Cargo Systems Manual (CSM): Hubble Space Telescope (HST)

All Flights

Mission Operations Directorate Operations Division

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Cargo Systems Manual (CSM): Hubble Space Telescope (HST)

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It is the sincere hope of all these individuals that the reader finds this document a useful reference throughout the life of the HST program.

PREFACE

This document has been prepared by the Payloads Operations Branch, Operations Division, Mission Operations Directorate (MOD), National Aeronautics and Space Administration (NASA), Lyndon B. Johnson Space Center (JSC), Houston, Texas, with editorial support from the United Space Alliance (USA) Integrated Documentation Support Department and technical support from the Space Shuttle Program, Space Shuttle Systems Integration Office. The Space Shuttle Systems Integration Office has reviewed the CSM to the level of detail contained herein and verified that all orbiter-to-payload avionics interfaces are accurately represented.

Comments concerning the general contents of this manual should be directed to DO5/John Osborn, 281-244-5568.

PURPOSE

The purpose of the Cargo Systems Manual (CSM) is to provide a payload reference document for payload and shuttle flight operations personnel during shuttle mission planning, training, and flight operations. The CSM will include orbiter-to-payload interface information and payload system information (including operationally pertinent payload safety data) that is directly applicable to the MOD role in the payload mission. The CSM is an accurate source of this information for use by flight operations personnel.

SCOPE

This document describes the HST vehicle and related support hardware to be flown on a given mission and the interface of that hardware to the orbiter. Though no detailed payload operating procedures are included, the document provides the basis of technical data for understanding those procedures documented in the HST Flight Operations Support Annex. Any technical data beyond that necessary to understand those procedures is provided for completeness in describing the HST system. In general, no ground equipment or ground operations descriptions are included. Further, detailed information on the HST ground-based operations system for controlling the orbiting vehicle is not provided.

DESCRIPTION

This document provides technical data on the HST vehicle only, and excludes descriptions of the Rigid Array Carrier (RAC), the Multi-Use Lightweight Equipment (MULE) carrier, the Second Axial Carrier (SAC), and Flight Support System (FSS). The document Cargo Systems Manual (CSM): Hubble Space Telescope Space Support Equipment Servicing Mission 3B, USA001948, provides a comprehensive source for technical data on the the FSS, and other space support equipment unique to HST servicing and refurbishment.

An overview of the HST vehicle is provided in Section 1, along with a listing of the customer and STS documentation used as the technical basis for the document.

The elements of the HST Support Systems Module (SSM) are described in Section 2. This includes all relevant data on STS interfaces and operations involved with this element of the HST vehicle. Section 2 breaks the SSM down into its component subsystems, including structures, mechanisms, electrical power, data management, communications, pointing control, thermal control, and safing.

The HST Optical Telescope Assembly (OTA) is described in Section 3. This section includes general descriptions of the operation of the OTA and its major subelements, as well as discussion of servicing operations pertaining to changeout of the Scientific Instruments (SIs).

Sections 4 and 5 describe in general terms the SIs in the HST complement and the SI control system, respectively. This data is provided in the HST CSM for general information only.

The HST Solar Array (SA) system is thoroughly treated in section 6. This description includes data on the array mechanisms, electronics, and operations, as well as discussion of servicing operations pertaining to the changeout of the SAs.

Sections 7 through 10 provide detailed information for servicing missions SM1, SM2, SM3A, and SM3B.

Several appendixes have been included to provide additional data for the reader. Appendix A provides supplementary data to support the main content of the CSM such as the itemization of mass properties of servicing components and the total weight of the HST prior to servicing, in addition to more detailed information about the nickel-hydrogen batteries. Appendix B delineates change control procedures for the HST CSM. Appendix C provides a substantial list of acronyms and abbreviations used throughout the document.

SOURCE/CHANGE TRACKING TABLE

This CSM incorporates the following changes:

DN: J-HST-103-1, C-HST-103-2 through C-HST-103-8, C-HST-109-1 through C-HST-109-45

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This CSM is based on the following engineering drawings, CCCD drawings, and source documents:				
Engineering drawings				
Drawing no.	Rev	Eos		
VA72-200186	Е	E01		
VD72-200186	А	B01-06		
VS72-200186	А	B01-06		
Source documents				
Phase 3 Safety Data Package 06/08/01				
Customer Drawings 10/01/91				
Annex 1 Drawings 10/2/01				
FSSR	FSSR 1/30/02			
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1.0 INTRODUCTION

1.1 BACKGROUND

The Edwin P. Hubble Space Telescope (HST) (Figure 1.1-1 and Figure 1.1-2) is an astronomical observatory placed by the *Space* Shuttle in a circular orbit approximately 330 nautical miles (n. mi.) (611.5 km) in altitude, with an orbital inclination of 28.45° and orbital period of approximately 95 minutes. This position lies well above the obscuring layers of the atmosphere. The HST *can* see close to the edge of the observable universe, estimated to be about 14 billion light-years distant. Looking so far in distance is equivalent to looking back in time at the possible birth of galaxies and quasars billions of years ago. Table 1.1-1 summarizes the characteristics of the HST.

The observatory was named in honor of American astronomer Edwin P. Hubble (1889 to 1953). His accomplishments include the observation (along with Milton Humason) that the greater the distance between two galaxies, the greater their relative velocity of separation. This is known as "Hubble's Law" and is expressed as $v = H_0D$, where v is the recessional velocity in km/s, H_0 is the "Hubble constant" (observed to be 71 ± 4 (km/s)/Mpc), and D is the distance from the galaxy to the observer in megaparsecs (Mpc). This law helped support the (then) newly proposed Big Bang theory via direct observation. Among his many other accomplishments, Hubble observed that the Andromeda Nebula was a separate galaxy from the Milky Way, developed a system to classify galaxies, and discovered the main belt asteroid, 1373 Cincinnati.

The objective of the HST project was to orbit a high-quality optical 2.4-m telescope system by the *Space* Shuttle for use by the astronomical community in conjunction with National Aeronautics and Space Administration (NASA). The scientific objectives of the HST are to determine the constitution, physical characteristics, and dynamics of celestial bodies; *examine the formation, structure, and evolution of stars and galaxies; study the history and evolution of the universe; and provide long-term, space-based research for optical astronomy.*

HST is a Ritchey-Chretien Cassegrain designed reflector telescope. Like its groundbased counterparts, the HST was designed as a general-purpose instrument, capable of utilizing a wide variety of scientific instruments at its focal plane. This multipurpose characteristic allows the telescope to be effectively used as a national facility, capable of supporting the astronomical needs for an international user community and, hence, making contributions to human needs. By using the Space Shuttle to provide Scientific Instrument (SI) upgrading and subsystems maintenance, the useful and effective operational lifetime of the HST is extended. The Space Shuttle placed the HST in its operational orbit and provides both planned and unplanned maintenance of the vehicle *in the form of Servicing Missions (SMs).* Upon conclusion of these missions, on-orbit checkout of the HST begins and continues for weeks after deployment during the Servicing Mission Observatory Verification (SMOV). The performance of the SIs is evaluated, calibration is performed, and techniques are developed for using the observatory to collect scientific data. Completion of SMOV certifies that the HST is ready for full-time science operations.



Figure 1.1-1. HST (as deployed on STS-31)

1.1-2



Figure 1.1-2. HST post deploy on STS-109

1.1-3

Parameter	Dimension
Stowed configuration	
Length	43 ft 6 in.
Diameter	14 ft 0 in.
Deployed configuration	
Length (aperture door opened)	53 ft 10 in.
Diameter (Solar Array (SA) and High-Gain Antenna (HGA) deployed)	41 ft 10 in.
Mass of spacecraft	26,222 lb
Orbit altitude	315 nautical miles (n. mi.)
Orbit inclination	28.45°
Propulsion	None
Attitude control	Reaction wheels and magnetic torquers
Slew rate	6 deg/min
Settling time	2 min
Pointing stability	0.007 arc second
Solar array (two)	
Length	23 ft 4 in.
Width	7 ft 10 in.
Power	4730 W
High-gain antenna (two) (+V3, -V3)	
Dish diameter (effective)	4 ft 2 in.
Boom length (from HST centerline)	19 ft 4 in.
Left Hand Circular (LHC) polarization	
Low-gain antenna (two) (fwd, aft)	
Small spiral cone	Base-to-tip - 8 in.
LHC polarization	Base dia. = 3 in.
Science data storage	$1 \times 10^{\circ}$ bits
Data rate	
Science data (HGA-to-TDRSS)	1 Mbps
Engineering data	500 bps 4, 32 kbps, 1 Mbps*
	(HGA to TDRSS or LGA to GSTDN)
Transmitter frequency	
S-band Multiaccess (MA) (two)	2287.5 MHz
S-band single access (two)	2255.5 MHz
Receiver frequency	
S-band MA (low-gain antenna)	2106.4 MHz
Command receive rate	1 kbps,
	125 bps

Table 1.1-1. Hubble Space Telescope characteristics (post STS-109/SM3B)

*Recorder dump rate.

Normal operations start with the observations selected by the Space Telescope Science Institute (STScI) on the basis of peer-reviewed research proposals. The Call for Proposals invites the astronomical community to propose for observing time on HST in a given cycle, nominally 1 year in duration. Scientists are provided an overview of HST's expected capabilities for that observing cycle. A committee, organized by the STScl, reviews and evaluates the submitted proposals and recommends a list of programs for preliminary approval. Upon final approval, the observations (called visits) in a proposal are assigned scheduling windows. These windows represent times that an observation could be conducted on HST. They are subject to change; however, STScI does try to hold to these windows whenever possible. The actual observations are scheduled on week-long calendars as stored commands in computer "command loads." Approximately 5 days before the start time of a calendar's execution, the command loads to be uplinked to HST are sent to Goddard Space Flight Center's (GSFC's) Flight Operations Team (FOT) for final checking and verification. Individual loads are uplinked daily by the FOT to the HST Spacecraft (486) and Payload (NASA Standard Spacecraft Computer, Model 1 (NSSC-1)) computers.

The Space Telescope Operations Control Center (STOCC) represents the focal point for all mission operations, including vehicle command and control; determination of operating constraints and restrictions; vehicle health status and monitoring; anomaly analysis; and contingency control of the HST. The FOT represents the real-time commanding and monitoring aspect of the STOCC and is currently located at the STScI at the Johns Hopkins University in Baltimore, Maryland. The White Sands Ground Terminal receives data from the HST spacecraft and forwards the information to the NASA GSFC Facility Packet Processor II (PACOR-II). PACOR-II translates the data into observations and forwards it to the STScI facility.

1.1.1 Why Space Telescope

Man's concept of the universe has drastically changed in the past century. Improved observation, instrumentation techniques, and analytical tools have provided opportunities for evolving a clearer understanding of the universe. With each significant improvement in observation capability, deeper insights into the composition, evolution, and processes of the universe have been made possible. Indeed astronomy as a science has yielded outstanding contributions to the understanding of physical processes and formulation of physical laws (e.g., gravity) that affect all aspects of our lives. Much has been learned, but much remains to be understood, including the extent and geometry of the universe; the past history and the future of the universe; and the many diverse and violent physical processes that occur in various stars, galaxies, and other celestial objects.

The 200-in. aperture Hale telescope at Palomar Mountain, California, can recognize individual galaxies several billion light-years away. However, like all earthbound devices, the Hale telescope has limited resolution because of the blurring effect that the atmosphere of the Earth causes due to its turbulence and light scattering. The wave-length region observable from the surface of the Earth is limited by the atmosphere to the visible part of the spectrum.

Unlike ground-based telescopes, the 2.4-m HST possesses, and can effectively utilize, an optical quality of such precision that its resolving power is limited primarily by the defraction limit of the optics. The HST has been taken into Earth orbit by the *Space* Shuttle where it is unhindered by atmospheric distortion and absorption. *Being in Earth orbit allows* it to see objects with a resolution approximately 7 to 10 times better than that obtainable with even the largest telescopes on Earth and over a wavelength region that reaches far into the Ultraviolet (UV) and Infrared (IR) portions of the spectrum. Objects at 7 billion light-years, for example, can be seen with the HST with as much detail as objects at 1 billion light-years can be seen with the best earthbound telescopes.

NASA and the scientific community have *evolved* the technological and operational capabilities that are needed to utilize the technology effectively. These technical advancements have occurred as a natural result of other orbiting astronomical satellite programs such as Orbiting Astronomical Observatory (OAO), Orbiting Solar Observatory (OSO), and *Skylab's* Apollo Telescope Mount (ATM), as well as through supporting research and technological activities. In addition, a number of space telescope mission definition and design activities were conducted within the agency and by contractors. Based on need and state of technology, the HST is the next logical step in astronomy.

HST is part of the Great Observatories series of spacecraft developed by NASA to cover the electromagnetic spectrum. Other vehicles in this series include the Compton Gamma Ray Observatory, the Chandra X-ray Observatory, and the Spitzer Space Telescope (observes thermal infrared wavelengths). Reference Figure 1.1-3.



Figure 1.1-3. NASA's Great Observatories

By combining images from across the electromagnetic spectrum using orbital and groundbased observatories, the scientific community and the public at large have a better understanding of the physical processes that make up the objects in the universe. This capability is demonstrated in Figure 1.1-4.

