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Mars Observer

JPL Contract 959444

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PHASE 0

SAFETY REVIEW

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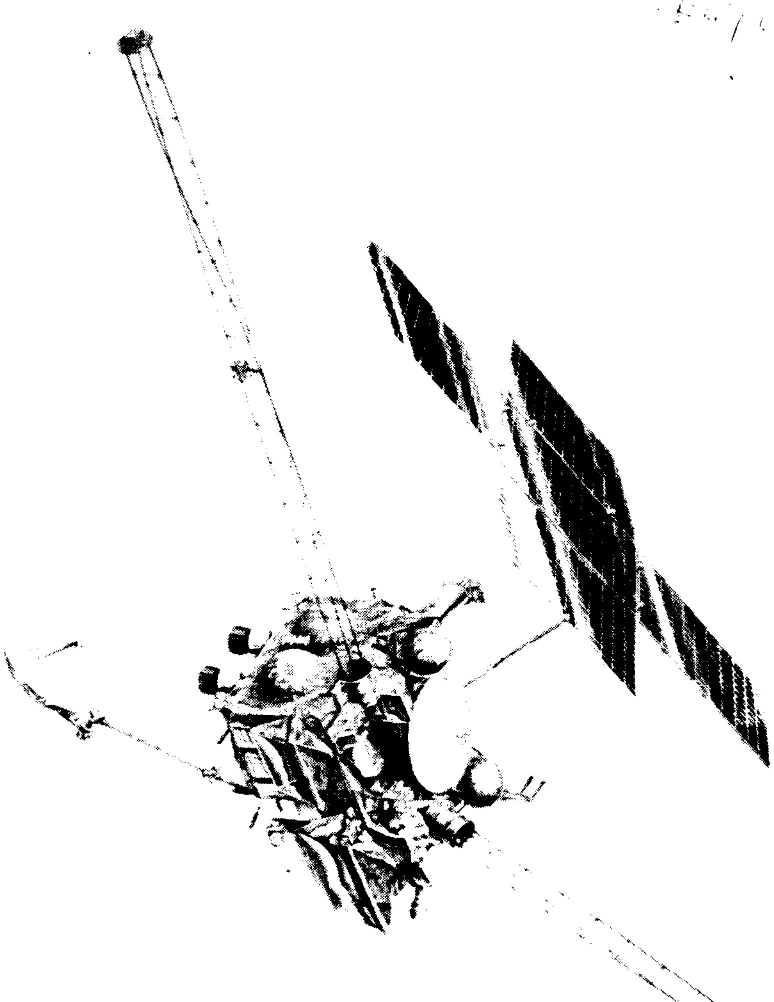

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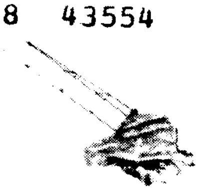

Astro-Space Division

A DIVISION OF RCA AEROSPACE AND DEFENSE

This report was prepared for the Jet Propulsion Laboratory, California Institute of Technology, sponsored by the National Aeronautics and Space Administration.

MARS OBSERVER

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION/ SPACE TRANSPORTATION SYSTEM PHASE 0 SAFETY REVIEW DATA PACKAGE

TG-27886

November 17, 1986

Prepared By: RCA Astro-Electronics
P.O. Box 800
Princeton, New Jersey 08540

Prepared For: California Institute Of Technology
Jet Propulsion Laboratory
Pasadena, California 91109

Contract No. 957444
DRD NO. SA004 & SA007

George Pace
Manager
Mars Observer Spacecraft
JPL

Larry Montgomery
Manager - Flight Projects
System Safety Office
JPL

PREFACE

This Phase Zero NASA Safety Review Data Package has been prepared by RCA Astro-Electronics Division for the California Institute Of Technology Jet Propulsion Laboratory. RCA has made maximum use of the experience gained with the NASA safety process on previous spacecraft programs in the preparation of this package. This package also reflects the continuing effort at RCA to refine our spacecraft safety packages. We believe that this package is totally responsive to the NASA safety requirements (as detailed in NHB 1700.7A, and KHB 1700.7A). This package has been prepared as outlined in JSC-13830A and the Mars Observer safety analysis has been performed using the guidelines in JSC-11123, as specified in JSC-13830A. RCA is proud to present the Mars Observer Phase Zero NASA Safety Review Data Package.

Dr. Ron C. Maehl
Manager
RCA Mars Observer Program

P. F. Kaskiewicz

Nicholas P. Ruggeri
Product Assurance Administrator
RCA Mars Observer Program

Paul F. Kaskiewicz
Systems Engineering Manager
RCA Mars Observer Program

C. Larsen

Charles R. Larsen
Manager
RCA Space Transportation Group

David D. Huff
Lead Engineer - Mars Observer
RCA Space Transportation Group

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ACRONYMS AND ABBREVIATIONS

4πSS	4π Steradian Sun Sensor
AACS	Attitude and Articulation Control Subsystem
ASC	American Satellite Corporation
ASE	Airborne Support Equipment
ATN	Advanced TIROS-N
BCA	Battery Charge Assembly
BRU	Battery Reconditioning Unit
BVR	Battery Voltage Regulator
C&DH	Command and Data Handling
CAD	Computer Aided Design
CIU	Controls Interface Unit Annex
CMU	Command Master Unit
CPC	Controls Power Converter
CPU	Central Processing Unit
CSA	Celestial Sensor Assembly
CSIE	Checkout Station Interface Electronics
CXU	Controls Interface Unit Annex
DBS	Direct Broadcast Satellite
DET	Direct Energy Transfer
DOD	Department of Defense
DSS	Dynamic Spacecraft Simulator
DTR	Digitial Tape Recorders
EASE	Electrical Airborne Support Equipment
EDF	Engineering Data Formatter
EGSE	Electrical Ground Support Equipment
EMI	Electromagnetic Interference
EMR	Electromagnetic Radiation
EVA	Extra-Vehicular Activity
GPC	General Purpose Computer
GRS	Gamma Ray Spectrometer
GSE	Ground Support Equipment
HPF	Hazardous Processing Facility
HZ	Hertz
ITOS	Improved TIROS
JSC	Johnson Space Center
KBPS	Kila-Bits Per Second
KHB	Kennedy Handbook
KSC	Kennedy Space Center
LSE	Launch Site Equipment
MAG	Magnetometer
MASE	Mechanical Airborne Support Equipment
MDM	Multiplexer - Demultiplexer
MGSE	Mechanical Ground Support Equipment
MHSA	Mars Horizon Sensor Assembly
MLP	Mobile Launch Platform
MMH	Monomethyl Hydrazine
MON-1	Nitrogen Tetraoxide + 0.8% Nitrous Oxide
MSFC	Marshall Space Flight Center
MVS	Majority Vote Sequencer

ACRONYMS AND ABBREVIATIONS

NHB	NASA Handbook
PC	Power Converter
PDI	Payload Data Interleaver
PDS	Payload Data Subsystem
PMD	Propellant Management Device
PMV	Power Master Unit
POCC	Payoload Operations Control Center
ROM	Read Only Memory
PPF	Payload Processing Facility
PRA	Pyrotechnic Relay Assembly
PROM	Programmable Read Only Memory
PSE	Power Supply Electronics
RAM	Random Access Memory
REA	Reaction Engine Assembly
RLE	Remote Located Equipment
RSS	Rotating Service Structure
RWA	Reaction Wheel Assembly
S/C	Spacecraft
SAD	Solar Array Drive
SCP	Standard Controls Processor
SCS	Spacecraft Checkout Station
SCU	Signal Conditioning Unit
SEPET	System Electrical Performance Evaluation Test
SIU	Shuttle Interface Unit
SRM	Solid Rocket Motor
SSP	Standard Switch Panel
SSV	Space Shuttle Vehicle
STE	Special Test Equipment
STS	Space Transportation System
TBD	To Be Determined
TBS	To Be Supplied
TCE	Thermal Control Electronics
TLM	Telemetry
TUTS	T-0 Umbilical Test Set
VPF	Vertical Processing Facility
XSU	Cross Strap Unit

1.0 The Mars Observer Spacecraft

The Mars Observer is a planetary exploration spacecraft that will be used to explore Mars. The spacecraft is being built for the California Institute Of Technology Jet Propulsion Laboratory (JPL) by RCA Astro-Electronics Division. RCA, as the prime contractor, provides the spacecraft bus and its associated subsystems and conducts the system integration and test activities for the entire flight system. The scientific payload is provided by JPL to RCA for integration with the spacecraft bus. The scientific payload subcontracts are managed by JPL. This package is directed toward the RCA provided bus and spacecraft subsystems while the scientific payload is discussed in an appendix to this package to be supplied by JPL.

1.1 Mission Overview

The Mars Observer Program has as its primary objectives a study of the geochemistry, atmospheric dynamics, atmosphere/surface interactions, seasonal variations and magnetic field characteristics of Mars.

The instrument complement is a gamma ray spectrometer, a radar altimeter, an imaging system, a pressure modulated infrared radiometer, a thermal emission spectrometer, a magnetometer, an ultra-stable oscillator (for radio science) and a visual and infrared mapping spectrometer.

These instruments were selected for their synergistic complementarity as well as their individual capabilities with the aim of providing an overall, comprehensive global data set to allow the systematic interdisciplinary study of Mars required to achieve the overall program goals.

The Mars Observer spacecraft was specifically designed with the objective of accommodating the science requirements of this set of instruments, consistent with the requirement to utilize existing hardware and designs as dictated by the overall Planetary Observer philosophy. The hybrid RCA Satcom/TIROS/DMSP design fulfills both of these requirements simultaneously.

The strategy to be used for MO is to operate all instruments essentially continuously, thereby providing a global data set from which seasonal variations and a comprehensive understanding of planetary morphology can be derived. This includes study of both large scale features at several spectral bands and very fine scale features (down to 1 meter surface resolution by the Mars Observer Camera) in regions of interest.

Another key issue is the question of the volatile content of the Martian surface and its seasonal variation. A sun-synchronous orbit was specifically chosen to eliminate diurnal variations of observations so that longer term effects can be unambiguously assessed.

In addition, the MO spacecraft will carry a sensitive magnetometer to determine if Mars has an intrinsic planetary field and, if so, its magnitude. This is a key scientific topic not only for the understanding of Mars but also for comparative planetology.

1.2 Spacecraft Description

The system design for the Mars Observer Spacecraft is based predominantly on the Structure, Propulsion, and Thermal subsystems, and the Solar Array assembly of the RCA Satcom Ku-Band spacecraft, and the Attitude and Articulation Control, Command and Data Handling, and Power Subsystem of the RCA Advanced TIROS-N and DMSP Block 5D-2 spacecraft. The TIROS and DMSP programs are long running Atlas-launched RCA spacecraft with proven on-orbit performance. The only Mars Observer assembly-level items without flight heritage on RCA spacecraft are the Bipropellant portion of the Propulsion Subsystem and the High-Gain Antenna (HGA) assembly, a component of the Telecommunications Subsystem. The majority of the components for these items are however flight qualified and have been flight proven in several spacecraft applications.

A pre-assembly view of the Mars Observer Spacecraft is shown in Figure 1-1. The spacecraft mass is 723.4 Kg, and propellant mass is 1345.9 Kg. The TOS adaptor mass is 40.1 Kg. The total spacecraft mass at liftoff (less the TOS upper stage) is 2155.0 Kg (this figure includes a margin of 45.6 Kg, which will be used). The paragraphs below provide an overview of the various spacecraft subsystems. Safety critical spacecraft subsystems are described in more detail in section 2.0 of this package.

1.2.1 Communications

The Mars Observer Telecommunications Subsystem is all new to meet the unique requirements of the MO mission. The Telecommunications Subsystem of TIROS/DMSP and Satcom K are not applicable to the MO mission.

The design incorporates two GFP NASA X-band transponders (NXTs) and two GFP command detector units (CDUs). It also includes two 40W (rf output) traveling wave tube amplifiers (TWTAs), a 1.5-meter diameter HGA with 2-axis gimbals and associated redundant control electronics, a hybrid mixer, passive rf components (switches, filters, etc), an assembly of six low-gain antennas (LGAs), and a medium-gain antenna (MGA).

Apart from the GFP, only the NXTs are generically new to the RCA candidate spacecraft buses.

The Telecommunications Subsystem provides the DSN-compatible X-band communications to and from the Earth for radiometric tracking, telemetry, commanding, and radio science. Command access through any of the three antenna systems is continuously available. Transmission mode (low-, medium-, or high-gain) is selectable by ground command.

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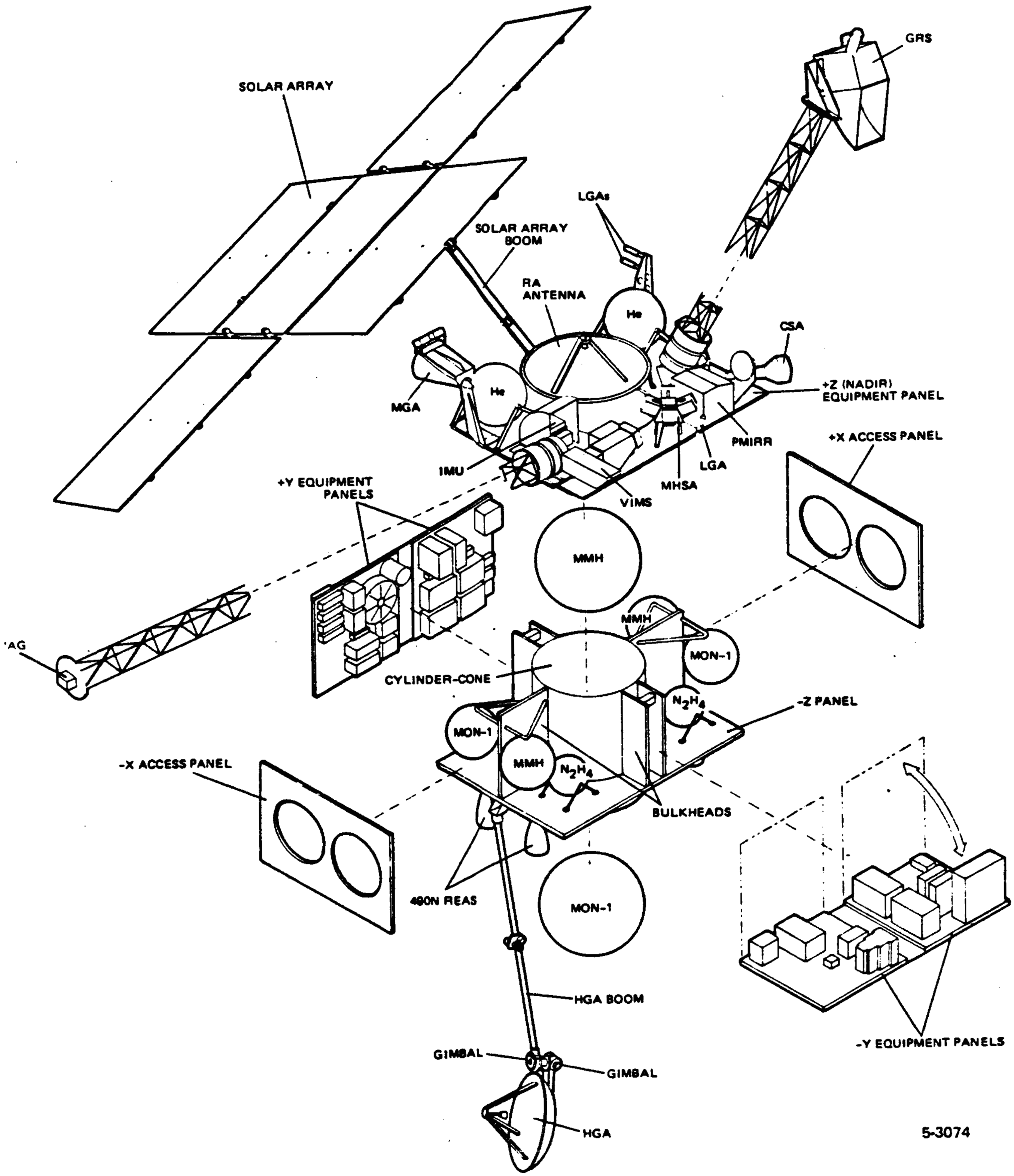


Figure 1-1. Pre-Assembly View of the Mars Observer Spacecraft

1.2.2 Command And Data Handling

The Command And Data Handling (C&DH) Subsystem is a direct application of flight proven TIROS/DMSP equipment, modified to incorporate the GFP Payload Data Subsystem (PDS).In the command role to control and reconfigure the spacecraft, it receives demodulated incoming signals, performs on-board computation, executes both received and computer-generated commands, and provides precision clocks for use throughout the spacecraft. In the telemetry role, the C&DH Subsystem synchronizes, times, collects, digitizes, multiplexes, formats, routes, stores, and plays back science instrument data, command verification, CPU memory dump, and payload and spacecraft engineering data.

1.2.3 Attitude And Articulation Control

The Attitude and Articulation Control Subsystem (AACS) is derived from the DMSP Block 5D-2 design, in particular that of Flight S-15.

The AACS, in conjunction with the control segment of the C&DH Subsystem and the Propulsion and Pyrotechnics Subsystem, provides all functions associated with attitude sensing and control, appendage articulation, powered flight guidance, and navigation throughout all phases of the mission. It is comprised of an Inertial Measurement Unit (IMU) which includes four gyros and a redundant accelerometer package, a Mars horizon sensor assembly (MHSA), a 6-slit strapdown celestial mapping sensor (CSA), a 4π steradian sun sensor (4πSS), four reaction wheel assemblies (RWAs), and software.

1.2.4 Electrical Power

The power subsystem is derived from the TIROS/DMSP boost-discharge Direct Energy Transfer (DET) design. The basic DET design has been tailored to meet the environmental and load requirements associated with the Mars Observer mission. The power subsystem provides the required generation, storage, control and distribution of electrical power for the spacecraft during all mission phases.

The Mars Observer Solar Array has two deployment configurations. For the cruise and orbit insertion phases of the mission, the solar array is partially deployed with the boom still stowed. During the mapping and quarantine phases the solar array is fully deployed. The mission phases are discussed in section 1.5.

1.2.5 Thermal Control

The thermal design of the Mars Observer Spacecraft is derived from the RCA TIROS/DMSP, and Satcom Ku-Band Spacecraft. This basic, flight proven, design has been tailored to the widely varying thermal environments associated with the Mars Observer mission. Thermal control is accomplished by passive means where possible, with minimum resort to quasi-active devices (such as heaters and louvers).

The thermal design of the Mars Observer Spacecraft has been carefully optimized with a special view to power consumption, utilization of external fluxes, and minimum mass. The required thermal control is accomplished by the judicious application of multilayer insulation blankets, heaters, thermal control electronics, insulating and conducting structural elements, radiators, surface finishes, tapes, and louvers as well as configuration design of the spacecraft constituent elements, thermal coupling of body-mounted science GFP with the equipment enclosure, and use of the permanently space-viewing side (-Y) for radiative detector cooling and heat shedding.

1.2.6 Structure

The Structure, Cabling, and Mechanisms for the Mars Observer Spacecraft bus are derived from the primary structure of Satcom Ku-Band. The rationale for this choice is to minimize mass, to ensure compatibility with the TOS upper stage, to reduce thermal heater load requirements by nesting propulsion tanks inside the structure in close proximity to electronic equipment, and to use an existing flight qualified structure. The only major changes form the Satcom Ku-Band are in the Solar Array geometry and deployment sequencing.

1.2.7 Propulsion

The Mars Observer Propulsion Subsystem provides all the attitude control, trajectory corrections, and orbit insertion and trim maneuvers necessary to reach Mars and to extensively map the martian surface. Accomplishment of these maneuvers is performed by two independent propulsion systems, one employing hydrazine, the other bipropellants. The attitude control, Mars-orbit trim system uses flight-proven, long-life, RCA Satcom hydrazine blowdown propulsion system design and hardware. A state-of-the-art pressure-fed, Earth-storable, bipropellant system provides the desired higher performance for major mission maneuvers.

The propulsion systems proposed are typical of those built by RCA for its communications and environmental spacecraft. RCA has flown or has in-build more than 27 communications-spacecraft hydrazine propulsion systems identical to the blowdown system proposed. Two of these systems have each demonstrated more than 8-1/2 years of flawless operation prior to propellant depletion and satellite retirement from service. For the TIROS/DMSP weather spacecraft, RCA has built, or has in process, 25 pressure-fed regulated multi-engine, high-thrust, hydrazine ($\geq$ 450N)/ nitrogen propulsion systems.

All the propulsion subsystem components are flight qualified and most have been flight-proven on RCA spacecraft with the exception of the 0.94-m (37-in.) diameter propellant tank, which will be qualified for Mars Observer. Qualification is an extension of RCA's existing tank design technology. RCA was the first to verify use of capillary propellant management devices for propellant control on synchronous spacecraft and has qualified a series of tanks of increasing diameter since RCA Satcom-F1 flight in 1975.

1.3 Scientific Payload

The spacecraft scientific payload description and hazard reports are provided by JPL in an appendix to this safety package.

1.4 TOS/STS Interface

The Mars Observer spacecraft will use the high performance Transfer Orbit Stage (TOS) which will be GFE (provided by MSFC). The TOS program is managed by Orbital Sciences Corporation (OSC) with design, development, production, and operations being handled by the Martin Marietta Corporation. The Mars Observer/TOS/STS interface is described in the following sections.

1.4.1 Electrical Interface

Electrical interfaces between the Mars Observer spacecraft and the TOS, as shown in figure 1-2, fall into two categories:

- o Spacecraft-to TOS avionics interfaces.
- o Spacecraft-to-Spacecraft EASE using the TOS stage as a pass-through

The spacecraft EASE is comprised of a Power Master Unit (PMU) and a Command Master Unit (CMU). Interfaces between the spacecraft EASE and the TOS EASE consist solely of an Orbiter power feed from the TOS EASE terminal board and is fused by the spacecraft EASE.

All spacecraft to TOS electrical interfaces use circular separation connectors located at the spacecraft separation joint. A wiring harness on the spacecraft adaptor links these connections with the adaptor-to-TOS connectors. RCA will be supplying this wiring harness and both matching halves of the connectors.

The spacecraft-to-TOS avionics interfaces consist of connections from the spacecraft to the TOS Majority Vote Sequencer (MVS). After SRM burnout, redundant signals from the TOS Majority Vote Sequencer called "system Pre-Arm" and "System Arm", will drive latching relays of the same names housed in the spacecraft which pre-arm and arm the spacecraft pyrotechnic, thruster fire, and thruster latch valve command buses. Shortly before the prescribed spacecraft/TOS separation time, a similar "Separation Ready" signal will be sent from the MVS to the spacecraft computer which will subsequently command previously armed pyro-actuated propellant isolation valves to open. The spacecraft computer will then command the spacecraft/TOS Marmon clamp to separate. Redundant design techniques employed in the MVS combined with the "three inhibit" design of spacecraft pyrotechnic and propulsion control circuits precludes the possibility of a two-failure hazard as is prescribed by NHB 1700.7A. These functions will also tolerate a single-point failure in the MVS or the spacecraft without jeopardizing mission success.

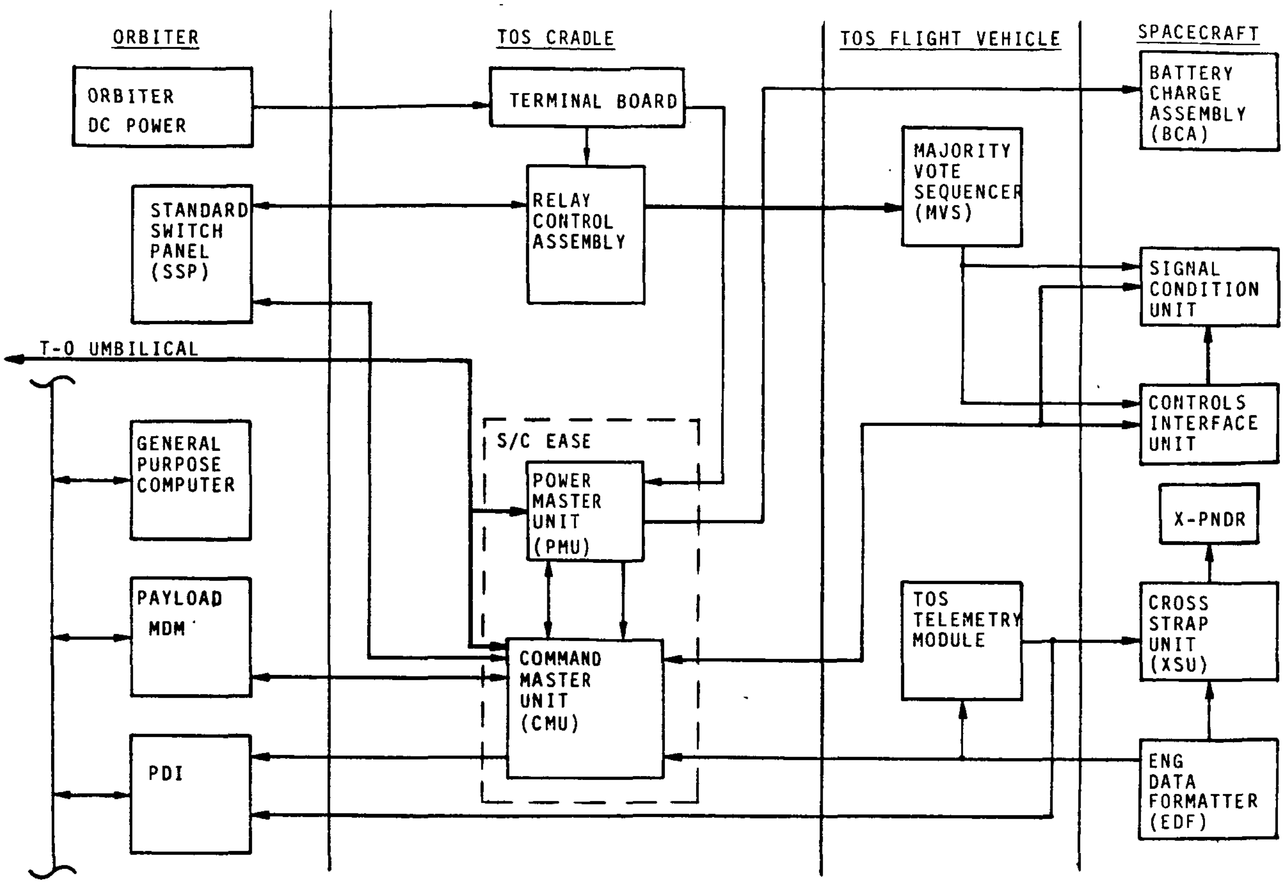


Figure 1-2. Spacecraft/TOS/Orbiter Electrical Interfaces.

The baseline is to not include a telecommunications package on the TOS flight vehicle and to use the X-band transmitter and digital tape recording facilities on the Mars Observer spacecraft to transmit and/or record TOS telemetry after deployment from the Orbiter. As shown in figure 1-2, a serial telemetry stream (at 1.5 kbps) of engineering data is sent from the spacecraft Cross-Strap Unit (XSU) to the TOS telemetry encoder where the stream is combined with TOS telemetry (which is at 14.5 kbps) to produce a combined spacecraft and TOS stream at a 16-kbps rate. The combined stream will be sent back to the XSU where it will be routed to the X-band transmitter and the spacecraft digital tape recorder. The combined stream will also be sent from the TOS telemetry encoder to the Orbiter Payload Data Interleaver (PDI) for crew display and transmission to the ground. Aware of PDI requirements on the format of telemetry frames, RCA and OSC are developing strategies for creating a combined-stream format compatible with the PDI. The TOS encoder/spacecraft electrical interfaces, which will subsequently be defined in detail, will minimize coupling with other spacecraft/TOS circuits in the EMI/EMC environment.

The spacecraft will also pass electrical interfaces through the TOS stage to the spacecraft EASE. Serial spacecraft commands (at 2 kbps) will be sent from the spacecraft EASE to the spacecraft upon crew initiation via the Orbiter Standard Switch Panel (SSP) or GPC Keyboard/MDM, or upon transmissions from the Ground Support Equipment (GSE) via the T-0 umbilical. Serial spacecraft telemetry (at 2 kbps) will be sent across the TOS to the spacecraft EASE for the PDI or for transmission to the GSE over the T-0 umbilical. A discrete command from the SSP, through the spacecraft EASE and through the TOS ASE will be supplied to the spacecraft to allow safing of all hazardous spacecraft functions simultaneously. Orbiter power conditioned by the spacecraft EASE will be provided to the spacecraft through the TOS. Maximum anticipated power to be delivered to the spacecraft is 300W. Power feeds on the TOS ASE, on the TOS itself, and on the spacecraft are being sized according to the requirements of JSC-07700, Volume XIV. Signal feeds are being designed and shielded to accommodate the EMI/EMC requirements of the same document, and TOS and spacecraft grounding will be to a single point on the TOS ASE.

