ³	0.3 nT
Turnon/Turnoff of dc Power	38,600 G-cm ³	61.8 nT*

* 50.4 percent of the 16.1 kW power could be turned off and this would cause a 100 nT field change at 7.3 meters.

9.5 SPACECRAFT CHARGING AND ELECTROSTATIC INTERFERENCE

Spacecraft charging and electrostatic interference, as with magnetics, are generalized aspects of the electromagnetic compatibility problem. Again, since these disciplines are not covered by the standard EMC technology, they are covered in this section.

9.5.1 Spacecraft Charging

Spacecraft charging problems arise mainly because of the inclusion of high energy electron and ion beam accelerators as important instruments on AMPS payloads. During a 2-ampere electron beam emission experiment, for instance, current continuity requires that a 2-ampere neutralizing current be collected from the ambient plasma. If adequate quantities of plasma electrons are not collected, the remainder must be made up by displacement currents. This implies possible chargeup of the Orbiter to multikilovolt levels. Many differential charging situations are then possible which could then result in serious arc discharge EMI problems. Even when chargeup and arc discharges do not occur, the neutralizing return currents involve unusual paths starting from the exterior surfaces of the Orbiter, its structural members, and finally back to the accelerator power supply. Thus, the standard EMC analyses must be expanded to cover this added dimension.

Whether or not high potential chargeup will actually occur is currently a subject of controversy within the scientific community. Figure 9-5 shows the results of a number of different calculations for a spherical satellite with different assumptions. The main point to be noted is that the potentials range from a fraction of a volt to over a million volts (with an assumed constant current source and unlimited voltage). Experimental evidence from electron beam experiments on rockets seem to indicate that chargeup does not occur. However, these were not at Orbiter altitudes, were not always at proposed energies, and certainly did not have the Orbiter configuration.

9.5.2 Accommodation of High Energy Accelerators

The proposals for the high energy accelerators have recognized the potential problems due to spacecraft charging and have incorporated a number of features to counteract them. The prime point made is that much

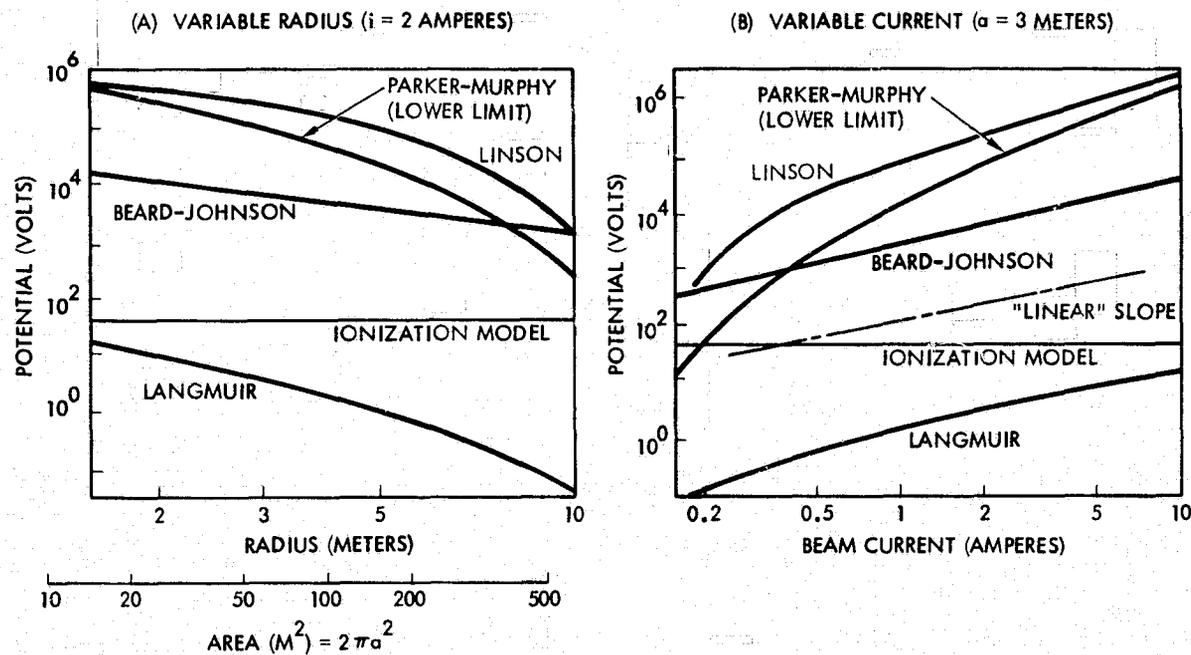


Figure 9-5. Spherical Satellite Charging Calculations Show a Wide Range of Potentials ($N = 2 \cdot 10^5/cc$)

"good science" can be accomplished at low current levels and that charging detectors would be an essential part of the diagnostic package for the experiment. Thus, a warning signal would be available if charging begins to occur. Over and above this, additional features such as a gas plume and a neutralizing beam accelerator (MPD Arc) have been proposed to permit operation at higher current levels.

It is proposed that additional chargeup diagnostics (retarding potential analyzers) be provided to monitor other parts of the Orbiter. Also, EMI noise monitors should be added to monitor the "unusual" neutralizing return-current paths and to detect and localize low level arc discharges if they should occur.

9.5.3 Electrostatic Requirements

Requirements on electrostatic environments are imposed by low energy particle and dc electric field detecting experiments. The latter, for example, requires a 10^{-4} volt/meter noise threshold over a 0 to 10 Hz frequency band. Low energy particle detectors generally specify a minimum exposed metallic surface area to assure that sufficient low energy plasma electrons are collected to maintain a sub-volt sensor reference

potential. Other potential electrostatic problem areas such as glass CRT faces and electrostatic chargeup, due to friction within the cabin and module, are not serious as long as potential problems are identified.

9.5.4 Sources of Electrostatic Interference

Spacecraft charging due to charged particle acceleration will certainly cause electrostatic interference. Besides accelerators, sources of electrostatic interference are those due to applied potentials, e.g., voltages applied to electrostatic analyzers, $\vec{v} \times \vec{B}$ voltages due to masts and booms, and charging or discharging due to photoemission, secondary emission, and the ambient plasma.