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Figure 1.1-4. Combined Hubble, Chandra and Spitzer image of Supernova SN1604

1.1-8

1.1.2 Specific Scientific Objectives

The overall scientific objectives are to gain a significant increase in our understanding of the universe through observations of celestial objects and events. The HST has been designed and developed to achieve the following specific scientific objectives:

- Precise determination of distances to galaxies out to expansion velocities of 1 X 10**4 km/sec and calibration of distance criteria applicable at cosmologically significant distances
- b. Determination of the rate of deceleration of the Hubble expansion of the universe, its uniformity in different directions, and possibly its constancy with time
- c. Testing of the basic reality of the universal expansion by determination of the surface brightness versus red-shift relation for distant galaxies
- d. Establishment of the history of the star formation and nuclear processing of matter as a function of position in nearby galaxies and determination of the variations from galaxy to galaxy
- e. Determination of the nature of stellar populations in the early stages of the galactic evolution, based on "lookback" observation of distant galaxies
- f. Estimation of the He/H ratio in quasars by observation of red-shifted He I and He II resonance lines
- g. Search for multiple-red-shift absorption line groups in the UV spectra of low-red-shift quasars
- h. Intercomparison of total spectra of high-red-shift quasars, low-red-shift quasars, and active galactic nuclei
- i. Resolution of densely packed nuclei of globular star clusters in search of massive black holes
- j. Identification and flux measurement in UV and optical wavelengths of faint X-ray sources and radio pulsars
- k. Resolution of the complex internal structure of Herbig-Haro objects to investigate their possible links to star formation
- I. High spatial resolution, IR observations of protostars
- m. Direct imaging and astrometric search for planetary companions of nearby stars
- n. Determination of bolometric luminosities of faint, hot stars for studies of stellar evolution

1.1-9

- o. Determination of composition, temperature, density, and ionization structure of the gas in the galactic halo, in high-velocity clouds, and in the intergalactic medium
- p. Precise mapping of the 100-um flux sources in compact H II regions
- q. Determination of composition of clouds in the atmospheres of Jupiter, Saturn, Uranus, and Neptune
- r. Resolution of surfaces of minor planets and asteroids
- s. Synoptic mapping of atmospheric features on Venus, Jupiter, Saturn, and Uranus for study of atmospheric dynamics
- t. Intensity measurements of atomic and molecular UV emission lines important to understanding the chemistry of comets

The primary objective of HST is to provide a high resolution, optical telescope system that meets these scientific objectives and performs them within, *and beyond*, the operational life of the HST. In principle, there is no reason the HST could not be operated for many decades. Because the SIs and some of the electronics are of modular design, they can be replaced with improved or different models. Periodic orbital maintenance and altitude reboost via the Space Shuttle orbiter have occurred. *There are no longer any plans to retrieve the HST and return it to Earth via the Space Shuttle*.
1.2 VEHICLE OVERVIEW (AFTER SERVICING MISSION 3B)

The HST weighs approximately *26,222 lb* (11,894 kg) and is 43.5 *ft* (14.3 m) long and 14 *ft* (4.3 m) in diameter. The major elements of the HST are the Support Systems Module (SSM), the Optical Telescope Assembly (OTA), *the SIs*, the Scientific Instrument Control and Data Handling (SIC&DH) subsystem, and the *Solar Arrays* (*SAs*). Figure 1.2-1 provides an exploded view of the HST vehicle.

1.2.1 Support Systems Module

The SSM is the spacecraft portion of the HST, supporting all other elements of the vehicle while providing all interfaces to the orbiter. The SSM also provides structural and electrical interface to the GSFC Flight Support System (FSS) for orbital servicing. The SSM contains a very precise pointing and stabilization control subsystem, thermal control subsystem, data management subsystem, and electrical power and communication subsystems. The electrical power to operate the HST is provided by batteries, recharged and supplemented by SAs during the sunlit portion of its orbit. The major structural components of the SSM are the Aperture Door (AD), the Light Shield (LS) and Forward Shell (FS) (which enclose the OTA), the SSM Equipment Section (ES) (housing the module subsystems and providing the primary load-bearing structure of the HST), and the Aft Shroud (AS) (which encloses the SIs, Fine Guidance Sensors (FGSs), Fixed Head Star Trackers (FHSTs), and Rate Sensor Units (RSUs)). Electrical interface to the orbiter during the SMs is provided through the AS. The SSM is detailed in Section 2.0.

1.2.2 Optical Telescope Assembly

The HST Primary Mirror (PM) contains push-pull actuators to correct the shape of the mirror surface. The Secondary Mirror (SM) position is also adjustable by ground command for tip, tilt, decenter, and despace.

When in operation, light from a star or other object travels through the aperture, down the assembly, and past the SM. It then strikes the PM and is reflected back to the SM, where it is narrowed and intensified into a small-diameter beam. (A *micro*meteoroid shield/Sun-shade protects these optics.) The beam passes through a 24-inch hole in the PM to the focal plane, almost 5 feet behind it. The focal plane is where the light originally captured by the PM is turned into a focused image. Parts of the image enter the apertures of the SIs and are transmitted as data. The images at the focal plane are not distorted by changes in the environment. These images, as well as other *science data* are converted to electronic digitized signals. The data are transmitted by means of HGAs at a rate of up to 1 Megabit per second (Mbps).



Figure 1.2-1. HST major structural elements

To minimize thermal distortion of the OTA structure during its mission, special composite materials were employed in its construction. The use of these materials has produced a structure whose dimensions are stable over a variable temperature range. However, the fabrication process used restricts the minimum temperature to 50° F for durations of 12 hours or more. Violation of temperature limits for certain OTA elements can render the HST unusable for *science data* acquisition.

1.2.3 Scientific Instruments

The initial complement of SIs on the HST included two imaging cameras, two spectrographs, a photometer, and the OTA FGSs used as astrometry SIs. *Subsequent servicing missions have installed newer and more capable SIs.* Table 1.2-1 provides a brief description of *the current SI complement.* The *Advanced Camera for Surveys (ACS)*, Space Telescope Imaging Spectrograph (STIS), Near Infrared Camera and Multi Object Spectrometer (NICMOS), and the Corrective Optics Space Telescope Axial Replacement (COSTAR) are mounted in the axial position of the OTA focal plane structure. The Wide Field/Planetary Camera 2 (WF/PC2) is radially mounted in the OTA, as are the FGSs. All SIs are ORUs, as are the FGSs. The SIs are detailed in Section 4.0.

The original WF/PC was replaced by WF/PC2 during STS-61/SM1. WF/PC2 is distinguished by its Field of View (FOV), spatial resolution, and wavelength range. It is able to cover the UV and blue regions of the electromagnetic spectrum as well as the red and near-Infrared (IR) regions. WF/PC2 has become a "workhorse" SI for the HST program, providing thousands of spectacular images for the scientific community and the public at large. WF/PC2 is due to be replaced by Wide Field Camera 3 (WFC3) during STS-125/SM4.

The mission of COSTAR, a second-generation HST instrument, was to place and align mirror corrective optics for the *first generation SIs* (the *Faint Object Camera* (FOC), the *Faint Object Spectrograph* (FOS), and the *Goddard High Resolution Spectrometer* (GHRS)). COSTAR replaced the original High Speed Photometer (HSP) *during the first Servicing Mission* (STS-61/SM1). COSTAR is no longer needed for observations as HST's SIs have corrective optics built into their individual designs to compensate for HST's PM aberration. COSTAR is scheduled to be replaced by the Cosmic Origins Spectrograph (COS) during STS 125/SM4.

The NICMOS is the first spaceborne astronomical designed to operate in the near-IR spectrum. The instrument uses a solid nitrogen cryogenic system to cool sensors to improve their sensitivity. The instrument consists of three cameras that can operate simultaneously and asynchronously at three different magnifications and fields of view. NICMOS replaced the FOS during STS-82/SM2.

The STIS is an extremely flexible advanced spectrograph. STIS is capable of a wide range of resolution and FOV. STIS replaced the Goddard High Resolution Spectrograph (GHRS) during STS-82/SM2. STIS is currently in a "safe" mode, awaiting repair during STS-125/SM4.

The OTA FGSs play the dual role of guiding the HST and being an astrometry SI. Only two of the three FGSs are required for pointing, making the third unit available for astrometric observations.

	Instrument (principal investigator)	Contractor	SSM location	Characteristics
WF/PC2	Wide Field/Planetary Camera 2	Jet Propulsion	Radial (-V ₃)	Two cameras (4 CCDs) + ascent/entry
	(Prof. J. A. Westphal - California Institute of Technology)	Laboratory (CA)		Purge gas
				Carousels of 48 filters/polarizers
				Far UV (1150 A) to near IR (11,000 A)
ACS	Advanced Camera for Surveys		Axial (-V ₂)	Three optical channels:
			(Pos 3)	WFC: 202 ² arcsecond FOV, resolution of 0.05 arcsecond/pixel for rapid survey
				HRC: 26x29 arcsecond FOV, resolution of 0.027 arcsecond/pixel
				SBC: 31x35 arcsecond FOV, resolution 0.032 arcsecond/pixel. Operates in far-UV spectrum (blocks visible light)
NICMOS	Near Infrared Camera Multi	Ball Aerospace	Axial (+V2)	FOV = 10 arcsecond
	(Dr. R. Thompson)		(Pos 2)	Modest spectral resolution across UV and visible wavelengths for faintest limiting magnitude
STIS	Space Telescope Imaging	Ball Aerospace	Axial (+V2)	FOV = 10 arcsecond for 2 modes
	Spectrograph (Dr. B. Woodgate - GSFC)		(Pos 1)	R = 2 x 104 in 105-320 n. mi. (UV) [up to 15 km/sec equivalent speed] should see Mv = 17
				R = 1.2 x 105 in 110-320 n. mi. (UV) [up to 2.5 km/sec equivalent speed]
COSTAR	Corrective Optics Space Telescope Axial Replacement (COSTAR)	Ball Aerospace	Axial (-V2) (Pos 4)	Optical subsystem fixed optical bench deployable optical bench

Table 1.2-1. HST complement SIs

1.2.4 Scientific Instrument Control and Data Handling

The SIC&DH subsystem provides a unified Command and Data Interface (CDI) between the five SIs and the SSM Data Management Subsystem (DMS). The SIC&DH hardware is located in bay 10 of the SSM ES and provides the overall control of SI operations and data routing. The major elements of the SIC&DH are the Control Unit/Science Data Formatter (CU/SDF), the Multiplexed Data Bus (MDB), the Bus Coupler Unit (BCU), the Remote Module (RM), the Power Control Unit (PCU), the Standard Interface (STINT) for computer, and the *NSSC-I*. The SIC&DH is further discussed in Section 5.0.

1.2.5 Solar Arrays

The HST SAs are affixed externally to the SSM FS. Each array is *composed* of *four rigid* solar *panels* and *control electronics*. An Electronics Control Assembly (ECA) governs the SA operation. The primary components of the ECA are the *Solar Array Drive Mechanism (SADM)* and the Solar Array Drive Electronics (SADE).

The deployment of each wing is manually performed via EVA crewmember while on the Remote Manipulator System (RMS) to reduce the amount of jitter produced by the SAs as they cool and heat up as the HST transitions between the day and night segments of its orbit. Each SA measures approximately 8 by 25 feet. The two SAs combined provide the SSM 4730 *W* average power to recharge the six nickel-hydrogen (NIOOH-H₂) batteries and *to* support the vehicle load each time it completes the Sun-side portion of its 95-minute orbit. Refer to Section 6.0 for additional information.

2.0 SUPPORT SYSTEMS MODULE

2.1 INTRODUCTION

The HST SSM provides the structural support, thermal control, electrical power, communications, data management, and pointing control in support of the OTA. Provisions are made in the SSM for on-orbit maintenance and planned SI replacements by EVA crewmembers while the HST is berthed to the GSFC FSS. Figure 2.1-1 provides a cutaway view of the HST, identifying the SSM with respect to the entire spacecraft. Figure 2.1-2 shows the HST under final assembly at the Lockheed Sunnyvale Vehicle Assembly and Test Area facility.



Figure 2.1-1. HST cutaway



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Figure 2.1-2. HST under final assembly (-V3 view)

2.1-2

2.2 STRUCTURES

The SSM design provides the structural support to mount components of the SSM and OTA subsystems. The SSM also provides the structural interface to the orbiter for deployment, servicing, and retrieval missions. The major elements of the SSM structure are the *Aperture Door (AD), the Light Shield (LS), the Forward Shell (FS), the SSM Equipment Section (ES), and the Aft Shroud (AS).* Figure 2.2-1 provides an exploded view of the SSM and identifies each major structural element. The physical configuration of the vehicle is depicted in Drawing 2.2 sheets 1 through 7.

2.2.1 SSM Structural Elements

2.2.1.1 Light Shield

The LS provides a protective envelope and stray-light shielding for the OTA. It is a cylindrical structure 120 inches in diameter and 153.2 inches long. The front section of the LS is 12.2 inches long and is of a semimonocoque (shell) construction. The rear (main) section is 113.1 inches long and is an integrally stiffened, machined aluminum/magnesium skin with longitudinal ribs. Annular rings with 10 optical baffles attached are part of both sections. The LS includes the following:

- a. Aperture Door (AD)
- b. Latches for both *High-Gain Antennas* (HGAs)
- c. Two forward latches for the Solar Arrays (SAs) (no longer used with SA3s)
- d. Two magnetometers
- e. Forward Low-Gain Antenna (LGA) and waveguide
- f. Two Sun sensors
- g. Visual acquisition reflectors
- h. EVA handrails
- i. Four foot-restraint supports
- j. Two scuff plates as berthing aids

The complete LS assembly is attached to the *FS*, as depicted in Figure 2.2-1 and Drawing 2.2. The function of the AD *was*/is to provide environmental contamination protection for the OTA during ground handling, ascent, and on-orbit operations. In addition, it protects the OTA and SIs from direct sunlight if the HST optical axis were to get closer to the Sun line than 20° . (AD closure in orbit is initiated by Bright Object Detectors (BODs) through the Mechanism Control Unit (MCU) whenever the Sun is within 35° of the +V1 axis.)

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Figure 2.2-1. SSM major structural elements (need updated figure for SA-3s)

2.2-2

The *AD* assembly consists of the honeycomb door structure (120 by 120 inches), two hinges, a latching mechanism mounted to the LS, and bidirectional rotary drives at the hinge and latch mechanisms (*Section* 2.3). The maximum travel of the AD from fully open*ed* to fully closed is 105.25°, with a travel time of 134 seconds. Figure 2.2-2 shows the AD installed on the vehicle.



Figure 2.2-2. Aperture door assembly

2.2.1.2 Forward Shell

The *FS* assembly is the primary structural support for the forward trunnion, which *was* used for HST mounting in the orbiter payload bay during *HST's deploy mission (STS-31)*. The FS is a 156.0-inch-long cylindrical structure, 121.2 inches in diameter, which surrounds the OTA but is not attached to it. A series of forged aluminum alloy rings provides stiffness to the sheet aluminum shell skin and attach points for accessory hardware mounting. The FS supports the following SSM components:

- a. SA panels (two)
- b. Aft SA latches (two, no longer used)
- c. Forward trunnion
- d. Magnetic Torquers (MTs) (four)
- e. HGAs and hinge assemblies (two)
- f. EVA handrails and foot restraint receptacles
- g. Orbiter RMS grapple fixtures (two)
- h. NOBLs (three)

The FS mates to the forward face of the SSM ES, as depicted in Figure 2.2-1 and in Drawing 2.2. The HGAs and their hinge mechanisms are mounted at the aft end of the FS, and the HGA stowage latches are mounted on the LS. The RMS grapple fixtures are EVA releasable. A keel camera target is provided near the +V3 axis of the FS, just forward of the HGA mount, as a berthing aid. The forward trunnion mounts through a spherical bearing system and is supported in the V3 axis with load carrying tubular members. The trunnion interfaced with an orbiter Payload Retention Latch Assembly (PRLA) when the HST was stowed in the payload bay *prior to deploy*. The external surface of the FS is covered with Multilayer Insulation (MLI) blankets. *NOBLs have subsequently been installed over bays 1, 6, 9 and 10 to mitigate the effects of MLI degradation. MLI patches have been applied to several areas on the telescope. Reference Figure 2.2-? for a map of these areas.*

2.2.1.3 SSM Equipment Section

The SSM ES is 61.25 inches long with an outside diameter of 168.0 inches and an internal barrel section diameter of 121.2 inches. The construction of the SSM ES consists of integrally stiffened aluminum frame panels and inner barrel, aluminum radial rib shear panels, 10 flat honeycomb core doors, and 2 curved stiffened panel doors.