1.4.2 Mechanical Interfaces

Figure 1-3 shows the spacecraft/upper stage cargo element for the Mars Observer mission. The spacecraft is cantilevered from the upper stage through a conical adapter and is located coaxially with the Orbiter cargo bay centerline. The allowable envelope for the cargo element, to include dynamic and thermal distortion effects, is the Orbiter payload clear envelope. The spacecraft lies well within this envelope.

The adapter provides the interface flange for the bolted attachment to the upper stage and, in addition, limits the conductive thermal coupling between the spacecraft and TOS. Figure 1-4 shows the preliminary design of the adapter which is currently baselined for the Mars Observer mission. The Mars Observer design uses the standard unmodified TOS structural interface. This 91.91-inch (2335-mm) diameter forward ring frame structure contains 120 threaded mounting holes on a 90.6-inch (2301-mm) base circle. Indexing (clocking) of the attachment holes is achieved by a 1° offset in the position of the hole on the $+X_{MO}$ $(+Z_T)$ axis, as shown in the figure. This indexing controls the MO/TOS coordinate system relationship to that shown in Figure 1-5.

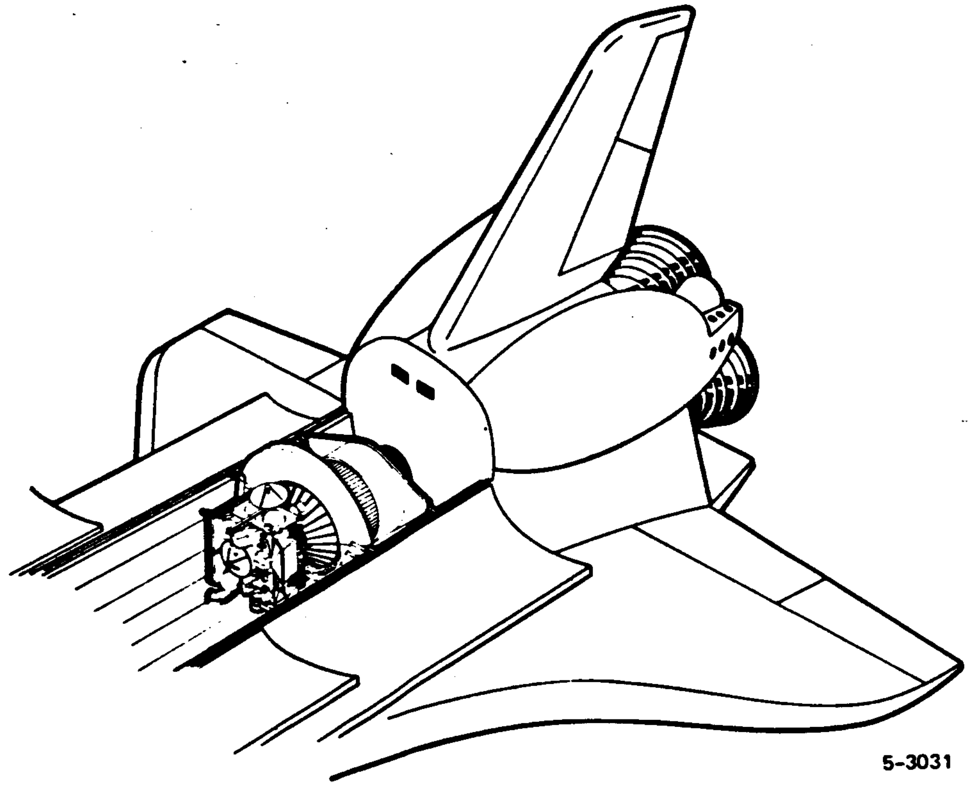


Figure 1-3 Mars Observer/TOS Cargo Element

To ensure adapter/TOS interface compatibility, the attachment holes in the RCA adapter will be controlled by a drill fixture supplied by OSC, and an adapter/TOS fit check will be performed at OSC as part of the spacecraft/upper-stage integration effort. The flight attachment hardware will be supplied by OSC and installed by OSC personnel with support from RCA. Access to the attachment bolts is gained from the spacecraft side of the interface.

Analysis of the loading across the structural interface will be performed during Mars Observer program development to verify the integrity of the mechanical joint. The effect of the loading transferred to the TOS forward ring frame from the conical spacecraft adapter (kick loads) will be analyzed by OSC and the adequacy of the structure verified. The ring frame is sized to support a cantilevered mass of up to 9,000 lb (19,841 kg) with a CG located up to 100 inches (2540 mm) forward of the interface. As the Mars Observer spacecraft and adapter weigh 2155 kg and have a combined CG located 1255 mm forward of the interface, no problems are anticipated.

The Electrical Airborne support Equipment (EASE) for the Mars Observer spacecraft (namely the Power Master Unit and the Command Master Unit) are mounted on the TOS aft cradle ASE at the two locations shown in Figure 1-6. At these locations, the spacecraft ASE lies well within the Orbiter payload envelope. The spacecraft ASE is mounted on the "hatband" secondary structure which is the standard method of attaching flight hardware to the TOS aft cradle primary structure. This allows flexibility in the attachment of Mars Observer avionics with no impact on the design of the primary (cradle) structure.

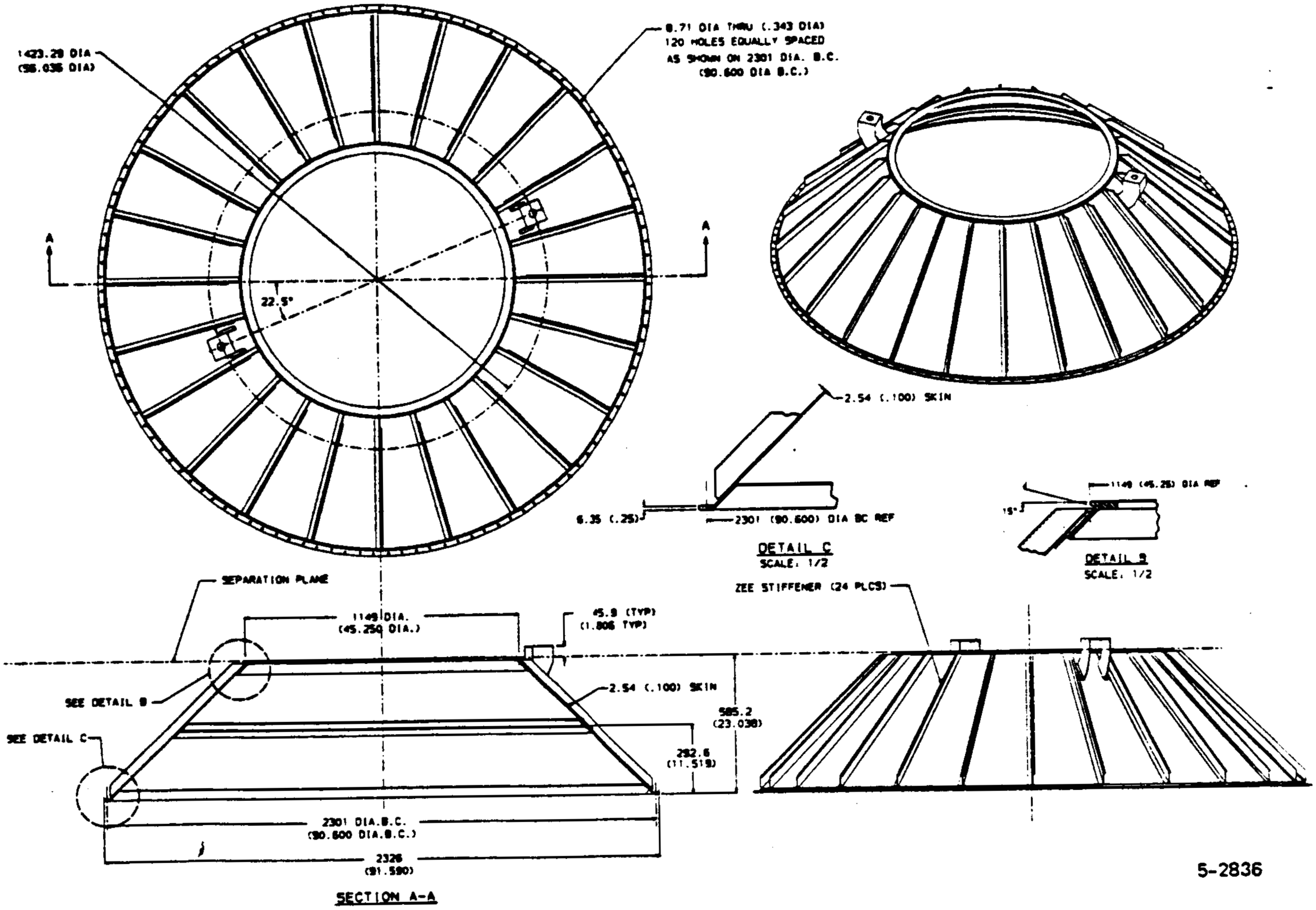
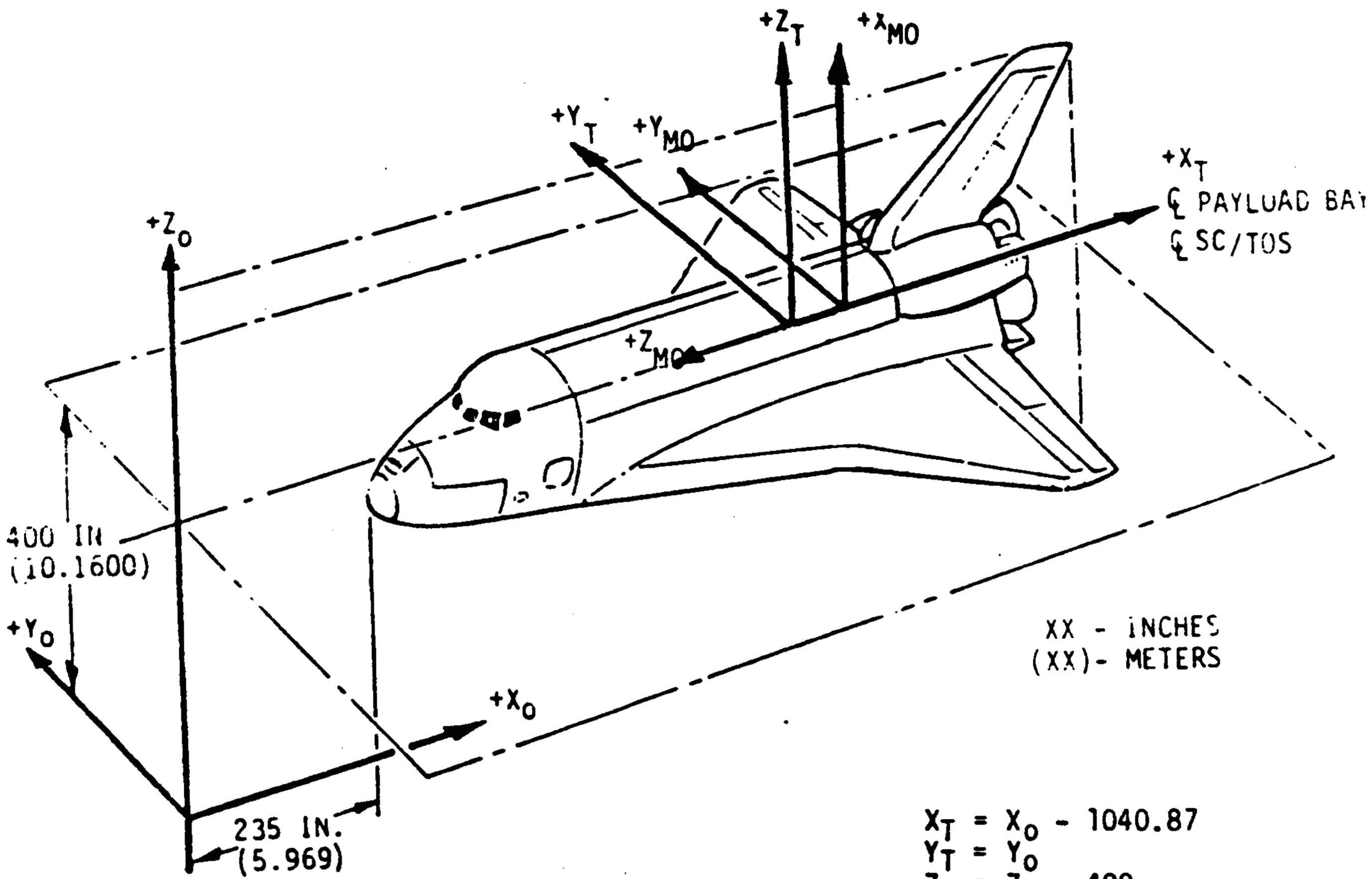


Figure 1-4. Spacecraft/TOS Adaptor



$X_T = X_O - 1040.87$
$Y_T = Y_O$
$Z_T = Z_O - 400$

$X_{MO} = Z_T$
$Y_{MO} = Y_T$
$Z_{MO} = 99 - X_T$

NOTE: These are structural axes not flight control axes

Orientation and Labeling:

The Standard Subscript for TOS is "T", Mars Observer Spacecraft is "MO", and the Orbiter is "O".

Origin of the TOS coordinate system is 123.96 inches forward (in the $-X_O$ direction) of the spacecraft Forward Ring Frame interface.

Origin of the spacecraft coordinate system is at the separation plane between the spacecraft and its TOS adapter. This separation plane is located at X_T = 99 inches.

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Figure 1-5. Spacecraft/TOS/Orbiter Coordinate System.

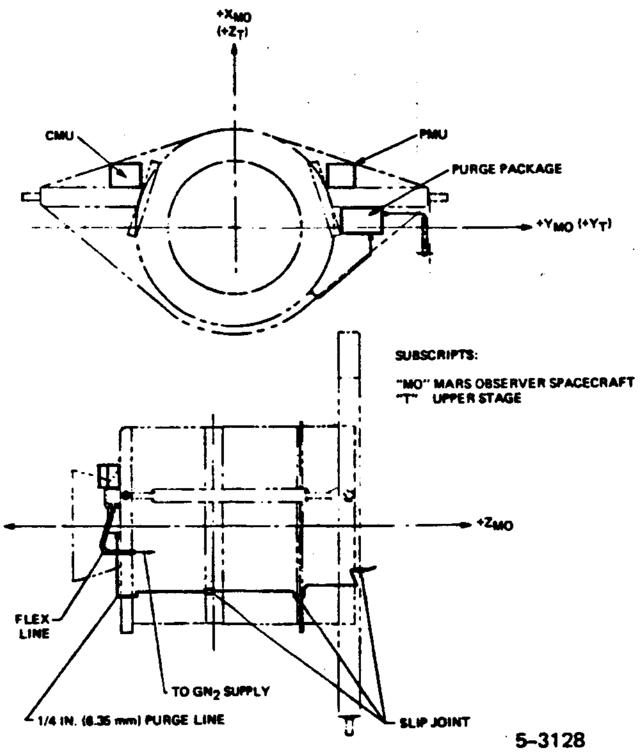

Figure 1-6. Equipment Mounting on TOS Cradle

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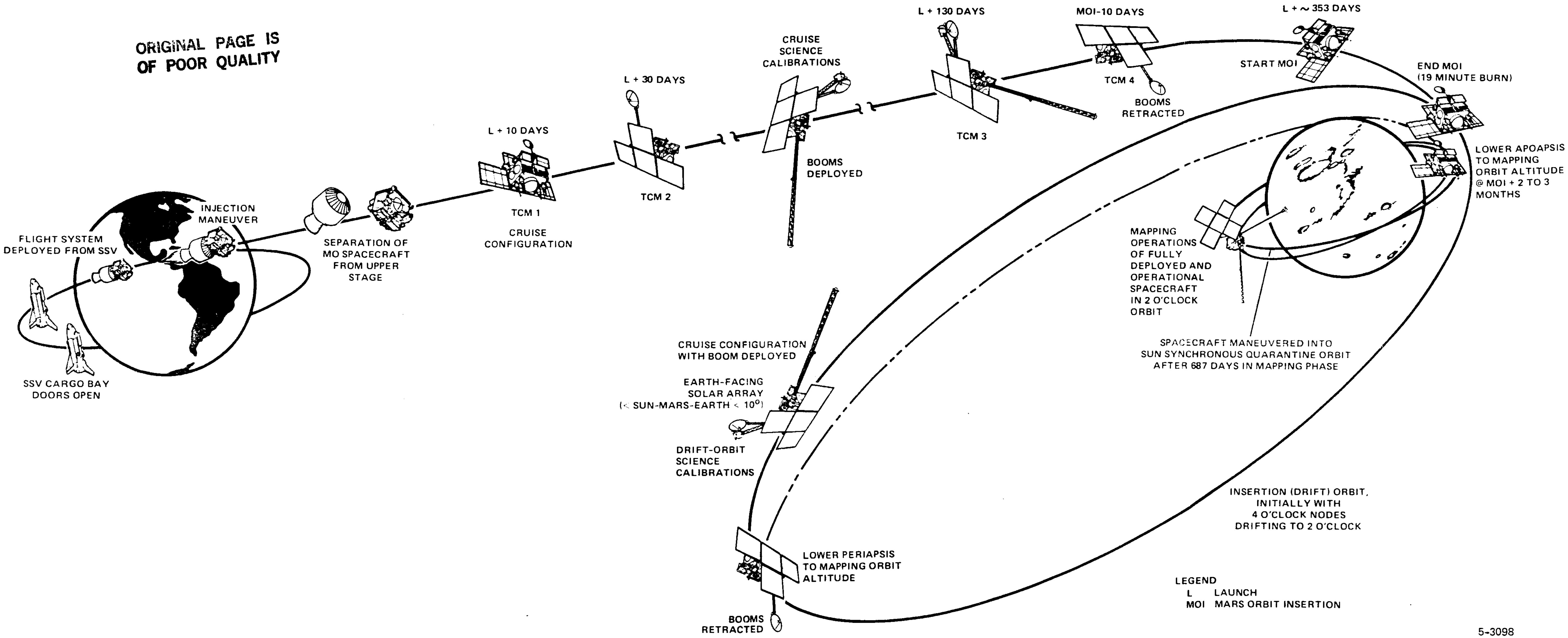
1.5 Mission Scenario

An overview of the Mars Observer mission is depicted in figure 1-7. The mission comprises five basic phases: launch, cruise, orbit insertion, mapping, and quarantine. Figures 1-8 and 1-9 illustrate these phases.

Launch can occur on any day between August 10 and September 9 during the Mars opportunity of August-September, 1990. Following separation from the upper stage, the spacecraft will enter the cruise phase which extends for approximately 1 earth year. During cruise up to four trajectory correction maneuvers may be required. Tracking and communications functions are provided by the Deep Space Network (DSN) in this phase and throughout the mission. While in cruise, the spacecraft bus must maintain a benign environment for the payload, make provision for instrument calibrations as necessary, and ensure that non-gravitational accelerations are minimized to permit reliable emphemeris predictions.

The specified interplanetary trajectory (Type II) will take the spacecraft slightly outside the orbit of Mars. To minimize injection energy requirements, in-plane insertion will be used. This will result in a capture orbit with nominally a 4 am ascending node with a north polar approach. The arrival date will be between August 12 and August 30, 1991. The nominal mission plan calls for completion of the orbit insertion phase and the beginning of mapping orbit operations in mid-October 1991, before the second conjunction period.

Following mapping orbit insertion the spacecraft will be configured for science mission operations. Again, in this phase, the minimization of unmodeled nongravitational accelerations is an important requirement. At the end of the mapping phase in October 1993 the altitude will be increased to a stable quarantine orbit. While, formally, this completes the mission, nothing in the bus design precludes operations being continued into the quarantine phase if additional useful scientific data can be obtained. For this reason a sun synchronous quarantine orbit will be used.



1 FOLDOUT FRAME

2 FOLDOUT FRAME

Figure 1-7. Mission Scenario

1.5.1 Launch And Ascent

Prior to lift-off, the spacecraft CPU memories will be loaded with their flight load packages, dumped and file compared using hard-wired links through the GSE. Command and telemetry links will be provided through the T-0 umbilical. The spacecraft status at lift-off is as follows:

- o Batteries connected (bus on Orbiter power)
- o Command and Telemetry via STS
- o All safety related items -SAFE
- o Ascent configuration of redundant equipment selected
- o RF transmitting elements - OFF
- o CPU memories loaded
- o CPUs on-line executing Idle Mode software
- o CPU Stored Command Table - OFF
- o IMUs - ON
- o RWAs - ON
- o All deployables stowed
- o All propulsion items OFF and SAFE
- o ASE - ON for communications and power via STS
- o Tape Recorder - On

The spacecraft and its ASE will be largely quiescent during launch and ascent to parking orbit with the CPUs controlling the RWAs running at low speed to avoid damage from the launch loads.

The communications link through the ASE and STS system provides a 2000 bps communication capability for transmission of engineering data via real-time link or for recording and playback later for analysis at JSC and JPL. In the launch configuration, the spacecraft poses no hazard to the Orbiter through flight system deployment, or in the event of an abort, through return and landing operations including the use of secondary or contingency sites. Safety monitoring is provided of all critical functions in the aft-crew compartment.

1.5.2 On-Orbit And Deployment

After reaching the desired orbit altitude, the Orbiter's payload bay doors will be opened between 1 and 3 hours after lift-off. A series of early spacecraft system checks can then be accomplished by the ASE through the Orbiter's standard switch panel. After initial checkout, the spacecraft will be configured in a quiescent mode to await the start of the deployment sequence. The RWAs will be switched off and both batteries will be placed on trickle charge (C/60 rate) on Orbiter power via the ASE.

Data from the spacecraft will be provided via the ASE for review in the aft-crew compartment. These data will be primarily safety-related to show the status of all Propellant and Pyrotechnic Subsystem elements. The sensing system associated with these safety-related items will be sufficiently redundant to assure unambiguous interpretation of the spacecraft safety status.

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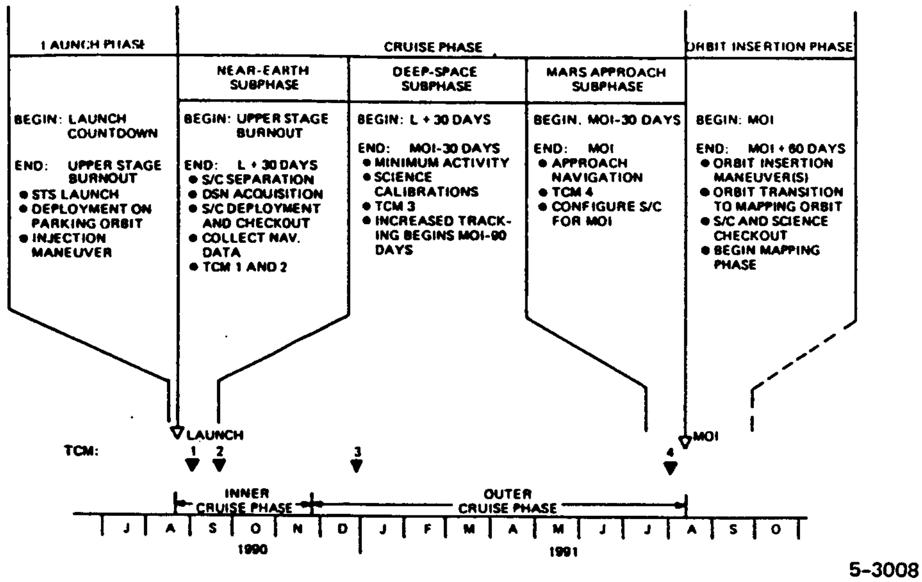


Figure 1-8. Mars Observer Mission Phases - Launch to Orbit Insertion.

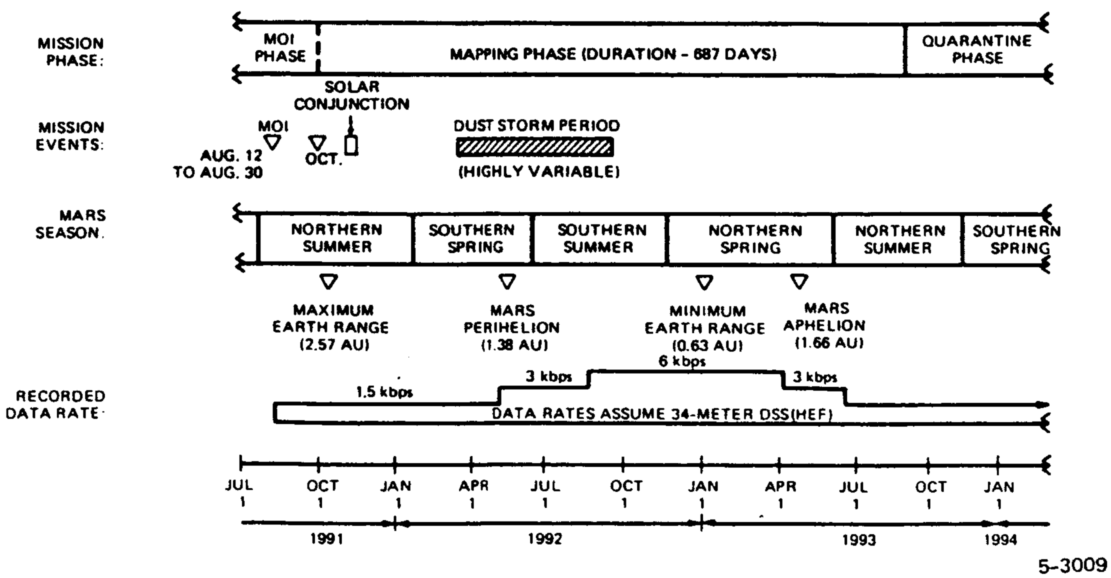


Figure 1-9. Mars Observer Mission Phases - Orbit Insertion to Quarantine.

During parking orbit operation, the Orbiter payload bay will normally be oriented towards the earth. For initialization of the upper-stage guidance system, the Shuttle will be maneuvered to ground-specified orientations for short periods of time and then returned to its nominal attitude. The spacecraft itself does not require these maneuvers, however, the opportunity may be taken to test the IMU concurrently. The thermal design of the spacecraft will accommodate these maneuvers and can support the spacecraft in this quiescent state for several days in the event the primary deployment opportunity cannot be satisfied.

Based on the achieved Orbiter state vector, the upper-stage injection burn orientation and timing will be computed on the ground and relayed to the Orbiter crew via JSC. At the appropriate time, the Orbiter will maneuver to the required upper-stage burn attitude and the reference gyros will be initialized. This reference attitude will be maintained in upper stage memory for use after deployment. If necessary, the reference attitude can be updated from the POCC and the changes verified by telemetry via the STS communications system.

Prior to deployment from the Orbiter payload bay, an autonomous Go/No-Go checkout will be initiated by a single Standard Switch Panel Command. Using self-test routines (SELTS), various state-of-health functions will be performed by both CPUs. At the conclusion of the checkout, a Go/No-Go indication is provided to the Orbiter general-purpose computer display.

Following successful completion of the Go/No-Go check, with verification of the on-board indication from the ground, the deployment sequence will be initiated by the flight crew. The upper stage will be armed by command from the aft-crew compartment; with the Orbiter in a stable position, the crew will activate the separation device. The Orbiter will not be constrained to the injection burn attitude for deployment since after initialization the orientation is retained by the upper stage. A separation velocity of at least 1 ft/sec will be imparted to the flight system relative to the Orbiter, and a separation signal provided by the Orbiter. An on-board spacecraft timer which starts at separation will activate the Propellant and Pyrotechnics Subsystem when safe Orbiter-spacecraft separation has been achieved. Deployment will occur between 45 minutes and 1 hour before upper-stage ignition. The flight system is compatible with earth or Orbiter shadowing during deployment operations. No EVA activities are required as part of the deployment sequence; all crew activities can be performed from the aft-crew compartment. No spacecraft control is required during this sequence; all Safe and Arm functions for the flight system are performed and verified on-board the Orbiter.