9.5.5 Accommodation of Electrostatic Interference Requirements

As with magnetics and standard EMC, an analytical electrostatic modeling approach is recommended for the reasons outlined in Section 9.4.3 for magnetic modeling. A spacecraft charging model incorporating the accelerator forcing function is recommended for the Orbiter. For the sensitive low energy particles and electric field detectors, placement on board remote subsatellites, as for the other sensitive electromagnetic sensors, is the most sensible approach. As with the other sensors, this is a requirement for the implementation of some experiments. A separate charging/discharging model should be generated for the subsatellite. The dc isolation of booms to eliminate $\vec{v} \times \vec{B}$ potential, shielding, and the selection of surface materials for conductivity and photoemission are some of the techniques which must be applied to the subsatellite to meet its electrostatic requirements.

9.6 SAFETY ANALYSIS

This section presents the results of the safety analyses during the AMPS Phase B study.

9.6.1 Safety Requirements and Shuttle/Spacelab Provision

During the study significant changes occurred relative to the safety requirements for Shuttle payloads. The safety requirements that were applicable at the first of the study changed significantly. The STS Payload Safety guidelines became available toward the end of the study. These two factors had a significant influence on the Flight 1 and 2 design. Also, there were major changes to the Spacelab system. Both new documents brought into focus the levels of risks that are acceptable. Some of the significant changes that occurred are:

- Caution and Warning (C&W) function may be monitored from a ground station and not just by the Shuttle Orbiter.
- Only emergency parameters requiring rapid man-in-the-loop response will be allowed to be monitored on the Orbiter C&W system.
- The Spacelab had a major change in the C&W system design.
- The required hazard controls for barium-thermite, ordnance, and electrical systems are more preventive in nature rather than requiring hazards to be controlled after they occur. Some other hazards must be controlled after they occur.
- A general definition of the level of failure/fault or human error protection required before catastrophic hazards can occur. The level of accident risks that are acceptable have been indirectly defined for many known hazardous conditions.

To help deduce, indirectly, the levels of accident risks that are considered acceptable, the guidelines in the STS payload safety guidelines handbook were assessed, for they indirectly establish a baseline or level of accident risk that is considered acceptable. To assess the guidelines they were separated into two categories: those that minimize the chance of hazards occurring and those that assure control of hazards and prevent damage to Orbiter or injury to personnel if they did occur. By separating the guidelines in this way, a determination could be made as to whether NASA will require major design effort into preventing hazards or preventing and controlling hazards once they occur. When just hazard

prevention methods are used, an apparent higher level of risk is being taken than would be taken if hazard prevention and occurrence controls were both used.

Relative to AMPS payloads, a review of the safety guidelines and requirements reveals the following:

- a) There is no guideline or requirement that states that more than one barium-thermite canister when inadvertently ignited while in the cargo bay must be controlled such that catastrophic conditions will not result. This is not to say that the effects of burning barium-thermite should not be minimized as much as possible. The above conclusion is based on the fact that the payload safety guidelines does not provide any guidelines for control of a large solid rocket motor in the cargo bay once the motor has fired. For this conclusion it is assumed that the barium-thermite instrument setup is equivalent to a fairly large solid rocket motor in hazard potential.
- b) The majority of the safety design guidelines for electrical hazards are controlled mostly by hazard prevention methods. Electrical fires in the cargo bay are mainly controlled at the STS system level (e.g., ground purges), except that electrical power may be shut off to the payload which minimizes the chance of a fire propagating.
- c) We cannot depend on just preventive measures. The safety design guideline document does indicate a desire for control of the following hazards once they occur because risk of Orbiter damage or personnel injury is too great without them.
 - Control of flammable gases in cargo bay by requiring detection system and purging of the cargo bay
 - Control of ignition sources for flammable vapors
 - Control of toxic gases in the Spacelab
 - Control of fires emanating from lasers
 - Control of cryogenic hazards by requiring remote loading and loading at the last minute before launch.

For a list of the significant STS payload safety guidelines that have the potential for large design impact relative to AMPS payloads, refer to Table 9-6. These guidelines are not requirements, they only identify some ways to control hazards. Other hazard control measures can be chosen, but equivalent levels of risk probably should be maintained.

Table 9-6. Significant STS Payload Safety Guidelines/Requirements Relative to the AMPS Flights 1 and 2

1) Pressure System

- Protect vessels from external hazards (para. 3.11.1.2f): temperature, shrapnel, collision, etc.
- Safely release pressure in case of failure (para. 3.11.1.1b)
- Monitor tank pressure, temperature, or strain
- Operate pressure system remotely (para. 3.11.1.12d)
- Control release of pressurant into cargo bay under normal and failure conditions (para. 3.11.1.2m)
- Contain all hazardous payload effluents vented during ground servicing or deservicing at the Shuttle vehicle (para. 3.11.3.1h)
- Warn Orbiter crew of imminent pressure vessel failure if injury would result (para. 3.11.1.12h)
- Consider helium low pressure gas test after installation into the cargo bay (para. 3.11.3.1g)
- Protect against shrapnel in case of failure. Consider fiberglass-wound tanks, pressure, relief devices, use of shrapnel-proof barrier, etc. (para. 3.11.1.2h).