The SSM ES is detailed in Drawing 2.2 sheet 7. The radial rib panels, which extend from the inner barrel to the outer diameter, separate the ES into 12 bays. Two of these bays provide the structure supporting the aft trunnions. The trunnions support the HST structurally in the V1 and V3 axes. A keel trunnion is provided as well, to accommodate

loads in the V2 axis. Eight of the remaining 10 bays have external flat honeycomb, hinged doors, while the two Reaction Wheel Assembly (RWA) bays (bays 6 and 9) have curved, thermal closeout covers. Subsystem components are mounted on the interior of the bay doors and the surface of the inner barrel, as depicted in Drawing 2.2 sheet 7. The ES aft bulkhead provides the OTA structural interface through a six-point mounting system and interfaces for supporting the AS. The FS, OTA ES, and SA diode boxes interface to the ES forward bulkhead. The \pm V2 trunnion bays provide Ground Support Equipment (GSE) interfaces. Also provided in the +V2 trunnion bay is the Astronaut Control Panel (ACP) shown in *Drawing* 2.2 sheet 3, which allows an EVA crewmember to control HST main and essential bus power (Section 2.4).

2.2.1.4 Aft Shroud

The AS is a cylindrical structure with an inside diameter (at the inside of the outer skin) of 168.0 inches and a length of 138.0 inches. The assembly is closed at the aft end with a flat honeycomb bulkhead attached through ring segments. The front end is attached to the SSM ES aft bulkhead. The structure provides protection and access to the five SIs, three FGSs, three RSUs, and three FHSTs. Access doors are provided for EVA crewmembers to perform a changeout of any of these components. The doors are held open with tethers. Figure 2.2-3 shows the -V2 axial doors.

The AS aft bulkhead (Figure 2.2-4 and Drawing 2.2 sheet 5) provides the interface structure for the FSS, which includes three berthing pins (towel racks), an umbilical tower, and an alignment target. SI nitrogen purge vents are provided through the aft bulkhead, and flexible tubes carry the purge gas from the intake fittings to Quick-Disconnect (QD) fittings on three of the axial SIs and the WFPC2. Provisions are made for stowing the purge lines during SI changeout. In addition, provision is made within the AS for future installation of a cryogenically cooled SI. Ducting through the aft bulkhead will not permit appreciable light leaks into the AS.

An electrical terminal board is incorporated into the aft bulkhead, which is routed along the exterior of the AS aft bulkhead to a set of cables that merge the two connectors into the main umbilical interface. Also located on the AS aft bulkhead is the deployment umbilical (two connectors), originally for the deployment umbilical and FSS umbilical systems (Drawing 2.2 sheet 5), which when released by the Umbilical Disconnect Mechanism (UDM), is retained against the AS by a retraction mechanism. In the contingency that the HST is to be returned on a maintenance mission (currently not planned), the redundant cable can be unplugged manually from the FSS umbilical tower and interfaced to a UDM mounted to the FSS berthing platform.

In addition to the umbilical system, the AS aft bulkhead provides mounting for the aft LGA. The antenna wave-guide is supported internal to the AS (Drawing 2.2 sheet 6). Foot restraint sockets and handrails are present on the AS outer and inner surfaces in support of EVAs.

NCS info related to AS bulkhead here...



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Figure 2.2-3. View along -V2 axis of HST during assembly

2.2-6



Figure 2.2-4. AS aft bulkhead (need updated image showing NCS conduit)



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Figure 2.2-4. Concluded

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NOTES: THE OTA EQUIPMENT SECTION IS FURTHER DETAILED IN SECTION 3. THE SSM EQUIPMENT IS FURTHER DETAILED IN SECTION 2.2.1.3. THE AFT SHROUD IS FURTHER DETAILED IN SECTION 2.2.1.4

2 HST INTERNAL 7

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2.3 SSM MECHANISMS

The SSM mechanisms include systems designed to deploy/stow or latch/release the SSM appendages through a combination of hinge and latch drives. The SSM appendage mechanisms are controlled by the STOCC via the MCU and operate with redundant actuators of a common design for their drive trains. Each is equipped with EVA override interfaces for contingency operations. The SSM also has a system for precise pointing of the HGAs at a Tracking and Data Relay Satellite (TDRS). The HST Antenna Pointing System (APS) is a redundant, Two-Axis Gimbal (TAG) control system for orienting each HGA. The APS is controlled by the STOCC via the Gimbal Electronics Assembly (GEA). This section provides an overview of both mechanical systems and their associated control electronics.

2.3.1 SSM Appendage Mechanism Hardware

2.3.1.1 SSM Latches

The latch mechanisms *are available* to secure all appendages (AD and HGAs) to the SSM during rendezvous operations with the orbiter and release them for deployment. *Note that with the installation of the new SA3s on SM3B, the SSM latches for the SAs are no longer used.* Figure 2.3-1 shows a typical SSM latch. The latch mechanism is driven by a three-phase stepper motor through a four-bar linkage mechanism, as depicted in Drawing 2.3 sheet 1, and consists of torsion rods, drive gears, a control linkage (clevis), a pawl, microswitches, and support mounts.

Each HGA latch, as shown in Drawing 2.3 sheet 2, consists of two similar latching mechanisms located on either side of the HGA dish. Figure 2.3-2 shows the HGA latch with the antenna dish removed, for clarity. As shown, only one latch has a motor-driven actuator and is, therefore, called the drive latch. The slave latch is driven by a torque tube from the drive latch. The latches interface with a latch fitting that extends from the back of the HGA dish. This latch fitting secures the HGA mast to the SSM. The remainder of the SSM latches are single units similar to the HGA drive latch. The aft SA latches, shown in Figure 2.3-3, are designed to react to ascent loads in three directions. All other latches include provisions to permit movement from thermal expansion differences between the appendages and the SSM.

Each latching mechanism is equipped with at least two sets of microswitches to indicate operating status. Latch microswitches are activated when the pawl assembly has reached the End of Travel (EOT) (i.e., the latch is fully open or fully closed (preload)). An additional set of microswitches is employed on the drive and slave HGA latches as well as on the aft SA latches to indicate the appendage berthing status (i.e., ready-to-latch indicators). The MCU provides the microswitch status data to the DMS (Section 2.3.1.4).

EVA Operations

The SSM latch mechanisms can be latched and unlatched using an EVA override capability. The latch protection guard is configured from NORMAL to CHGOUT. To drive the latch, the 7/16-inch hex head can be rotated 1-1/3 revolutions at a rate not to exceed 6 deg/sec (the rate at which a second hand travels around a clock). It may require a stronger initial force to begin movement, but care must be taken not to overtorque the system, as this may cause overspeeding of the motor. Additionally, if the 6 deg/sec rate is exceeded during drive, it may be possible to jam the drive mechanism. Nominally, it requires approximately 36 in-lb torque to manually drive the latch in either direction (Figure 2.3-4).

2.3.1.2 SSM Hinge Mechanisms

The SSM hinge drives open/close the AD and deploys/stows the HGA. The AD hinge is shown in Figure 2.3-5; the HGA hinge is shown in Figure 2.3-6. The SSM hinge is a motor-driven mechanism containing angular contact-paired bearings, torsion rod, rubbing friction elements, rotary drive actuator, and microswitches. The HGA and AD hinges are operated by STOCC via the MCU, which also provides AD position data from a single potentiometer mounted to the door hinge mechanism. A typical hinge configuration is depicted in Drawing 2.3 sheet 1.



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Figure 2.3-1. SSM latch (HGA)

2.3-2



SHEET 1 OF 5



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FORWARD LATCH (light shield)

J

(SA LATCH FITTING SHOWN DETACHED FROM LATCH) VIEW ON ARROW C

Р	0	N	M	↓ L	K



G

F

н

SOLAR ARRAY STOWED CONFIGURATION 🗊



VIEW ON ARROW E

SOLAR ARRAY LATCHES 🕞

E

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				7
	AFT LATCH (FORWARD SHELL)			6
		SA SECONDARY DEPLOYMENT MECHANISM		5
	7, 628 +.003 003	ATCH FITTING TCH N	•	←
		LATCH FITTING GUIDE		3
BERTHED CROSWITCHES (2)		NOTESI D SOLAR ARRAY LATCHES NO SOLAR ARRAY 3 2 CCM 1		2
	(sa latch fitting shown detached from latch) VIEW ON ARROW D	SIGNATURE DATE DSON DR J. NEELY 05/18/96 ENGR P. RIECK ENGR APP APP CADS CDS APP	AIGNINUEZ NATIONAL AERONAUTICS AND SPACE ADMINISTRATION LYNDON B. JOHNSON SPACE CENTER HOUSTON, TEXAS SSM LATCH AND HINGE DETAIL HST CSM DWG NO ALL 126.5 X 34 SHEET 2 OF 5 JSC-29029	1



sfoc_fl2705_012.tif

(a) Overview Figure 2.3-2. HGA latch





Drive latch

Slave latch

(b) Thermal covers installed

Figure 2.3-2. Continued

2.3-8





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Drive latch

Slave latch

(c) Thermal covers removed

Figure 2.3-2. Concluded

2.3-9





Forward latch (light shield)

Aft latch (forward shell)

(a) Overview

Figure 2.3-3. SA latches, no longer used with SA3s installed (microswitch protection guards not shown)

2.3-10



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(b) SA berthed aft latch overview (no longer used post STS-109/SM3B) Figure 2.3-3. Continued

2.3-11



sfoc_fl2705_019a.tif

(c) SA aft latch berthing microswitches (changeout position) (no longer used post STS-109/SM3B)

Figure 2.3-3. Concluded

2.3-12



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Figure 2.3-4. SA forward latch EVA drive interface (no longer used post STS-109/SM3B)



(a) Overview



sfoc_fl2705_022.tif

(b) Active hinge (sunshield released)

Figure 2.3-5. AD hinge

2.3-14



sfoc_fl2705_023.tif

(Some thermal insulation removed)

Figure 2.3-6. HGA hinge

2.3.1.3 Rotary Drive Actuator

The rotary drive actuator is common to all mechanisms and contains redundant, threephase, dc, bidirectional, rotary, 1.5° stepper motors with a 200:1 harmonic drive speed reducer. Each motor is operated from a current source of 0.8-amp maximum and is capable of providing an unpowered holding torque of 100 in-lb (minimum) and a powered torque of 500 in-lb (minimum). Each drive is controlled through the MCU. The rotary drive actuator is detailed in Drawing 2.3 sheet 1.

2.3.1.4 Mechanism Control Unit

The MCU is an internally redundant unit that generates the three-phase motor currents and control signals that determine motor direction, speed, and distance for the SSM appendage mechanisms. Command signals to the MCU are initiated from the STOCC for appendage deployment. AD closure signals may be initiated by the Pointing and Safemode Electronics Assembly (PSEA) upon entry into contingency safe mode or when the BODs indicate the +V1 axis is pointing too close to the Sun.

Physical Description

The redundant units of the MCU (A and B) are mounted in a single enclosure. The unit is 24 by 10.5 by 7.25 inches in size and weighs approximately 41.9 pounds. The enclosure has *one* MCU electronics section in each end, with each section comprising 10 plug-in printed circuit boards. The MCU is tunnel-mounted in SSM ES bay 7 using 10 5/16-inch hexhead bolts with captive washers, torqued to 45 in-lb maximum. Figure 2.3-7 depicts the enclosure, circuit board layout, and connector layout.

Electrical Power

The redundant units of the MCU are independently powered from separate Power Distribution Units (PDUs). PDU-3 provides power to MCU-A, and PDU-4 powers MCU-B. Each MCU uses 10.5 watts on orbit when only one is powered. When both MCUs are powered, the unit uses 17 watts.

The primary MCU control electronics are connected to the primary motor winding of each of the 10 actuators. The redundant MCU control electronics are connected to the redundant motor windings. Each unit is independent of the other except for limited cross-strapping in the MCU command circuitry.

Command/Data Interfaces

The MCU exchanges commands and Telemetry (TLM) with the SSM DMS to operate and monitor the SSM appendage mechanisms. Specifically, MCU-A receives High-Level Discrete (HLD) (pulsed) signals and serial digital command messages from Data Interface Unit (DIU)-4. In addition, discrete, analog, and serial digital TLM messages are output by MCU-A to the same DIU. Similar interfaces are provided between MCU-B and DIU-5, with the only exception being that MCU-B does not output analog signals to that unit.




Figure 2.3-7. SSM ES bay 7 and MCU enclosure detail

2.3-17

The DIU HLD commands provide power on/off control of the control power supply and link power supply, as shown in Drawing 2.3 sheet 3. The MOTOR OPERATE discrete command to the MCU may originate from either a DIU or the PSEA. The PSEA issues the MOTOR OPERATE command only when a bright object is detected to be too close to the telescope optical axis or upon entry into hardware safe mode.

The serial digital command messages received by an MCU (A or B) allow the STOCC to select the mechanisms to be operated; their direction, speed, and distance of travel; and the hardware serial digital TLM message format used to respond to DIU polling for microswitch status information. As shown in Drawing 2.3 sheet 3, the 16-bit command message provides for selecting the primary winding of one mechanism and redundant winding of another. To accomplish this, only one link power supply (i.e., MCU-A or MCU-B) may be powered at a time, and a portion of the link command processing circuitry (that corresponding to the 8-bit redundant mechanism selection code) is crossstrapped to the redundant MCU. Therefore, to command two mechanisms simultaneously, both the MCU-A and MCU-B control power supplies must be commanded on because MCU-A drives the primary windings on each mechanism and MCU-B drives the redundant windings. The hardware serial digital TLM message format selection code of the 16-bit command message is not applicable to the HST use of the MCU, since both selectable formats are identical. The format information simply reads the microswitch data from the switch data multiplexer in a specified order during DIU polling of the MCU.

Each MCU provides discrete status signals and serial digital TLM messages to its corresponding DIU to provide the STOCC with EOT/Beginning-of-Travel (BOT) and ready-for-latch information for each appendage. The placement of microswitches for each appendage is summarized in Figure 2.3-8. Each SSM latch (HGA drive and slave latches, SA forward and aft latches, and AD latch) is internally equipped with microswitch pairs to indicate the OPEN or LATCHED status of the mechanism. Each HGA latch pair is provided primary and redundant microswitch pairs mounted to the latch fitting guide of the drive and slave latches to indicate that the HGA is positioned properly for latch closure. Similarly, each aft SA latch is provided a microswitch pair on the latch guide to indicate that SA is properly positioned for latch closure. Finally, the AD hinge is instrumented with a potentiometer to provide an analog signal to DIU-4 corresponding to the angular door position. This potentiometer is nonredundant.