If the deployment is aborted for any reason after the sequence has started but before separation, the upper-stage ASE will have the capability to restow the flight system in a safe condition for the rest of the Orbiter mission including return and landing. The spacecraft will provide a positive indication that its deployment sequence has stopped and that all propulsive elements are in a safe state. In the extremely unlikely event that it is not possible to restow the spacecraft or to confirm that the flight system is in a safe state, the flight system will be ejected by the upper-stage ASE.

The ASE will be designed so that following deployment of the flight system it will not impose any limitations on Orbiter attitude or extended mission operations.

Spacecraft activiation will be performed by an on-board timer which starts at separation from the Orbiter and ensures that for the minimum separation velocity, a safe distance will have developed before any propulsion or pyrotechnic elements are activated.

When the spacecraft is in the Orbiter payload bay, all telemetry and command links will be via the ASE and STS system to the ground. There is no requirment to deploy any antenna or other appendage on the spacecraft until after separation and activation. After deployment of the flight system from the Orbiter bay, all communication from the spacecraft will be directly to the ground. No spacecraft/STS links will be required.

2.0 Safety Critical Spacecraft Systems

The following sections provide more detailed discussions of subsystems that have been identified at phase zero as safety critical during ground or flight operations. All components of the spacecraft and it's airborne support equipment are located in the cargo bay, there are no components in the crew compartment.

2.1 Communications

The communications subsystem block diagram is shown in Figure 2-2 and the communications antenna locations on the spacecraft are shown in Figure 2-1. Antenna coverage is provided by a steered high-gain antenna (HGA), a fixed medium-gain antenna (MGA) or a set of wide coverage low-gain antennas (LGA), depending on the phase of the mission and operational conditions. The HGA is used to monitor telemetry during thruster firing, check spacecraft attitude prior to and after thruster firing, and for communications with the spacecraft during the mapping phase. During maneuvers the HGA will be uncaged to permit motion around one gimbal axis while the HGA boom remains partly stowed. The medium-gain antenna is used for communications with the spacecraft during the outer portion of the cruise phase and during the early Mars orbit phase, except during maneuvers in which case the HGA is used. The low-gain antennas are used from the time of Shuttle separation through the inner portion of the cruise phase, except for maneuvers. One LGA is used to transmit flight system telemetry during the time from shuttle deployment to TOS separation. The low-gain antennas are also available throughout the mission for emergency communications with the spacecraft, should this become necessary.

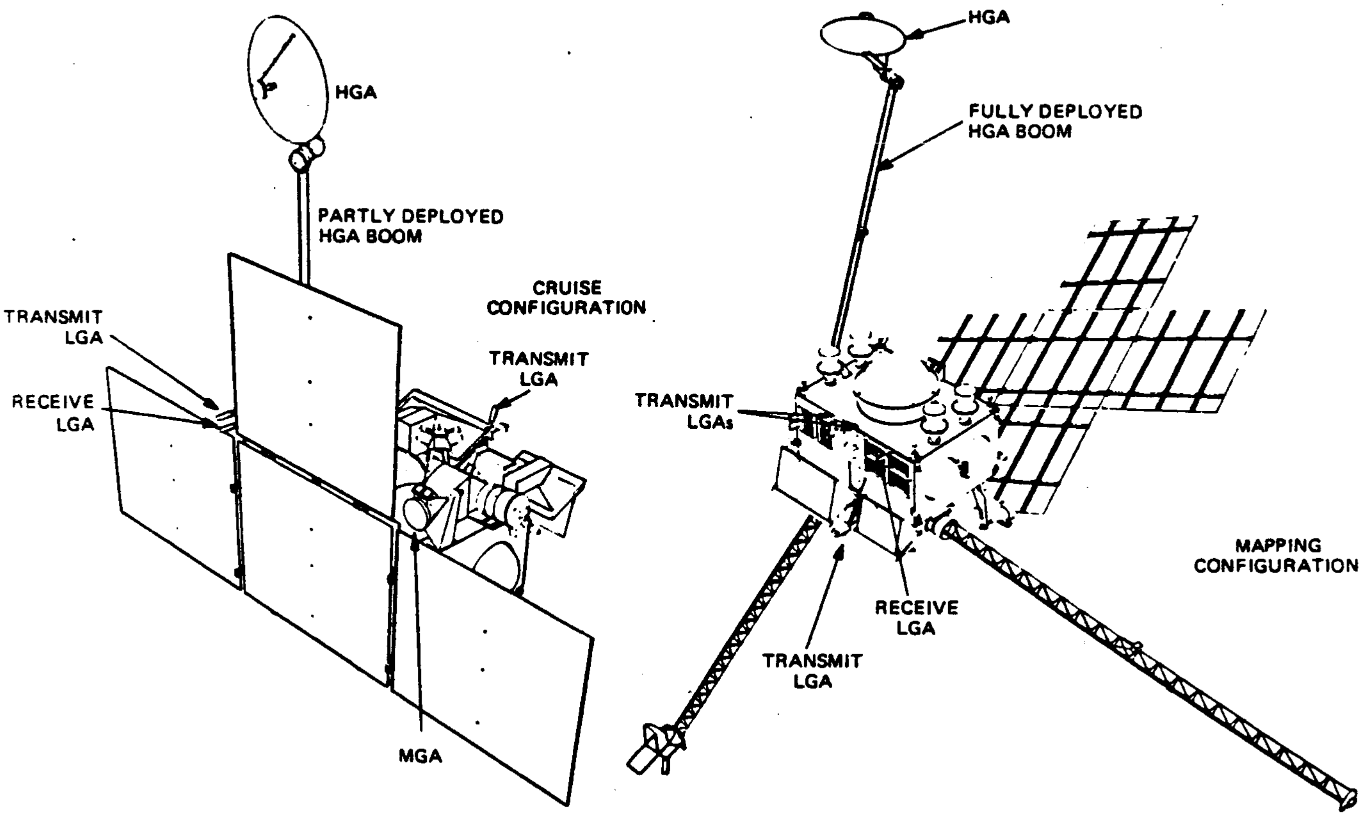
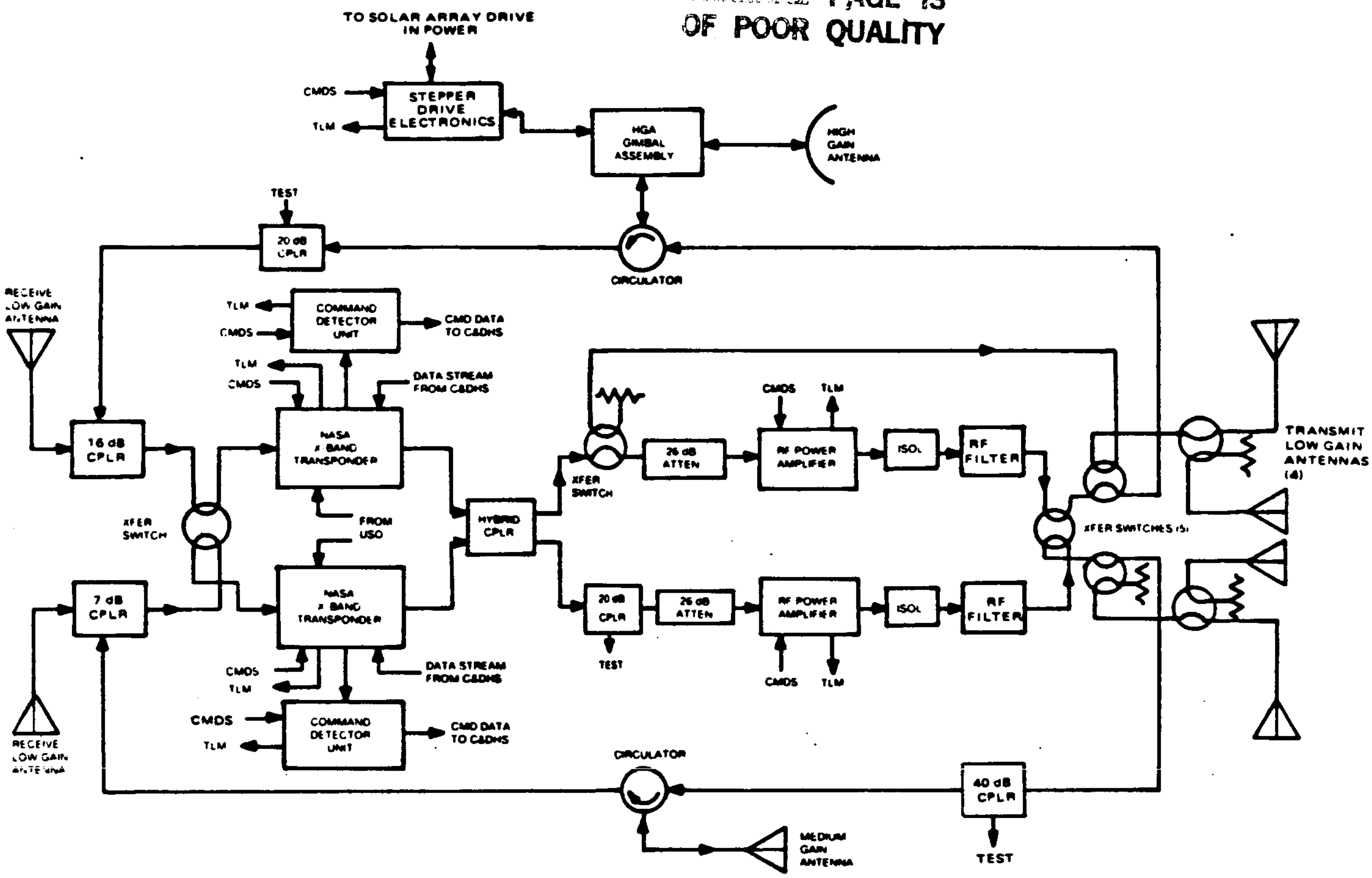


Figure 2-1. Antenna Locations



KEY COMPONENT CHARACTERISTICS

- TWO COMMAND DEMODULATOR UNITS (CDUs)
  - GFP
  - 16 kHz SUBCARRIER INPUT FROM NXT
  - COMMANDS AT 7.8125 bps AND 31.25 bps
- TWO NASA X-BAND TRANSPONDERS (NXT)
  - GFP
  - DEMODULATED COMMAND SUBCARRIER TO CDU
  - COHERENT TURNAROUND OR ONE-WAY TRANSMISSION
  - ULTRA-STABLE OSCILLATOR INPUT
  - PHASE MODULATED BY TELEMETRY & RANGING
  - TWO-WAY OR ONE-WAY RANGING
- TWO POWER AMPLIFIERS
  - 40W RF OUTPUT @ 110W INPUT (EACH)
  - TRAVELING WAVE TUBE AMPLIFIER (TWTAS)
  - EPC & TWT PACKAGED SEPARATELY
  - 6 dBm RF INPUT
- SIX LOW GAIN ANTENNAS (LGAS)
  - 2 dBi GAIN
  - 4 TRANSMIT, 2 RECEIVE
  - HEMISPHERIC BEAM
  - RIGHT-HAND CIRC. POL.
- ONE MEDIUM GAIN ANTENNA (MGA)
  - XMIT GAIN 28.5 dBi; REV GAIN 20.5 dBi
  - 6° BEAMWIDTH BETWEEN 1 - dB POINTS
  - LEFT-HAND CIRC. POL.
- ONE HIGH GAIN ANTENNA (HGA)
  - 1 m DIAMETER REFLECTOR
  - XMIT GAIN 36.3 dBi; REV GAIN 34.3 dBi
  - 1.5° BEAMWIDTH BETWEEN 1 - dB POINTS.
  - LEFT-HAND CIRC. POL.
- TWO GIMBALS
  - PERMANENT MAGNET STEPPER MOTORS
  - HARMONIC DRIVE GEARS
  - ROTARY WAVEGUIDE JOINTS
  - 0.0075° STEPPING GRANULARITY
  - OPTICAL SHAFT ENCODERS
- ONE STEPPER DRIVE ELECTRONICS (SDE)
  - GENERATES STEPPER PULSES
  - DRIVES BOTH GIMBALS AND SAD
  - COMMANDED FROM COMPUTER

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Figure 2-2. Communications Subsystem Block Diagram.

The uplink signal can include carrier only, command modulation, ranging modulation, or command plus ranging. In any event, it is received by the receive section of the transponder, which demodulates the command and ranging signals (if present). Each demodulated command baseband is sent to a command detector unit, which supplies a detected command data stream to the Command And Data Handling (C&DH) Subsystem for decoding. Demodulated ranging signals are remodulated onto the transmit carrier in the exciter section of the transponder. During two-way communications, the transmit carrier frequency is coherently related to the uplink carrier frequency with an 880 to 749 ratio. If no uplink is present, the transmit carrier is derived from an internal auxiliary oscillator with the same nominal frequency as used in two-way operation. For radio science measurements, an ultra-stable oscillator (part of the GFP payload) can be switched in to replace the auxiliary oscillator. The downlink signal can be modulated by the turn around ranging signal or the telemetry data system from the C&DH subsystem or both. The transponder can be commanded also to provide a one-way ranging modulation signal instead of two-way ranging. At any given time, only one of the two transponders has its exciter section energized.

The output from the energized exciter goes through a 3-db hybrid coupler which feeds the signal to both rf power amplifiers, only one of which is energized, The rf power amplifier provides a 40-watt output to an isolator and rf filter.

A network of rf transfer switches connects the transmit signal to the selected antenna. The network is configured to provide either rf power amplifier access to each of the antennas. For communications while in the Shuttle orbit, the rf power amplifier is by-passed and the transponder output is switched directly to one of the low-gain antennas. Rf circulators are used to permit simultaneous receive and transmit access to the high-and medium-gain antennas.

The subsystem is controlled by commands from the C&DH Subsystem. Discrete commands control the transponder operating modes, the command detector bit rate, the transmit switch configuration, and the on/off status of the rf power amplifiers. Serial commands from the C&DH Subsystem control the high-gain antenna pointing. Telemetry points monitor the operating status and temperatures of the subsystem components. Serial telemetry data provide signal-to-noise ratio (SNR) estimates from the CDU, and antenna pointing information from the HGA.

2.2 Command And Data Handling

The Command And Data Handling (C&DH) Subsystem block diagram is shown in Figure 2-3.

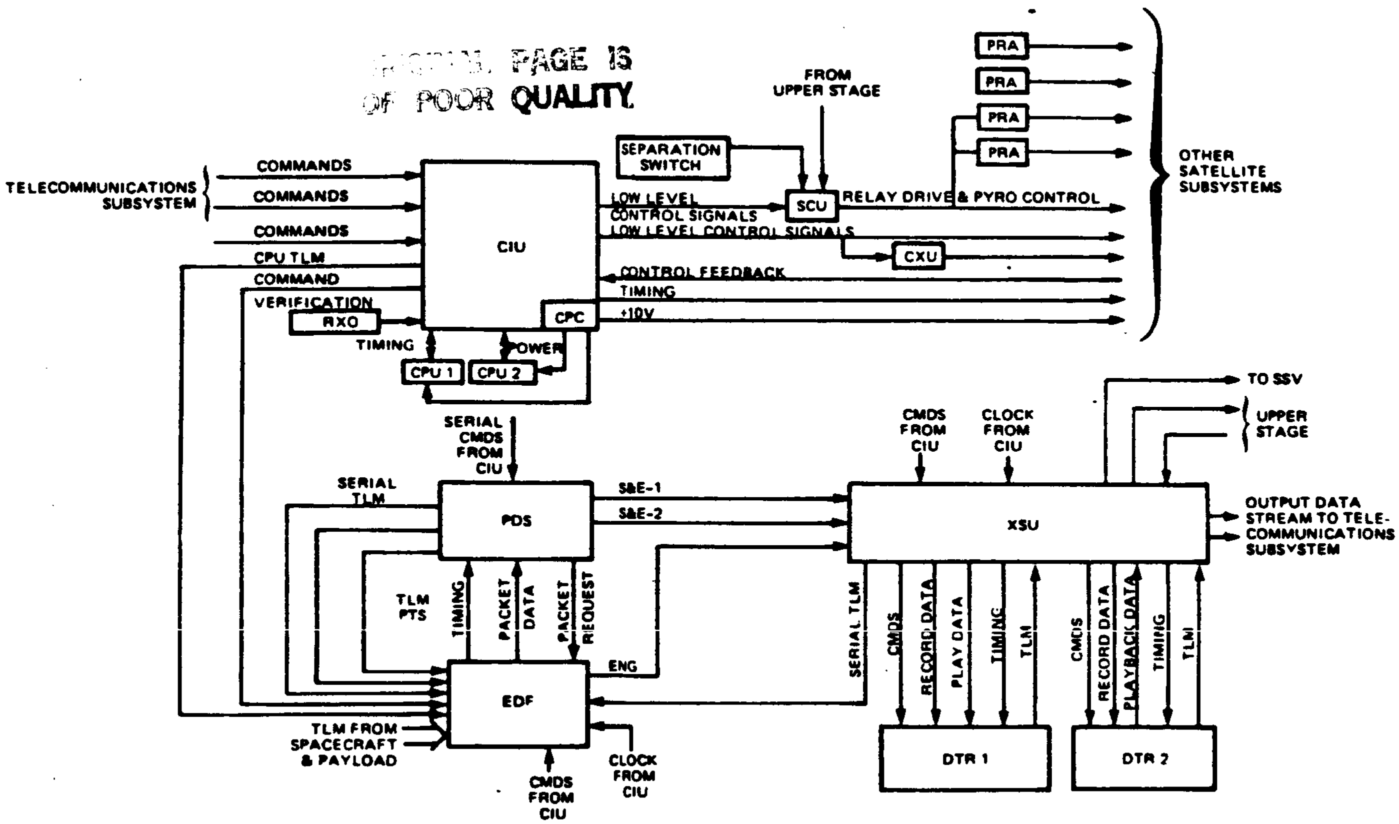
All control functions are accomplished by dual (redundant) on-board central processing units (CPUs) each with 64K words of random access memory (RAM) and 256 words of read-only memory (ROM) for bootstrapping, a controls interface unit (CIU), a controls interface unit annex (CXU), a signal conditioning unit (SCU), and four pyrotechnics relay assemblies. The CPUs provide both data-processing flexibility and computational capability. Each CPU has sufficient RAM for autonomous operation for up to 60 days for the Mars Observer mission. The CPU, via the CIU, provides all control and computation functions for all modes of attitude determination and control, powered flight guidance and navigation, command decoding, checking, storage and distribution, power management, failure diagnosis (including self-diagnosis), and redundancy switching. Identical software packages reside in each CPU with provision for autonomous switching between CPUs (via a watchdog timer) in the event of a self-check failure indication. Single-bit error correction and two-bit error detection are special features of the design.

The CPU Flight software comprises a set of interrelated modules grouped into five main categories: Command and Control, Guidance and Navigation, Self-Test, Redundancy Management, and Executive. The software may be modified or changed out via uplinks. Low level commands are distributed by the CIU and CXU; those requiring high-current drive capability are conditioned in the SCU, and the Pyrotechnics Relay Assemblies (PRAs).

The CIU links all command-and-control-elements of the C&DH Subsystem, and interconnects the computers with all other spacecraft components. It provides direct hardware access (bypassing the CPU) for certain critical commands. Included in the CXU is 8k words of programmable read-only memory (PROM) for the safe-system mode firmware, which is CPU independent. A redundant crystal-controlled oscillator (RXO) provides a 5.12-MHz source for all spacecraft timing functions. A controls power converter (CPC) provides the necessary tightly regulated power (+10 Vdc) for the command and control units.

To accommodate the data handling requirements, three additional redundant blocks are included: the engineering data formatter (EDF) together with the GFP payload data subsystem (PDS); the cross-strap unit (XSU); and two 1.5 x 10^9 bits capacity Odetics digital tape recorders (DTR). This system synchronizes, times, collects, digitizes, multiplexes, formats, routes, stores, and plays back science instrument data, command verification, CPU memory dump, and payload and spacecraft engineering data. The PDS also applies Reed-Solomon error coding to the data streams.

Four data streams (Science and Engineering 1 and 2, spacecraft bus engineering, and spacecraft bus and upper-stage engineering) are input to the XSU. By ground command, any one of the four streams can be selected



KEY COMPONENT CHARACTERISTICS

- TWO CENTRAL PROCESSING UNITS (CPUS), EACH:
  - 64k WORDS RAM
  - 256 WORDS ROM (BOOTSTRAPPING)
  - SINGLE ERROR CORRECTION, DOUBLE ERROR DETECTION
  - 52 INSTRUCTIONS
  - 2.34 MICROSECOND CYCLE TIME
  - SINGLE AND DOUBLE PRECISION
- CONTROLS INTERFACE UNIT (CIU)
  - COMMAND DECODING
  - CPU INTERFACE
  - 224 DISCRETE COMMANDS
  - 19 SERIAL BUFFERS
  - CLOCK GENERATION
  - ATTITUDE CONTROL ELECTRONICS
- CONTROLS INTERFACE UNIT ANNEX(CXU)
  - 8k WORDS PROM FOR FIRMWARE FOR ATTITUDE INITIALIZATION, SAFE-SYSTEM, CRUISE ORIENTATIONS, ...
  - 192 ADDITIONAL LOW-LEVEL COMMANDS
- SIGNALS CONDITIONING UNIT (SCU)
  - HIGH LEVEL COMMAND INTERFACES
  - PRE-ARM AND ARM RELAYS FOR PYROTECHNICS
  - THRUSTER FIRE SIGNALS
- PYROTECHNIC RELAY ASSEMBLY (PRA)
  - FIRE RELAYS FOR PYROTECHNICS
  - COMPLETELY SHIELDED
  - 4 VOLTS
- REDUNDANT CRYSTAL OSCILLATOR (RXO)
  - 5.12 MH_z TIMING SOURCE
  - STABILITY 1 PART IN 10^6 OVER 3 YEARS
- PDS
  - GFP
- ENGINEERING DATA FORMATTER (EDF)
  - ENGINEERING PACKETS TO PDS
  - ENG DATA STREAM AT 1500, 200, 10 bps
  - TRANSFER FRAME FOR ENG DATA
- CROSS STRAP UNIT (XSU)
  - OUTPUT STREAM SWITCHING
  - CONVOLUTIONAL ENCODING
  - SUBCARRIER OSCILLATOR AND MODULATOR
  - AMPLITUDE CONTROL
  - DATA INTERFACE WITH UPPER STAGE
- TWO DIGITAL TAPE RECORDERS (DTRS), EACH:
  - 1.5 X 10^9 BITS STORAGE
  - BER 1 IN 10^6

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Figure 2-3. C&DH Subsystem Block Diagram.

for transmission. In addition, all but the S&E-2 stream can be simultaneously recorded on either of the two DTRs. Recorded data can be selected for transmission instead of real-time telemetry. The XSU is a flexible routing control unit providing, complete interchangeability of input data streams, output channel, and DTR usage, by ground command. The EDF is a general-purpose microprocessor-controlled telemetry processor, whose output format can be modified by command. It collects engineering data and generates packets for the PDS and for direct transmission. It is a derivative of the DMSP/ATN Programmable Information Processor introduced on DMSP in 1981 and flown on DMSP Block 5D-2 flights F6 and F7.

Launch by the SSV and injection by the TOS necessitate new launch-vehicle telemetry interfaces. To save launch mass, the telemetry handling system of the OSC TOS has been removed. Before deployment of the flight system from the Orbiter, TOS telemetry and spacecraft telemetry are routed separately, via the electrical airborne support equipment (EASE) into the Payload Data Interleaver (PDI) of the SSV. Following deployment, spacecraft telemetry is routed from the EDF to the telemetry encoder in the OSC TOS, the majority vote sequencer (MVS), and is interleaved with TOS telemetry. The mixed telemetry stream is then passed back to the XSU for transmission to the ground by the X-band Telecommunications Subsystem, using an NXT directly coupled to an LGA.

2.3 Electrical Power

The Power Subsystem provides the required generation, storage, control and distribution of electrical power for the flight system during all phases of the Mars Observer mission. This subsystem includes a photovoltaic solar cell array, two nickel-cadmium secondary batteries (with KOH electrolyte pressurized at 45 psi), power conditioning electronics, distribution, and an automated battery energy management capability. Elements of the subsystem are connected in a direct energy transfer (DET) configuration, using solar array sequential partial shunt regulation and a battery voltage boost-regulator scheme.

Figure 2-4 presents the Mars Observer (MO) Power Subsystem functional block diagram. The Power Subsystem consists of the following components:

- Solar Array (SA)
- Solar Array Drive (SAD)
- Array Drive Electronics (Part of SDE)
- Power Supply Electronics (PSE)
- Battery Charge Assembly (BCA)
- Batteries (B)
- Battery Reconditioning Unit (BRU)
- Power Converter (PC)
- Fuse Board Unit (FBU)

The Power Subsystem Electronics (PSE, BCA, PC) accepts power inputs from the solar array and the batteries (via BVR) and provides a regulated output at 28 Vdc $\pm 2\%$. The PSE and BCA control the charge current to the batteries to within selected limits, and when full array output power is not required reduces solar array current output via a redundant shunt system. During periods when array power is unavailable or is insufficient, the batteries will discharge through the BVR to maintain the bus at the regulated voltage.

The BCA passes current from the solar array bus to each battery. The charge current is limited by the mode control (a part of the PSE) based on the charge rate selected by ground command and by the sensed V/T feedback signal from the battery.

A PWM boost regulator supplies energy from a bus supplied by all the batteries to the 28-volt regulated bus during eclipse periods or when the solar array ouput is not adequate to supply load current demands. The battery bus voltage is boosted as required to maintain the regulated bus at 28 volts. A "short" failure in one cell of the 17 cells can be accommodated by the BVR.

A partial shunt regulator reduces the current output from the solar array when full output is not required for satisfying load requirements and for charging the batteries. The partial-shunt configuration consists of 18 switched sections and two parallel linear sections. Each section is connected across a single solar cell circuit of 31 series and 15 parallel cells. The shunt operation is accomplished by electrically mismatching

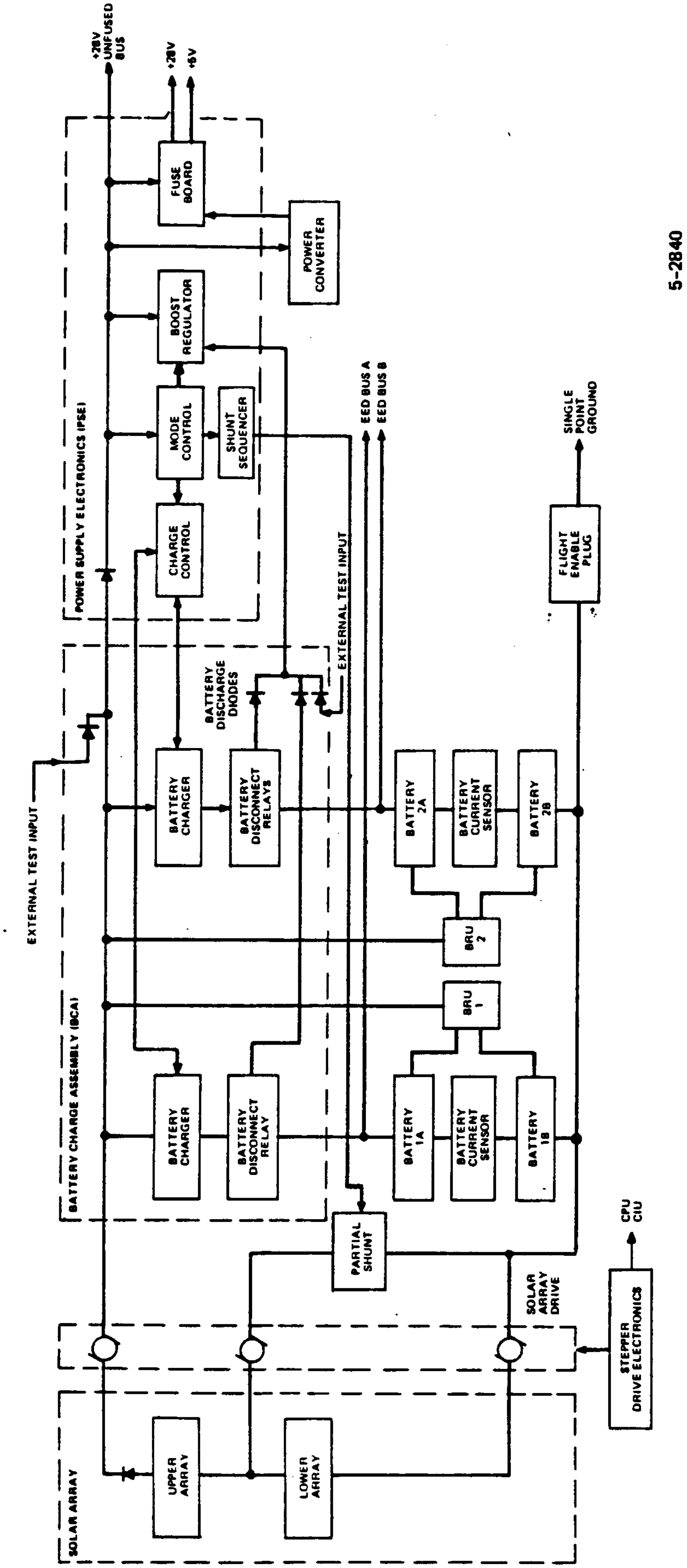

Figure 2-4. Power Subsystem Block Diagram.

selected portions of the solar cell circuits controlled by the partial linear shunt section. The remaining cell circuits are controlled by switched partial-shunt regulators which operate in either the "OFF" or "ON" (saturated) modes. The binary shunt regulators are activated as the linear shunt section approaches its fully loaded condition. The Mode Control is the control element for power regulation and causes the appropriate mode to be selected. It senses voltage change on the 28V bus and applies drive signals to the appropriate regulatory elements to maintain the 28 volts within the $\pm 2\%$ tolerance.