2) Cryogenics

- Cannot release hazardous gases into payload bay (STS Payload Safety Requirements, para. 5.1.14)
- Must contain vented hydrogen and methane gas with the Orbiter while on the ground (para. 3.11.3.1h)
- Provide hazardous gas detection equipment (para. 3.3.3.1f)
- Load and check out cryogenics remotely (para. 3.3.3.2c)
- Equivalent to explosive proofing electrical equipment must be provided for all electrical equipment in the payload bay (KSC-STD-E-0002)

Table 9-6. Significant STS Payload Safety Guidelines/Requirements
Relative to the AMPS Flights 1 and 2 (Continued)

2) Cryogenics (Continued)

- Cryogenic liquids loading will be performed as late as possible during preparation for launch (KMI1710.1B/SF)
- Liquid hydrogen should be stored in closed containers under 3 to 10 psig to prevent backflow of air into system (para. 3.3.3.1a)

3) Pyrotechnic/Solid Rocket Motor (SRM)

- Must haul safe and arm device with safing capability (para. 3.12.1.1a)
- SRM initiator should be installed at the latest feasible point during ground preparation for launch (para. 3.12.3.1c and 3.13.3.2b)
- Pyrotechnic exhaust products should be contained or controlled (para. 3.13.1.1k)
- Pyrotechnic device housing should be designed to prevent debris damage because of explosion (para. 3.13.1.1j)

4) Radiation

- Possibly provide a backup (emergency) communication link that will not be affected by EMI (para. 3.14.1.3-)
- Equipment producing radiation hazardous to the crew should be shielded (para. 3.14.1.1c)

5) Laser

- Incorporate adequate means to prevent accumulation of hazardous gases (para. 3.10.1.2dd)
- Locate laser experiments involving potential dangerous heat sources should be in place where fire protection is provided (para. 3.10.2)
- Do not use quartz for flash tubes (para. 3.10.1.2h)
- Use low-powered lasers to align target, optics, filters, etc., (para. 3.10.1.2v)
- View laser beam with closed circuit TV or optical comparator only (para. 3.10.1.2w)

Table 9-6. Significant STS Payload Safety Guidelines/Requirements
Relative to the AMPS Flights 1 and 2 (Continued)

6) Payload Level Safety Requirements

- For hazardous functions, the design must consider the consequences of inadvertent occurrence. The level of redundancy will depend on the hazard potential of the payload. In special cases, dual or greater redundancy may be required for adequate hazard control. An arm and execute action with provisions to return to a safe condition is required for intentional occurrence (para. 5.1.3 of payload safety requirements document).
- Hazard detection and safing functions must be capable of being tested for proper functioning from the Orbiter, from Spacelab, or ground.
- Hazardous situations which require a rapid response should, preferably, be corrected by automatic systems within the payload design. Use of this display and command capability should be limited safety conditions requiring rapid, man-in-the-loop response which cannot or logically should not be handled by design and/or operational provisions.
- The status of safing system and the indication of anomalous conditions occurring within a payload which do not meet the above criteria should be handled in the same manner as general payload telemetry and command and control.
- Interface with the caution and warning and safing control system as required in paragraphs 14.1.7, 14.1.2, 11.1.3.1, and 14.1.3.1 of JSC 07700, Volume XIV, Revision D.

9.6.2 AMPS Flight 1 and 2 Safety

9.6.2.1 Safety Analysis Approach

The AMPS Flights 1 and 2 payload which consists of the FSE and instruments have been thoroughly reviewed and analyzed to determine where potential hazards may exist, the magnitude and nature of the hazards, and the potential effects of the hazards when uncontrolled. All the above was accomplished by first performing an energy survey. See Table 9-7 for a typical example. From the data developed in the energy survey, many of the hazards and their potential energies could be determined. After the hazards were identified, they were evaluated to determine if they would have catastrophic effects if they came into being and were uncontrolled.

Table 9-7. Energy Source Survey, Flight 1

Energy Source	Functional Description	Characteristics	Remarks
LIDAR Two laser beams	Lidar performs sounding to determine altitude profile of ozone in the earth's atmosphere	<ul style="list-style-type: none"> ● 1 joule beam power ● Radiates away from the Orbiter through the Z axis ● 0.02 resolution at Sodium D line (5890A) ● UV near 300 nanometers ● 2 kW from common energy storage required ● Beam diameter 10 mm ● Laser pulse duration 1 μsec ● Laser pulse power: 1,000,000 W/μsec ● Frequency-doubled dye laser ● Wavelength 30.16, 310.9 nanometers ● Laser efficiency 0.1 percent ● Beamwidth 10^{-3} rad (4 arc-min) ● Pulse repetition rate: 1 pps each ● Maximum beam power 20 kW 	<ul style="list-style-type: none"> ● Shall be designed to be fail-safe; capable of being safed under various conditions ● Cover over cylinder when not firing laser ● Lidar is rigidly mounted ● Glass lasing tube temperature should be kept at less than 70°C ● Possibly cryogenic cooling required
Electron Accelerator		<ul style="list-style-type: none"> ● Beam energy 2 to 40 KeV ● Maximum beam current 2 amps ● Pulse length 0.01 to 1 sec ● Energy storage 40 kJ ● Peak power levels of 20 kW ● Alignment with B-field to ± 2 deg 	<ul style="list-style-type: none"> ● Lockout prevents firing if B-field is not correct ● Base mounted to pallet center or end hard points ● Dedicated cables to NIM-CAMAC in module ● Levels I and II diagnostics part of accelerator

The next very critical stage in the safety analysis was to determine how the hazards identified would be controlled. In all cases, the first effort made was to eliminate the potential hazard by design. For example, many hazards to personnel in the Spacelab were eliminated by placing all the highly hazardous equipment on the pallets outside the Spacelab. Another example was the sizing of the xenon gas release vessels on Flight 1. These bottles were sized so that if one did rupture, the cargo bay would not be overpressurized. In general, the hazard controls identified were specified according to the hazard reduction precedence found in paragraph 5.2.3 of the STS Payload Safety Requirements document.

For potential hazards that remained after efforts were made to eliminate them, hazard controls were established based on the minimum hazard controls found in the STS Payload Safety Requirements document and the KSC Ground Safety Requirements document. In addition, the STS payload safety guidelines were used as a baseline for determining acceptable levels of risk on hazards that were not specifically controlled by the STS Payload Safety Requirements document. Section 9.6.1 of this report discusses the acceptable levels of risk.

9.6.2.2 AMPS Flight 1 and 2 Specific Hazard Controls

For the AMPS Flight 1 and 2, 39 types of hazards have been reviewed and evaluated, and hazard controls have been established at the system level.

The specific system level hazard controls for the more significant hazard sources for Flights 1 and 2 are shown in Table 9-8 and Table 9-9. Each of these figures lists the instrument or flight support equipment of concern; the hazardous aspects of the equipment; the significant safety controls or attributes; and (under the remarks column) various measures that allude to the potential energies that could be involved in the hazard.