Functional Description

The nominal MCU activation sequence involves commanding on the MCU-A link power supply CDI and the MCU-A and B control power supplies. As stated above, only one link power supply is required to operate both halves of the MCU. A STOCC serial digital command via DIU-4 then loads all zeros in the 16-bit MCU-A command register, and the current regulators of both MCU halves are enabled, also by DIU HLD.



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Figure 2.3-8. MCU microswitch placement summary

To operate an SSM mechanism, a serial digital command (two 48-bit command words, as depicted in Drawing 2.3 sheet 3) is issued to MCU-A, which loads the command registers with the relevant data on which motor is to be operated, its direction, speed, and distance of travel. A single command sequence can load operational data for two motors (one primary and one redundant motor winding), provided the other unit (MCU-B) is powered and two separate mechanisms are selected. The selection of the primary and redundant windings of the same motor is prohibited. No motor operation occurs until an MCU-A MOTOR OPERATE discrete command is received from the DIU. Once issued, the selected instructions in the MCU Programmable Read-Only Memories (PROMs) operate the motor as specified. The MCU-B MOTOR OPERATE command to operate the second selected motor via its redundant winding. Safe-mode AD CLOSE signals also issue a MOTOR OPERATE command and operate whatever mechanism selection is currently loaded in the MCU command registers.

When motor operation has terminated and been verified, the next command load can be sent to the MCU. Sending a new command load while motors are still operating results in immediate termination of ongoing motor operation. When all appendages have been successfully deployed, a partial CLOSE command is sent to each latch mechanism to relax pressure on and prevent damage to the latch microswitches. The MCU-B current regulator is then disabled and MCU-B control power turned off. The STOCC loads the MCU-A command registers with AD closure data and issues the AD hinge connect command to enable safe-mode door closure capability.

The MCU is shown schematically in Drawing 2.3 sheet 3.

2.3.2 Antenna Pointing System

The HST APS is used to position the HGAs to point and track the appropriate TDRS at the scheduled times for downlink of science and engineering data. The APS consists of a *Two-Axis Gimbal (TAG)* and a *Gimbal Electronics Assembly (GEA)*. The SSM includes two complete APSs with cross-strapping to improve reliability. Redundant inputs to each GEA are provided from redundant DIUs, and GEA outputs are cross-strapped to each TAG. Each GEA has single-string electronics, but redundancy is enhanced by the output cross-strapping. Each TAG has redundant torquers and resolvers for each gimbal, and both have noncontacting Radio Frequency (RF) rotary joints that connect from the X-axis gimbal to the HGA and from the Y-axis gimbal to the mast/waveguide. This interface thus provides an RF path via waveguide and coaxial cable from the S-Band Single-Access (SSA) and MA transmitters through RF transfer switches to the HGA. Each GEA is capable of commanding an X- and Y-axis gimbal and of processing position and instrumentation data.

Pointing control of the HGA is provided by the onboard software of the 486 flight computer. Control segments are initiated by time-tagged, stored-program commands. Pointing is normally maintained so that the HGA boresight is never more than 1° off the true bearing to the TDRS. Antenna pointing is open loop, requiring no feedback from the TDRS. The 486 issues pointing commands to the GEAs, which specify the axis,

rotation, and torque for each HGA movement. Both axes of a TAG can be moved simultaneously and/or independently. If the DMS is in bypass mode (i.e., 486 is not in control), ground-generated torque commands may be passed by the Data Management Unit (DMU) directly to the GEAs without 486 processing. However, this results in limited control that is not proven to provide the resolution to stow.

Table 2.3-1 summarizes APS performance specifications.

Table 2.3-1. APS performance summary

Mechanical stops	
X-axis	<u>+</u> 102° + 0, -2°
Y-axis	<u>+</u> 96° <u>+</u> 1°
Slew rate	
Angular rotation velocity	30 <u>+</u> 2 deg/min. (min)
Maximum output torque	20 oz-in
Simultaneous motion in both axes	Yes (either direction)
	<u>+</u> .35°
Pointing error (Root Sum Square (RSS) of all error sources (3 sigma values))	
Position information (from GEA, both axes)	<u>+</u> 0.2°
X-axis SM S/W limits	102°

2.3.2.1 Two-Axis Gimbal

The TAG consists of two orthogonal gimbal drives (X-axis and Y-axis) that each employ internally redundant drive torquers and resolvers, a mechanical interface between the gimbal assembly and the antenna, limited-angle-cable flex capsules, and noncontacting RF rotary joints. The operational characteristics of the TAG are summarized in Table 2.3-2.

Minimum holding torque ^a	
Command torque scale factor, oz-in./digital command LSB	0.009770396
Command linearity 0-10 percent max. torque range (1 percent max. increment range), scale deviation factor 10-100 percent max. torque range (10 percent max. increment range), scale duration factor	+25 percent +25 percent
Electrical duty cycle Commanded at full torque (20 oz-in.) Commanded torque reversals at max. torque ^b	72 hr 72 hr
RF performance (RF connector and rotary joint) Operating frequency Minimum Maximum Power level ^c	2245 MHz 2298 MHz
Voltage Standing Wave Ratio (VSWR) TAG RF losses	1.5:1 0.6 dB

Table 2.3-2. Two-Axis Gimbal operational characteristics

The operational software stop limits for the gimbal mechanism follow:

HGA software stops

-HGA		+HGA	
+Xo	86°	+Xo	86°
+Yo	75 °	+Yo	75°
-Xo	86°	-Xo	86°
-Yo	93°	-Yo	93°

However, these limits do not permit stowage of the antennas. The software limits will be increased for the \pm Xo coordinates to allow proper positioning of the gimbals and antenna dish for stowage.

-HGA		+HGA	
+Xo	95°	+Xo	95°
+Yo	75 °	+Yo	75°
-Xo	95°	-Xo	95°
-Yo	93°	-Yo	93°

^aTAG provides holding torque to prevent angular movement greater than 7.5° under a no-power condition during HST maneuvers.

^b40-Hz rate reversals.

^cContinuous operation with up to 20 W CW RF power at 2298 MHz over specified frequency range.

Physical Description

The TAG uses direct-drive, brushless, dc, gimbal control technology. The brushless dc motor system, as shown in Drawing 2.3 sheet 4, consists of a permanent magnet rotor, a wound stator, and a method of sensing rotor position (resolver). Each gimbal axis has redundant motors and resolvers. Electronics in the GEA perform the required commutation by applying the proper excitation to the windings, as determined by the resolver. RF signals are routed through the gimbals by coaxial RF rotary joints.

2.3.2.2 Gimbal Electronics Assembly

Two identical GEAs are provided for control of HGA pointing. Each contains a single set of electronics capable of simultaneously servicing a TAG X-axis gimbal and a TAG Y-axis gimbal. Redundancy is provided by cross-strapping the two GEAs with the two TAGs. Configuration of the GEA/TAG stringing is performed by the STOCC via DIU HLDs. Nominally, the +V3 TAG is driven by GEA-2, and the -V3 TAG by GEA-1.

Physical Description

Each GEA is a separate enclosure housing 16 removable circuit card assemblies containing interface and control logic, power bridge drivers for two gimbals, secondary power supply, and Electromagnetic Interference (EMI) filters. The box measures 7.25 by 13.56 by 7.2 inches and weighs 14.9 pounds. Each unit is equipped with five connectors residing on one end of the enclosure. GEA-1 is mounted in SSM ES bay 1, and GEA-2 is mounted on the SSM bay 7 door, as shown in Figure 2.3-9. The GEAs are not classified as ORUs.

Electrical Power

GEA-1 is hardwired to the HST main buses via PDU-1, and GEA-2 is hardwired via PDU-2. On each GEA, power enters the unit through connector J5. The PDU circuit is protected by a 15-A fuse.

Functional Description

The GEA is functionally represented in Drawing 2.3 sheet 5. Gimbal movement is accomplished by a multiword Real-Time Command (RTC) to the 486 flight computer specifying the gimbal, maneuver duration, final angle desired, maneuver shaping parameter, and a flag to indicate an update of the initial angle if the angle is not an absolute value. (Command angles are expressed in radians.) The 486 computes the required torque commands and sends them at a 10-Hz rate via the DIU to the GEA command input module, which passes them on the appropriate bridge driver control and bridge driver modules. Each bridge driver powers one phase of a two-phase permanent magnet motor. (The two phases are referred to as sine and cosine windings.) Thermistors on each gimbal axis provide temperature data for TLM.

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(a) SSM bay 1



(b) SSM bay 7

Figure 2.3-9. GEA installation

2.3-29

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2.3.3 Safety Considerations

Inadvertent hazardous operation of an HST appendage (AD, HGA, or SA) from faulty HST hardware components, erroneous commanding, or uncommanded motion from external loads exists only for RMS grapple operations, FSS berthing and deployment operations, FSS positioning operations, and EVA operations. The MCU has been designed to be single-fault tolerant through redundant MCU-A and MCU-B systems. Drawing 2.3 sheet 3 shows the two inhibits that will be in place at all times to prevent inadvertent operation of the mechanisms.

The actuator motors for AD and HGA have control and operating circuits designated as the MCU. The MCU has a redundant system (A and B). Drawing 2.3 sheet 3 shows the two inhibits in Sections K7 and all of I-H.

For the SM, it is a given that the HST main bus, link power supply and control power supply are powered and two inhibits remain for control of the HST mechanisms.

First, the MCU must receive a digital command from the register selecting appendage direction, distance, and speed. The register is zeroed after each use of the MCU as a precautionary measure. As long as the register is zeroed; direction, distance, and speed will be zero and no motor can be inadvertently selected.

The AD motor select relay is an exception to the above reasoning. After the AD is commanded opened or closed, the AD Motor 11 "Latching Relay" (K11) remains selected even after register power is removed. The K11 relay must be deselected by command.

Second, the motor operate command must be issued to activate the bridge switching circuitry providing cyclic switching to the motors.

The drive motors are so designed that to produce motion, the three motor input leads have to be switched between a positive power source and ground in a cyclic manner. Each switching cycle produces a single step of motor rotation. Each stepped motion of the motor corresponds to 0.0075 degrees of rotary drive output shaft rotation.

The cyclic switching required for motor rotation is produced by a 3-phase generator that receives its power from either primary or secondary control power supply. When the control power supply is initially commanded on and control power is sent to the 3-phase generator, no signal will exit the 3-phase generator because it is automatically reset to OFF by a command from the control power supply.

The corresponding primary or redundant motor relay, primary or redundant direction, distance, speed, and primary or redundant motor operate commands must be sent in sequence to result in mechanism operation.

Events required to activate an actuator motor

- 1. MCU configuration command given to select mechanism, distance, speed, and direction
- 2. Motor operate command given to the 3-phase generator (selected motor now operates)

2.4 ELECTRICAL POWER SUBSYSTEM

The Electrical Power Subsystem (EPS) of the HST provides electrical power generation, storage, and distribution to the entire complement of telescope components. Power generated by a set of SAs simultaneously supplies the HST load and energy storage systems during the sunlight portions of each orbit. During the eclipse period, energy stored in the six nickel-hydrogen (NiH₂) batteries is redistributed to the HST equipment. Battery charging circuits recharge the batteries in each orbit during nominal observation attitudes, although failure to achieve complete recharge is acceptable in one orbit for each 24-hour period. After off-nominal roll maneuvers, the batteries are recharged to the maximum operation capacity within five orbits. The average depth of discharge over a 24-hour period does not exceed 20 percent with one battery out.

The EPS contains provisions for autonomous power management as well as instrumentation to allow for manual power management. Multilevel voltage steps for battery charging are provided. The STOCC has the capability of raising or lowering the cutoff voltage. Orbiter power is received by the EPS through umbilicals. Control of the HST main and essential buses is provided from the orbiter Flight Deck (panel R1, and the Standard Switch Panels) as well as the ACP on the SSM ES.

The major elements of the EPS are two sets of SAs mounted on booms capable of pointing them to the Sun; six NiH2 batteries, each with a Charge Current Controller (CCC) and a Voltage/Temperature (V/T) Improvement Kit (VIK); one Power Control Unit (PCU) providing switching for the main and essential buses; and four PDUs for interfacing the EPS to the end-item subsystems. All components are replaceable during on-orbit servicing. An overview of the HST EPS is provided in Drawing 2.4 sheet 1.

Circuit protection (fusing) is provided in the distribution circuits to ensure safe and reliable operation of the EPS. Fusing is rated to protect HST wire harnesses and EPS equipment from short circuits occurring in the end-item equipment. Separate power lines from redundant circuits are routed through separate connectors to supply redundant power to this equipment. Switching services provided in the PCU and PDUs are capable of switching and isolating power sources.

The HST power return buses are referenced to a single point on the SSM structure via a Single-Point Ground (SPG). The SPG can be connected to the orbiter through the shuttle umbilical and is designed to carry the maximum expected fault current. Actual electrical connection to the orbiter is provided via any of three umbilical systems on the telescope AS. The deployment umbilical provides two connectors (J105 and J106) that *are* not used during servicing missions. The FSS *main* umbilical is routed along the AS aft bulkhead to the J101 connector mounted atop the umbilical tower. Both sets of umbilicals originate from the terminal board on the aft bulkhead. For the servicing missions, the terminal board provides an identical pair of connectors (J107 and J108) to the FSS umbilical tower. The umbilical tower integrates these two connectors into one receptacle (the J101 connector) for interfacing with the umbilical mate/demate mechanism on the FSS fails, the two connectors (J107 and J108) leading to the

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J101 connector can be manually demated and connected to a UDM similar to that flown for the deploy mission. The J107 and J108 connectors have retractable tethers that pull them back to a seated position after they are disconnected from the UDM. The AS umbilical system is depicted in Drawing 2.2 sheet 5.

2.4.1 EPS Hardware

2.4.1.1 Solar Array

During Servicing Missing (SM) 1, the original SA1s were replaced with SA2s. SA2 was similar in design to SA1 with a few redesign exceptions. The SA1 and SA2 design made them susceptible to temperature-induced jitter. During SM3B new, rigid SAs, or SA3s, were installed. These arrays are intended to provide for the long-term power needs of the HST while reducing the jitter experienced with the previous flexible arrays designs.

The HST SA3 system comprises two symmetrically designed wings. Each wing is equipped with four panels attached to a Panel Support Structure (PSS) and a Solar Array Drive Mechanism (SADM) that slews the array to face the Sun when it is deployed. The weight of one SA is 640 pounds. The SA output is passed to the SSM EPS via two diode boxes for battery charging and distribution to the HST load as shown in Drawing 2.4 sheet 1. The system consists of two wings and control electronics to allow positioning of the SA solar cell panels for optimum solar illumination. The SA system is capable of delivering approximately 4730 watts of power to HST.