Failure-detection circuits continuously monitor the operation of the charge controllers, shunt control amplifiers, boost regulator, and mode control. If a failure is detected, automatic switchover from the primary to backup side is implemented. Either the primary or backup function may also be selected by ground command. All automatic switching functions can be overridden by ground command.

Battery state-of-charge is automatically controlled using a C/D - augmented V/T limit technique which utilizes Standard Controls Processor (SCP) software and Power Subsystem hardware. Battery overcharge is rigorously controlled to minimize battery degradation. Battery cell reconditioning will be performed just prior to Mars orbit charge/discharge cycles to ensure proper cell voltage balance and maximum battery performance.

The spacecraft receives power from the Orbiter via the Power Master Unit (PMU), which is a piece of EASE mounted on the TOS cradle. The power required when connected to the Orbiter is TBS at phase one.

2.4 Thermal Control

The Mars Observer thermal design makes maximum use of passive thermal control where possible and only resorts to the use of active thermal control elements when necessary. Passive thermal control elements include multilayer insulation blankets, radiators, structural insulators and conductors, tapes, and surface finishes. Active thermal control elements include combinations of Satcom-type Thermal Control Electronics (TCE) units, louvers, and heaters.

During pre-launch, conditioned air within the cargo bay will maintain benign average temperature levels (this temperature is TBD) which the Mars Observer Spacecraft will be compatible with. After launch, the Orbiters cargo bay doors will open no sooner than 1 hour and no later than 3 hours after liftoff. In the latter case, spacecraft component temperatures may reach 30°C prior to opening of the cargo bay doors. Temperature rise due to internal dissipation in this relatively warm environment is attenuated by the thermal capacitance of the spacecraft. Once the cargo bay doors are open, the spacecraft is subjected to the Orbiter +ZLV orientation as well as multiple allowable excursions to +Z Solar for 30 minutes and to +Z Deep Space for 90 minutes. Spacecraft recovery time from these excursions is TBD. Thermal analyses of the Mars Observer/Space Shuttle Vehicle (SSV) in these scenarios have shown that heater power is required to maintain the spacecraft at the minimum allowable temperatures within the SSV environment.

In the event of an abort situation and touchdown at a primary or secondary landing site, a conditioned air purge is made available within 15 minutes, providing a means for removal of heat absorbed during SSV reentry. At a contingency landing site (where no cooling is available), the hottest transient temperature reached by any localized part of the cargo bay is 96° C (per STS83-0425). The spacecraft is able to withstand the payload bay environment, maintaining component temperatures and tank pressures within safe limits.

A major issue is the pressure buildup in the fuel tanks due to increasing temperatures. However, the 5% tank ullage is sized to permit adequate margin for a steady-state temperature of 70° C, which is well above the overall average payload bay liner temperature during a post-contingency landing.

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2.5 Structure, Cabling, and Mechanisms

The structures principal function is to serve as the spacecraft's physical interface with the launch vehicle and to support the other spacecraft subsystems. The structure also externally supports and aligns the payload sensors, solar array, antennas, and thrusters. The external configuration of the spacecraft in it's mission mode is presented in Figure 2-5. The Mars Observer launch configuration is shown in figure 2-6.

The general arrangement of the Mars Observer structure is shown in Figure 1-1. As in all the Mars Observer predecessor communications spacecraft the center structure assembly, extending from the top panel (nadir-facing panel) to the separation plane, is the primary load path of the spacecraft. This cylindrical assembly is joined to the side panels by

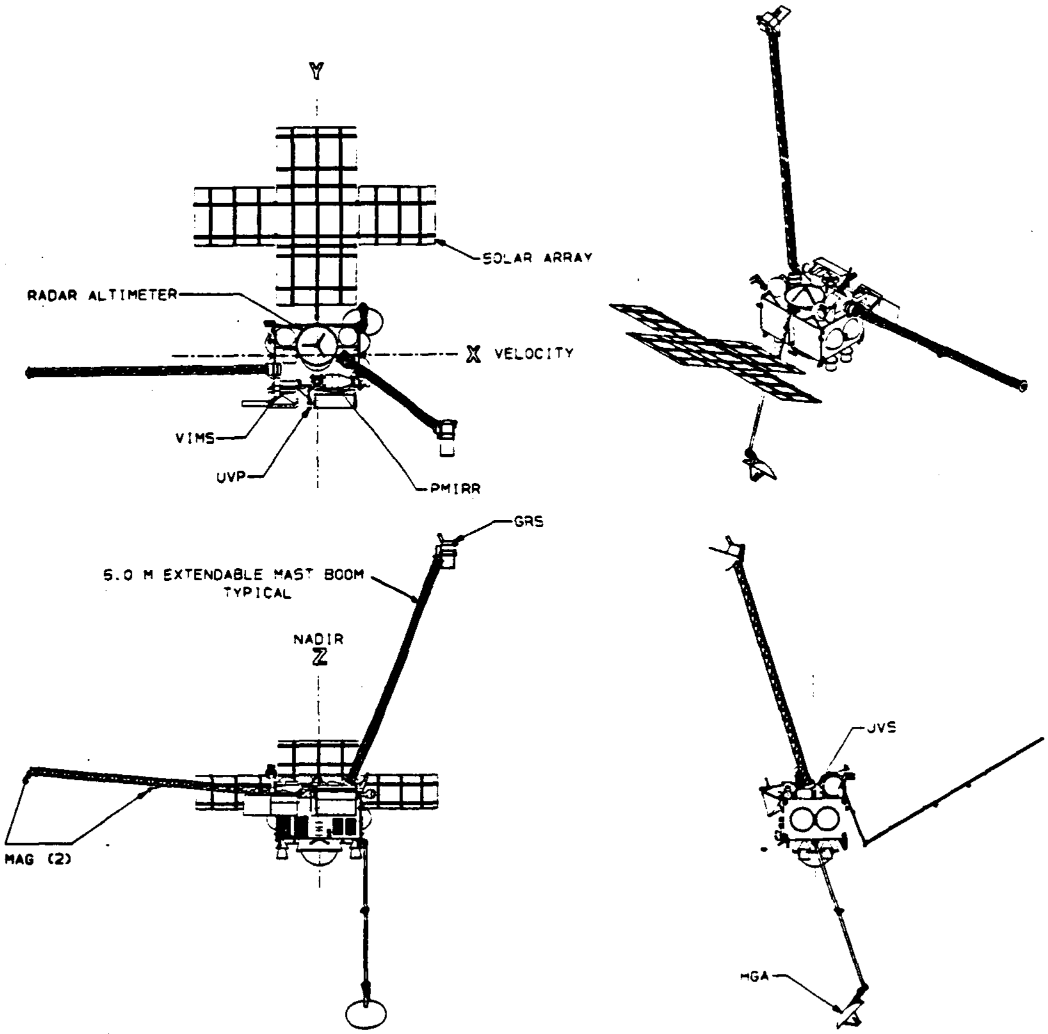

Figure 2-5. Mars Observer Mission Configuration.

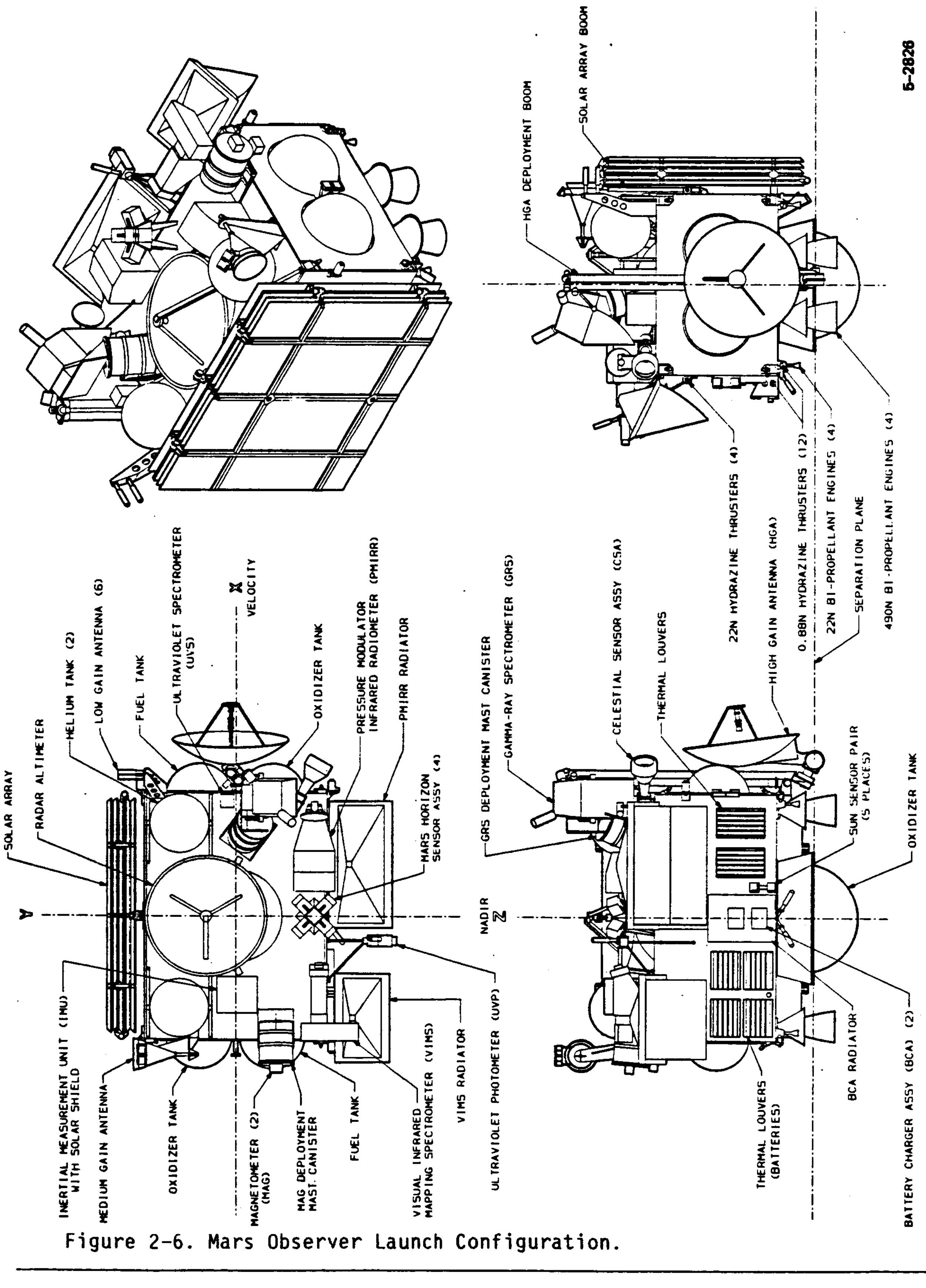

Figure 2-6. Mars Observer Launch Configuration.

means of bulkheads. At the base panel, the center structure changes to a conical section and terminates in a separation ring that is the interface with the TOS adaptor. The center structure assembly directly supports two large propellant tanks within the cylinder. It is also the support for other Propulsion Subsystem components that are brazed into half-systems in a separate fixture and then joined to each other and to the center structure assembly. Struts extend from the cylinder and bulkheads to internal bipropellant and monopropellant tanks to react lateral loads. Thrust loads from these tanks are reacted by shear fittings in the bulkheads (four bipropellant tanks) and in the cylinder (monopropellant tanks). The equipment mounting panels are divided into halves separated by chambers, one of which is occupied by the solar array boom during launch. The four equipment mounting panels support the bulk of the electronic equipment. The velocity and anti-velocity panels are designed as structural shear panels that can be readily removed thus allowing access during all phases of integration and test. The top panel (nadir-facing) and the base panel complete the box structure and transfer lateral louds to the center structure.

The Mars Observer structure supports a 102-kg payload which compares favorably with the 170-kg payload of the Satcom Ku-Band Spacecraft for which the structure was originally designed. The Mars Observer payload is located principally on the nadir surface of the spacecraft. Support brackets that extend from sensor attachments to structural "hard points" are provided where required for each sensor. The nadir-facing panel is shared by attitude reference sensors and pressurant tanks.

The forward end of the adapter mates with the separation ring and clamp band developed for Satcom Ku-Band, and the aft end terminates in a flange containing a bolt circle on a diameter of 90.6 inches. This bolt circle is the actual structural interface with the upper stage. It is controlled by drill fixtures built by the upper stage contractor. The fixtures are "mastered" to ensure matching of holes, and fit check will be performed at the upper stage contractor's facility.

Two separation connectors provide the electrical interface between the spacecraft and adapter. The adapter cable is actually part of the upper stage cable assembly with no additional connector between adapter and stage. Both halves of the separation plane connectors will be procured by RCA who will supply the stage halves to the stage contractor.

No direct mechanical interface exists between the Mars Observer and the STS, because the spacecraft is totally supported by the upper stage. However, the volume, natural frequency, and center of gravity of the spacecraft must satisfy STS constraints.

The spacecraft on the upper stage is oriented horizontally and coaxially in the STS. The circle that circumscribes the spacecraft has a diameter of 136 inches, well within the 180-inch allowable envelope.

Cantilevered from the upper stage, the spacecraft, on its conical adapter, has a fundamental frequency of 16.7 Hz. This value is typical of the cantilevered Satcoms and is amply greater than the 6 Hz minimum dictated by the STS control system. The fundamental frequency of the entire cargo element (spacecraft, upper stage, and ASE) is greater than 7 Hz.The weight of the cargo element is well below the 29,478 kg (65,000 lb) limit of the STS. Its location must be chosen to meet the c.g. limits of the STS when the mass properties of co-manifested payloads are identified.

2.5.1 Solar Array

The solar array is located on one side of the spacecraft to accomodate the fields of view of the radiative coolers of both the PMIRR and VIMS. The solar panels and their associated boom are stowed during launch against the side of the spacecraft. They are retained by cables through the panels and at their edges. Figure 2-8 shows the stowed solar array and boom and figure 2-7 shows the solar array restraint. Release of the solar

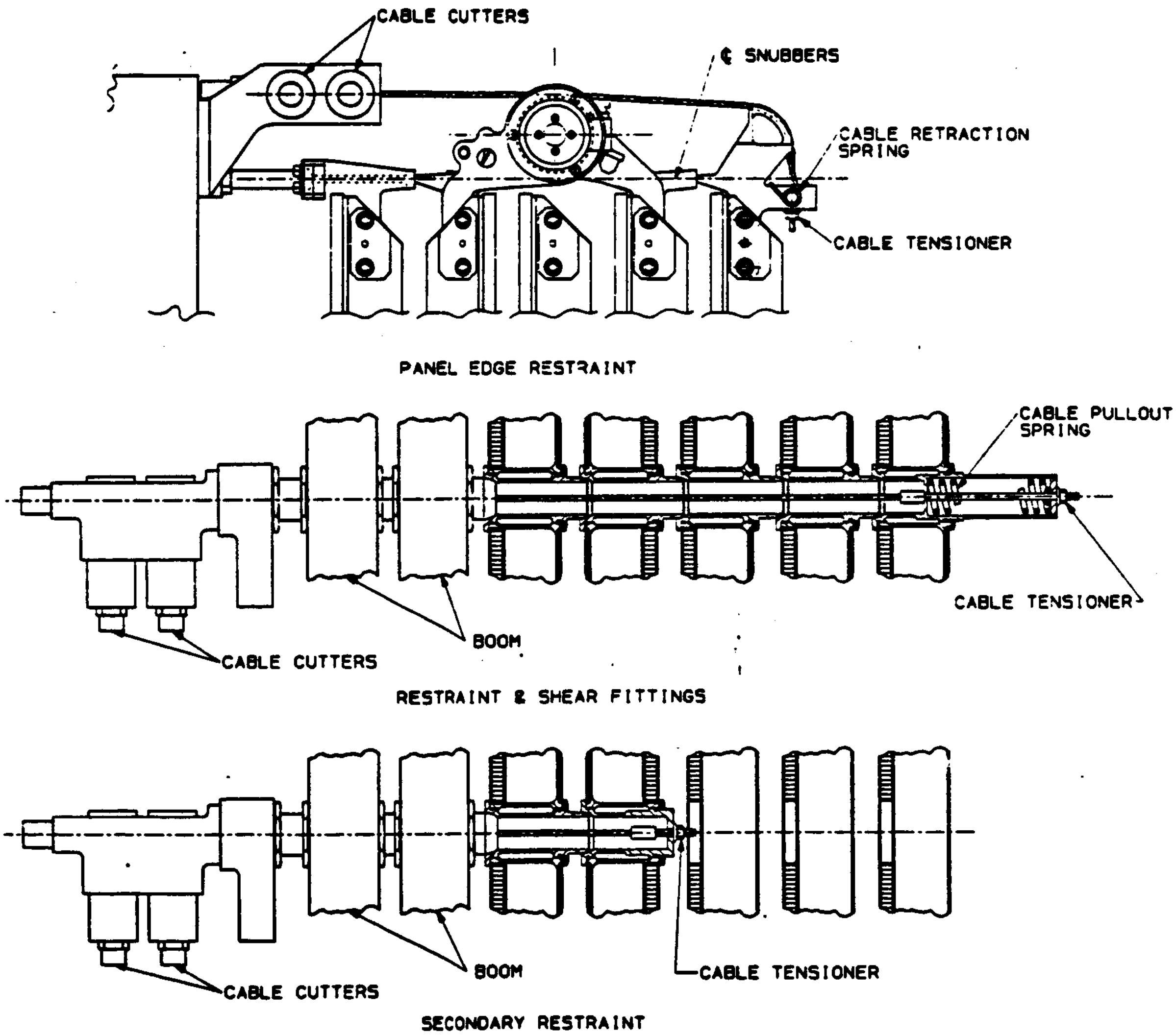

Figure 2-7. Solar Array Restraint.

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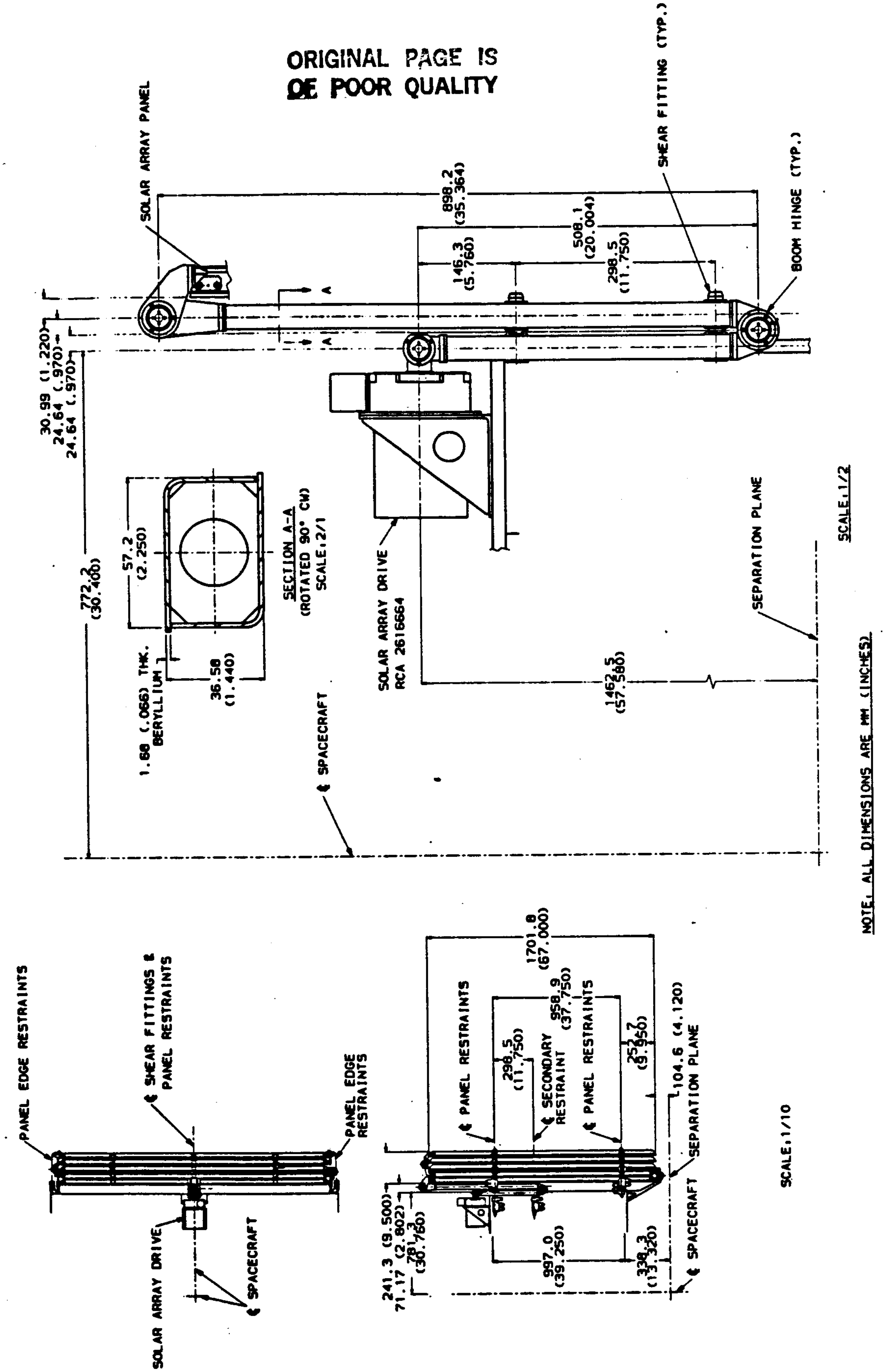

Figure 2-8. Stowed Solar Array and Boom

array is effected by the firing of redundant cable cutters. The solar array deployment sequence is illustrated in figure 2-9. Deployment of the array to the cruise configuration is initiated by firing the cable cutters that sever the six primary restraint cables. The three unrestrained panels then deploy one after the other by means of a passive sequencing system, into a planar array at the side of the spacecraft. Deployment is energized by springs and damped by viscous dampers at each hinge axis. The elements of the deployment system are identical to those flight proven on Satcom Ku-Band.

2.5.2 Instrument Booms

The Gamma Ray Spectrometer (GRS) and Magnetometer (MAG) are mounted on identical booms that must have the ability to be extended and retracted for calibration during cruise, retracted for MOI burn, and finally extended for the mapping mission.

Coilable longeron booms have been chosen for these functions because of their extension/retraction capability, deployment repeatability, light weight, stiffness, and compact stowage. The 22.9 cm (9-inch) diameter (circumscribing circle) boom used on the Voyager, NOVA, and Dynamics Explorer (DE) spacecraft has been selected as the baseline for the Mars Observer mission. The boom retraction technique was used on the S_3 and Solar Maximum missions. The booms are self energizing and are released from their canisters by severing their restraint cables with redundant pyrotechnic cable cutters. Deployment energy is attenuated by using the retraction motor to limit the rate of extension of the lanyard attached to the tip of the boom. Retraction is accomplished by a brushless DC gear motor which winds the lanyard onto a spool. To retract the fully extended boom requires a torsional moment to be applied to the end to initiate the coiling of the longerons. This action is effected by a bridle attached to the end plate which is pulled by the lanyard.

The magnetometer boom is required to support two such instruments, one at the tip and the other near the mid-point. The latter instrument will be located outside the cross-section of the boom. This technique, previously demonstrated on Voyager, allows the boom to be coiled and compressed into its canister as a single continuous element, because the mid-point instrument is not in the way, as it would be with a centered instrument. The externally mounted magnetometer is supported by a bracket from the boom. As the boom is compressed into its canister for stowage, the bracket passes through a slot in the canister, and the instrument itself is seated externally.

Boom length is six meters, and its lowest natural frequency when supporting the GRS at its tip is 0.32-Hz. This is well above the minimum required by the attitude control system. The strength of the booms is ample to sustain the thrust levels (0.015 g) experienced during firing of the four 22.2N (5-lb_f) velocity correction thrusters and by the four 22.2N bipropellant thrusters used for insertion into quarantine orbit.

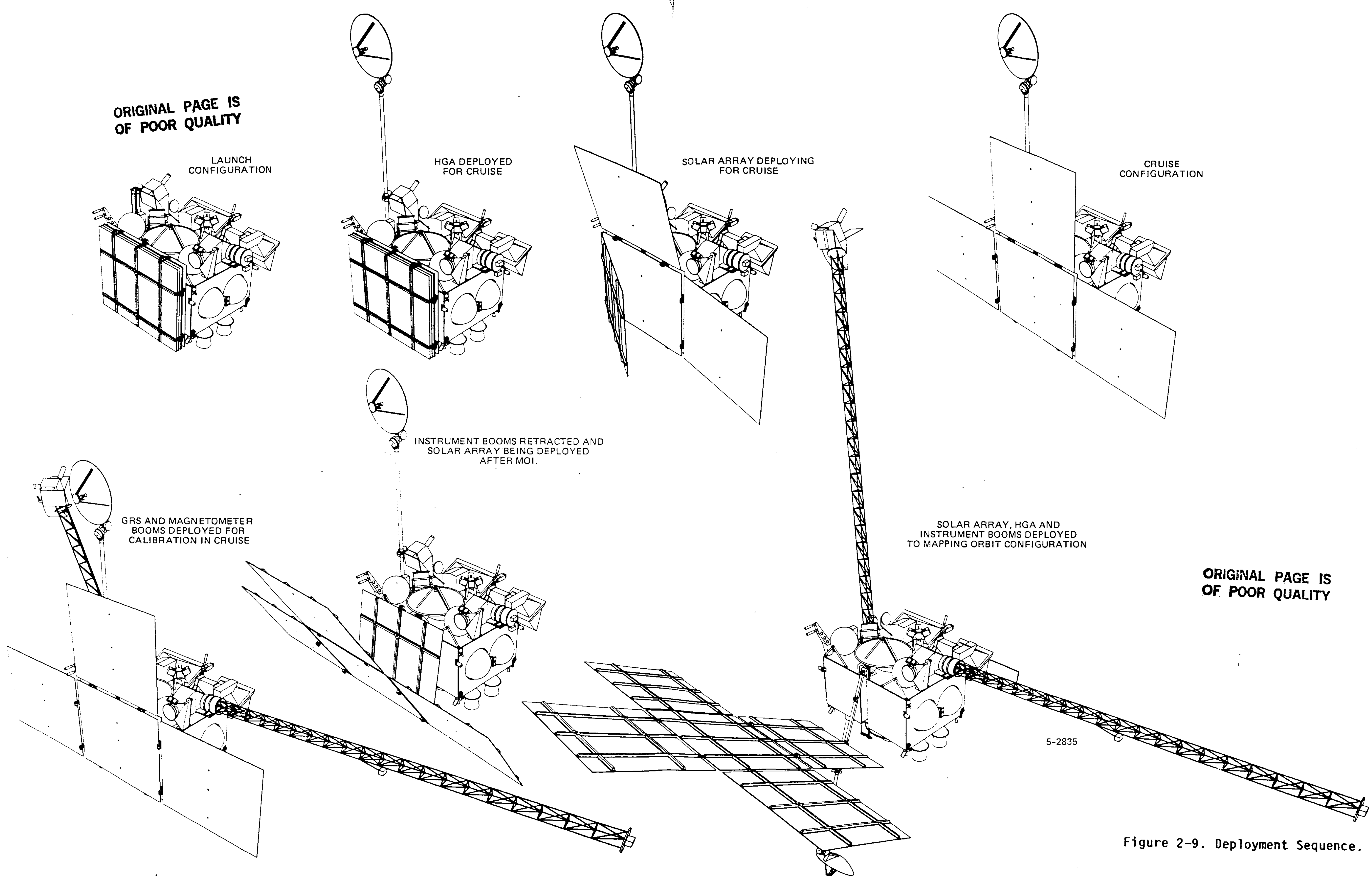

Figure 2-9. Deployment Sequence.