The majority of the hazards are controlled by preventive measures which consist of providing elements in the design which assure that there is a very low risk that a hazard will occur. In addition to providing preventive measures, a caution/warning and emergency system, in conjunction with a safing system, is used to detect hazards that have occurred

or about to happen and provides a means of safing the system to prevent damage to the Orbiter or injury to personnel. See Section 9.7 of this report for a description of the caution and warning system.

Several of the hazards are on both flights as shown in Table 9-9. The electron accelerator is on the first flight only.

Because the STS payload safety guidelines handbook does not cover all the types of hazards (barium-thermite, solid hydrogen, etc.) that are found on Flights 1 and 2, the acceptable levels of risk were deduced by contacting Johnson Space Center and discussing the hazards and by similarity. For example, the levels of risk peculiar to a solid rocket motor (SRM) were considered similar to the levels of risk for barium-thermite; therefore, the levels of risk associated with safety controls for the SRM were considered acceptable for controlling the hazards of barium-thermite canisters.

Table 9-8. AMPS Flight 1 Significant Safety Attributes

Instruments/FSE and Hazardous Aspects	Significant Safety Attributes	Remarks
<p>1. <u>Lidar</u></p> <p>Flash lamps, thermal, flammable fluids, fire, pressure, laser beams, EMI, x-rays, ultraviolet radiation, high voltages, electrical shorting, corona, reflected laser light, loose or weak equipment, failure to retract cover over lidar before reentry</p>	<p><u>Laser Beam</u></p> <p>Points away from the Orbiter through the Z axis</p> <p>None of the laser equipment (except controls) is located in the Spacelab</p> <p>Laser is fixed and not gimballed</p> <p>Caution, warning, and safing provided</p> <p>Detail design will be per the payload safety guidelines</p> <p><u>Dye (Alcohol) Reservoir</u></p> <p>Dye tank designed to withstand crash landing loads without a hazard being created</p> <p>Dye tank designed to be fail-safe (double-wall tank will be provided)</p> <p>Fire detectors and fire suppressants provided</p> <p>C&W and safing provided</p> <p>All electrical power turned off before reentry</p>	<p>1 joule beam power</p> <p>Laser pulse duration 1 μsec</p> <p>Laser pulse power, 1 million W/μsec</p> <p>150 to 250 liters of ethyl alcohol dye</p>

C-4

Table 9-8. AMPS Flight 1 Significant Safety Attributes (Continued)

Instruments/FSE and Hazardous Aspects	Significant Safety Attributes	Remarks
<p>1. <u>LIDAR (Continued)</u></p>	<p><u>High Voltage Power Supply</u></p> <p>This instrument-provided power supply will be required to conform to all the appropriate STS payload safety design guidelines</p>	<p>2.5 kv power supply (1 kW)</p>
<p>2. <u>Electron Accelerator</u></p> <p>High voltages, x-rays, electron beam, static charging, electrical shorts, high pressures, corona, fire, high voltage/current electrical discharges, EMI, loose or structurally weak equipment, flammable fluids in isolation transformer</p>	<p><u>High Voltage Power Supply</u></p> <p>Connected to single point ground</p> <p>Located on pallet away from all personal contact during flight</p> <p>Deenergized on ground and during launch and landing</p> <p>C&W and safing capability provided</p> <p>STS payload safety guidelines followed during detail design</p> <p>1 atmosphere nitrogen environment provided for capacitors</p> <p>Capacitor containers and electrical circuits designed to be fail-safe</p>	<p>500-volt capacitors converted up to 30 kv and 10 amps</p>

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Table 9-8. AMPS Flight 1 Significant Safety Attributes (Continued)

Instruments/FSE and Hazardous Aspects	Significant Safety Attributes	Remarks
<p>2. <u>Electron Accelerator (Continued)</u></p>	<p><u>Electron Beam</u></p> <p>Beam direction is strictly controlled through accelerator control system</p> <p>Magnetometer (on the 5-meter boom) used to sense improper position of earth's magnetic field when accelerator is fired. Sensing signals fed into the AMPS payload C&W system</p> <p>Sensors placed on the pallet to detect any excessive charging of the payload or the Orbiter. EMI sensors provided</p> <p>Power levels lower for Flight 1 than for other flights</p> <p>Power levels on accelerator gradually increased so that all trends may be monitored that may lead to a detrimental condition</p> <p>C&W and safing provided</p>	<p>The power level for Flight 1 is 20,000 volts, and on later flights it will be increased to 50,000 volts</p>

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Table 9-8. AMPS Flight 1 Significant Safety Attributes (Continued)

Instruments/FSE and Hazardous Aspects	Significant Safety Attributes	Remarks
<p>3. <u>Gas Releases</u></p> <p>High pressure xenon, deployment of gas vessels, ordnance devices, structural strength during various environments</p>	<p><u>High Pressure Vessels</u></p> <p>Pressure vessels to be designed to leak before bursting</p> <p>Dumping of all gas in one vessel when cargo bay doors are closed does over-pressurize the cargo bay</p> <p>Ordnance designed to STS payload safety guidelines</p> <p>Each vessel is spring ejected from cargo bay at very low velocity</p> <p>Xenon is released from vessel 1-1/2 to 3 hours after being launched from the Orbiter</p>	<p>The pressure in each of the five xenon gas pressure vessels is 50 atmospheres</p>
<p>4. <u>SIPS - Cryogenic Canister</u></p> <p>Solid hydrogen, liquid nitrogen, liquid helium, gaseous hydrogen, high rates of boil-off, ordnance, fires, explosions, ignition sources, liquid air, rotation, stowage, retraction of SIPS, structural strength, asphyxiation, excessive pressures</p>	<p><u>Cryogenics</u></p> <p>Remote loading of cryogenics in cargo bay at launch pad</p> <p>Normally vented overboard prior to landing</p> <p>Liquid helium flown to keep H₂ solid upon abort landing for 24 hours. Chemical pump to contain H₂ effluents when on ground and during landing</p>	<p>There is 7 kg of solid H₂</p> <p>There is a 10-hour time limit on chemical pump after abort from orbit. The 10-hour time limit is after the 24-hour time limit for the liquid helium</p> <p>Purging is not provided 15 minutes after landing. Same risk is taken for the Orbiter LH₂ fuel cell</p>