The SA physical configuration is depicted in Figure 2.4-1 and Figure 2.4-2. The wing mechanical functions and characteristics are discussed in Section 6.0.







Solar Array 3

Figure 2.4-2. SA3 (back side)

The individual solar panels are procured directly from the Iridium commercial production line and are built using a 5056 Al honeycomb core assemblies with front and backside graphite epoxy face sheets. The front face sheet is covered with Kapton sheeting and is then populated with solar cells. The rear face sheet of the panel is co-bonded with a Tedlar coating. The honeycomb core edges are covered with Tedlar tape. Each panel weighs 33.5 pounds.

Each of the four solar panels on an SA wing contains 1000 6.5- by 5.5-cm Gallium Arsenide on Germanium solar cells, shown in Figure 2.4-3. These solar cells are wired within the solar panels into 20 circuits of 50 series-connected cells each. These 20 circuits are grouped electrically into 10 strings of two parallel-connected circuits, as shown in Figure 2.4-4. Each string is capable of producing a maximum short circuit current (Isc) of 2.4 amps. The power from each solar panel string is carried individually through redundant wiring all the way to the SA wing Diode Box Assembly 2 (DBA2), where the strings are electrically combined into individual Solar Panel Assemblies (SPAs). See Section 2.4.1.2 for DBA2 detailed information.



Figure 2.4-3. SA3 solar cell detail



Figure 2.4-4. SA3 circuit detail

The electrical interface between each solar panel and its SA wing is a single 37-pin sub-D connector, which is supported by the interface connector-mounting bracket located on the backside of the panel (total of four panels per wing). Solar panel power passing through this sub-D connector is carried from the panel to the coupler through a wiring harness bundle consisting of 20 twisted pairs (i.e., 40 wires per bundle). This allows the power from each panel's 10 strings to be carried redundantly through two separate twisted wire pairs within the bundle; either pair is rated to carry the full string current by itself.

The four SA wing wiring harnesses are combined within the coupler and are passed on to the SADM flex harness, as shown in Figure 2.4-5. The flex harness design for SA3 is intentionally unchanged from SA1/SA2 because it is a proven design *that* ensures SADE torque compatibility. The overall design is unchanged. However, the flex harness wiring of circuits passing through it had to be changed relative to SA1/SA2 because the individual SPAs in SA1/SA2 are electrically wired within the SA blanket itself and carried through the interconnecting wiring as intact SPAs. In SA3, 40 individual strings are carried from the four solar panels (10 strings per solar panel) down the wing through the SADM flex harness, all the way to a DBA2, where the SPAs are then electrically formed.

2.4.1.2 Diode Box Assembly 2

The DBA2s are the second generation DBAs. The design is based on the original HST DBAs with design enhancements. In addition to providing diode isolation between the SAs and batteries, each DBA2 includes relays that are commanded by a respective Diode Box Controller (DBC). These additional relays could be used to recover from PCU relay failures and provide enhanced charge control capability. Control of these relays required the DBA2s to be connected to the HST P601 EPS test connector (at the EPS test panel). The DBA2s also include arc suppression networks to extend the life of the PCU relays and are equipped with wingtab connectors for easier EVA installation.

The DBA2 isolation diodes are JANTXV1N5816 high efficiency diodes. As mentioned above, the two diodes of a single SPA circuit are mounted on the same heatsink and in electrical contact with it. Isolation from the other heatsinks is provided by an insulating plate onto which the heatsinks are bolted.

Within the DBA2, the SPAs are electrically formed and are then fed to the PCU. The 40 solar panel strings fed to each DBA2 from their respective SA wings are wired into SPAs as follows:

- a. Three 5-string SPAs: D, E, and AA
- b. Six 4-string SPAs: A, B, C, BB, CC, and DD
- c. One 1-string SPA: EE

The SPA assignments are configured to protect the HST wire deratings and the derating of the PCU and DBA2 relays.

Each DBA2 contains 28 relays, as shown in Drawing TBD: 10 latching disconnect relays, 9 nonlatching charge control relays, and 9 latching bypass relays. A set of three relays (one disconnect, one charge control, and one bypass) is designated for each of the nine SPAs that provide battery charging. The \pm EE SPAs are not connected to a battery and, therefore, have only disconnect relays in the DBA2. The disconnect relays will be used to electrically disconnect the SAs from HST and may be used to deadface the SA connectors. The charge control relays are provided for each SA SPA, except the \pm EE SPAs, and are used to manage battery charging. The bypass relays will be used to recover charge control relays that have failed in the disconnect position. The operation of these relays will be described in Section 2.4.1.3.

An Arc Suppression Network (ASN) is included in the DBA2 for each SPA section to eliminate arcing across the contacts of the PCU relays. The way the ASN operates is as follows: In sunlight, with charge control relays closed, a capacitor is charged to the SPA voltage level. The charge control relay starts to open, and as the contact resistance increases (arcing across the contacts), the SPA voltage increases, thus causing the capacitor to absorb the SPA current, which limits the rate of voltage rise, causing the arc to extinguish. Then the capacitor charges up to the open circuit

SPA voltage. Once orbit night arrives, the capacitor discharges through the 100 k Ω resistor down to the bus voltage.

Thermal Control

Heat generated by the DBA2 power diodes is radiated out of the unit through the silverized Teflon-coated heatsink plates. The inside face of the shield surrounding the plates is painted black. All other external surfaces are taped with aluminized Kapton except where they are covered by the mounting bracket.

Active thermal control of the DBA2 is provided during HST mission phases when the SA3 is not generating power. Survival heaters are installed on each of the 10 power diode heatsink plates in a DBA2 and controlled by two parallel sets of two series-connected thermostats. Heater and thermostat placement is shown in Drawing 10.3 *sheet* 4. Each heater element is approximately 256 ohms. A DBA2 survival heater circuit consists of the thermostats controlling 10 heater strips connected in parallel. The instantaneous maximum power required by the DBA survival heaters is 80 watts (40 watts per DBA2). The survival heaters are powered off during nominal HST operations because heat generated by the power diodes warms the DBA2 while the SA is generating electricity.

2.4.1.3 Diode Box Controller

The DBC is an addition to the DBA2 that provides a means to recover from certain PCU relay failures. The DBC provides battery charge control external to the PCU using relays in the DBA2.

The DBC enables the STOCC to reconfigure the PCU and perform battery charge control in the DBA2/DBC (Software Charge Mode (SCM)). It provides an Autonomous Charge Mode (ACM) to provide battery charge control while in hardware sunpoint. The DBC provides an SCM to reduce usage of PCU relays during normal operation and provides disconnection of SA power for PCU changeout.

The DBC has three modes of operation: SCM, ACM, and Ground Command Mode (GCM). In SCM, the DBA2 relays are commanded by the 486 and EPS FSW based on battery voltage and temperature. ACM commands the DBA2 relays based on battery voltage and temperature also, but the commands originate at the DBC. GCM allows the DBA2 relays to be commanded in real time by the STOCC.

SCM is enabled by a Keep-Alive Signal (S K/A) sent from the DCE and by enabling the EPS charge control routine in FSW. The DCE S K/As are cross-strapped to both DBCs for redundancy using the cross-strap harness. The 486 monitors battery voltages and temperatures and compares them against a virtual V/T curve for chargeoff and charge-on determination. When a given battery has reached the chargeoff or charge-on V/T threshold, the 486 sends a relay specific pulse-width modulated signal to both DBCs to offline the respective SPA. The pulse-width command is decoded by the intended DBC and the specific DBC relay driver is commanded to open or close the

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respective nonlatching charge control relay in the DBA2. Loss of the S K/A from the DCE results in the disabling of SCM, and the CC relays in the DBA2 are autonomously closed.

ACM is enabled by the loss of the Autonomous Mode Disable (AMD) signals from the DCE, which are cross-strapped to both DBCs. Nine individual DBC hardware charge controllers, one for each SPA (except the EE), compare the respective temperature compensated battery voltage against a fixed chargeoff and charge-on threshold. When a given battery reaches the chargeoff or charge-on threshold, the hardware charge controller commands the respective relay driver to open or close the respective nonlatching charge control relay in the DBA2. Resumption of the AMD signals from the DCE results in the disabling of ACM.

Enabling the ACM required the EVA crew to install the V/T signal jumper plug on the cross-strap harness. This provides the battery voltage signals to the DBC. In addition, the crew installed six VIK jumper plugs to provide battery temperature signals to the DBC.

GCM is a subset of SCM and permits the following: the opening of each DBA2 charge control relay; the collective opening of the D, E, and AA DBA2 charge control relays in each or both DBA2s; the collective closing of all DBA2 charge control relays in each or both DBA2s; the capability to collectively open or close all 10 SPA disconnect relays in each or both DBA2s; the capability to collectively open or close all 9 bypass relays in each or both DBA2s. GCM requires the S K/A signals to be enabled, the 486 to be in control, and the critical command enable for disconnect and bypass relay operations.

Because of a limitation of available telemetry channels, the collective status of the 10 disconnect relays and the collective status of the 9 bypass relays in the DBA2 are provided in telemetry by using one spare bilevel telemetry channel in each SADE. SADE-1 conditions the collective status of the 9 bypass relays, and SADE-2 conditions the collective status of the 10 disconnect relays. For each relay type, the telemetry indicates if all are open or one or more are closed. The status of the nine DBA2 charge control relays is provided in telemetry by using nine spare analog telemetry channels in the DCE.



Figure 2.4-5. SA wiring/harness overview

The SADM flex harness terminates in connectors that mate to the DBA2. The current SA1/SA2 DBA design utilizes ganged rack-and-panel Deutsch connectors on a common drive stud bracket, with the Y- and Z-ganged connectors dedicated to passing SA power to the DBA X-connector. The X-connector passes power from the DBA to the spacecraft. The W-connector is dedicated to SA and DBC command and telemetry and heater power (see Figure 2.4-6).



Figure 2.4-6. SADM to DBA2 detail



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<u>،</u>		SPA IDENTIFICATION			
	BATTERY	+ WING (+V2)		- WING (-V2)	
	NO.	BLANKET 1 = +S ₁	BLANKET 2 = -S ₁	BLANKET 1 = +S ₁	BLANKET 2 = -S ₁
	1	D	DD		-CC
	2	С	AA	-B	
	3	A	BB	-E	
	4	В		-C	- A A
	5	E		- A	-BB
	6		сс	-D	-DD
	BUS		EE		-EE

DSGN T. WISONG 12/06/85 LYNDON B. JOHNGON SPACE CENTER HOUS DR SSM ELECTRICAL ENGR P. RIECK SUBSYSTEM OVERVIE ENGR SA3/DBA2/PC	TONL TEXAS POWER W- POST		
DR SSM ELECTRICAL ENGR P. RIECK SUBSYSTEM OVERVIE ENGR SA3/DBA2/PC	POWER W- POST		
ENGR P. RIECK SUBSYSTEM OVERVIE	W- POST		
ENGR SA3/DBA2/PC	SUBSYSTEM OVERVIEW- POST		
	SA3/DBA2/PCU		
APP INSTALLATIO	INSTALLATION		
APP HST CSM DWG NO	1		
CADS ALL Z	+		
LTR PCN 126.5 X 34 SI	EET 1 OF 5		

Note: Cell data are taken from cells irradiated to equivalent 1 MeV electron flux: 2E13 el/cm ²	Cold case 44° C	Hot case 81° C
Cell working voltage (mV)	349	349
Cell working current (mA)	333	322.1
Voltage at SA/SSM interface (V)	34	34
Voltage losses (V)	2.0	2.0
Voltage at string ends (V)	37	37
Parallel cells of SA	460	460
Current lost factor	0.882	0.929
Working current of SA (A)	135.1	137.6
Calculated power output of SA (W)	4594	4680

Table 2.4-1. Solar array output

The array output has been calculated using the following blanket temperature data:

Maximum temperature	98° C
Minimum temperature	-85° C
Hot case design temperature	81° C
Cold case design temperature	44° C

2.4.1.4 Batteries

The HST is powered during orbital eclipse by six NiH₂ batteries. *These are due to be changed out on SM4 (reference Section 11 TBD).*

The design requirements for the NiH₂ batteries are

- a. To provide for continuous orbital power at 26.2 to 34.3 V dc
- b. To provide for up to five orbits of up to 30° off-nominal roll with Depth-of-Discharge (DOD) limited to 20 percent over any 24-hour period
- c. To provide at least 164 A-h for safe-mode or gravity gradient mode operations
- d. To provide an operational life of up to 60 months
- e. To provide maximum orbital power of 1250 watts at the OTA/SI interface 2 years into the mission with one battery failed

The battery characteristics are listed in Table 2.4-2.

Table 2.4-2. Battery characteristics

Weight, pounds (estimated)	153
Volume Length, inches Width, inches Height, inches	37.80 10.10 11.25
Capacity, A-h	90
Depth-of-discharge (nominal), percent	10.0
Life, requirement (estimated), years	5.0
Trickle charge (estimated time), minutes	5.0
Solar array charge power, watts Potential power savings (peak), watts	1519 721
Battery replacement (two times), years	15

The requirements are met by using six 88 Ah, NiH_2 batteries. Each battery consists of 22 active cells and electronics. The batteries are installed in the HST as two modules (three batteries each) on the doors of bays 2 and 3 (Figure 2.4-7). Batteries 1, 2, and 3 are located in bay 2 and batteries 4, 5, and 6 are located in bay 3.

A battery is composed of 23 cells, of which 22 are wired in series to a power connector as shown on Figure 2.4-8. Two connectors, one power (J1), as mentioned, and the other (J2) for telemetry and heater circuits are attached to the connector bracket. Each cell is clamped with a cast aluminum sleeve and then wrapped with a silicone coated fiberglass cloth. A dual element heater strip is bonded at the bottom and inside of each sleeve. The sleeve/cell assemblies are then fastened to a battery baseplate made from 0.125-inch aluminum. A support tube, the same diameter as the sleeve, is attached to the baseplate in the middle of the battery.

The cells that make up the battery are constructed with dry sintered nickel electrodes that are stacked with the negative electrodes, zirconium oxide cloth separators, and gas screens on a polysulfone core (Figure 2.4-9). Both positive and negative electrode leads exit through the center of the core and allow both terminals to be located on the same end of the cell pressure vessel. This design feature, along with other battery packaging allows the final Nickel-Hydrogen Battery Design Module to fit into the limited space of the HST bay. The cell pressure vessels are dome-shaped, about 3.5 inches in diameter and approximately 9.3 inches long. The cell terminals are each located at 45° from the vessel longitudinal axis. The cells are activated with 27 percent potassium hydroxide electrolyte designed to promote long life and high discharge voltage of the HST application.