FOLDOUT FRAME

2 FOLDOUT FRAME

Figure 2-10 illustrates two different coilable longeron booms that RCA has used to deploy instruments from previous spacecraft: the NOVA U.S. Navy navigation satellite and the Dynamics Explorer (DE) scientific satellite. The latter boom is the baseline for supporting the GRS and the MAG in the Mars Observer mission.

2.5.3 High Gain Antenna Deployment Mechanisms

The HGA boom design, its supports and geometry must be such as to afford the antenna a clear field of view (FOV) of Earth during the cruise phase and later during the mapping orbit phase. A key requirement is that the mapping orbit FOV be clear as the Mars Observer moves into and out of occultation at the Martian limb. Extensive geometry studies involving mission date, position in the orbit, solar array position, and gimbal axes orientations were conducted to ensure that this would be the case.

One output of the studies was the boom length requirement of 3.81 m (150 inches) in the mapping orbit. To satisfy this length the boom is fabricated in two sections joined by a 180° hinge. The inboard boom is hinged to the zenith side of the spacecraft, and when fully deployed in the mapping orbit, the entire boom extends in the zenith direction. Figure 2-9 shows the boom in its stowed, cruise and mapping orbit configurations.

The antenna is attached to the gimbal system which in turn is joined to the outboard end of the outboard boom. During launch the inboard boom is latched to the nadir end of the spacecraft structure. A boom-mounted fitting is pinned to a toggle link that is secured to a spacecraft fitting by a pair of pyrotechnic pin pullers (Figure 2-11). Actuation of either pin puller will produce a successful release. The same system was used to release the solar arrays on RCA's ITOS series of meteorology satellites. Adjustable standoffs stabilize the antenna itself during launch.

Deployment is initiated in the cruise phase by firing the pin pullers securing the outboard boom/gimbal assembly. Redundant laminated negator springs, assisted initially by short stroke kickoff springs, provide the deployment energy, and rotary viscous dampers limit the angular velocity. An end-of-travel latch incorporates a cam feature which eliminates backlash. In this position the gimbal system is operable, and the antenna has a clear view of earth during cruise. After injection into Mars phasing orbit, the large bipropellant engines are deactivated, and their loads are no longer applied to the spacecraft. The inboard boom's restraining pin pullers are fired and the entire HGA and boom are rotated to the zenith position using the same spring/damper/latch design as for the mid-point hinge. This assembly is the same temperature-controlled design described previously for the solar array. Rotary waveguide joints at the hinges provide RF continuity.

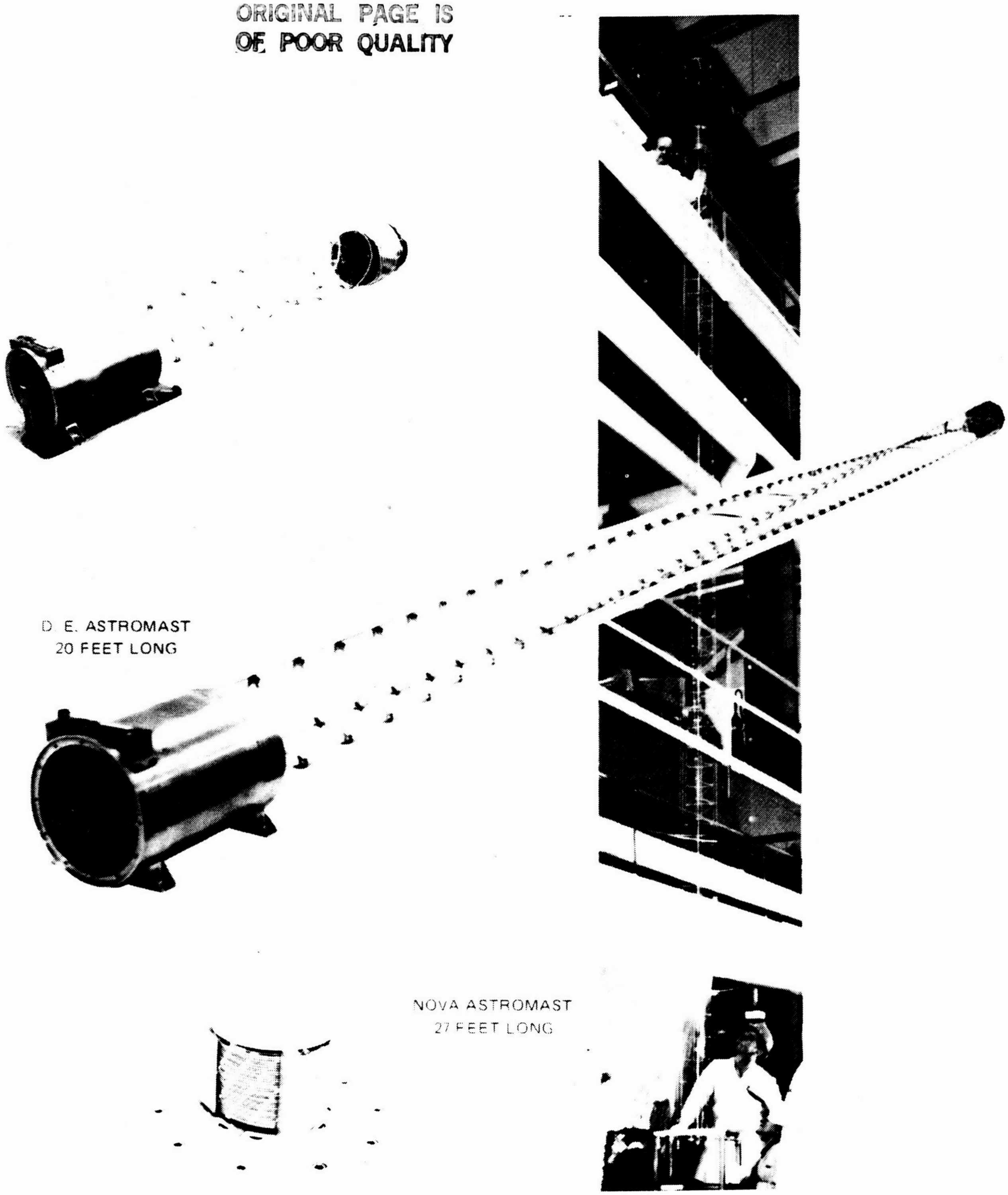

Figure 2-10. Nova and Dynamics Explorer Booms.

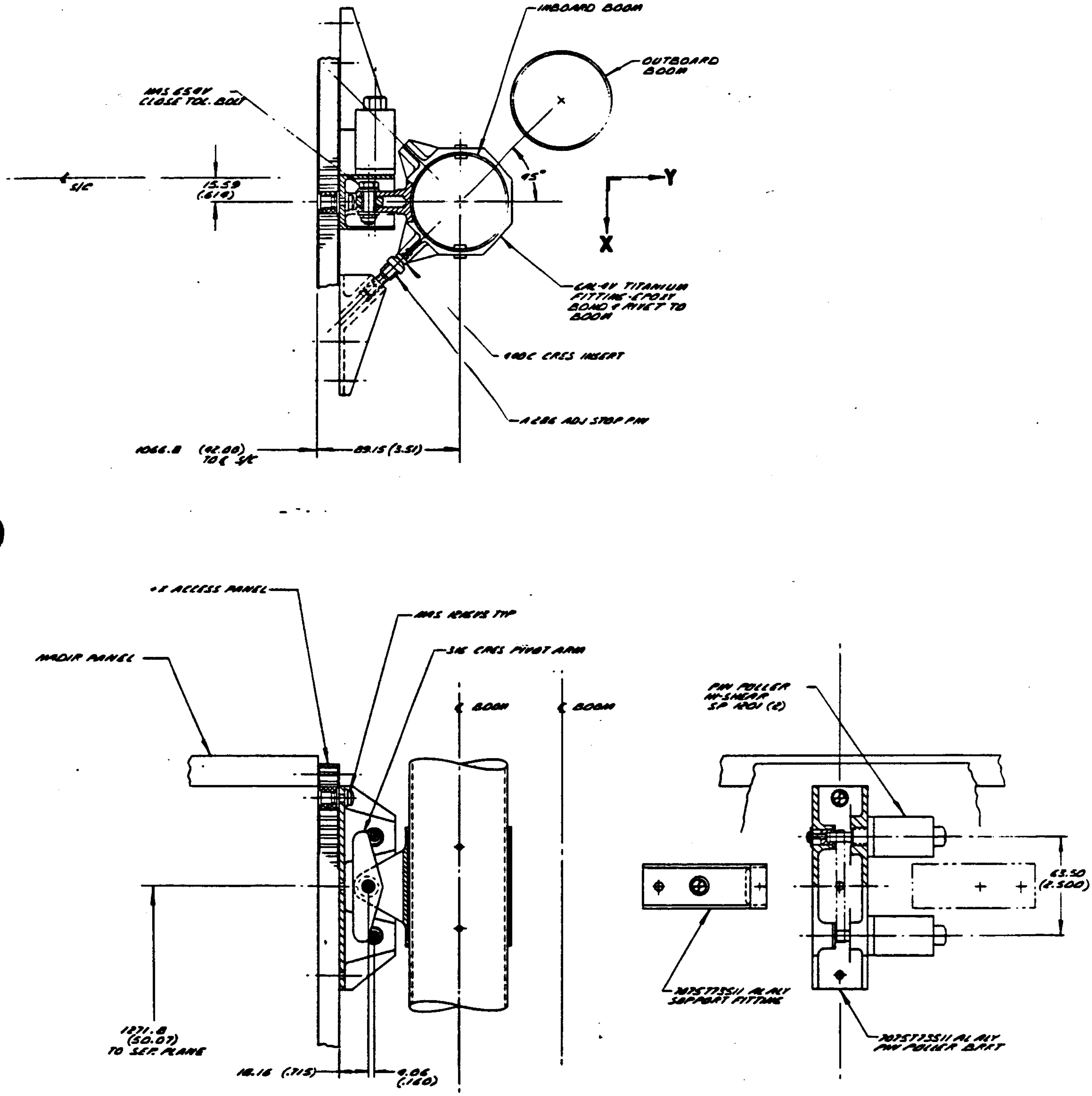

Figure 2-11. HGA Boom Restraint.

The HGA boom is a tubular graphite/epoxy laminate 6.35 cm (2.5 inches) in diameter x 1.52 mm (.060 inch) wall thickness. The material is cured in a quasi-isotropic pattern to approach a zero coefficient of thermal expansion. Sensitivity to thermal distortion is further reduced by enclosing the boom in a multilayer insulation blanket.

2.5.4 Separation Clamp Band

The separation clampband, identical to that of Satcom Ku-Band, joins the spacecraft and upper stage adapter from final integration until after separation from the upper stage is commanded. Two semicircular steel bands are tensioned by a pair of diametrically opposed bolts to apply radial pressure to 14 V-blocks which compress mating rings in the spacecraft and adapter. Each ring has a 15° ramp angle at the V-bloc interface, which produces an axial compressive force from the radial force. The combination of this ramp angle and 36,030N (8100 lb) of preload in the tensioning bolts produce an axial force great enough to not only overcome the tension forces produced by spacecraft bending but also to prevent gapping between the mating rings.

The band halves terminate in "bathtub" fittings which are joined by the separation bolts. Spherical washers on each bathtub prevent high bending loads from occurring in the separation bolt.

Separation is effected by firing a bolt cutter at each bolt, only one of which is required for successful release. The strain energy of the band combined with 10 springs on the adapter pull the band halves radially outward and free of the mated rings. A hard anodize finish on the clamped surfaces facilitates separation of the vee blocks. However, the contacting surfaces on the two rings are treated with irridite to ensure ample grounding between the spacecraft and adapter.

2.5.5 Cabling

The Mars Observer bus harness includes all cabling among bus components as well as between bus and GFP, between PDS and GFP body-mounted instruments, between NXT and CDU, and between the boom mounted instruments and the spacecraft. Flight harness fabrication will be fully compliant with the well-established RCA practices in design, materials, and component selection. Harness design will be governed by the following documents:

JPL D-1673	Mars Observer Performance Assurance Provisions
DOD-W-83575A	Wiring harness, space vehicle design and testing, General Specification for

MIL-STD-1541 Electromagnetic Compatability
Requirements for Space Systems.

These harness designs will be developed through the use of computer-aided harness analysis (CAESH). After initial routing is determined, this program will define the electrical performance of the harness in terms of resistance, voltage drops, and power losses. Additionally, complete mass properties data is provided in terms of bundle sized, weights, inertias, and segment CGs. The program works in conjunction with RCA's PRIME CAD system.

The Mars Observer harness set is divided into the following subassemblies:

- Bus Harness, which comprises two individual harnesses for improved partitioning and installation.

- Shunt Harness, very similar to the Satcom K design.

- RCS Harness, an extrapolation of Satcom K technology to reflect the additional RCS requirements of MO.

- Payload Harness, built to instrument requirements, which interconnects payload elements with each other and with bus components. (Note: the Payload Harness will be provided to JPL to allow instrument acceptance testing, utilizing the flight instrument cables.)

These harnesses are joined in the spacecraft with the interface connectors. This approach improves fit, shortens the design cycle, and greatly facilitates debug and test. All harnesses will be firmly supported by spacecraft structure to withstand the launch environment.

Copper conductors are the baseline for all harnesses. However, on-going trades will be performed to evaluate the weight benefits which might be realized by using aluminum conductors, a technique presently employed on several RCA communications spacecraft.

2.6 Propulsion

The Mars Observer Propulsion Subsystem is comprised of a bipropellant system for large delta-V maneuvers and a hydrazine system for the Mars mapping phase maneuvers. Schematic diagrams of the Mars Observer reaction control system are shown in figures 2-12 and 2-13. All active elements in the bipropellant and hydrazine systems are fully redundant. The system designs provide for isolation of any failed component by ground command, yielding a system with no single point failures. Thruster locations on the spacecraft are shown in figure 2-14. All thrusters are functionally redundant with latch valve isolation of thruster sets in the unlikely event of thruster leakage or failure. Propellant manifolding is designed to make all propellant available to any operating thruster set. Subsystem design includes redundant heaters, blankets, low emittance tape wrap, and thruster thermal protection which preclude any operational constraints due to the temperature environment. Long bipropellant system life is ensured by operation of this propulsion system in the blowdown mode after the Mars mapping orbit is achieved. The all-welded RCS construction provides leak free, structurally sound systems. The thruster requirements are well within proven steady-state burn and pulse capabilities.

2.6.1 Bipropellant Propulsion System

The bipropellant propulsion equipment consists of eight pressure-fed, liquid bipropellant, radiatively cooled, rocket engines and their propellant feed, storage, and pressurization systems. Monomethyl hydrazine (MMH) is used as fuel and Nitrogen Tetraoxide (N_2O_4) with 0.8% of nitrous oxide (NO), designated MON-1, as the oxidizer. The bipropellant system is designed to deliver more than 3.69×10^6 N-sec of impulse to the Mars Observer spacecraft.

Helium pressurant is stored at 28.96 MPa in two nominal 0.475m diameter 6Al-4-V titanium shell tanks with Kevlar 49 fiber load-bearing overwrap. Regulation of the helium is provided by two pressure regulators in series, one primary, the other standby. The regulator maintains the tanks at a maximum pressure of 1.79 MPa until after the Mars mapping insertion maneuver is completed. A pyrovalve then permanently isolates the helium supply so that the feed system is operated in a blowdown mode until mission completion. Propellants are stored in six spherical 6Al4V titanium tanks, comprising four 0.564-m diameter Satcom Ku-Band heritage tanks, and two 0.935-m diameter tanks. The latter tanks are mounted in the spacecraft center core allocated to the solid apogee motor in Satcom Ku-Band. The nominal operating oxidizer to fuel mixture ratio of 1.65 for the 490N and 22N bipropellant thrusters permits oxidizer and fuel storage in tanks of equal volumetric capacity. Propellant is fed to the thrusters from the 0.94-m diameter tanks which contain propellant management devices (PMD) to maintain a gas-free propellant supply at the tank outlet whenever outflow is required. Redundant check valves in the pressurant distribution manifold isolate the ullage volumes of the two 0.935-m diameter tanks to prevent propellant liquid and vapor from migrating up the helium lines and intermixing. Two pair of 0.564-m diameter tanks are

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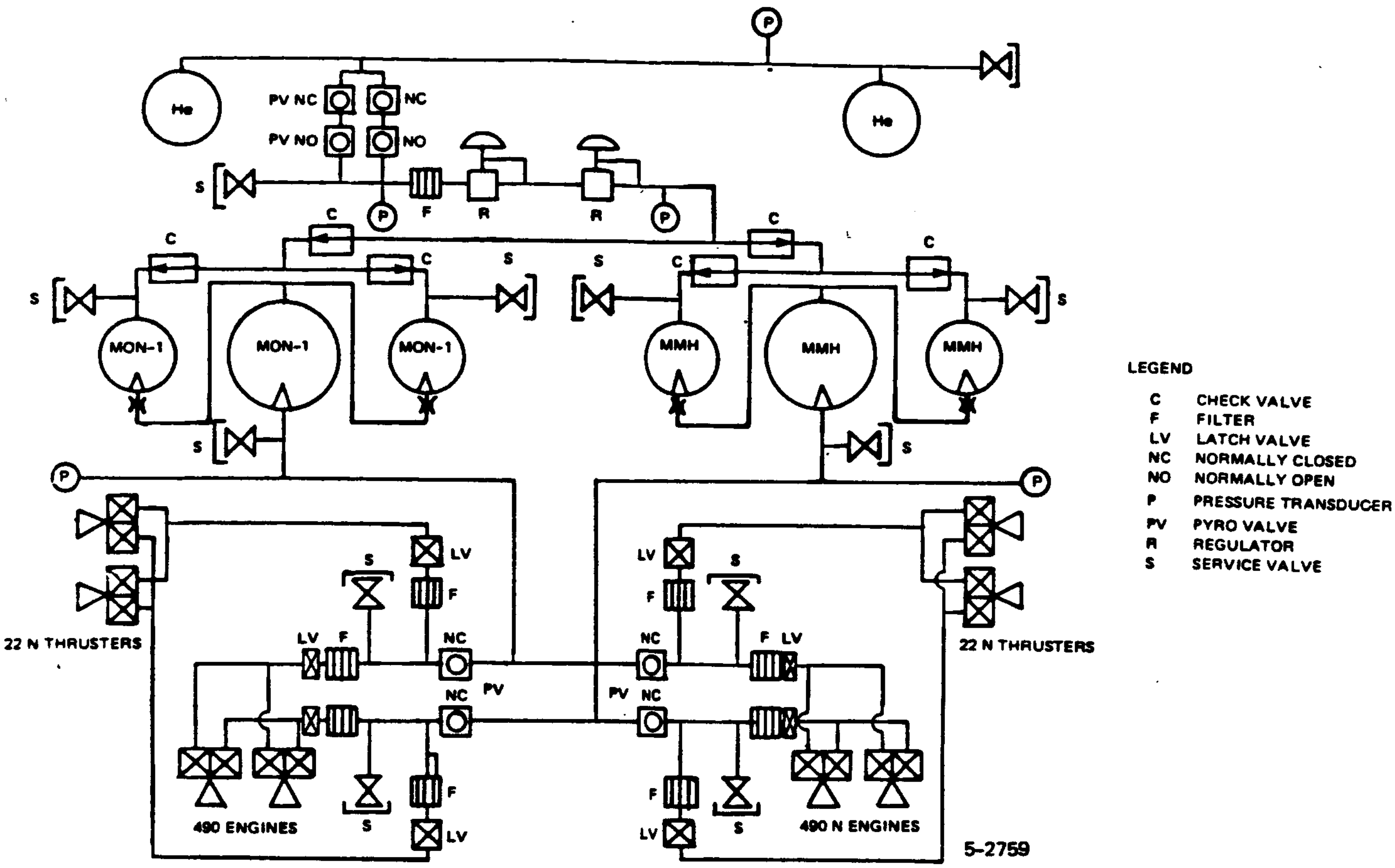

Figure 2-12. Bipropellant Propulsion System Schematic Diagram.

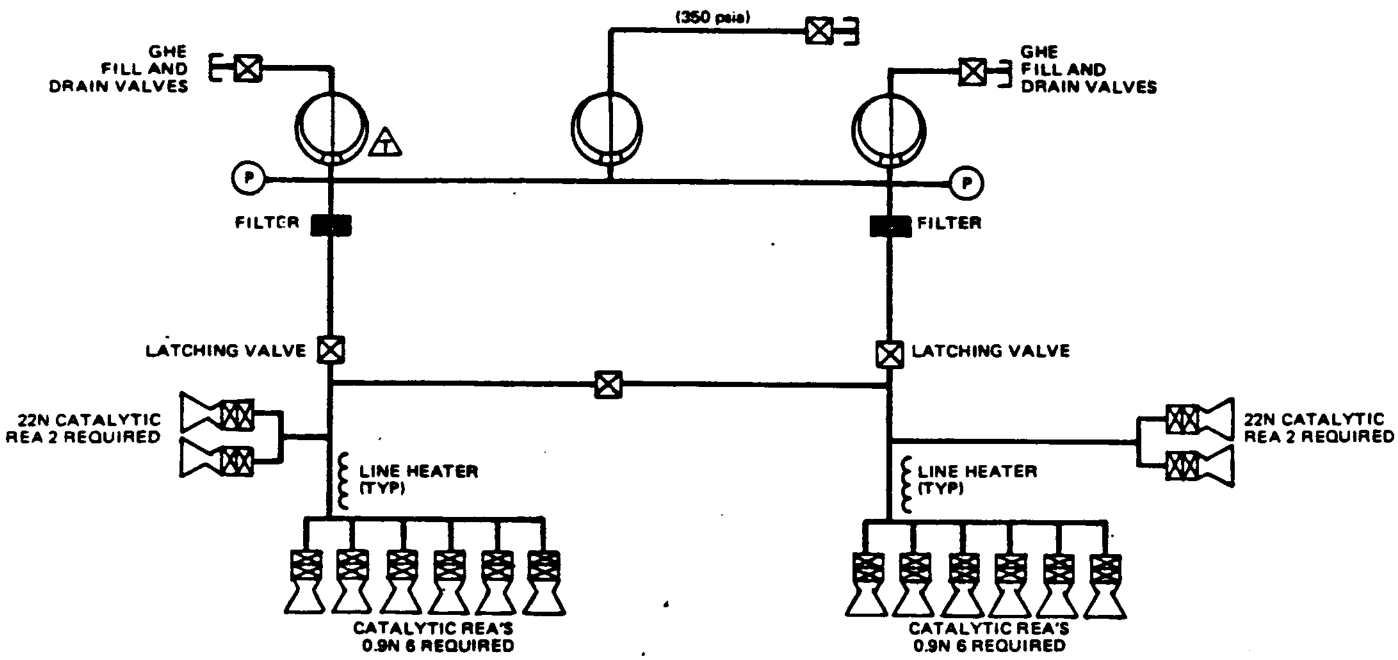

Figure 2-13. Hydrazine Propulsion System Schematic Diagram.

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interconnected on the gas side to equalize liquid flow from the tanks into the 0.935-m diameter tank. Liquid transfer between the 0.564-m diameter tanks is inhibited by check valves in the ullage lines. The flow of gas-free liquid into the 0.935-m diameter supply tank is ensured by the "bubble traps" designed into the 0.564-m diameter tank outlets. During thruster firings, the acceleration will settle liquid over the tank outlets. Four 490N and four 22N thrusters provide the system impulse capability. Nominal system operation is four 490N thrusters firing simultaneously for velocity-change burns. Three-axis control during these burns is accomplished by on-pulsing the appropriate 22N bipropellant and 22N hydrazine thrusters. For the trajectory correction maneuvers which adjust the arrival conditions at Mars (TCM 2 through 4), the 22N bipropellant thrusters are used to provide finer control. These thrusters are also used to inject the spacecraft into a sun synchronous, quarantine orbit at the end of the mission.

Positive containment of the pressurant during regulated propulsion system inactive periods is accomplished by grouping of parallel-redundant, normally-closed and normally-open pyrotechnic valves. Latching solenoid valves provide dual seal backup for the bipropellant thruster valves. Filters are placed immediately downstream of all pyro valves to trap any debris which might contaminate sensitive componments. Service valves which seal with redundant caps are located in the liquid and gas manifolds to facilitate system testing and launch servicing. Subsystem pressures and valve and tank temperature status are telemetered. Positive indication of thruster firing is provided by the thruster valve temperature readout and on-board accelerometers.

Thermal control is accomplished with redundant propellant line, tank and component heaters. Passive thermal control is also provided by thermal blankets and tape wrap on lines and components. Redundant element heaters are used on all valves and on the propellant manifolds. The thermal control design maintains propellant temperatures within design limits. Stainless steel heat shields surround the 490N thrusters to protect the spacecraft structures from overheating as a result of direct radiation from the hot combustion chamber during burns.

All joints between tubing, fittings, and components in the continuously pressurized portions of the propulsion subsystem are welded by the tungsten inert gas (TIG) method. This RCA fabrication technique provides proven, leak-free, structurally sound subsystems.

2.6.2 Hydrazine Propulsion System

The hydrazine system for Mars Observer is an extension of the propulsion subsystems flown to date on more than a dozen RCA Astro-built commercial communications satellites.

Three 0.44-m diameter spherical tanks contained in the hydrazine system operate in a blowdown mode. Each tank contains helium pressurant and hydrazine propellant. Low-g propellant positioning is ensured by means of surface-tension propellant management devices (PMDs). The propellant load is equally distributed among the three tanks, two of which are manifolded to the two redundant sets of attitude control and velocity change thrusters. Twelve 0.9N, hydrazine rocket engine assemblies (REAs) provide for momentum wheel desaturation and four 22N hydrazine REAs are used for 3-axis control during delta-V maneuvers and for Mars mapping, orbit-trim maneuvers. All hydrazine thrusters have series redundant propellant valves. The 0.9N REAs were developed and qualified for the NASA/JPL Mariner-Jupiter-Saturn (Voyager) program and are flown on all current RCA-built Satcom, Spacenet, GSTAR, DBS, and ASC communications satellites. The 22N thrusters have been qualified and flown on Voyager, the Global Positioning Satellite, and extensively on the Lockheed P-95 program.

Filtration of the propellant to the thrusters in the hydrazine system is provided by high-capacity, latch valve inlet filters. Servicing is provided by manual propellant and pressurant service valves on each tank. Subsystem instrumentation includes two pressure transducers that measure delivery pressure to the thrusters. All latch valves have position switches for positive valve state determination. The thrusters have redundant chamber heaters. Thermocouples on each thruster provide telemetry data for ground verification of thruster temperature and operation. Telemetry signals are also provided for tank, line, and thruster valve temperatures.

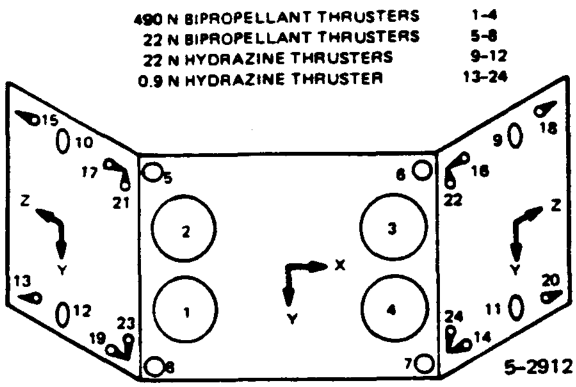


Figure 2-14. Spacecraft Axis Conventions and Thruster Locations.