Table 9-8. AMPS Flight 1 Significant Safety Attributes (Continued)

Instruments/FSE and Hazardous Aspects	Significant Safety Attributes	Remarks
<p>4. <u>SIPS - Cryogenic Canister (Continued)</u></p>	<p><u>Cryogenics (Continued)</u></p> <p>Launch and landing ground purges considered sufficient</p> <p>All power (except for monitoring system) turned off for reentry and landing</p> <p>Accumulated static charges discharged in orbit</p> <p>C&W monitoring and safing provisions provided</p> <p>Release of cryogenic boil-off into cargo bay does not overpressurize the cargo bay</p> <p><u>SIPS</u></p> <p>All ordnance designed per payload safety design guidelines</p> <p>SIPS ejected if it fails to retract</p>	<p>The electron accelerator will be discharged before landing</p> <p>All thermal insulation will be designed such that a static charge will not be accumulated</p>
<p>5. <u>Booms and RMS</u></p> <p>Inadvertent extension, jettison, static charging, operation of, stowage, loose item during landing, dynamics, pyrotechnics</p>	<p><u>Booms</u></p> <p>C&W monitoring and safing provided</p> <p>Boom jettisoned if it fails to retract or be stowed properly</p>	<p>The standard Shuttle RMS and a 5-meter boom are used on Flight 1</p>

Table 9-8. AMPS Flight 1 Significant Safety Attributes (Continued)

Instruments/FSE and Hazardous Aspects	Significant Safety Attributes	Remarks
<p>5. <u>Booms and RMS (Continued)</u></p>	<p><u>Booms (Continued)</u></p> <p>Boom grounded so that static charge will not accumulate</p> <p><u>RMS</u></p> <p>Orbiter-provided system. All design safety controls provided by the Orbiter</p>	
<p>6. <u>Equipment in Orbiter Aft Flight Deck (AFD)</u></p> <p>Flammable materials, toxic off-gassing, structural strength during crash landing, fire, CRT vacuum tube, EMI, electrical shock, mechanical hazards to personnel</p>	<p><u>Electrical and Mechanical Equipment</u></p> <p>Equipment provided in the AFD designed to conform to all payload safety design guidelines</p> <p>Present equipment proposed in general conforms to all payload safety guidelines</p>	

Table 9-9. AMPS Flight 2 Significant Safety Attributes

Instruments/FSE and Hazardous Aspects	Significant Safety Attributes	Remarks
<p>1. <u>LIDAR</u></p> <p>Same as for Flight 1</p>	<p>Same as for Flight 1</p>	<p>Same as for Flight 1</p>
<p>2. <u>SIPS - Cryogenic Canister</u></p> <p>Same as for Flight 1</p>	<p>Same as for Flight 1</p>	<p>Same as for Flight 1</p>
<p>3. <u>Gas Releases</u></p> <p>High temperatures, explosions, fires, all from barium-thermite gas release canisters; ordnance devices and initiators, high pressure gas, toxic gas, inadvertent deployments, structurally weak during crash landing</p>	<p><u>Barium-Thermite Gas Release System</u></p> <p>Barium-thermite system designed such that three failures or operator errors must occur before inadvertent operation can result</p> <p>Pyrotechnic subsystems and devices designed to meet the safety related provisions of JSC 08060A, "Space Shuttle System Pyrotechnic Specification"</p> <p>All barium-thermite canisters protected from external thermal hazards to prevent inadvertent ignition</p> <p>Caution, warning, and safing provisions provided</p>	<p>Barium-thermite when ignited releases a large volume of gas at approximately 3000°F. There are 20 canisters of barium-thermite, weighing a total of 704 pounds</p>

Table 9-9. AMPS Flight 2 Significant Safety Attributes (Continued)

Instruments/FSE and Hazardous Aspects	Significant Safety Attributes	Remarks
<p>4. <u>Booms and RMS</u></p> <p>Same as for Flight 1</p>	<p>Same as for Flight 1</p>	<p>There is a 5-meter boom as on Flight 1 and an additional 15-meter long boom with a 10-meter balloon test body on top of the 15-meter boom</p>
<p>5. <u>RF Sounder</u></p> <p>High voltages, 100-meter long dipole antenna, static charging, EMI, RF energy, inadvertent deployment, antenna structural stability (may buckle)</p>	<p><u>Antenna</u></p> <p>Orbiter position closely monitored to prevent buckling of antenna</p> <p>C&W and safing provided</p> <p>All electrical equipment grounded to prevent electrostatic charging</p> <p>Capability to jettison antenna provided in case of safety problem</p> <p>Controls for inadvertent extension of antenna provided</p> <p>Monitoring of RF and EMI provided to assure safe levels of radiation</p>	<p>10,000 watts peak power at 100 kHz to 20 MHz on 100-meter dipole antenna. The antenna is retractable</p>

Table 9-9. AMPS Flight 2 Significant Safety Attributes (Continued)

Instruments/FSE and Hazardous Aspects	Significant Safety Attributes	Remarks
<p>6. <u>Test Body</u></p> <p>Static charging, high voltages, high pressure gas, inadvertent deployment, dynamic problems</p>	<p><u>Test Body</u></p> <p>Means provided to eject test body in case of safety problem</p> <p>Fail-safe design provided</p> <p>C&W and safing capability provided</p> <p>Failure of pressure vessel will not result in overpressurization of cargo bay</p>	<p>The best body is an inflatable 10-meter diameter balloon that has an electrically conductive skin</p>

9.7 CAUTION AND WARNING

The caution and warning (C&W) requirements have been in a state of flux during this study and some major changes occurred toward the end of the study. The C&W requirements in the STS Payload Safety Requirements document changed significantly and the Spacelab C&W system changed significantly, both of which are discussed in this section.

Some of the changes to the payload C&W requirements are:

- The nomenclature has changed; it is now called "Hazard Detection and Safing" and not C&W.
- The status of safing systems and the indication of anomalous conditions occurring within the payload can be monitored by ground stations and not on the Orbiter C&W panel.
- The Orbiter C&W display is to be used to monitor safety conditions requiring rapid man-in-the-loop response which cannot or logically should not be handled by design and/or operational provisions.
- Only hazards that have a reasonable probability of occurrence are to be monitored (per JSC 11123 STS Payload Safety Guidelines Handbook, Section 3.2).
- Hardwiring of C&W parameters are not required.