SSM bay 2





Figure 2.4-7. Batteries installed in SSM ES bays 2 and 3

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Figure 2.4-8. Battery cell configuration



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Each battery module is housed in an aluminum enclosure and mounted on a 0.5-inch-aluminum baseplate that was specially machined to accept either replacement hardware of a NiCd or NiH₂ battery module. The module baseplate has 4 shear pins and 14 latches (J-hook, captive fasteners) for proper location and bay door mounting (Figure 2.4-10).

Three batteries are bolted to the module baseplate. The battery connector brackets are mounted on the module connector plate (part of the enclosure structure). A cover encloses the top of the batteries and is bolted to the support tubes to provide overall structural integrity. The inside surfaces of the enclosure and cover are lined with polyamide foam for short circuit prevention and protection. Two bright yellow handles are mounted on the cover for EVA handling. The module weighs approximately 470 pounds and can be replaced in orbit by disconnecting three power and three telemetry wing-tabbed connectors and by disengaging 14 ORU latches.



Figure 2.4-10. Battery module

Temperature control of the battery environment is provided by means of a controller, a thermistor, and a thermostat in each NiH₂ battery. A parallel backup system, operating between the limits of 23° and 31° F, is located within each battery. Normal battery temperature range is between the limits of 28° and 36° F.

Each battery has the following dedicated monitors and controls:

- a. One temperature monitor for telemetry
- b. Four charge control thermistors
- c. One primary and one backup heater system with controllers

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d. One current monitor

e. Two cell pressure monitors

The temperature monitor outputs a 0 to 5 V dc signal for telemetry. It uses a thermistor located on a cell vessel top near the center of the battery. The full temperature range of this monitor is 10° to 160° F, but is most accurate between 20° and 80° F.

Two charge control thermistors (primary and redundant) form part of the voltage divider network located in the CCC. They are used to control the KX1 and KX2 relays for battery charging. Two additional overtemperature cutoff thermistors (primary and redundant) are used for charge termination in the event of CCC failure, or if battery thermal runaway occurs. Charge cutoff temperatures are set for $82^{\circ} + 3/-2^{\circ}$ F and $73^{\circ} \pm 3^{\circ}$ F and $78^{\circ} \pm 3^{\circ}$ F, respectively.

Each battery is also provided a primary and a backup heater system with controllers that turn heaters on and off based on set-point temperatures. Each heater system has 23 strip heater elements connected in parallel. A bimetallic thermostat is included in each heater circuit to cut off heater power in case the controller fails in the closed position.

Each battery has a current monitor that operates within the range of -25A to +25A and outputs a 0 to 5 V dc analog signal for the telemetry. It measures the battery current at a point 3 inches from the last cell in the battery cell circuit.

Each battery also has two cell pressure monitors that measure the battery internal pressure from 0 to 1500 psi, and, when calibrated, they give an indication of the battery State of Charge (SOC). One is telemetered by DIU A side and the other by DIU B side. Therefore, although two pressure monitors are provided by each battery, only one can be monitored at any one time. The controls for the pressure monitor excitation reside in the PDUs, which supply the power. Battery discharge characteristics are provided in Figure 2.4-11.

The batteries are recharged by the SA through CCCs. Each battery is hardwired to three switchable SPAs, as shown schematically in Drawing 2.4 sheets 1 and 2. The three SPAs for each battery are geometrically selected within the SA to ensure equal energy distribution under varying sunlight and shadow conditions.

HST Launch Experience

- a. Prelaunch; 27º C; 68 hours; 20 percent discharge; no load
- b. Postlaunch; +2º C; 24 hours; 1 percent discharge; no load
- c. Postlaunch; solar array deployment; +2° C; 6 hours 40 minutes; 24 percent discharge; 1 kW load
- d. Postlaunch; solar array charging; +2º C; 4 hours to 100 percent charge; 1 kW load



Figure 2.4-11. Battery discharge characteristics



Figure 2.4-11. Battery discharge characteristics (concluded)

2.4.1.5 Charge Current Controller

The complete CCC/battery charge/discharge system comprises 18 SPAs (3 to each battery string) as power sources supplying the 6 NiH₂ batteries and the HST load. Each battery charge function is controlled by a dedicated CCC. The CCC provides multilevel, voltage-temperature control of battery charging through two independent channels, as shown in Drawing 2.4 sheet 2. One CCC is provided for each of the six batteries, and each operates independently of the others.

Each CCC measures 8.13 by 7.62 by 4.5 inches and weighs approximately 4.9 pounds. Each unit is colocated in the same SSM ES bay with its corresponding battery, mounted to the SSM ES inner barrel structure. The installation of the CCCs in SSM ES bays 2 and 3 is shown in Figure 2.4-7. The CCC consumes less than 4 W of power and can operate over a 24 to 32 V dc range.

The CCC is capable of compensating for battery aging by providing four levels of voltage/temperature on/off control to two relays, KX1 and KX2, resident in the PCU. (The "X" corresponds to battery string number.) The two channels are independently controlled with separate power excitation and battery reference voltage/temperature inputs. The CCC receives battery terminal voltage and temperature data from sensors within the battery case. Sensing circuits within the CCC react to this information

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and cause the KX1 and KX2 relay driver circuits to function within the battery reference voltage limits, specified by the STOCC, to satisfy the charging needs of the batteries. An overtemp cutoff circuit in each channel removes the charging source of the associated battery string at a preset temperature limit for each channel. All CCC modes of operation have command override capability in the event of circuit failure. The four on/off, voltage/temperature settings are each selectable by STOCC command via DIU-2, as shown in Drawing 2.4 sheet 2. A typical battery charge/discharge cycle is depicted in Figure 2.4-12.


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Orbital Period



Figure 2.4-12. Battery charge/discharge cycle

2.4.1.6 Voltage/Temperature Improvement Kits

Because of changes in battery and battery bay characteristics, the battery CCC voltage/ temperature curves are not low enough to preclude battery thermal instability during hardware safe-mode. The VIK modifies control signals from each battery to its respective CCC to lower the V/T curve levels. Battery charge cutoff at lower V/T levels provides adequate system SOC and minimizes battery thermal instability in hardware safe-mode. Six VIKs were installed in SSM bays 2 and 3 during SM3A, one for each HST battery.

Physical Description

Physically, the electronics for each VIK are individually packaged, one for each battery. Each VIK weighs approximately 5 pounds and consists of an electronics box, harnessing, and a gender changer. Figure 2.4-13 shows an overview of an uninstalled VIK. A gender changer is incorporated into the VIK design to convert the battery

connector from a scoop-proof pin to scoop-proof socket interface. This makes subsequent battery EVA activities (battery changeout on SM-4) more compliant with safety requirements, since there are two hot pins on each battery connector. Each VIK is restrained to other battery harnesses using Velcro straps. Figure 2.4-14 shows the VIK installed on one of the HST battery modules.







Figure 2.4-14. VIK installation

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Electrical Power

The VIK is a passive device that consists of a pair of relays (to switch the device in and out of the circuit), zener diodes, and resistors. The relays are powered from PDU relays that were previously assigned to provide redundant battery heater power. The isolation relays for VIKs 3, 5, and 6 are powered by PDU-2; for VIKs 1, 2, and 4 the relays are powered by PDU-4.

Thermal

The VIK operational limits are -10° to +30° C, with a minimum turn-on limit of -40° C. Survival limits are -60/+80° C.

Functional Description

The sole function of the VIK is to shift the voltage signal up from the battery to its respective CCC by approximately 700 millivolts. Put another way, the VIK makes the CCC "think" that the battery voltage is 700 millivolts higher than it actually is; so with the VIK installed, the battery charge cutoff is 700 millivolts lower than normal. Therefore, the VIK effectively shifts the CCC *V/T* level control range down by one full level (700 millivolts between levels).

In order for the batteries to be properly charged during normal operations with the VIK installed, the CCCs are set one level higher than they were prior to VIK installation. When the vehicle goes into safe-mode, the BIK, installed on SM2, resets the CCC to one charge level lower. This results in the batteries cutting off one level lower than they do today, and the heating problem is eliminated.

2.4.1.7 Battery Reconditioning

The voltage characteristic degradation in a NiH_2 battery is largely a reversible process. The characteristic can be restored to its original state by reconditioning whereby each cell in the battery is completely discharged for some predetermined period of time. Reconditioning is performed by individually loading and discharging each battery through a relay and resistor network. This is accomplished through reconditioning relays in the PCU, shown in Drawing 2.4 sheets 1 and 2.

2.4.1.8 Power Control Unit

The PCU provides control and distribution of electrical power from the HST SAs and batteries to the four PDUs. The physical dimensions for the PCU enclosure 44.7 inches *long* by 22.6 inches *wide* by 8.8 inches *high*. The height dimension includes bolts that extend below the PCU Baseplate for attaching the astronaut tether loop. These bolts fit into a cutaway area on the mounting surface for the PCU in Bay 4 of the HST. The unit weighs approximately 150 pounds and dissipates approximately 20 to 25 watts, depending on the HST load and configuration. It is mounted in bay 4 of the SSM ES on the rings of the inner barrel structure, as shown in Figure 2.4-15.



Figure 2.4-15. SSM ES bay 4 (PCU and PDUs)

The PCU was functionally depicted in Drawing 2.4 sheet 1. Portions of the unit are schematically detailed in Drawing 2.4 sheet 2, depicting the battery charge control circuitry; and the bus control switching circuitry of the PCU will be discussed in Section 2.4.2. Its circuitry consists primarily of relays, transfer switches, diodes, current and voltage monitors, bus bars, and printed wiring boards. Functionally, the circuitry can be divided into six categories, as described in the subsequent paragraphs.

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SA Trim Relays

Twenty latching, 25-A relays provide the capability to disconnect any of the 20 SPAs of the SA from the bus. These relays were used to disconnect all SA panels during deployment and retraction *prior to the installation of the SA3s*, and may be used for trim charging capability in contingency situations. Off-angle pointing of the array is the intended normal operating mode for limiting array output. Drawing 2.4 sheet 1 depicts the SA trim relays. Note that each of the six battery circuits draws power from three SPAs and all account for 18 of the 20 relays. The final two trim relays are not associated with a battery circuit, but will disconnect SPA +EE and SPA -EE from the diode bus.

The SA trim relays are controlled by the STOCC via DIU-2 as depicted in Drawing 2.4 sheet 2. Twenty DIU HLD commands are provided to disconnect each SPA individually. Six additional HLDs reconnect the 20 panels in 4 groups of 4 SPAs and 2 groups of 2 SPAs.

Currently, 3 SA trim relays have failed, resulting in the loss of 2 SPAs. K16 has failed in the open position, effectively removing the +DD SPA from the HST EPS. K44 has failed in the closed position, which has not resulted in a loss of the –AA SPA, but does have safety implications during the PCU changeout on SM-3B. K74 has failed shorted, which has removed the +EE SPA from the HST EPS.

CCC and Bypass Relays

Charge current from the SAs is controlled by 25-A, nonlatching relays in the PCU that are in turn controlled autonomously by each CCC for its battery circuit. No capability exists to directly command these relays from the ground. Control of their operation is achieved through configuration commands to the CCCs. These relays can be bypassed by CDI special discrete commands from the ground. The discrete commands go to two bypass relays in the battery circuit. Each bypass relay can be commanded on individually, but both are commanded off by a single HLD command from either DIU-2 or DIU-4. The bypass relays are for contingency operations in the event of a CCC failure, and the relays remain open during nominal operations.

Battery Reconditioning

The PCU contains the switching circuitry to permit battery reconditioning when required. Drawing 2.4 sheet 2 depicts the typical arrangement for battery 1. Four relays are provided in each battery-charging string to enable removal of a battery from the charging circuit, connection of the associated SPAs to a diode bus, and connection of the battery to 6.1-ohm and/or 50.7-ohm loads. The commands that switch the SPAs between the battery bus and the diode bus are CDI special discretes. The commands that provide switching of the reconditioning load resistors on/off are DIU HLDs via either DIU-2 (for batteries 1 through 5) or DIU-4 (for battery 6).

The reconditioning relays provide the additional capability to isolate a diode bus, which has been rendered unusable, and still retain the service of the SPAs that supply that bus. For example, if diode bus A were unusable, the service of batteries 1 and 2 would be lost. The reconditioning relays allow diode bus A to be isolated and allow the SPAs, which originally supplied diode bus A, to be transferred to diode bus B.

The battery discharge relays are interlocked so that either the 5-A or the 1-A load resistance can be connected to the battery, but not both at once. The resistors used to load the batteries are heater elements and are mounted on the SSM ES bulkhead near the area of the +V2 trunnion bay.

Voltage Limiting Circuit

A voltage limiting circuit is provided in each diode bus feeder to limit the voltage at the busing equipment to 32.0 V dc. When the diode bus voltage exceeds 32.72 ± 1 V dc, the bypass relay opens and inserts the diode in the circuit as a voltage-dropping element. The relay recloses, shunting the diodes, when the bus voltage drops to 30.15 ± 1 V dc. The STOCC can enable or disable the voltage control circuit by means of CDI special discrete commands. The circuit will be disabled only if a circuit failure occurs. A single CDI command will enable three of the circuits (i.e., two commands enable all six), and each can be disabled individually.

An additional backup shunt relay permits the ground-controlled bypass of the voltagedropping diode in each diode bus feeder in the event the voltage sensing circuit fails. The controls for this relay are provided via nine DIU-4 HLDs, six of which individually open the backup shunt relay. The backup shunts are closed in pairs (i.e., three commands for the six relays). These relays are open (shunt out) under nominal operating conditions.

Bus Separation Switching

The PCU provides three main buses (A, B, and C) for distribution of HST electrical power by the PDUs and an essential bus to provide continuous power to the CDIs and MA command receivers. Each of the three diode buses (A, B, and C) is connected to a corresponding main bus through transfer (motor) switches in the PCU. One switch connects the A buses, and one switch connects the B buses. Diode-bus-C-to-main-bus-C connections are made through primary and redundant switches, as depicted in Drawing 2.4 sheet 1, and are nominally open. The transfer switches employed are motor-driven, 100-amp switches requiring approximately 7 amps during their 50- to 60-ms transfer time. Because of the switching current requirements, pilot relays are used to control the operation of the transfer switches. Control over the external/internal bus configuration is provided to the flight crew from the SSP in the AFD or from an ACP mounted in the +V2 trunnion bay of the SSM ES. A complete description of the orbiter interface is provided in Section 2.4.2.

The diode bus interconnect switches (S7 through S10) are nominally closed and are opened by the 486 onboard computer in the event of a bus overcurrent or excessive

structure current condition. The bus overcurrent condition arises when the total bus current exceeds 125 amps for 10 seconds or 150 amps for 2 seconds. The excessive structure current condition occurs if structure current exceeds 15 amps for 10 seconds or 50 amps for 2 seconds. Under such conditions, the 486 will also close switches S3 and S4, and will connect diode bus C to main bus C. The PSEA will also open the interconnect switches and connect main bus C upon entry into hardware Sun-point mode for any reason. The 486 and PSEA commands are pulses, rather than sustained signals, to allow STOCC to reset the affected switches, whether or not HST is in safe-mode.