3.0 Ground Support Equipment

This section describes the Ground Support Equipment (GSE) that will be used at the launch site to support final build-up, pre-launch checkout, and launch operations of the spacecraft. This GSE falls into two distinct categories, the Electrical Ground Support Equipment (EGSE), and the Mechanical Ground Support Equipment (MGSE). The GSE design, operation and heritage are discussed for each piece of equipment. The aspects of GSE design and operation that are safety critical are also discussed. The EGSE is presented in section 3.1. The MGSE is presented in section 3.2. Section 4.0 gives an overview of the conceptual Mars Observer spacecraft launch site ground operations scenario.

3.1 Electrical Ground Support Equipment

The EGSE is used to perform pre-launch checkout of the spacecraft (and it's EASE) and to perform command and monitoring of spacecraft systems through the T-0 umbilical. This equipment is an application of existing EGSE used for past and present RCA STS launched spacecraft (ASC, Satcom, G-Star, and STC-DBS). RCA and NASA have gained a great deal of experience in working with this equipment which will help to minimize the effort required to integrate this equipment with the KSC ground support system. This experience base will also help to minimize EGSE related safety issues and ground processing delays due to EGSE problems.

3.1.1 Spacecraft Checkout Station (SCS)

The SCS is a computer-driven system that contains all of the test equipment necessary to initiate spacecraft commands and to process spacecraft telemetry. The SCS also includes the rf stimuli and measuring equipment needed to measure the performance of the spacecraft rf components. The SCS is used at the launch site to perform initial spacecraft checkout, perform launch pad commanding, perform initial rehearsals of launch pad procedures, and monitor spacecraft safety and health status.

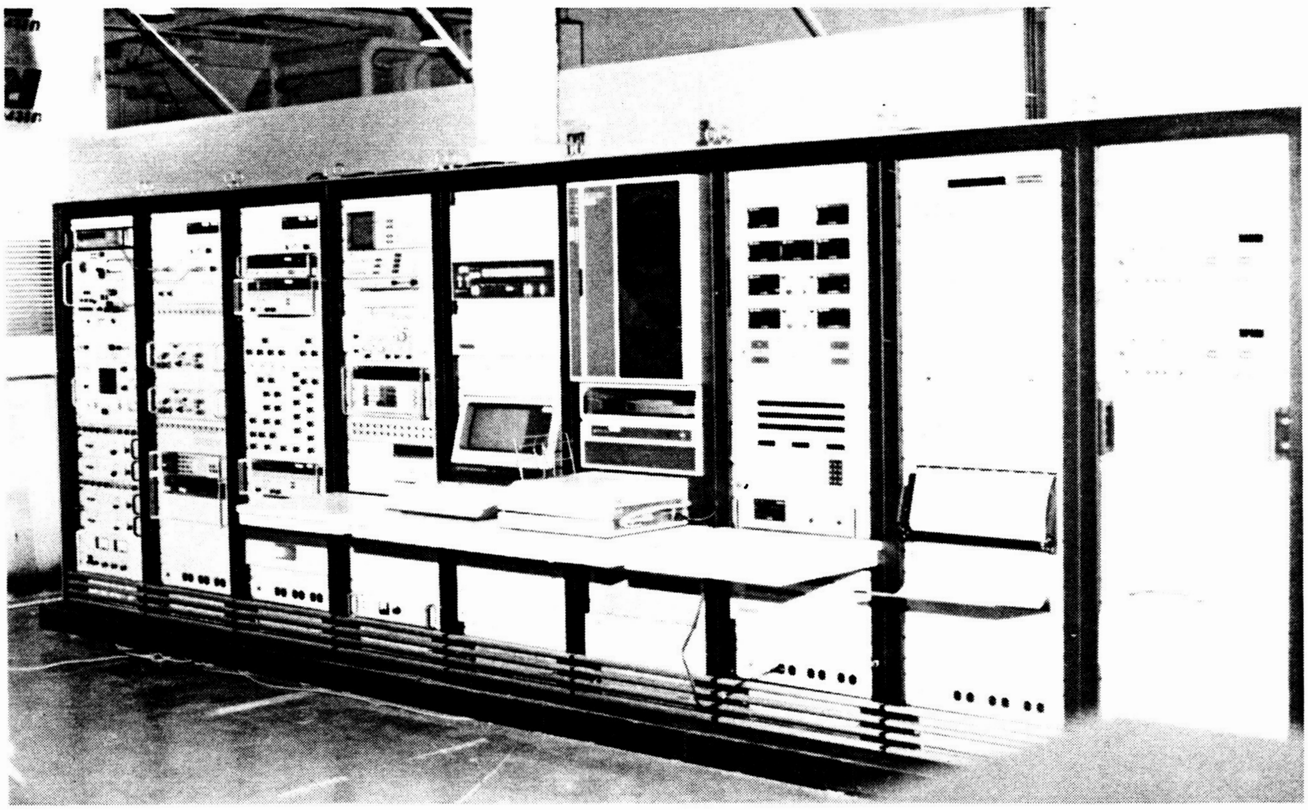


Figure 3-1. Spacecraft Checkout Station.

The initial spacecraft checkout consists of performing a System Electrical Performance Evaluation Test (SEPET) utilizing the SCS. This test is an abbreviated version of the three SEPETs that are performed at RCA during the spacecraft level verification phase. The initial SEPET is performed early in the spacecraft level verification phase, the Final SEPET is performed toward the end of this phase and just prior to spacecraft shipment to the launch site, and the Thermal Vacuum SEPET is performed while the spacecraft is undergoing thermal vacuum testing. These tasks will all be performed by the same SCS which will be used at the launch site for the Launch Site SEPET and pre-launch commanding.

The SCS is shown in figure 3-1 and a diagram of the SCS rack layout is TBS at phase one. The floor space requirements and electrical power requirements will also be supplied at phase one. The SCS will be installed in the Mars Observer Payload Processing Facility (PPF) two weeks prior to the start of spacecraft PPF operations. This time is required to set up and checkout SCS and data links to other Launch Site Equipment (LSE) and to the mission operations network.

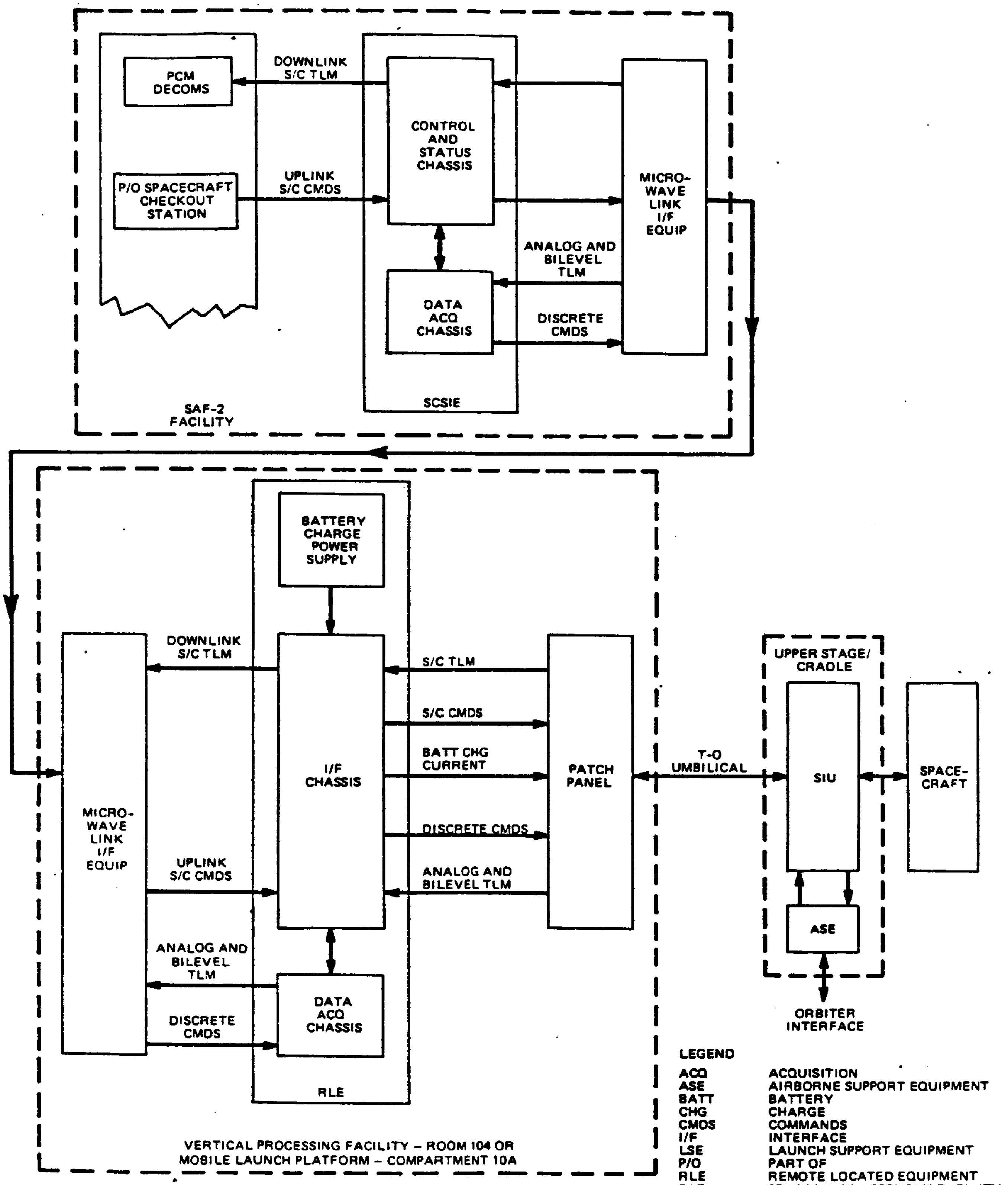
3.1.2 Launch Site Equipment

The primary purpose of the LSE is to provide an interface between the SCS and the Mars Observer spacecraft while the spacecraft is mated to the Upper Stage (TOS) in the Orbiter cargo bay. Other functions are to verify correct operations of the PPF/KSC interfaces and to provide spacecraft simulation such that all TOS-to-spacecraft interfaces can be verified prior to spacecraft integration with the TOS. The LSE consist of four separate subsystems: the SCS Interface Equipment (SCSIE), the Remote Located Equipment (RLE), the T-0 Umbilical Test Set (TUTS), and the Dynamic Spacecraft Simulator (DSS).

A block diagram of the LSE system, shown in Figure 3-2, depicts the basic interconnections used from the SCS through the RLE to the spacecraft. At Kennedy Space Center (KSC) the SCS and CSIE are located in the PPF. The RLE is mounted in a GFE cabinet in room 104 when the spacecraft is in the VPF, or in compartment 10A of the Mobile Launch Platform (MLP) when the spacecraft is in the orbiter bay. The interface between the PPF and VPF or the MLP consists of a video link supplied by NASA. In addition, the PPF supplies MUX-DEMUX equipment at both ends to handle a variety of signals over the link. The following signals will pass between the SCSIE and RLE over the link:

- o Telemetry - Two spacecraft baseband telemetry streams
- o Commands - One SCS-generated baseband command input to the spacecraft
- o Shuttle Interface Unit (SIU) and RLE discrete commands and bilevel status telemetry

Interconnections between the SCSIE, the video link, and the RLE are made via patch panels supplied by KSC. The interface between the RLE and the spacecraft is via the T-minus-0 (T-0) umbilical cabling and the SIU, located in the cradle. As in the video link, interconnections between the RLE and the T-0 umbilical are made via patch panels supplied by KSC. The



LEGEND

ACQ	ACQUISITION
ASE	AIRBORNE SUPPORT EQUIPMENT
BATT	BATTERY
CHG	CHARGE
CMDS	COMMANDS
I/F	INTERFACE
LSE	LAUNCH SUPPORT EQUIPMENT
P/O	PART OF
RLE	REMOTE LOCATED EQUIPMENT
SAF	SPACECRAFT ASSEMBLY FACILITY
S/C	SPACECRAFT
SCSIE	SPACECRAFT CHECKOUT STATION INTERFACE UNIT
SIU	SHUTTLE INTERFACE UNIT
TLM	TELEMETRY

Figure 3-2. LSE Block Diagram.

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connections between the T-0 umbilical and the SIU are via the Standard Mixed Cargo Harness (SMCH) and the cradle harness.

The TUTS connects to the TOS side of the T-0 umbilical prior to the spacecraft being integrated with the Orbiter. It is housed in a portable, ruggedized container, operates from 120 Vac, and provides the following functions:

- o Tests of SCSIE and RLE operation at RCA or the launch site prior to shipment or installation using test cabling.
- o Tests of PPF and RLE operation after installation and connection to KSC interfaces, to validate the launch site interfaces.
- o Provides a means of troubleshooting the LSE or cabling and a means of training personnel without jeopardizing the spacecraft.

The DSS tests the TOS-to-spacecraft interface with the SIU installed in the cradle while in the PPF. The DSS functions are housed in the same portable container as the TUTS, since they share some of the same circuitry and are not used at the same time. The DSS provides the following functions:

- o Simulate the spacecraft interface with the TOS.
- o Simulate the RLE interface with the T-0 umbilical.
- o Provide a means of testing the Cradle/SIU interfaces and the Cradle test set (Orbiter simulator) functions prior to mating with the spacecraft.

The following sections provide more detailed descriptions of the Mars Observer LSE.

3.1.2.1 SCS Interface Equipment (SCSIE)

The CSIE is housed in a sloped front, single-bay console, as shown in Figure 3-3 and is located in close proximity to the SCS cabinets. The SCSIE consists of two chassis: a chassis containing front panel displays, control switches and internal logic, and signal conditioning circuitry; and a data acquisition chassis. The data acquisitiion chassis multiplexes CSIE front panel switch commands onto a single output for transmission to the RLE for control of both RLE and SIU functions. It also demultiplexes analog and bilevel telemetry transmitted from the RLE for presentation on front panel displays. The CSIE provides the following functions:

- o Receive spacecraft PCM telemetry from RLE via microwave link and relay to the SCS after signal conditioning.
- o Receive spacecraft commands from the SCS, condition the signal, and transmit to RLE.

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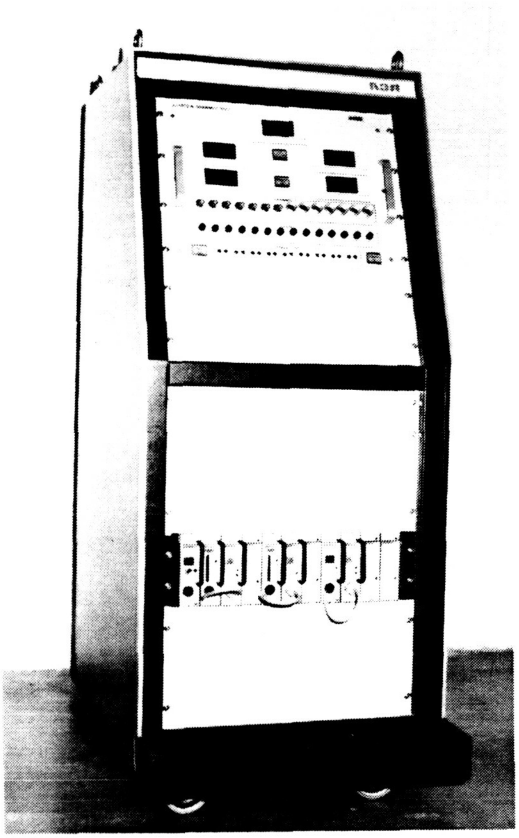

Figure 3-3. SCSIE

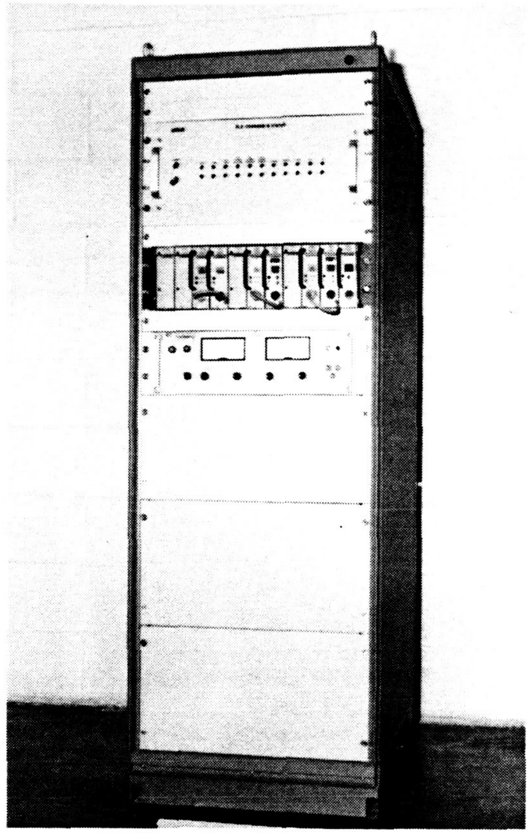

Figure 3-4. RLE

- o Display voltage and current telemetry received from RLE for battery charge power supply located in RLE.
- o Receive and display RLE and SIU bilevel status telemetry.
- o Transmit discrete front panel switch commands for RLE and SIU to RLE.

3.1.2.2 Remote Located Equipment

The RLE consists of a standard 0.48-meter (19-inch), rack-mountable chassis as shown in Figure 3-4, designed for installation in a GFE rack cabinet in room 104 at the VPF or compartment 10A at the MLP. The RLE consists of three chassis: a chassis containing logic, relays, and signal conditioning circuitry; a 10 A battery charge power supply; and a data acquisition chassis. The data acquisition equipment multiplexes both analog and bilevel telemetry inputs from the RLE and SIU onto a single output for transmission to the CSIE. It also receives the discrete front panel switch commands generated at the CSIE and demultiplexes the commands into parallel relay closure type outputs for control of RLE or SIU functions. The RLE provides the following functions:

- o Receive SIU bilevel status telemetry via the T-0 umbilical and transmit to the CSIE.

- o Receive spacecraft baseband telemetry from SIU via the T-0 umbilical, condition the signal, and transmits to CSIE.
- o Monitor battery charge power supply voltage and current, and transmit to CSIE for display.
- o Receive discrete front panel switch commands from CSIE for SIU control, and transmit to SIU via the T-0 umbilical.
- o Provide up to 10A of battery charge current to the SIU via the T-0 umbilical, remotely controlled from the CSIE.

3.1.2.3 T-0 Umbilical Test Set (TUTS)

The TUTS is housed in a portable container for ease of transport (see Figure 3.5). It mates with the TOS side of the T-0 umbilical and provides an accurate simulation of the spacecraft umbilical interfaces. Figure 3-6 depicts the TUTS connected in the LSE KSC test configuration. It is self-contained and operates from 120 Vac. In conformance with KSC safety requirements it contains no internal power sources (batteries). All voltages within the unit can be turned off by a single front panel switch, and the unit can be used at least 30 feet from the Orbiter. The TUTS provides the following functions:

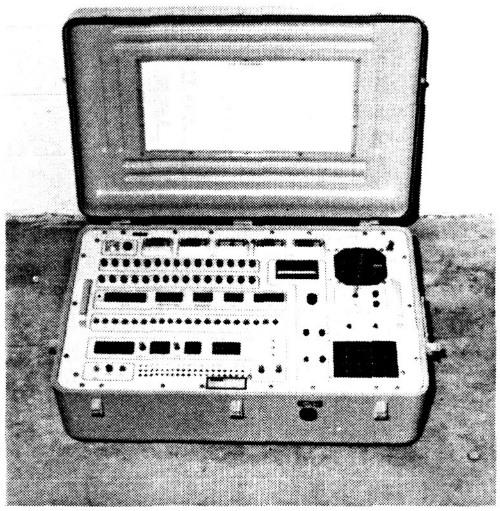

Figure 3-5. DSS/TUTS.

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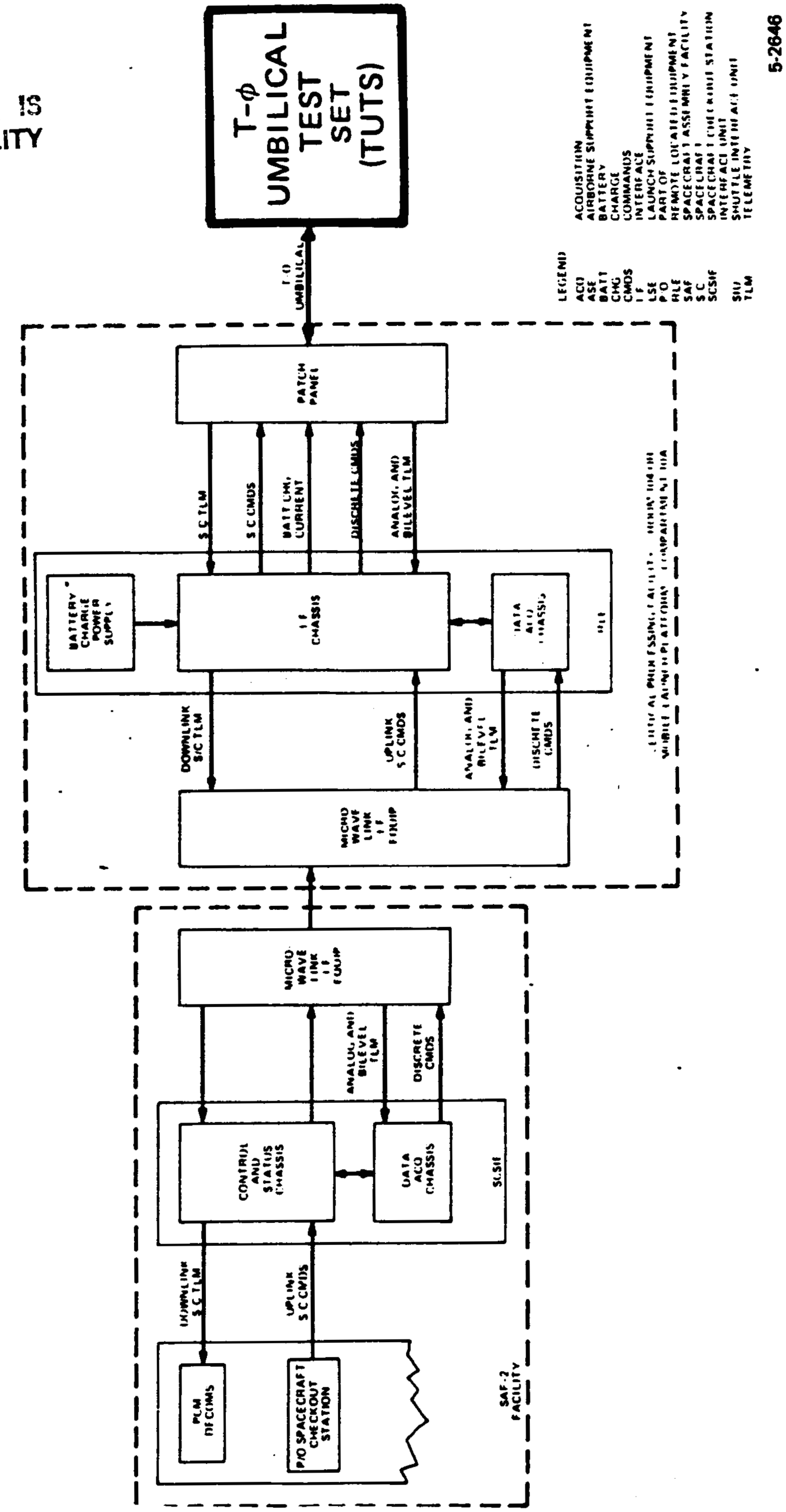

Figure 3-6. TUTS Test Configuration.

- o Generate simulated spacecraft PCM telemetry.
- o Decode and display SCS-generated spacecraft commands.
- o Provide resistive load for battery charge current from RLE power supply.
- o Provide resistive load and status indication of discrete commands generated by the RLE for the SIU.
- o Generate simulated SIU bilevel status telemetry via front panel switches.
- o Provide test points for all umbilical signals.

3.1.2.4 Dynamic Spacecraft Simulator (DSS)

The DSS is housed in the same portable container as the TUTS. It mates with the TOS/SIU and associated Test Set as shown in Figure 3-7. The DSS provides the following functions to simulate the spacecraft interface with the TOS:

- o Generate simulated spacecraft PCM telemetry.
- o Provide test points for strip charting commands to the spacecraft.
- o Provide a resistive load to simulate the spacecraft load on the SIU +28V dc/dc converter.
- o Provide status indicators and resistive loads for TOS timer discrete commands to the spacecraft.
- o Simulate spacecraft safety talk-back circuits using front panel switches.

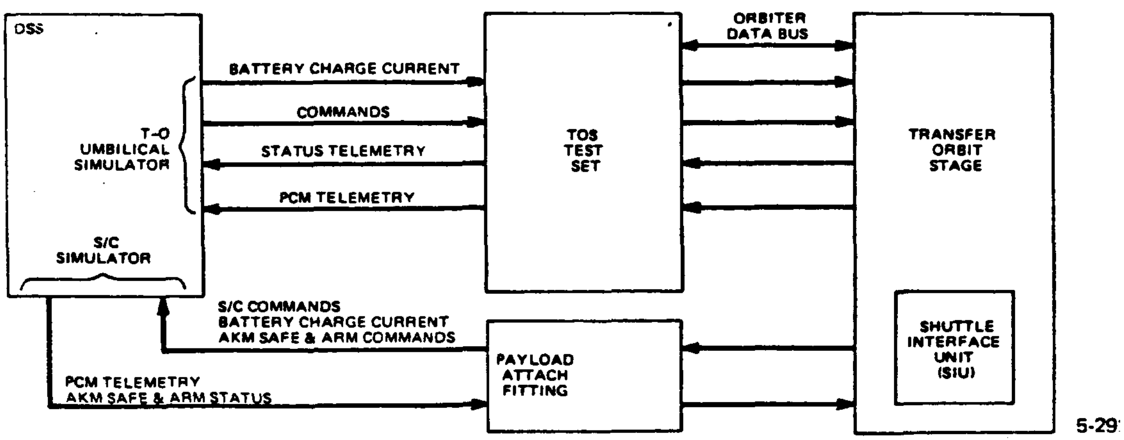

Figure 3-7. DSS Test Configuration.

- Provide simulation of spacecraft separation jumpers.

The DSS provides the following functions to simulate the T-0 umbilical interface to the TOS:

- Provide indications of SIU bilevel status telemetry.
- Provide test points for monitoring spacecraft telemetry with an oscilloscope.
- Generate uplink commands to spacecraft (via SIU) using front panel switches.
- Provide up to 10 A of battery charge current to SIU
- Provide simulation of discrete commands to the SIU using front panel switches.

3.1.2.5 T-0 Umbilical Simulator

TBS at Phase One

3.1.3 Stray Voltage And Bridgewire Checker

This piece of special test equipment (STE) performs two safety-related functions. These tests are performed in the Electromechanical Test Facility at Cape Canaveral Air Force Station (CCAFS). In the first mode, it is connected in turn to the various spacecraft pyrotechnic device firing terminals via a selector switch. This permits measurement of any stray or induced voltages produced across these terminal while other spacecraft tests are being conducted (such voltages could cause premature firing of the pyrotechnic devices).

In the second mode, the STE is connected to the same firing terminals via the selector switch; however, now the resistance across those terminals is measured to ensure the integrity of the pyrotechnic devices safety bridgewires. This STE is small, battery powered, and housed in a rugged container. It incorporates a digital voltmeter and a digital, precision, low-current ohmeter. The design is derived from the current TIROS program unit, modified for the Mars Observer interfaces.

3.1.4 Ordnance Device Simulator

This STE is connected to the pyrotechnic device firing terminals to replace the firing squibs during the launch site SEPET. It uses precision circuit breakers to simulate the burnout of the squib when sufficient current is present for sufficient time. This STE is small,externally powered, and housed in a rugged container. The self-test mode provides verification before use. This design is derived from the current TIROS program unit.

3.1.5 Thruster Simulator

This STE is connected in lieu of the thrusters during propulsion system and attitude and articulation control system tests that are conducted as a part of the launch site SEPET. It displays and records changes of state in firing circuits to allow verification of thruster sequences. This STE is small, externally powered, and housed in a rugged container. The design is an adaptation of a DMSP unit.

3.2 Mechanical Ground Support Equipment

The MGSE consists of the various fixtures required to handle the spacecraft at the launch site. As with the EGSE, the MGSE is an application of existing designs used on past and present RCA STS launched spacecraft (ASC, Satcom, G-Star, and STC-DBS).