The above changes were clarified after discussions held with the JSC Space Shuttle safety office at which time it was determined that there is a definite JSC desire to limit the amount of C&W monitoring by the flight crew. JSC is requesting that only "emergency" parameters which require rapid man-in-the-loop response be monitored by the Orbiter C&W.

In addition to the changes to the Orbiter C&W requirements, there were major changes to the Spacelab C&W provisions. The Spacelab C&W panel in the manned module no longer provides space for monitoring C&W parameters from the Spacelab payload (e.g., AMPS, Labcraft). The Orbiter can monitor a maximum of 50 C&W parameters from the payload (i.e., sufficient wiring and interfaces are provided). The Spacelab will be using 12 (per the Spacelab PDR-A data) of these parameters and the remaining 38 may be used by Spacelab payload. The Spacelab C&W panel provides C&W indications for 12 parameters and these are all presently

being used by the Spacelab. Therefore, the AMPS or Labcraft payloads will have to provide their own C&W system.

TRW has developed a system that is compatible with all the above conditions and it is depicted in Figure 9-6. This C&W system was developed with the following groundrules in mind:

- Emergency parameters only monitored on Orbiter C&W system
- Caution and warning parameters (not emergency parameters) monitored on a payload dedicated panel at the Orbiter aft flight deck (AFD)
- In-flight safing and configuring for landing performed at the payload station at the AFD
- Redundancy required for all safing operations
- Maximum utilization of the Spacelab and Orbiter C&W system required.

Our approach on AMPS C&W monitoring, as indicated in Figure 9-6, is to monitor all emergency parameters requiring rapid man-in-the-loop response by the Orbiter C&W panel. The primary mode for monitoring of caution and warning parameters (not emergency parameters) will be by a hardwired payload dedicated C&W system. Ground monitoring (via the Orbiter Engineering Data Handling System and TDRS) of C&W parameters will be used as a backup to AMPS primary C&W system.

The emergency parameters that are to be monitored during flight by the Orbiter C&W system for AMPS Flight 1 are: fire on the pallet or in the lidar, excessive EMI, excessive charging of Orbiter, and inadvertent ignition of barium-thermite canister.

The caution and warning parameters that are to be monitored by the AMPS dedicated C&W system for Flight 1 are shown in Table 9-10. A quick summary reveals that there are 61 warning parameters to be monitored and 73 caution parameters to be monitored. This large number of parameters does not conflict with the 38 parameters allocated to the payload because the Orbiter C&W system is not being used to monitor the parameters shown in Table 9-10.