2.4.1.9 **Power Distribution Unit**

The HST electrical power is distributed to a majority of the SSM and OTA equipment via four PDUs. The PDUs are identical to each other and include latching and nonlatching relays, bus bars, diodes, fuses, current sensors, and voltage monitors. Most of the fuses are mounted internal to the box, but fuses in circuits that supply equipment classed as ORU are installed in fuse modules external to the PDU for ease of replacement on orbit. Each PDU has three fuse plug modules equipped with wingtabs for ease of changeout by an EVA crewmember.

Each PDU is 10 by 5.4 by 17.5 inches, and each is installed on the door of SSM ES bay 4, as shown in Figure 2.4-16, using six 0.20-inch diameter bolts per unit. Each unit weighs approximately 29 pounds and dissipates 5.2 watts of electrical power. The PDUs are designed and built by Lockheed Missiles and Space Company (LMSC).

During the SM-1 mission, each of the P15 fuse plugs (one per PDU) *was* changed out. The 3-amp fuses were changed out with 5-amp fuses to overcome underrated fusing to the RGA/ECU circuits served by the P15 fuse plugs. The 3-amp fuses have to carry from 2.0 to 3.3 amps for 27 seconds.

Circuit Description

Each PDU has six distribution buses, with two buses that are supplied from each main bus in the PCU (A1 and A2, B1 and B2, C1 and C2) by two individually fused feeders. Individual circuits are then in turn supplied from an A, B, and C bus through diodes and fuses and supplied to the individual end items (Drawing 2.4 sheet 1). Main bus C is unpowered during nominal operations. Some of the circuits that are operated by STOCC command include latching or nonlatching relays. These relays are commanded by STOCC using DIU HLDs or CDI special discretes.

Table 2.4-3 provides a summary of all PDU circuits, including PDU connector number and circuit rating. If the circuit is supported by a fuse plug, the corresponding fuse plug number is also provided.



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Figure 2.4-16. PDUs installed on SSM bay 4 door

PDU 1 relay	Item controlled or function	PDU 1 conn.	CKT rating
K1	STIS Redundant Power	J4	20A/P16
K2	SADE 1 OLD Power	.]4	20A/P17
K3	(Spare)	.14	20A/P16
K4	FOC Primary Power	15	200/P16
K5	Main Ring CGG Heaters - OTA Rus 9	18	207/110
KG	PWA 1 Main Rue A or B Soloct	110	20A 20A/D17
	RWA 1 Main Bus A 01 D Select	110	30A/F17
	ED/TOE Bug 2 Drimony Dower Heater		30A/F17
KO	EP/ICE Dus 3 Filliary Fower Redier	J5	20A
K9	EP/TCE Bus 2 Redundant Power	JS	
K10	EP/ICE Bus 6 Primary Power	15	10A/P16
K11	EP/ICE Bus 5 Primary Power	J <u>5</u>	10A
K12	Forward Shell Primary Heater Power	J/	10A
K13	(Spare)	J8	3A
K14	(Spare)	J7	10A
K15	SIC&DH Power Redundant	J6	10A/P16
K16	BAT 3 Heater Primary Power	J9	3A/P15
K17	RWA 1 Primary Heater	J8	3A
K18	SIC&DH Redundant Heaters	J6	10A/P16
K19	DCE Primary Power	J6	10A
K20	FHST 1 Redundant Heater	J9	3A/P15
K21	SDM +Wing Survival Heaters	J7	3A
K22	RSU 3 Ch. 5 Operational Heaters	J4	5A/P15
K23	RSU 1 Ch. 1 Operational Heaters	J4	5A/P15
K24	MA Transmitter 1 Power	.17	3A
K25	RSU 1 Survival Heater Ch 1	.19	5A/P15
K26	FHST 1 Power	.19	34
K20	FHST 1 Primary Heater	17	30
K28	SSPs 1 and 3 Primary Heaters	10	34
K20	STIS PILL Drimory Dowor	10	3A 2A/D15
K29	BAT 5 Drimony Hostor	JO	3A/F13 2A/D15
K30	Chara	J9 10	SA/FIS
K31	(Spare)	JO	ЪА
K3Z	(Spare)	J7	
K33	FOC RIU Primary Power	J8	3A/P15
K34	CSS Operational Primary Heaters	19	3A
K35	RWA 1 Redundant Heater	19	3A
K36	BAY 4 Door Primary Heater	19	3A
K37	MSS 1 Power	J9	3A
K38	BAT 6 Primary Heater	J8	3A/P15
K39	Diode Box Primary Heaters	J8	5A
K40	Battery 1 Pressure Transducer Power	J6	10A
K41*	DISABLE +Wing SDM Survival Heaters	J7	3A
K42	OTC SIC&DH Control Redundant	J6	10A
K43	(Spare)	J9	3A
K44	FHST 1 Shutter Power	J9	3A
(H)	GEA 1 Power	J4	15A
ÌΉ)	ICU Bus A Power	J7	3A
(H)	MTE 1A Power Primary	J7	3A
ÌΗ)	MTE 3A Power Primary	J7	3A
ÌΗ	PSEA MCE-A Power	J8	3A
ίHί	RGA 1 Electronics Control Unit Ch. 1	J8	5A/P15
ίH	RGA 3 Electronics Control Unit Ch. 5	J8	5A/P15
ÌHÌ	Data Management Unit A Power	J6	10A
(H)	DIU 1 Element A Power	J7	3A
λ Ή	DIU 2 Element A Power	J7	3A
λΗ	DIU 4 Element A Power	J7	3A
λΗ	DILLS Flement A Power	.17	34
		51	0/1

Table 2.4-3. PDU circuit assignments[k1]

H = Hardwire. * = Nonlatching.

PDU 2 relay	Item controlled or function	PDU 2 conn.	CKT rating
K1	STIS Primary Power	J4	20A/P16
K2	SADE 2 OLĎ Power	J4	20A/P17
K3	(Spare)	J4	20A/P16
K4	FOC Redundant Power	15	20A/P16
K5	(Spare)	.18	204
K6	RWA 2 Main Bus A or B Select	110	304/P17
	RWA 2 Main Bus A of D Select	110	20A/D17
	ED/TCE Pue 2 Pedundent Dewer Heater		30A/F17
KO	EP/ICE Dus 3 Reduituditi Power Heater	J5	20A
N9	EP/TCE Dus 2 Primary Power mealer	JO	
K10	EP/TCE Bus 6 Redundant Power	J2	10A/P16
K11	EP/ICE Bus 5 Redundant Power	J <u>5</u>	10A
K12	Forward Shell Redundant Heater Power	J/	10A
K13	(Spare)	J8	3A
K14	(Spare)	J7	10A
K15	SIC&DH Power Primary	J6	10A/P16
K16	VIK-3 Disable	J9	3A/P15
K17	RWA 2 Primary Heater	J8	3A
K18	SIC&DH Primary Heaters	J6	10A/P16
K19	DCE Redundant Power	J6	10A
K20	FHST 2 Primary Heater	J9	3Å/P15
K21	SDM -Wing Survival Heaters	.17	3A
K22	RSU 3 Ch 6 Operational Heaters	.19	5A/P15
K23	RSU 1 Ch. 2 Operational Heaters	19	54/P15
K24	(Spare)	03	57/115
K24	(Spare) RSU 2 Survival Haster Ch. 6	10	5 A /D15
K20	ROU S SUIVIVAI REALEI CII. 0	J9 17	3A/F13
K20	FIGIZ Power	J7	3A 2A
NZ7	FHST 2 Redundant Heater	J7	3A
K28	ESTR 2 BAY 7,8 Primary Heaters	19	3A
K29	STIS RIU Redundant Power	J8	3A/P15
K30	VIK-5 Disable	19	3A/P15
K31	(Spare)	J8	5A
K32	(Spare)	J7	
K33	FOC RIU Redundant Power	J8	3A/P15
K34	CSS Operational Redundant Heaters	J9	3A
K35	RWA 2 Redundant Heater	J9	3A
K36	BAY 4 Door Redundant Heater	J9	3A
K37	MSS 2 Power	J9	3A
K38	BVIK-6 Disable	J8	3A/P15
K39	SADM/PDM Primary Heaters	J8	5A
K40	Battery 2 Pressure Transducer Power	J6	10A
K41*	DISABLE -Wing SDM Survival Heaters	J7	3Å
K42	OTC SIC&DH Control Primary	J6	10A
K43	(Spare)	.19	3A
K44	EHST 2 Shutter Power	.19	3A
(H)	GFA 2 Power	14	154
		17	20
	MTE 1B Dower Redundant	17	34
	MTE 28 Dower Redundant	17	24
		10	24
	POEA IVICE-D FUWEI DCA 1 Flastronica Control Linit Ch. 2	JO	
	RGA 1 Electronics Control Unit Ch. 2	JØ	5A/P15
	RGA 3 Electronics Control Unit Cn. 6	JQ	5A/P15
	Data Management Unit A Power	JD	IUA
(H)	DIU 1 Element A Power	J/	3A
(H)	DIU 2 Element A Power	J/	3A
(H)	DIU 4 Element A Power	J/	3A
<u>(H)</u>	DIU 5 Element A Power	J/	3A

Table 2.4-3. PDU circuit assignments (continued)

H = Hardwire. * = Nonlatching.