3.2.1 Spacecraft Workstand

The spacecraft workstand provides a fixed bench that positions the body of the spacecraft at a convenient height for mechanical work activities while in the PPF. The spacecraft is attached to the workstand via a separation ring, as shown in figure 3-8. The existing spacecraft workstand from the Satcom program will be used for Mars Observer.

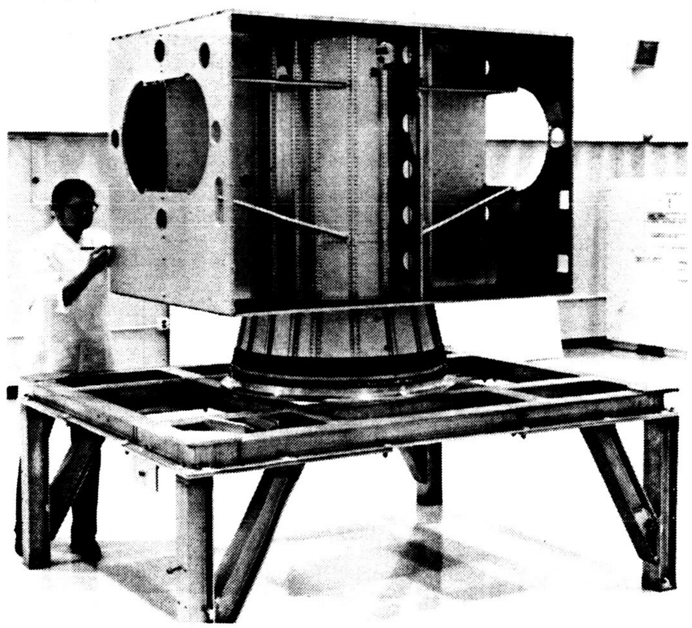

Figure 3-8 Spacecraft Workstand.

3.2.2 Spacecraft Vertical Lifting Fixture

Prior to mating with the TOS upper stage, all spacecraft vertical lifting will be accomplished with the aid of the spacecraft Vertical Lifting Fixture. Hardpoints are designed into the bus structure that direct lifting loads into the center core. Lifting is accomplished through rods that thread into the hardpoints and attach to an overhead spreader suspended from the PPF/HPF bridge cranes. The rods are equipped with ball joints at each end to avoid bending moments. The Mars Observer program will make use of the same lifting fixture that was used on the Satcom Ku-Band program. In addition to use at the launch site, this fixture will support all spacecraft vertical lifting operations at RCA.

3.2.3 TOS Adaptor Simulator

The TOS Adaptor Simulator is used to provide an interface between the spacecraft and the spacecraft workstand (see section 3.2.1) prior to mating with the TOS upper stage. This is the same adaptor that will be used at RCA to interface the spacecraft with the workstand and test fixtures. The Adaptor Simulator is shown mated to the spacecraft workstand in figure 3-8. The Adaptor Simulator for Mars Observer will be similar to the Satcom equivalent, with greater depth to provide clearances for the oxidizer tank and 110 lb_f thrusters.

3.2.4 Workhorse Marmon Band

This Marmon Band is a flight-type band that is used for handling and test only. The actual flight band is used for the TOS upper stage interface fit check and for the final mating of the spacecraft and TOS.

3.2.5 Spacecraft Shipping Container

An aluminum reusable shipping/storage container will be designed for Mars Observer spacecraft that will consist of a base, a suspension system, and a cover. The separation plane between cover and base will be low to afford maximum access to the spacecraft during installation and removal. The container will employ casters for mobility and extendable jacks that will off-load the casters during installation and removal operations and during storage periods. The suspension system is attached to the base through highly damped elastomeric shock absorbers, sized so that the natural frequency of the suspended item is significantly less than the first mode of the spacecraft. The suspension platform will contain an adapter that presents an interface to the spacecraft which is identical to that of the flight TOS adapter. A flight-type Marmon band will secure that spacecraft in an upright (Nadir panel down) position. No other attachments are made to the spacecraft, thus avoiding redundant load paths and/or indeterminate loads. The cover fully surrounds the spacecraft and attaches to the base through a bolted, gasketed joint. The cover is sized to provide generous clearance around the spacecraft to preclude impacting during transportation and inadvertent shock loading. Other features of the container are:

- Thermostatically controlled heaters and an air conditioning system to maintain temperatures within prescribed limits during transportation.
- Insulation lining the cover and base to minimize heat transfer.
- A nitrogen supply and purge system used during transportation and periods of storage.
- Shock, temperature, and humidity recorders.

- Relief valves to maintain pressure equilibrium during temperature or altitude transients.

- Provisions to enable handling by crane or forklift.

- All materials within the container chosen to maintain cleanroom conditions.

In addition, a separate gasoline powered generator will be supplied to power the heaters and air conditioning system during highway transportation.

3.2.6 Propulsion Subsystem Ground Servicing Equipment

The propellant loading operation is made up of four phases in the following sequence:

1. Hydrazine transfer from shipping drums to the N_2H_4 vessel. This operation does not require the spacecraft and may be performed as a parallel operation to pre-loading launch preparations.
2. NTO loading (Figure 3-9).
3. N_2H_4 loading (Figure 3-10).
4. Pressurant loading (Figure 3-11).

The propellant load carts that will be used for bipropellant and large spacecraft will be designed and built by RCA. The same design philosophy and manufacturing processes as used in a flight system will be adhered to when constructing the carts. All loading equipment will be tested prior to use at the launch facility. Testing will include proof pressure testing, and a functional test using NTO, N_2H_4, and GHE for the respective load carts.

A schematic of the NTO load cart is presented in Figure 3-9. The NTO load cart consists of four major subassemblies:

1. NTO source
2. Transfer panel subassembly
3. Molecular sieve
4. NTO drain tank

The NTO source is baselined as the government furnished tank in which the NTO is supplied by Vicksburg Chemical. The transfer panel contains all valves and gauges required for loading operation. The molcular sieve is a titanium column containing LZ-Y82 pellets, a synthetic zeolite. The sieve serves to remove iron oxides in the NTO. The sieve uses a vent line with a sight glass to release any gas bubbles which might form both when fluid initially enters the load cart manifold and during the NTO loading operation. Since the venting gas will contain NO_2, N_2, and NTO fumes, it will be routed, using flex lines, to a facility vapor scrubber for decontamination. After being filtered, the NTO is loaded into the spacecraft via the spacecraft service valves. Propellant gauging during both the NTO and N_2H_4 loading is accomplished by weight. The spacecraft will be mounted directly onto the scale platform. The weight will be monitored real time during the loading operation.

Once the propellant tanks are loaded to the predicted requirement, the service valves are closed. This method will enable an accuracy ot ± 0.5 pound of the entire propellant load. The scale is intrinsically safe for National Electrical Code Class I, Division 1, Group C and D location enabling it to be used in a hydrazine, monomethylhydrazine, and nitrogen tetroxide loading operation.

The high pressurant load panel is shown schematically in Figure 3-10, is used to both load gaseous helium pressurant and conduct the pressurant system proof test at the launch site. It utilizes an explosion proof pnuematic pump controlled by a high pressure control panel. The panel contains all valves and temperature and pressure monitors for the pressurant loading. Monitoring is continuous throughout the loading operation to avoid overheating the spacecraft pressurant components.

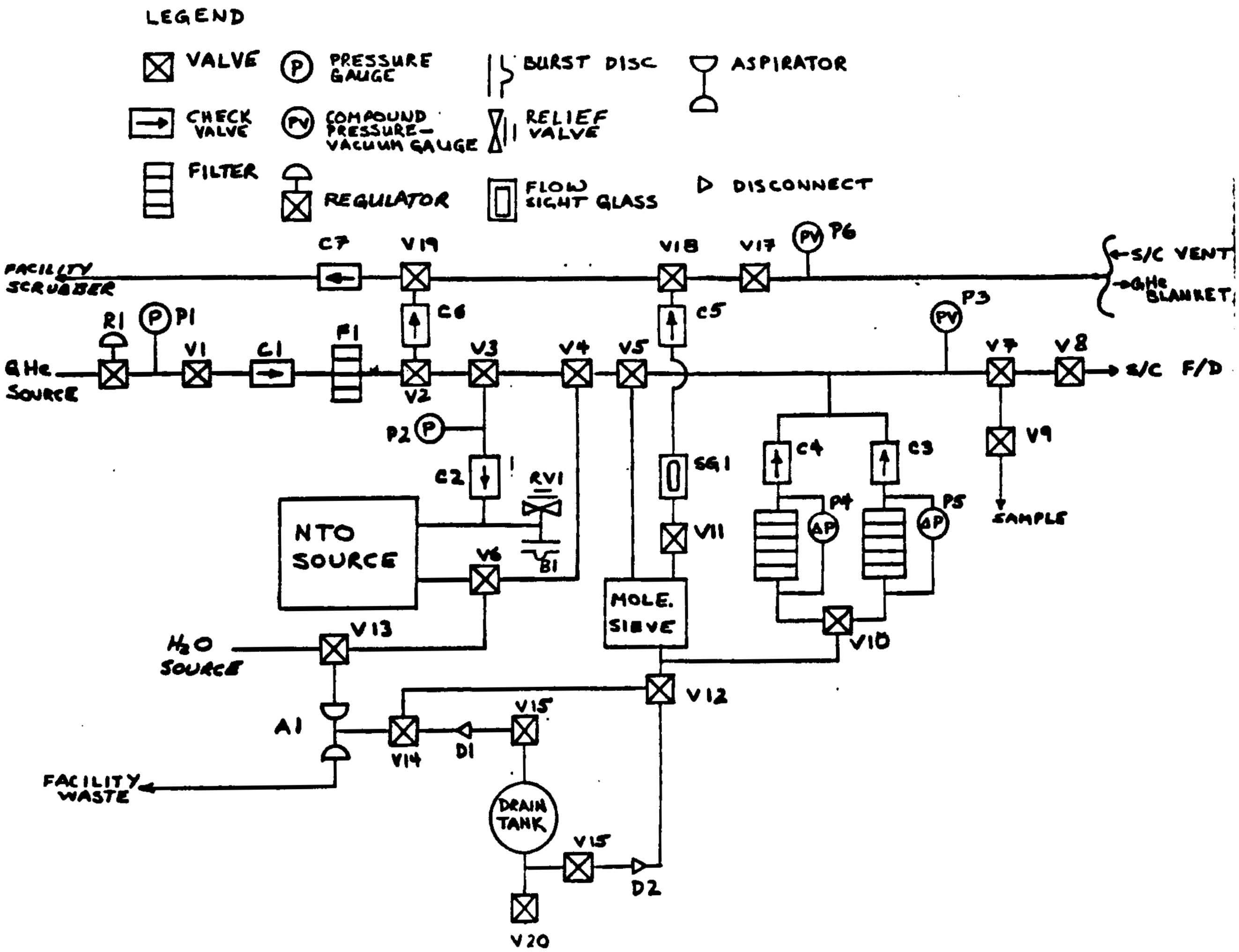

Figure 3-9. NTO Load Cart.

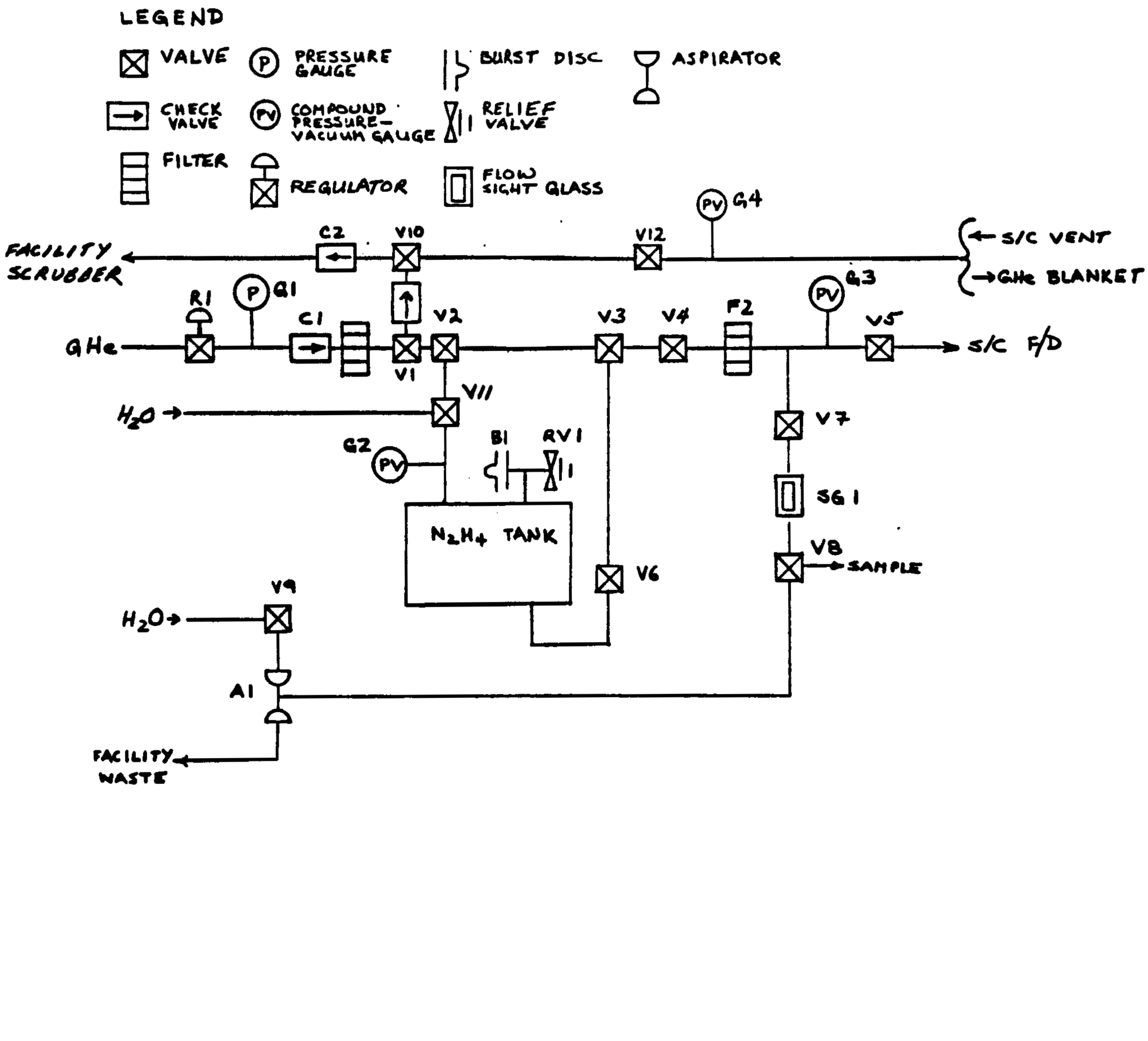

Figure 3-10. Hydrazine Load Cart.

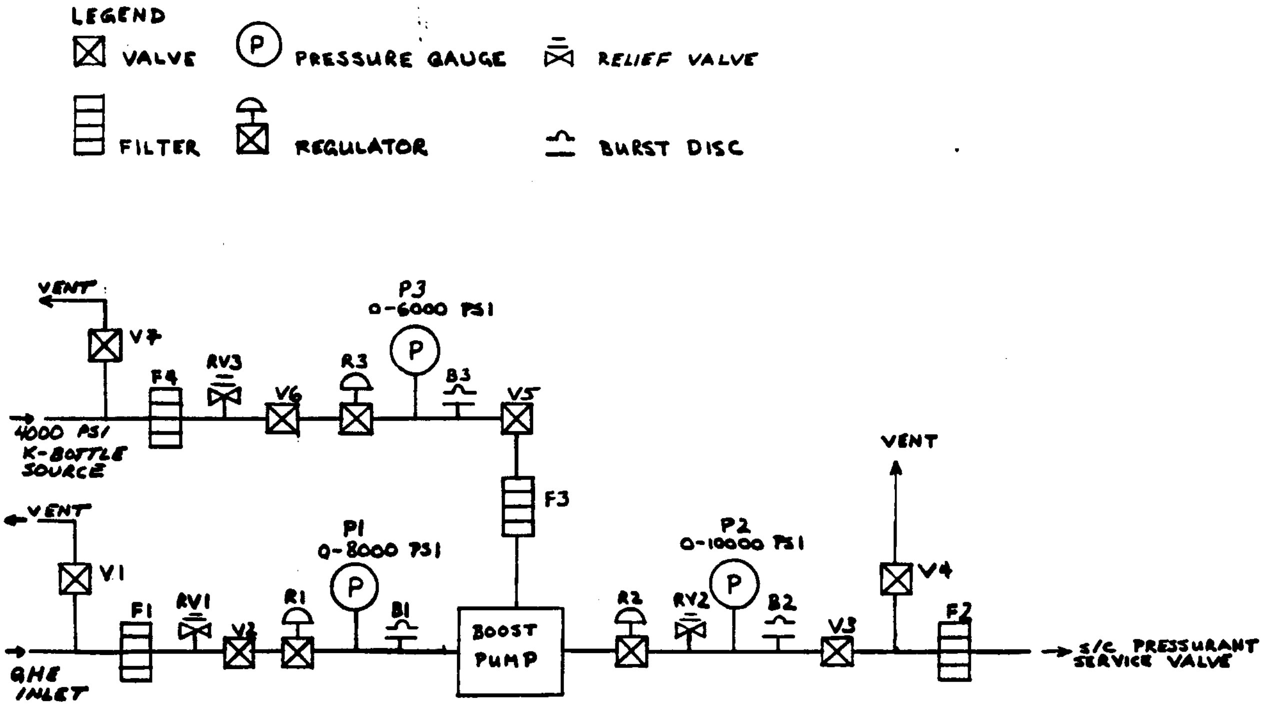

Figure 3-11. Pressurant Load Cart Schematic.

4.0 Ground Operations Scenario

The launch operations sequence is shown in Figure 4-1. This figure shows the events asscociated with the spacecraft that will take place from the time the spacecraft arrives at the Eastern Test Range (ETR) through to launch. The sequence of events occur in the following four stages:

1. Operations at the Spacecraft Checkout Facility (Payload Processing Facility)

2. Operations at Explosive Safe Facilities

3. Operations at Vertical Processing Facility

4. Operations at Launch Complex

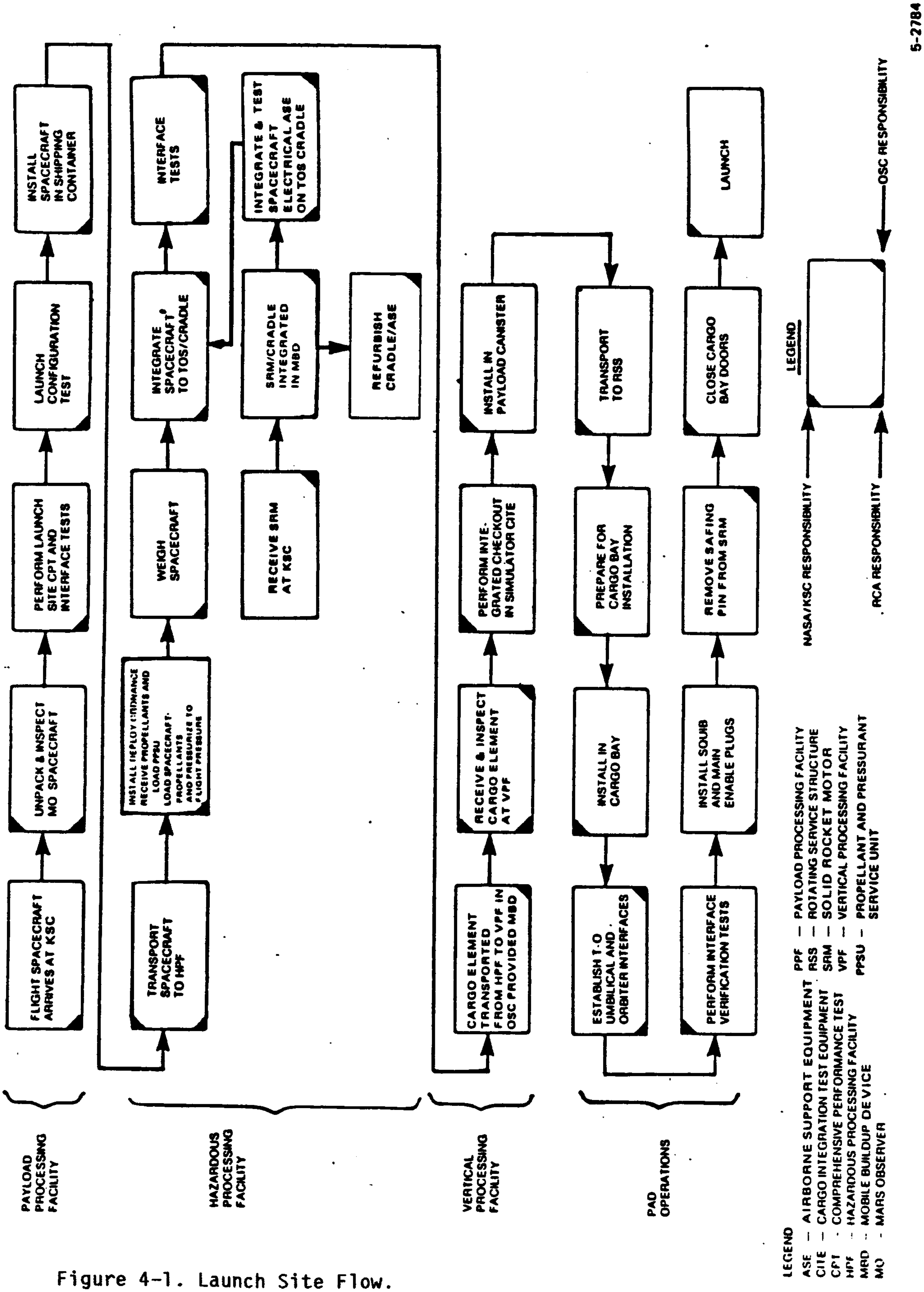

Figure 4-1. Launch Site Flow.

4.1 Prelaunch Operations

The Mars Observer Spacecraft will be transported to the KSC via surface carrier in shipping containers and delivered to the Payload Processing Facility (PPF) for payload checkout and assembly. The Ground Support Equipment (GSE) for the Mars Observer spacecraft and the Airborne Support Equipment (ASE) will be delivered via air-ride truck.

4.1.1 Payload Processing Facility (PPF) Operations

Operations at this facility are primarily associated with the electrical checkout of the spacecraft as it arrives at ETR. The Spacecraft Checkout Station (SCS) will be located at this facility for all launch site operations.

Activities at the PPF will include:

a. Unpack and inspect
b. Perform launch site Standard Electrical Performance and Evaluation Test (SEPET)
c. Perform launch configuration test
d. Perform squib circuit stray voltage test
e. Perform spacecraft battery capacity test
f. Perform spacecraft propulsion system pressure test

During processing at the PPF, the STS will conduct an inspection of the spacecraft for sharp edges/corners/surfaces or protrusions which may damage an EVA crew members suit or associated equipment. Where possible, corrective action (tape, covers, standard teflon guard material, etc.) will be implemented. This inspection will be coordinated with JPL and corrective actions will be approved by JPL.

The Mars Observer spacecraft will then be transported to the Hazardous Processing Facility (HPF)

4.1.2 Payload Ordnance Processing Area Operations

The Mars Observer ordnance will arrive at KSC via temperature-controlled air-ride van and be delivered to the Eastern Space and Missile Center (ESMC) contractor for storage in the payload ordnance processing area until scheduled for ordnance activities. RCA will supply KSC approved procedures to test and inspect all ordnance items. After approval by the KSC Safety Office, these procedures will be performed in the appropriate payload ordnance processing facility. The LSSM will schedule test dates and delivery of the ordnance to the facility with the ESMC contractor.

4.1.2.2 Electrical-Mechanical Testing (EMT) Facility

The Mars Observer bridge-wire resistance checks will be performed in the EMT facility. The ESMC contractor will accomplish the actual testing or inspections based on RCA procedures.

4.1.2.3 Missile Research Test Building (MRTB)

Ordnance receiving inspections will be accomplished by RCA in the MRTB with ESMC contractor support. The TOS SRM receiving inspection and cold conditioning are also performed in this facility.

4.1.3 Hazardous Processing Facility (HPF)

The payload will enter the HPF for payload-unique hazardous operations. RCA is responsible for the hazardous operations performed in the HPF (after procedure approval) and for coordinating test activities and required support with the assigned LSSM. KSC is responsible for providing and scheduling the necessary and agreed-upon support. Hazardous operations will only be performed in the presence of a safety representative. Activities at the HPF shall include:

a. Unpack and inspect
b. Weigh spacecraft dry
c. Load Propellants and pressurize with helium to flight pressure
d. Weigh spacecraft (final, wet)
e. Install ordnance

The spacecraft will then be given to OSC for mating to its upper stage.

4.1.4 Vertical Processing Facility (VPF) Operations

The payload will be moved into the VPF airlock using an electric tug or the snatch blocks. All crane operations in the VPF will be performed by KSC cargo integration contractor personnel. After transportation, the container is cleaned and removed, and the cargo element will be moved into the high bay for rotation and insertion in the Vertical Payload Handling Device (VPHD).

The Mars Observer/TOS cargo element will then be installed into the Cargo Integration and Test Equipment (CITE) and mechanical and electrical connections will be made in preparation for CITE testing.

The purpose of CITE is to verify the interfaces to the Orbiter. The detailed requirements of the interfaces to be tested come from JSC as part of the Operation and Maintenance Requirements Specifications (OMRS) File II. Test plans and testing of the RF and subsystem test will be prepared by RCA in conjunction with NASA and the KSC cargo integration contractor.

Integration and handling activities at the VPF will be under the overall management of NASA and KSC cargo integration contractor with all spacecraft/upper stage owners/contractors participating in test procedure development and implementation. RCA will be responsible for spacecraft-unique testing and for coordinating test activities with the LSSM. KSC is responsible for providing and scheduling agreed-upon support services.

During the cargo integration testing with CITE, safety control and monitor functions will be exercised. All hardwire connections from the Aft Flight Deck (AFD) and T-0 umbilicals will be verified. An abbreviated mission simulation will include prelaunch and predeployment sequences. Data/command links will be exercised and verified. The power transfer from Orbiter to payload/carrier will be demonstrated. Power-on and power-off stray voltage tests will check for stray voltage. The data will be reviewed in realtime prior to CITE disconnection. A Mars Observer end-to-end test with the remote POCC is planned to be conducted.

The entire cargo will then be transferred from the VPHD into the payload canister for transportation to the pad. Environmental control will be provided enroute. The payload GSE will be transported separately by KSC.

4.1.5 Orbiter Processing Facility (OPF) Operations

The shuttle integration contractor is responsible for all installation and integration of the AFD equipment and other ASE with the Orbiter at the OPF. These procedures will be developed by the shuttle integration contractor with input from the user.

4.1.6 Launch Pad Operations

At the pad, the transporter will position the canister at ground level below the Rotating Service Structure (RSS) crane. The canister will be hoisted to the Payload Changeout Room (PCR) door level and locked into position. The PCR environmental seals will be inflated, and the space between the canister and PCR doors purged. The PCR and the canister doors will be opened sequentially; the cargo will be transferred out of the canister by the Payload Ground Handling Mechanism (PGHM) and into the PCR. The PCR and canister doors will be closed and the canister lowered and removed. A transportation inspection of the cargo will be performed.

4.1.7 Orbiter Integrated Operations

After the crawler transporter moves the shuttle to the pad and the Mobile Launch Platform (MLP) is positioned on the pad, the RSS will be rotated into position for transfer of the cargo into the Orbiter. The PCR environmental seals will be inflated; the cavity between the PCR doors and the Orbiter cargo bay doors will be purged, and the doors opened sequentially. The cargo will be installed in the Orbiter cargo bay by the shuttle integration contractor using the PGHM and secured.

Prior to connecting the cargo-to-Orbiter electrical interfaces, power-off continuity and isolation/power-on voltage tests will be run to verify proper power system conditions. Then the interfaces will be mated and the elements will be powered to verify the interfaces. The testing will include interface verification and end-to-end tests.

The same series of tests specified in paragraph 4.1.5 shall be repeated on the cargo unit when it is installed in the cargo bay. At the completion of the verification testing the flight arming plugs will be installed, and final ordnance connections made. Then the payload GSE will be moved into the PCR, the Orbiter doors closed, and the RSS rotated to the launch position.