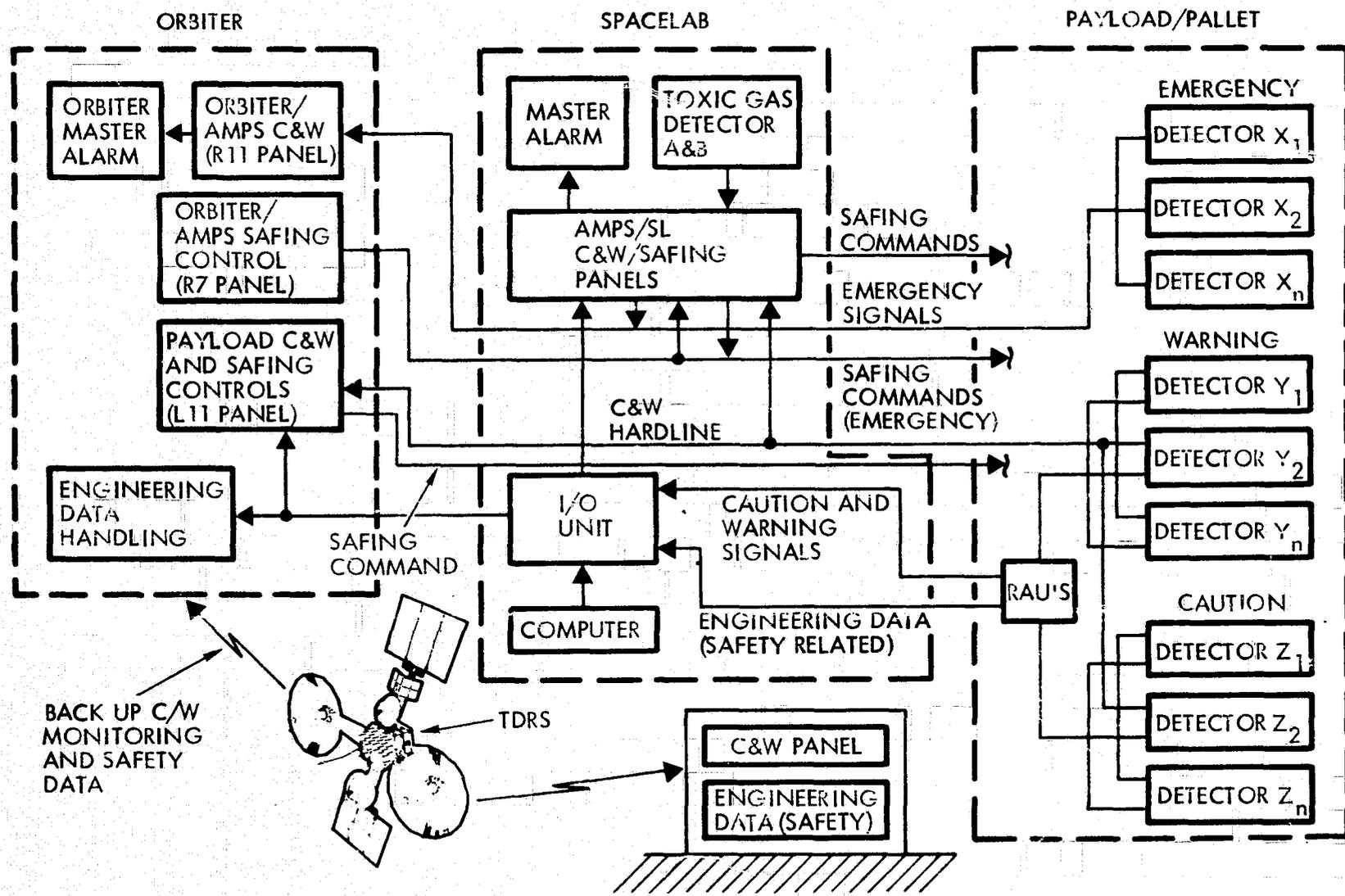


Figure 9-6. Caution/Warning and Safing System

Table 9-10. AMPS Flight 1 Caution and Warning Parameters

Category/Parameters	Warning	Caution
1) Excessive temperatures		
a) Low power processor temperature transducer		1
b) Six capacitor bank units - one temperature transducer per bank		6
c) Freon loop		1
d) LIDAR (power supplies and lasing cavity)	2	2
e) Electron accelerator (isolation transfer/control grid, power processor, high power processor)	3	
f) Electrical control equipment in Spacelab module		10
g) Cryogenic tanks temperatures	3	
h) Gas temperatures in pressure vessels		7
2) Thermal protection		
a) Loss of Freon coolant	1	
3) Fire/Smoke		
a) Fire detectors on pallet (emergency monitoring only)		
b) In Spacelab module, fire and smoke sensors	3*	
4) Stowage of payload for reentry		
<u>Pallet segment No. 1:</u>		
a) Replace ESP on cradle and lock	1	
b) Resecure RMS	1	
c) Command power off to ESP	1	
Accelerator		
a) Purge Level I gas	1	
b) Retract Level III	1	
c) Remove accelerator power	1	
d) Close and lock accelerator cover		1

* Spacelab provided

Table 9-10. AMPS Flight 1 Caution and Warning Parameters (Continued)

Category/Parameters	Warning	Caution
Common energy storage		
a) Remove CES power	1	
<u>Pallet segment No. 2</u>		
SIPS		
a) Boil off cryogen		1
b) Close canister covers	1	
c) Close star tracker optical covers	1	
d) Reorient, retract, and lock SIPS gimbals	1	
e) Power-off all SIPS electronics	1	
<u>Pallet segment No. 3</u>		
LIDAR		
a) Remove power from laser		1
b) Secure optics	1	
c) Close and lock cover	1	
d) Transfer dye fluids and flush dye system		1
e) Purge reservoir area with inert gas		1
5-Meter mast		
a) Retract mast and lock	1	
b) Remove power		1
Each gas release remaining		
a) No actions required		
5) Excessive pressure		
a) Xenon pressure vessels (5)		5
b) N ₂ pressure tank on accelerator		1
c) Chemical pump on cryogenic system		1
d) Cryogenic dewars	3	

Table 9-10. AMPS Flight 1 Caution and Warning Parameters (Continued)

Category/Parameters	Warning	Caution
6) Vacuum		
a) Cryogenic dewar walls		3
b) Accelerator vacuum	1	
c) SIPS/canister vacuum		1
7) Loss of pressure		
a) Common energy storage	6	
b) Ullage pressure of dye tank		1
8) Pyrotechnics		
a) Xenon gas releases (one per vessel) (armed monitor)	5	
b) Lidar cover pyrotechnics (arm monitor) (assume redundant pyrotechnics)	2	
c) SIPS pyrotechnics (arm monitor)	2	
d) 5-Meter mast (assumed redundant pyrotechnics) (arm monitor)	2	
9) Experiment operations		
a) RMS/ESP operation		
b) SIPS operation		
● Rotation control		1
● Inadvertent deployment (armed)	1	
c) Lidar cover operation (armed)	1	
d) Accelerator beam control		
● Magnetometer - failure	1	
● Magnetometer - operation parameters	2	
e) Electrostatic charging		
● E&B (3)	3	
● Line noise monitor (5)		5
● Retarding potential analyzer (2)		2
● Failure of electron accelerator static discharge device		1

Table 9-10. AMPS Flight 1 Caution and Warning Parameters (Continued)

Category/Parameters	Warning	Caution
f) Lidar operation		
● Cover opened		1
● Inadvertent firing (around 2 lasers) (armed)	2	
● Mirror misalignment	2	
g) Gas release system deployment		
● Inadvertent deployment		5
● Inadvertent timer actuation	5	
h) 5-Meter boom		
● Inadvertent extension	1	
10) Hazardous gases		
a) H ₂ in cargo bay	1*	
b) Ethyl alcohol vapors	1*	
11) Toxic gases		
a) Spacelab module (2 sets of instrument racks)	2	
12) Electrical power		
a) Electron accelerator		
● Charging circuit for common energy storage		6
● Voltage control in instrument high voltage power processor	2	
b) Lidar power supply		
● Spacelab power supply J-bos		
Monitor circuit breaker position	3	
Monitor circuit breaker failure (not presently provided by the Spacelab power distribution box)		3
● Lidar power conversion equipment		3
*Orbiter provided		

Table 9-10. AMPS Flight 1 Caution and Warning Parameters (Continued)

Category/Parameters	Warning	Caution
13) Accelerator system a) Failure of isolation transformers b) Inadvertent firing <ul style="list-style-type: none"> ● Inadvertent charging of capacitor ● Inadvertent discharging of capacitor 	1	6

9.8 RELIABILITY

Our reliability analysis was directed toward achieving reliability goals and developing a preliminary model.

9.8.1 Reliability Design Goals

The AMPS mission and hardware complements were analyzed in order to assess the capability of achieving the following reliability goals:

- No single point failure (SPF) will result in injury to the crew, or the loss of a major portion of the objectives for the AMPS mission
- A single failure will not result in:
 - a) Nullification of system redundancy
 - b) A requirement for extensive or immediate crew intervention
- No second failure in the same hardware element will preclude successful abort of the mission
- A failure of an instrument or FSE will not propagate into other instruments or other FSE.

The results of these analyses indicate that these goals can be achieved.

9.8.2 Reliability Model

A preliminary reliability model (Figure 9-7) was developed for Flight 1. Analysis of this model indicates the presence of two critical links: the SIPS and the ESP. The ESP is critical to the Environmental Observations experiment while the SIPS is critical to the Solar Flux Radiation experiment. Although the ESP and SIPS are used in the Electron Accelerator and Beam Plasma experiments, each serves as a redundant path to the other and therefore neither is critical by itself. Also, for the Minor Constituents experiment, a redundant path exists for the SIPS.

Further analysis of this model in conjunction with the experiment timelines led to the conclusion that a full 7-day experiment sequence can be completed even with the failure of a major item of equipment (Figure 9-8). This type of redundancy and workaround analysis is being performed on a continuing basis at TRW using computer simulations to optimize the expected value from each AMPS flight.

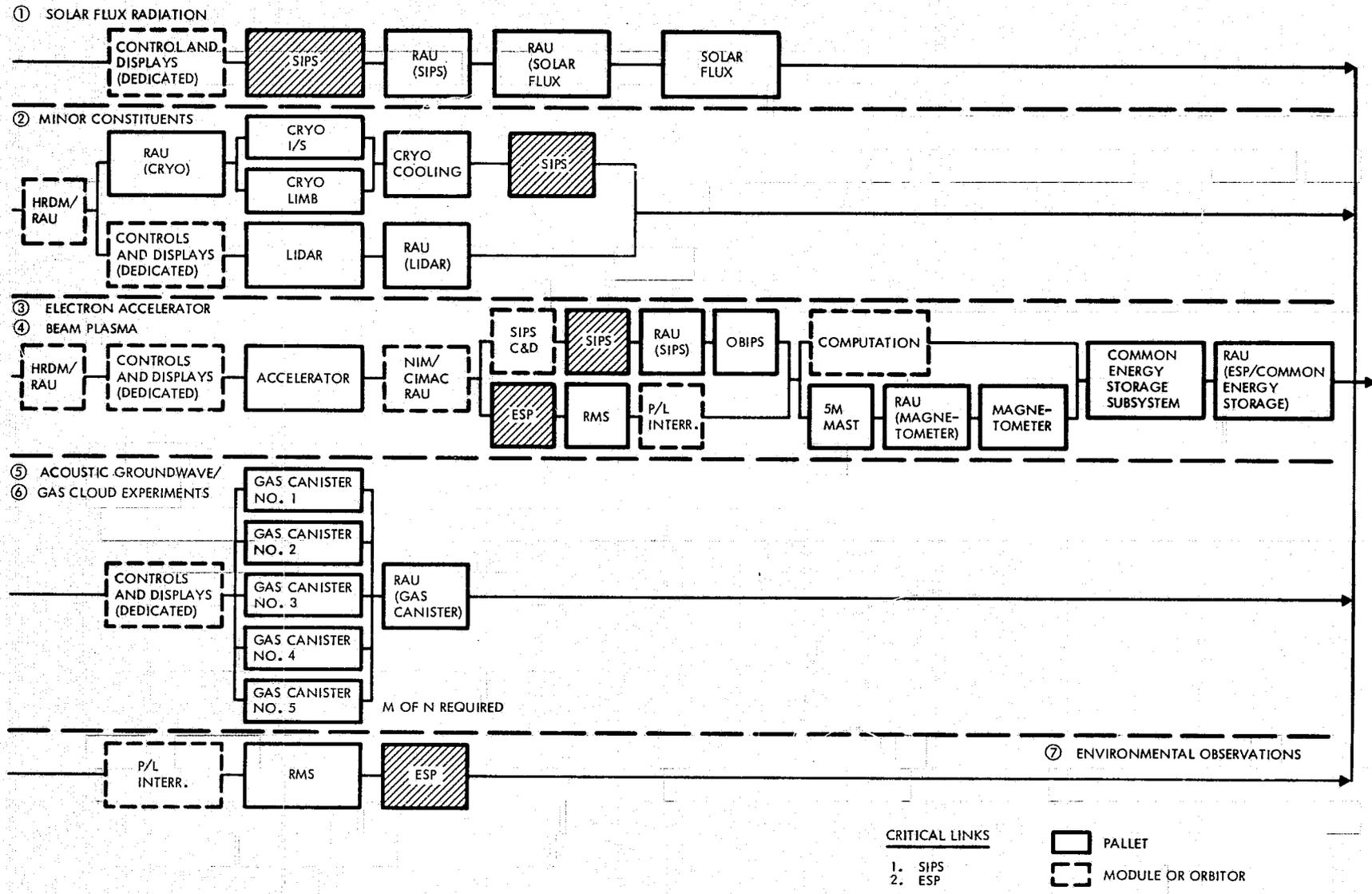


Figure 9-7. Reliability Model

A. CONTINGENCY MODE OPERATIONS - HOURS		FAILED ITEM			
EXPERIMENT	ALL UP	LIDAR	ESP RMS	SIPS	ACCEL.
SOLAR FLUX RADIATION	9	9	9	0-9	9
ENVIRONMENTAL OBSERVATIONS	62	62	0-62	62-86	62-88
OZONE DENSITY - LIDAR	26	0-26	26	26	26
MINOR CONSITUENTS - CRYO	41	41	41-103	0-41	41-67
ACOUSTIC GRAVITY WAVE GAS CLOUD EXPANSION	5	5	5	5	5
ACCELERATOR - FIRST GENERATION	37	37	52-37	0-37	0-37
ACCELERATOR - BEAM PLASMA	15	15	0-15	52-15	0-15
TOTAL EXPERIMENT TIME*	169	169	169	169	169

* SOME EXPERIMENTS ARE CONDUCTED SIMULTANEOUSLY.

B. MAXIMUM EXPERIMENT/INSTRUMENT OPERATING TIME

INSTRUMENT	MAXIMUM TIME - HOURS	LIMITATION
SOLAR FLUX MONITOR	9	MEASUREMENT REQUIREMENTS
ESP	≈ 96	BATTERY LIFE
LIDAR	26	DYE LIFE
GAS RELEASE	5	RELEASE OPPORTUNITIES
ACCELERATOR	52	NIGHT FIRING OPPORTUNITIES
CRYO INSTRUMENTS	154	CRYOGENICS

Figure 9-8. Experiment Contingency Analysis