After the cargo bay doors are closed, the spacecraft batteries will be brought up to a fully charged state. The fully charged state will be maintained by trickle charging. Control for charging will be through the T-0 umbilical. The primary power for charging will be from the Orbiter power bus with a backup through the T-0 umbilical.

Shortly before launch, the momentum wheels will be turned on at low rotational speed (220 RPM). This activity shall be done by commands through the T-0 umbilicals as controlled by the spacecraft checkout station located at the PPF.

4.2 Postlaunch Operations

When pad access has been granted following the launch, the payload GSE will be packaged by the user for removal from the PCR. KSC will provide the handling and transportation support necessary to return the GSE to the PPF.

At completion of activities in the PPF, RCA will package (with KSC assistance if required) all material. KSC will provide personnel and equipment to load the GSE onto transportation for return to the PPF facility.

4.3 Postlanding Operations

When the Orbiter lands at the KSC Shuttle Landing Facility, a payload bay purge of conditioned air will be connected and the Orbiter will be towed to the OPF. After Orbiter safing, the payload bay doors will be opened, and the ASE mounted on the TOS cradle will be disconnected, removed, and installed in transportation containers. KSC will transport the equipment to a previously designated KSC facility for refurbishment or return it to RCA.

4.4 Contingency Operations

4.4.1 Prelaunch Abort

Approval for conducting nonscheduled payload activities will be the result of a realtime decision involving the program participants.

The payload is safed by:

TBS at phase 1

4.4.2 Return to Launch Site Abort

In the event of an abort landing on KSC, the postlanding activities will be subject to orbiter contingency operations/planning based on the nature of the abort. The nominal planning calls for services to be established to the Orbiter within 15 minutes after landing. The Orbiter will then be towed to The OPF, safed, and payload bay doors opened. The cargo will be removed, safed, and placed in the canister. Then it will be taken to the Vehicle Assembly Building, rotated to the vertical, and taken to the VPF for cargo installation into the VPHD. The cargo will be deintegrated and the individual payloads returned to their respective HPF.

4.4.3 Secondary and Contingency Landing Sites

The overall activities required to safe the Orbiter, remove the payload, and return the Orbiter from a secondary/contingency landing site are outlined in KTV-PL-0014, "KSC Off-Site Operations Plan."

The payload is safed by:

TBS at phase 1

5.0 Requirements Matrix

The Mars Observer Requirements Matrix is shown in figure 5-1. The requirements matrix descriptive data can be found on the pages that immediately follow figure 5-1.

STS PAYLOAD SAFETY REQUIREMENTS APPLICABILITY MATRIX

1. PAYLOAD/SUBSYSTEM: Mars Observer	2. DATE: 8/15/86	3. PHASE: 0	4. PAGE 1 OF 1

6. NHB 1700.7A REQUIREMENTS

	5. PAYLOAD ELEMENT	200 GENERAL	201 FAILURE TOLERANCE	202.1 CRITICAL HAZARDS	202.2 CATASTROPHIC HAZARDS	202.2A SOLID PROPELLANT	202.2B LIQUID PROPELLANT	202.2C DEPLOYMENT/SEPARATION	202.2D BAY DOOR CLOSURE	202.2E PAYLOAD RF XMTR	203 RETRIEVAL OF PAYLOAD	204 HAZARD DETECTION & SAFING	205 P/L CONTINGENCY RETURN	206 FAILURE PROPAGATION	207 REDUNDANCY SEPARATION	208.1, 2 & 3 STRUCTURAL	208.4 & 5 PRESSURE SYSTEM	208.6 DECOMPRESSION	208.7 SEALED CONTAINERS	209.1 HAZARDOUS MATERIALS	209.2 CABIN MATERIALS	209.3 FLAMMABLE MATERIALS	209.4 OFFGASSING MATERIAL	210 PYROTECHNICS	211 DESTRUCT SYSTEMS	212.1 IONIZING RADIATION	212.2 NONIONIZING RADIATION	213 ELECTRICAL SYSTEMS	214 VERIFICATION REQMTS.	215 HAZARDOUS PROCEDURES	216 REFLOWN HARDWARE	217 EXTRAVEHICULAR ACTIVITY	218 SERIES PAYLOAD	219 FLAMMABLE ATMOSPHERE	7. SAFETY ANALYSIS BY: (NAME, ORG, INITIAL)
1	Electrical	—	—	—	—	—	—	—	—	—	—	—	—	X	—	—	—	—	—	—	—	—	—	—	—	—	X	X	—	—	—	—	—	X	
2	Environmental	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	X	X	—	—	—	—	—	—	—	X	—	—	—	—	—	—	
3	Materials	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	X	—	X	—	—	—	—	—	—	—	—	—	—	—	—	
4	Mechanical	—	—	—	X	—	—	X	X	—	—	—	X	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	
5	Pressure Sys.	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	X	—	—	X	—	—	—	—	—	—	—	—	—	—	—	—	—	—	
6	Propulsion	—	—	—	—	—	X	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	X	—	—	—	—	—	—	—	—	—	—	
7	Pyrotechnics	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	X	—	—	—	—	—	—	—	—	—	—	
8	Radiation	—	—	—	—	—	—	—	—	X	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	X	—	—	—	—	—	—	—	
9	Structures	—	—	—	—	—	—	—	—	—	—	—	—	—	—	X	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	

8. LEGEND:

— NOT APPLICABLE	/ APPLICABLE: NO HAZARD IDENTIFIED
X APPLICABLE: HAZARD IDENTIFIED	■ WAIVER DEVIATION REQUIRED

9.

	IMPLEMENTATION PERSONNEL	DATE	PAYLOAD ORGANIZATION PERSONNEL	DATE
PREPARED BY:	D [signature]	8/15/86		
REVIEWED BY:				
APPROVED BY:				

JSC Form 1090 (APR 83)

Figure 5-1. Mars Observer Requirements Matrix.

STS PAYLOAD SAFETY REQUIREMENTS
APPLICABILITY DESCRIPTIVE DATA

MATRIX ELEMENT	COMMENTS
1-219	Refer to PHR 1 - Arcing of exposed powered electrical circuits and devices or payload temperatures greater than 352°F could cause ignition of flammable atmosphere in cargo bay.
1-213 206	Refer to PHR's 2, 3, - Electrical faults such as shorts, grounds, overvoltages, or arcing resulting in damage to STS circuitry or malfunctioning of electrical thermal control devices causing ignition of adjacent Orbiter or payload flammable material.
1-212.2	Refer to PHR 4 - Improper shielding and/or filtering of harnesses and spacecraft components causing spurious non-ionizing radiation in Orbiter Cargo Bay which exceeds levels specified in JSC 07700 Vol. XIV, Attachment 1.
2-209.1 -213	Refer to PHR's 5,6 - Loss of heaters which prevent propellant from freezing causing rupture of containers and/or lines and release of hydrazine. Heater failure causing temperature and/or pressure increases and formation of ignition sources.
2-208.7 -209.1	Refer to PHR 7 - Excessive pressure buildup of sealed containers causing rupture of hardware releasing debris.
3-209.3	Refer to PHR 8 - Use of flammable materials in payload design or improper separation of them from ignition sources resulting in Orbiter or other payload damage.
3-209.1	Refer to PHR 9 - Use of non-compatible materials that may react and cause rupture of propellant tanks or lines and/or NiCd battery pressure vessel containers.
4-205	Refer to PHR 11 - Failure of electrical circuitry resulting in unsafe conditions in the event of a contingency return of the payload and Orbiter.
4-202.2	Refer to PHR 12 - Failure of rotating equipment which causes the release of debris.

STS PAYLOAD SAFETY REQUIREMENTS
APPLICABILITY DESCRIPTIVE DATA

MATRIX ELEMENT	COMMENTS
4-202.2C -202.2D	Refer to PHR 13 - Premature deployment of payload appendages (antennas, solar arrays, booms) would prevent payload bay door closure and/or safe landing of the Orbiter.
5-209.1	Refer to PHR 14 - Release of propellant from spacecraft tanks and/or lines due to leakage causing corrosion.
5-208.4 -208.5	Refer to PHR 15, 16 - Structural failure causing rupture of the propellant tanks and/or lines that releases debris or propellant damaging the Orbiter. Structural failure or electrical faults of the NiCd cells causing rupture releasing debris and/or electrolyte.
6-202.2B	Refer to PHR 17 - Premature firing of propulsion subsystem thrusters that causes collision of spacecraft with Orbiter or release of propellant in payload bay.
7-210	Refer to PHR 18 - Premature firing of initiators could cause release of appendages resulting in collision.
8-202.2E -212.2	Refer to PHR 19 - Inadvertent operation of payload transmitters in excess of levels defined in JSC 07700 Vol. XIV, Attachment 1 is a catastrophic hazard.
9-208.1 -208.2 -208.3	Refer to PHR 20 - Failure of structure or appendages may cause damage to the Orbiter. Refer to PHR 10 - Breakage of payload structure due to stress corrosion may damage the Orbiter.

6.0 Phase 0 Hazard Reports

The Phase 0 Hazard Reports have been prepared as outlined in JSC-13830A, Implementation Procedure for STS Payloads System Safety Requirements, Revision A, dated May 16, 1983. The Mars Observer Safety Analysis has been performed using the guidelines of JSC-11123, Space Transportation System Payload Safety Guidelines Handbook, with Change 1, dated September, 1978. The results of this analysis are presented here in the Phase 0 hazard reports.

Section 6.1 presents the Phase 0 hazard reports for flight operations. Section 6.2 presents the Phase 0 hazard reports for ground operations.

6.1 Flight Payload Hazard Reports

The Mars Observer Phase O Flight Payload Hazard Reports are dispositioned as follows.

Table of Contents

PAYLOAD HAZARD REPORT		NO. FLIGHT: 1 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Electrical	HAZARD GROUP: Fire	DATE: October 1986
HAZARD TITLE: Ignition of Flammable Atmosphere		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 219		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Payload ignition of flammable atmosphere in cargo bay during abort and contingency landing of the STS.

HAZARD CAUSES:

1. Arcing of exposed powered electrical circuits and devices produced by making and/or breaking of electrical circuits or electrostatic discharge.
2. Payload Temperatures greater than 352°F.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 2 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Electrical	HAZARD GROUP:	DATE: October 1986

HAZARD TITLE: Damage to STS Circuitry

APPLICABLE SAFETY REQUIREMENT:	HAZARD CATEGORY:
NHB 1700.7A - 213, 206	[ ] CATASTROPHIC
	[X] CRITICAL

DESCRIPTION OF HAZARD:

Fault in Payload damages STS circuitry. Hazard is critical since payload bay power is fused by Orbiter.

HAZARD CAUSES:

1. Internal payload electrical faults such as shorts, grounds, overvoltages, etc. that could cause overloading of STS circuitry.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 3 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Electrical	HAZARD GROUP: Fire	DATE: October 1986
HAZARD TITLE: Thermal Ignition Source		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 213		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Generating sufficient heat from payload electrical circuits to ignite adjacent orbiter or payload flammable material.

HAZARD CAUSES:

1. Internal shorts or arcing in electrical circuitry.
2. Malfunction of electrical thermal control devices.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 4 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Electrical	HAZARD GROUP: Radiation	DATE: October 1986

HAZARD TITLE: Susceptibility To Electromagnetic Radiation

APPLICABLE SAFETY REQUIREMENT:	HAZARD CATEGORY:
NHB 1700.7A - 212.2	X CATASTROPHIC
	CRITICAL

DESCRIPTION OF HAZARD:

Spurious non-ionizing radiation from the Orbiter, copassenger payloads, or the MO spacecraft while in Orbiter Cargo Bay exceeding levels specified in JSC 07700 Vol. XIV, Attachment 1 during normal payload operation.
(NOTE: Radiation from antennas is covered in Payload Hazard Report 19, "Excess Transmitter Radiation.")

HAZARD CAUSES:

1. Radiation of energy from the Orbiter, copassenger payloads, or the spacecraft components which adversely effects the MO spacecraft due to improper shielding or improper filtering of the MO spacecraft components.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 5 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Environmental Control	HAZARD GROUP: Temperature Extremes	DATE: October 1986
HAZARD TITLE: Temperature Extremes (Low)		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 209.1a, 213		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Low temperature extremes could result in the release of hazardous material (propellant).

HAZARD CAUSES:

1. Loss of heaters that could allow freezing of payload fluids to such an extent that containers and/or lines could rupture. Propellant tanks or lines are the only possible items.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT	NO. FLIGHT: 6 GROUND:
PAYLOAD: Mars Observer	PHASE: 0

SUBSYSTEM: Environmental Control	HAZARD GROUP: Temperature Extremes	DATE: October 1986

HAZARD TITLE: Temperature Extremes (High)

APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 209.1a, 213	HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

High temperature extremes could cause the release of hazardous material (propellant) or the formation of ignition sources

HAZARD CAUSES:

1. Failure of a heater in On position that could cause high temperatures, resulting in propulsion system pressure increases or the formation of ignition sources.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 7 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Environmental Control	HAZARD GROUP: Contamination	DATE: October 1986
HAZARD TITLE: Rupture of Sealed Containers During Ascent or Descent		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 208.7, 209.1		HAZARD CATEGORY: [ ] CATASTROPHIC [X] CRITICAL

DESCRIPTION OF HAZARD:

Rupture of sealed containers releasing debris. The sealed containers are the RWA's and small relays internal to subsystem boxes.

HAZARD CAUSES:

1. Excessive pressure buildup in sealed containers.
2. Undetected flaws or cracks.
3. Physical damage during handling.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 8 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Materials	HAZARD GROUP: Fire	DATE: October 1986
HAZARD TITLE: Flammability of Materials		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 209.3		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Orbiter Damage or other payload damage due to ignition of flammable materials.

HAZARD CAUSES:

1. Use of flammable materials in payload design.
2. Improper separation of flammable materials from potential ignition sources.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

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PAYLOAD HAZARD REPORT

PAYLOAD HAZARD REPORT	NO. FLIGHT: 9 GROUND:
PAYLOAD: Mars Observer	PHASE: 0

SUBSYSTEM: Materials	HAZARD GROUP: Fire, Contamination, Corrosion	DATE: October 1986

HAZARD TITLE: Hazardous Materials Compatibility

APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 209.1	HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Reaction of containing materials with hazardous fluids, specifically hydrazine tanks and NiCd battery pressure vessel containers.

HAZARD CAUSES:

1. Use of non-compatible materials that may react.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 10 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Structures	HAZARD GROUP: Corrosion	DATE: October 1986
HAZARD TITLE: Stress Corrosion Structural Weakening		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 208.3		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Damage to the Orbiter due to breaking of structural parts of the payload.

HAZARD CAUSES:

1. Stress Corrosion of payload structural materials weakening them past their design limits.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT	NO. FLIGHT: 11 GROUND:

PAYLOAD: Mars Observer	PHASE: 0

SUBSYSTEM: Mechanical	HAZARD GROUP: Loss of Orbiter Entry Capability	DATE: October 1986

HAZARD TITLE: Contingency Return Considerations

APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 205	HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Spacecraft failure prior to or during contingency return or abort causing damage to the Orbiter.

HAZARD CAUSES:

1. Spacecraft may be in deployment mode configuration and otherwise electrically unsafe.
2. Buildup of temperature within the orbiter bay.
3. Possible high landing loads.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 12 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Mechanical	HAZARD GROUP: Collision	DATE: October 1986
HAZARD TITLE: Rotating Equipment		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 202.2		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Debris from rotating equipment failure could damage Orbiter. The RWA's and gyro assemblies are possible rotating equipment that could fail.

HAZARD CAUSES:

1. Failure of rotating equipment (reaction wheels and/or gyro assemblies) during STS launch or deployment operations releasing debris.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT	NO. FLIGHT: 13 GROUND:
PAYLOAD: Mars Observer	PHASE: 0

SUBSYSTEM:	HAZARD GROUP:	DATE:
Mechanical	Collision	October 1986

HAZARD TITLE: Premature Deployment

APPLICABLE SAFETY REQUIREMENT:	HAZARD CATEGORY:
NHB 1700.7A - 202.2C, 202.2D	X CATASTROPHIC
	CRITICAL

DESCRIPTION OF HAZARD:

Premature deployment of spacecraft appendages that may prevent payload bay door closure and/or safe landing of Orbiter.

HAZARD CAUSES:

1. Premature release of solar panel restraint bands.
2. Premature deployment of the high gain antenna.
3. Premature deployment of an instrument boom.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 14 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Pressure Systems	HAZARD GROUP: Contamination Corrosion Fire	DATE: October 1986

HAZARD TITLE: Leakage of Hazardous Fluids

APPLICABLE SAFETY REQUIREMENT:	HAZARD CATEGORY:
NHB 1700.7A - 209.1a	X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Release of propellant (MON-1, MMH, N_2H_4) from spacecraft's propulsion subsystem damaging the Orbiter.

HAZARD CAUSES:

1. Leakage of propellant from seals, valves, or connections due to material defect, degradation, undetected damage, or weld failure.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT	NO. FLIGHT: 15 GROUND:

PAYLOAD: Mars Observer	PHASE: 0

SUBSYSTEM: Pressure Systems	HAZARD GROUP: Contamination Corrosion Fire	DATE: October 1986

HAZARD TITLE: Rupture of Pressure Vessels

APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 208.4, 208.5	HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Rupture of pressure vessels releasing debris or propellant (MON-1, MMH, N_2H_4) into the payload bay that could damage the Orbiter.

HAZARD CAUSES:

1. Structural failure of propellant tanks or lines due to material defects, degradation, undetected damage, weld failure, or insufficient factor of safety in the design causing rupture rupture of tanks and/or lines which could result in the release of propellant.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 16 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Pressure Systems	HAZARD GROUP: Contamination Corrosion	DATE: October 1986

HAZARD TITLE: Rupture of NiCd Batteries

APPLICABLE SAFETY REQUIREMENT:	HAZARD CATEGORY:
NHB 1700.7A - 208.4, 208.5	X CATASTROPHIC
	CRITICAL

DESCRIPTION OF HAZARD:

Rupture of the NiCd battery cells, releasing debris and/or electrolyte.

HAZARD CAUSES:

1. Structural failure of the NiCd cells.
2. Overpressurization caused by internal or external short, cell reversal, overcharging, temperature extremes, or battery plate insulator failure.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 17 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Propulsion	HAZARD GROUP: Collision	DATE: October 1986
HAZARD TITLE: Propulsion Subsystem		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 202.2b(1), 209-1a, 213		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

1. Premature firing of the propulsion subsystem thrusters.
2. Heating of propellant to hazardous temperature levels.

HAZARD CAUSES:

1. Failure of the propulsion subsystem or inadvertent commanding that allows flow of propellant to the thrusters.
2. Relay failure causing a latch valve coil to be continuously energized. This would heat the latch valves and raise the temperature of the propellant adjacent to them.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 18 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Pyrotechnics	HAZARD GROUP: Collision	DATE: October 1986
HAZARD TITLE: Inadvertent Firing of Initiators		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 210		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Firing of initiators could cause inadvertent release of restraints on appendages, causing collision of the spacecraft with the Orbiter.

HAZARD CAUSES:

1. Premature firing of initiator due to stray EMI.
2. Failure in fire control circuitry.
3. Explosion of initiator.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT	NO. FLIGHT: 19 GROUND:

PAYLOAD: Mars Observer	PHASE: 0

SUBSYSTEM: Communications	HAZARD GROUP: Radiation	DATE: October 1986

HAZARD TITLE: Excess Transmitter Radiation

APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 202.2e, 212.2	HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Exposure of the Orbiter to RF radiation in excess of JSC 07700 Vol. XIV.

HAZARD CAUSES:

1. Inadvertent operation of RF transmitters causing radiation from transmitters in excess of permitted levels.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: 20 GROUND:
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Structures	HAZARD GROUP: Collision	DATE: October 1986
HAZARD TITLE: Structural Failures		
APPLICABLE SAFETY REQUIREMENT: NHB 1700.7A - 208.1, 208.2, 208.3		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Failure of the primary spacecraft structure or failure of critical appendages causing movement of the payload or appendages in the orbiter bay which may cause orbiter damage.

HAZARD CAUSES:

1. Insufficient factor of safety
2. Fracture growth

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

6.2 Ground Payload Hazard Reports

The Mars Observer Phase 0 Ground Payload Hazard Reports are dispositioned as follows.

Table of Contents

PAYLOAD HAZARD REPORT		NO. FLIGHT GROUND: 1
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Electrical	HAZARD GROUP: Electrical Shock	DATE: October 1986
HAZARD TITLE: Electrical Hazards		
APPLICABLE SAFETY REQUIREMENT: KHB 1700.7A - 4.3.2		HAZARD CATEGORY: X CATASTROPHIC; CRITICAL

DESCRIPTION OF HAZARD:

Potential electrical shock hazard to operating personnel.

HAZARD CAUSES:

1. Inadequate fault protection/detection in the Spacecraft Checkout Station (SCS) Design, other GSE, or the spacecraft.
2. Improper installation, assembly, operator error, etc.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: GROUND: 2
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Mechanical	HAZARD GROUP: Collision Injury	DATE: October 1986

HAZARD TITLE:
Failure of Lifting and/or Handling Equipment During Ground Operations

APPLICABLE SAFETY REQUIREMENT:	HAZARD CATEGORY:
KHB 1700.7A - 4.5.1	X CATASTROPHIC
	CRITICAL

DESCRIPTION OF HAZARD:

Inadvertent dropping of Spacecraft could result in equipment damage and/or personnel injury during handling operations.

HAZARD CAUSES:

1. Handling equipment failure due to insufficient factor of safety.
2. Improper rigging and/or procedures in using handling equipment.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT

PAYLOAD HAZARD REPORT	NO. FLIGHT: GROUND: 3
PAYLOAD: Mars Observer	PHASE: 0

SUBSYSTEM: Propulsion	HAZARD GROUP: Injury Contamination Illness	DATE: October 1986

HAZARD TITLE:
Release of Propellant

APPLICABLE SAFETY REQUIREMENT: KHB 1700.7A - 4.3.3, 4.3.7	HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Rupture and/or leakage of the propulsion subsystem or propulsion GSE tanks and/or lines releasing propellant/pressurant (MON-1, MMH, N_2H_4, Helium) which could cause personnel injury or illness from debris or depletion of breathable atmosphere.

HAZARD CAUSES:

1. Leakage of propellant through propulsion subsystem/propulsion GSE valves or connections.
2. Fluid spills due to improper loading procedures.
3. Structural failure of tanks and/or lines of the propulsion subsystem/propulsion GSE.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: GROUND: 4
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Materials	HAZARD GROUP: Contamination	DATE: October 1986

HAZARD TITLE: Loss of Habitable/Breathable Atmosphere

APPLICABLE SAFETY REQUIREMENT:	HAZARD CATEGORY:
KHB 1700.7A - 4.4.2.2	X CATASTROPHIC
	CRITICAL

DESCRIPTION OF HAZARD:

Inhalation or ingestion of beryllium or other toxic fumes and/or particulates leading to personnel illness.

HAZARD CAUSES:

1. Welding, cutting, or machining operations performed on beryllium components.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: GROUND: 5
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Pyrotechnic	HAZARD GROUP: Injury Fire	DATE: October 1986
HAZARD TITLE: Inadvertent Ignition of Pyrotechnics		
APPLICABLE SAFETY REQUIREMENT: KHB 1700.7A - 4.3.5.2, 4.3.5.3, 4.3.5.4		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Inadvertent ignition of ordnance could result in equipment damage, personnel injury or fire.

HAZARD CAUSES:

1. Inadvertent application of power to EED's due to improper installation/ handling, inadvertent command, operator error, or test equipment failure.
2. Stray voltages on ordnance power lines during checkout.
3. Electrostatic discharge from static charge build up.
4. Electromagnetic Interference (EMI) from spacecraft or in the vicinity of the spacecraft.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT	NO. FLIGHT: GROUND: 6
PAYLOAD: Mars Observer	PHASE: 0

SUBSYSTEM:	HAZARD GROUP:	DATE:
Communications	Radiation	October 1986

HAZARD TITLE: Hazardous Levels of Radiation

APPLICABLE SAFETY REQUIREMENT:	HAZARD CATEGORY:
KHB 1700.7A - 4.3.4	[ ] CATASTROPHIC [X] CRITICAL

DESCRIPTION OF HAZARD:

Communication Transmitter emits level of radiation hazardous to personnel.

HAZARD CAUSES:

1. Personnel present within range of hazardous radiation when transmitter is On.
2. Inadvertent transmitter activation.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT	NO. FLIGHT: GROUND: 7
PAYLOAD: Mars Observer	PHASE: 0

SUBSYSTEM: Electrical	HAZARD GROUP: Fire Explosion	DATE: October 1986

HAZARD TITLE: Ignition of Flammable Materials.

APPLICABLE SAFETY REQUIREMENT: KHB 1700.7A - 4.3.2.3, 4.3.9	HAZARD CATEGORY:
	X CATASTROPHIC
	CRITICAL

DESCRIPTION OF HAZARD:

Ignition sources in the presence of flammable material during spacecraft electrical checkout operations could cause a fire or explosion resulting in damage to equipment, facilities and/or personnel injury.

HAZARD CAUSES:

1. Electrical fault resulting in ignition of flammable materials.
2. Use of flammable materials in spacecraft design.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: GROUND: 8
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Pyrotechnics Propulsion	HAZARD GROUP: Injury Fire	DATE: October 1986
HAZARD TITLE: Postlanding Operations - Aborted Mission		
APPLICABLE SAFETY REQUIREMENT: KHB 1700.7A - 4.3.5, 4.3.7		HAZARD CATEGORY: X CATASTROPHIC; CRITICAL

DESCRIPTION OF HAZARD:

Premature ignition of ordnance during removal of a cargo element after an aborted mission could result in equipment damage, personnel injury, or fire.

HAZARD CAUSES:
1. Inadvertent application of power to EED's.
2. Stray voltages on ordnance power lines.
3. Inability to return safety inhibits that may have been removed during deployment operations to a safe configuration.
4. Extreme temperatures at contingency landing sites causing over pressurization of propellant tanks, lines, or sealed containers or freezing of propellant.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: GROUND: 9
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Electrical	HAZARD GROUP: Corrosion Contamination	DATE: October 1986
HAZARD TITLE: NiCd Batteries Hazards		
APPLICABLE SAFETY REQUIREMENT: KHB 1700.7A - 4.3.2, 4.3.3		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Rupture of the NiCd battery cells, releasing debris and/or electrolyte causing fire contamination and/or personnel injury..

HAZARD CAUSES:

1. Structural failure of the NiCd cells.
2. Overpressurization caused by internal or external short, cell reversal, overcharging, temperature extremes, or battery plate insulator failure.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: GROUND: 10
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Propulsion	HAZARD GROUP: Injury Contamination, Fire	DATE: October 1986
HAZARD TITLE: Inadvertent Firing of Thrusters or Leakage of Thrusters		
APPLICABLE SAFETY REQUIREMENT: KHB 1700.7A - 4.1.3		HAZARD CATEGORY: X CATASTROPHIC; CRITICAL

DESCRIPTION OF HAZARD:

1. Inadvertent firing of thrusters could result in personnel injury, equipment damage, or fire.
2. Leakage of propellant through thrusters.

HAZARD CAUSES:

1. Failures in RCS system or inadvertent commands that allows flow of propellant to the thrusters.
2. Failure of seats in valves.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		

PAYLOAD HAZARD REPORT		NO. FLIGHT: GROUND: 11
PAYLOAD: Mars Observer		PHASE: 0
SUBSYSTEM: Mechanical	HAZARD GROUP: Injury Contamination	DATE: October 1986
HAZARD TITLE: Rotating Equipment		
APPLICABLE SAFETY REQUIREMENT: KHB 1700.7A - 4.3.6		HAZARD CATEGORY: X CATASTROPHIC CRITICAL

DESCRIPTION OF HAZARD:

Debris from rotating equipment (RWAs) failure could damage Orbiter, or cause injury to personnel.

HAZARD CAUSES:

1. Failure of rotating equipment (rotating wheel assemblies) during testing releasing debris.

HAZARD CONTROLS:

SAFETY VERIFICATION METHODS:

STATUS OF VERIFICATION:

APPROVAL	PAYLOAD ORGANIZATION	STS
PHASE I		
PHASE II		
PHASE III		