K1 NICMOS Primary Power J4 20A/P16 K2 486 B Power J4 20A/P17 K3 WF/PC Redundant Power J4 20A/P16 K4 COSTAR Redundant Power J5 20A/P16 K5 (Spare) J8 20A K6 RWA 3 Main Bus A or B Select J10 30A/P17 K7 RWA 3 Main Bus C Select J10 30A/P17 K7 RWA 3 Main Bus C Select J10 30A/P17 K8 EP/TCE Bus 4 Redundant Power J5 20A K10 EP/TCE Bus 7 Primary Power Heater J5 10A/P16 K11 Battery 5 Pressure Transducer Power J5 10A K12 LS/FS Deploy/Retrieval Heaters J7 10A K13 (Spare) J8 3A K14 SSA Transmitter 1 Power J9 3A/P15 K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A/P16 K17 RWA 3 Primary Heater J9	PDU 3 relay	Item controlled or function	PDU 3 conn.	CKT rating
K2 486 B Power J4 20Å/P17 K3 WF/PC Redundant Power J4 20Å/P16 K4 COSTAR Redundant Power J5 20Å/P16 K5 (Spare) J8 20Å K6 RWA 3 Main Bus A or B Select J10 30Å/P17 K7 RWA 3 Main Bus C Select J10 30Å/P17 K8 EP/TCE Bus 4 Redundant Power J5 20Å K9 EP/TCE Bus 7 Primary Power J5 10Å/P16 K11 Battery 5 Pressure Transducer Power J5 10Å/P16 K11 Battery 5 Pressure Transducer Power J7 10Å K13 (Spare) J8 3Å K14 SSA Transmitter 1 Power J7 10Å K15 (Spare) J6 10Å/P16 K16 BAT 4 Heater Primary Power J8 3Å K17 RWA 3 Primary Heater J8 3Å K18 (Spare) J6 10Å/P16 K19 SADE 1 Power J6 10Å K20 BAT 2 Primary Heater Ch. 3 J9 SÅ/P15	K1	NICMOS Primary Power	J4	20A/P16
K3 WF/PC Redundant Power J4 20Å/P16 K4 COSTAR Redundant Power J5 20Å K5 (Spare) J8 20Å K6 RWA 3 Main Bus A or B Select J10 30Å/P17 K7 RWA 3 Main Bus C Select J10 30Å/P17 K7 RWA 3 Main Bus C Select J10 30Å/P17 K7 RWA 3 Main Bus C Select J10 30Å/P17 K7 RWA 3 Main Bus A or B Select J10 30Å/P17 K8 EP/TCE Bus 1 Primary Power Heater J5 20Å K10 EP/TCE Bus 7 Primary Power Heater J5 10Å K11 Battery 5 Pressure Transducer Power J5 10Å K12 LS/FS Deploy/Retrieval Heaters J7 10Å K13 (Spare) J6 10Å/P16 K14 SSA Transmitter 1 Power J8 3Å K17 RWA 3 Primary Heater J8 3Å K18 (Spare) J6 10Å/P16 K17 RWA 3 Primary Heaters J9 3Å/P15 K22 RSU 2 Survival Heater Ch. 3	K2	486 B Power	J4	20A/P17
K4 COSTAR Redundant Power J5 20Å/P16 K5 (Spare) J8 20A K6 RWA 3 Main Bus A or B Select J10 30Å/P17 K7 RWA 3 Main Bus C Select J10 30Å/P17 K8 EP/TCE Bus 4 Redundant Power J5 20Å K9 EP/TCE Bus 1 Primary Power Heater J5 20Å K10 EP/TCE Bus 7 Primary Power J5 10Å/P16 K11 Battery 5 Pressure Transducer Power J5 10Å K12 LS/FS Deploy/Retrieval Heaters J7 10Å K13 (Spare) J6 10Å/P16 K14 SSA Transmitter 1 Power J8 3Å K17 RWA 3 Primary Heater J8 3Å K18 (Spare) J6 10Å/P16 K19 SADE 1 Power J6 10Å/P16 K19 SADE 1 Power J6 10Å K20 BAT 2 Primary Heater Ch. 3 J9 SÅ/P15 K21 SDM +Wing Retrieval Heater Sh. 3 J9 </td <td>K3</td> <td>WF/PC Redundant Power</td> <td>J4</td> <td>20A/P16</td>	K3	WF/PC Redundant Power	J4	20A/P16
K5 (Spare) J8 20A K6 RWA 3 Main Bus A or B Select J10 30A/P17 K7 RWA 3 Main Bus C Select J10 30A/P17 K8 EP/TCE Bus 4 Redundant Power J5 20A K9 EP/TCE Bus 1 Primary Power Heater J5 20A K10 EP/TCE Bus 7 Primary Power J5 10A/P16 K11 Battery 5 Pressure Transducer Power J5 10A K12 LS/FS Deploy/Retrieval Heaters J7 10A K13 (Spare) J6 10A/P16 K14 SSA Transmitter 1 Power J8 3A K14 SSA Transmitter 1 Power J6 10A/P16 K15 (Spare) J6 10A/P16 K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A K20 BAT 2 Primary Power J6 10A K21 SDM + Wing Retrieval Heaters J7 3A K22 RSU 2 Survival Heater Ch. 3 J9	K4	COSTAR Redundant Power	J5	20A/P16
K6 RWA 3 Main Bus A or B Select J10 30A/P17 K7 RWA 3 Main Bus C Select J10 30A/P17 K8 EP/TCE Bus 4 Redundant Power J5 20A K9 EP/TCE Bus 1 Primary Power Heater J5 20A K10 EP/TCE Bus 7 Primary Power J5 10A/P16 K11 Battery 5 Pressure Transducer Power J5 10A K12 LS/FS Deploy/Retrieval Heaters J7 10A K13 (Spare) J8 3A K14 SSA Transmitter 1 Power J9 3A/P15 K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A/P16 K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A K20 BAT 2 Primary Heaters J9 SA/P15 K21 SDM +Wing Retrieval Heaterss J9 SA/P15 K22 RSU 2 Survival Heater Ch. 3 J9 SA/P15 K24 MA Transmitter 2 Power	K5	(Spare)	.18	20A
K7 RWA 3 Main Bus C Select J10 30A/P17 K8 EP/TCE Bus 4 Redundant Power J5 20A K9 EP/TCE Bus 4 Redundant Power J5 20A K10 EP/TCE Bus 7 Primary Power J5 10A/P16 K11 Battery 5 Pressure Transducer Power J5 10A K12 LS/FS Deploy/Retrieval Heaters J7 10A K13 (Spare) J8 3A K14 SSA Transmitter 1 Power J6 10A/P16 K15 (Spare) J6 10A/P16 K16 BAT 4 Heater Primary Power J8 3A K15 (Spare) J6 10A/P16 K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A K20 BAT 2 Primary Heaters J9 3A/P15 K21 SDM + Wing Retrieval Heaters J7 3A K22 RSU 2 Survival Heater Ch. 3 J9 5A/P15 K23 RSU 2 Operational Heaters J9	K6	RWA 3 Main Bus A or B Select	.110	30A/P17
NameHyperbolicJonJonJonJonJonJonK8EP/TCE Bus 4 Redundant PowerJ520AK10EP/TCE Bus 7 Primary Power HeaterJ510A/P16K11Battery 5 Pressure Transducer PowerJ510AK12LS/FS Deploy/Retrieval HeatersJ710AK13(Spare)J83AK14SSA Transmitter 1 PowerJ710AK15(Spare)J610A/P16K16BAT 4 Heater Primary PowerJ83AK17RWA 3 Primary HeaterJ83AK18(Spare)J610A/P16K19SADE 1 PowerJ610A/P16K19SADE 1 PowerJ610A/P16K20BAT 2 Primary HeatersJ93A/P15K21SDM +Wing Retrieval Heaters Ch. 3J95A/P15K22RSU 2 Operational Heaters Ch. 3J95A/P15K24MA Transmitter 2 PowerJ73AK25RSU 3 Survival HeaterJ73AK26FHST 3 Primary HeaterJ73AK27FHST 3 Primary HeaterJ73AK28SSR 1,3 Redundant HeatersJ93A/P15K31SSA Transmitter 1 Overtemp ControlJ710AK32CSAT 3 Redundant HeaterJ93AK34(Spare)J83A/P15K34SSA 7FHST 3 Redundant HeaterJ93AK35RWA 3 Redundant HeaterJ93AK36 <t< td=""><td>K7</td><td>RWA 3 Main Bus C Select</td><td>110</td><td>304/P17</td></t<>	K7	RWA 3 Main Bus C Select	110	304/P17
K9 EP/TCE Bus 1 Primary Power J5 20A K10 EP/TCE Bus 1 Primary Power J5 10A/P16 K11 Battery 5 Pressure Transducer Power J5 10A K12 LS/FS Deploy/Retrieval Heaters J7 10A K13 (Spare) J8 3A K14 SSA Transmitter 1 Power J7 10A K15 (Spare) J6 10A/P16 K16 BAT 4 Heater Primary Power J9 3A/P15 K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A/P16 K20 BAT 2 Primary Heater J9 3A/P15 K21 SDM +Wing Retrieval Heaters J9 SA/P15 K22 RSU 2 Operational Heaters Ch. 3 J9 SA/P15 K23 RSU 2 Operational Heater Ch. 1 J9 SA/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater J9<	K Q	ED/TCE Bus / Redundant Power	15	201
NS EP/Tote Bus 7 Primary Power JS 20A K10 EP/Tote Bus 7 Primary Power JS 10A/P16 K11 Battery 5 Pressure Transducer Power JS 10A K12 LS/FS Deploy/Retrieval Heaters J7 10A K13 (Spare) J8 3A K14 SSA Transmitter 1 Power J7 10A K15 (Spare) J6 10A/P16 K16 BAT 4 Heater Primary Power J9 3A/P15 K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A/P16 K19 SADE 1 Power J6 10A K20 BAT 2 Primary Heaters J9 SA/P15 K21 SDM +Wing Retrieval Heaters Ch. 3 J9 SA/P15 K22 RSU 2 Survival Heater Ch. 1 J9 SA/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater J7 <td< td=""><td>KO</td><td>ED/TCE Bus 1 Primary Power Heater</td><td>15</td><td>207</td></td<>	KO	ED/TCE Bus 1 Primary Power Heater	15	207
N10EPT/TOE Bus / Finitally PowerJ310A/P 10K11Battery 5 Pressure Transducer PowerJ510AK12LS/FS Deploy/Retrieval HeatersJ710AK13(Spare)J83AK14SSA Transmitter 1 PowerJ710AK15(Spare)J610A/P16K16BAT 4 Heater Primary PowerJ93A/P15K17RWA 3 Primary HeaterJ83AK18(Spare)J610A/P16K19SADE 1 PowerJ610A/P16K19SADE 1 PowerJ610AK20BAT 2 Primary HeatersJ73AK21SDM +Wing Retrieval HeatersJ73AK22RSU 2 Survival Heater Ch. 3J95A/P15K23RSU 2 Operational Heaters Ch. 3J95A/P15K24MA Transmitter 2 PowerJ73AK25RSU 3 Survival Heater Ch. 1J95A/P15K26FHST 3 PowerJ73AK27FHST 3 Primary HeaterJ93A/P15K30BAT 1 Primary HeaterJ93AK32COSTAR RIU Redundant HeatersJ93AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK36<	K10	ED/TCE Dus T Fillindiy Fower fielder	15	20A 10A/D16
K11Datery of ressure frainsuber PowerJ310AK12LS/FS Deploy/Retrieval HeatersJ710AK13(Spare)J83AK14SSA Transmitter 1 PowerJ710AK15(Spare)J610A/P16K16BAT 4 Heater Primary PowerJ93A/P15K17RWA 3 Primary HeaterJ83AK18(Spare)J610AK19SADE 1 PowerJ610AK20BAT 2 Primary HeatersJ93A/P15K21SDM +Wing Retrieval HeatersJ73AK22RSU 2 Survival Heater Ch. 3J95A/P15K23RSU 2 Operational Heaters Ch. 3J95A/P15K24MA Transmitter 2 PowerJ73AK25RSU 3 Survival Heater Ch. 1J95A/P15K26FHST 3 PowerJ73AK27FHST 3 Primary HeaterJ73AK28SSR 1,3 Redundant HeatersJ93A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 Overtemp ControlJ710AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant HeatersJ93AK36Computer Redundant HeatersJ93AK36Computer Redundant HeatersJ93AK36Computer Redundant HeatersJ93AK36Computer Redundant HeatersJ93AK37FHST 3 Redundant Heaters <td>K10 K11</td> <td>Battony 5 Processor Transducor Dowor</td> <td>15</td> <td>10A/F 10 10A</td>	K10 K11	Battony 5 Processor Transducor Dowor	15	10A/F 10 10A
K12LS/FS Deploy/Retrieval heatersJ710AK13(Spare)J83AK14SSA Transmitter 1 PowerJ710AK15(Spare)J610A/P16K16BAT 4 Heater Primary PowerJ93A/P15K17RWA 3 Primary HeaterJ83AK18(Spare)J610A/P16K19SADE 1 PowerJ610A/P16K20BAT 2 Primary HeatersJ93A/P15K21SDM +Wing Retrieval HeatersJ9SA/P15K22RSU 2 Survival Heater Ch. 3J9SA/P15K23RSU 2 Operational Heaters Ch. 3J9SA/P15K24MA Transmitter 2 PowerJ73AK25RSU 3 Survival Heater Ch. 1J9SA/P15K26FHST 3 PowerJ73AK27FHST 3 Primary HeaterJ73AK28SSR 1,3 Redundant HeatersJ93A/P15K30BAT 1 Primary HeaterJ93A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 Overtemp ControlJ710AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant HeatersJ93AK36Computer Redundant HeaterJ93AK36Computer Redundant HeatersJ93AK36Computer Redundant HeatersJ93AK36Computer Redundant HeatersJ93AK36Computer Redundant Heaters	K11	LS/ES Deploy/Detrioval Heaters	17	10A
N13 (Spare) J6 J7 K14 SSA Transmitter 1 Power J7 10A K15 (Spare) J6 10A/P16 K16 BAT 4 Heater Primary Power J9 3A/P15 K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A K20 BAT 2 Primary Heaters J9 3A/P15 K21 SDM +Wing Retrieval Heaters J9 5A/P15 K22 RSU 2 Survival Heater Ch. 3 J9 5A/P15 K23 RSU 2 Operational Heaters Ch. 3 J9 5A/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater Ch. 1 J9 5A/P15 K26 FHST 3 Power J7 3A K27 FHST 3 Primary Heater J7 3A K28 SSR 1,3 Redundant Heaters J9 3A/P15 K30 BAT 1 Primary Heater J8 3A/P15 K31 SSA Transmitter 1 and 2 Primary Heaters J8 SA </td <td></td> <td>(Spara)</td> <td>J7 10</td> <td>10A 2A</td>		(Spara)	J7 10	10A 2A
K14 SSA Transmitter 1 Power J7 10A K15 (Spare) J6 10A/P16 K16 BAT 4 Heater Primary Power J9 3A/P15 K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A K20 BAT 2 Primary Heaters J9 3A/P15 K21 SDM +Wing Retrieval Heaters J9 5A/P15 K21 SDM +Wing Retrieval Heaters Ch. 3 J9 5A/P15 K23 RSU 2 Operational Heaters Ch. 3 J9 5A/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater Ch. 1 J9 5A/P15 K26 FHST 3 Primary Heater J7 3A K27 FHST 3 Primary Heater J7 3A K28 SSR 1,3 Redundant Heaters J9 3A/P15 K30 BAT 1 Primary Heater J9 3A K32 SSA Transmitter 1 and 2 Primary Heaters J8 SA/P15 K31 SSA Transmitter 1 Overtemp Control		(Spare)	JO	3A 404
N15 (Spare) Jo 10A/P to K16 BAT 4 Heater Primary Power J9 3A/P15 K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A K20 BAT 2 Primary Heaters J9 3A/P15 K21 SDM +Wing Retrieval Heaters J7 3A K22 RSU 2 Survival Heater Ch. 3 J9 5A/P15 K23 RSU 2 Operational Heaters Ch. 3 J9 5A/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater Ch. 1 J9 5A/P15 K26 FHST 3 Power J7 3A K27 FHST 3 Primary Heater J7 3A K28 SSR 1,3 Redundant Heaters J9 3A/P15 K30 BAT 1 Primary Heater J9 3A/P15 K31 SSA Transmitter 1 and 2 Primary Heaters J8 SA K32 SSA Transmitter 1 and 2 Primary Heaters J8 SA K33 COSTAR RIU Redundant Power J	K14	SSA Transmiller i Power	J	
R16 BAT 4 Heater Primary Power J9 3A/P15 K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A K20 BAT 2 Primary Heaters J9 3A/P15 K21 SDM + Wing Retrieval Heaters J9 5A/P15 K21 SDM + Wing Retrieval Heaters Ch. 3 J9 5A/P15 K23 RSU 2 Survival Heater Ch. 3 J9 5A/P15 K23 RSU 2 Operational Heaters Ch. 3 J9 5A/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater Ch. 1 J9 5A/P15 K26 FHST 3 Primary Heater J7 3A K27 FHST 3 Power J7 3A K28 SSR 1,3 Redundant Heaters J9 3A/P15 K30 BAT 1 Primary Heater J9 3A/P15 K31 SSA Transmitter 1 overtemp Control J7 10A K33 COSTAR RIU Redundant Power J8 3A/P15 K34 (Spare) J9<	K15	(Spare)	JD	
K17 RWA 3 Primary Heater J8 3A K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A K20 BAT 2 Primary Heaters J9 3A/P15 K21 SDM +Wing Retrieval Heater Ch. 3 J9 5A/P15 K22 RSU 2 Survival Heater Ch. 3 J9 5A/P15 K23 RSU 2 Operational Heaters Ch. 3 J9 5A/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater Ch. 1 J9 5A/P15 K26 FHST 3 Power J7 3A K27 FHST 3 Primary Heater J7 3A K28 SSR 1,3 Redundant Heaters J9 3A/P15 K30 BAT 1 Primary Heater J9 3A/P15 K31 SSA Transmitter 1 and 2 Primary Heaters J8 SA K32 SSA Transmitter 1 Overtemp Control J7 10A K33 COSTAR RIU Redundant Power J8 3A/P15 K34 (Spare) J9 3A K35 RWA 3 Redundant Heater J9 <td>K16</td> <td>BAT 4 Heater Primary Power</td> <td>19</td> <td>3A/P15</td>	K16	BAT 4 Heater Primary Power	19	3A/P15
K18 (Spare) J6 10A/P16 K19 SADE 1 Power J6 10A K20 BAT 2 Primary Heaters J9 3A/P15 K21 SDM +Wing Retrieval Heaters J7 3A K22 RSU 2 Survival Heater Ch. 3 J9 5A/P15 K23 RSU 2 Operational Heaters Ch. 3 J9 5A/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater Ch. 1 J9 5A/P15 K26 FHST 3 Power J7 3A K27 FHST 3 Primary Heater J7 3A K28 SSR 1,3 Redundant Heaters J9 3A/P15 K30 BAT 1 Primary Heater J9 3A/P15 K30 BAT 1 Primary Heater J9 3A/P15 K31 SSA Transmitter 1 and 2 Primary Heaters J8 SA K33 COSTAR RIU Redundant Power J8 3A/P15 K34 (Spare) J9 3A K35 RWA 3 Redundant Heater J9 3A K36 Computer Redundant Heater J9 3A<	K1/	RWA 3 Primary Heater	J8	3A
K19SADE 1 PowerJ610AK20BAT 2 Primary HeatersJ9 $3A/P15$ K21SDM +Wing Retrieval HeatersJ7 $3A$ K22RSU 2 Survival Heater Ch. 3J9 $5A/P15$ K23RSU 2 Operational Heaters Ch. 3J9 $5A/P15$ K24MA Transmitter 2 PowerJ7 $3A$ K25RSU 3 Survival Heater Ch. 1J9 $5A/P15$ K26FHST 3 PowerJ7 $3A$ K27FHST 3 Primary HeaterJ7 $3A$ K28SSR 1,3 Redundant HeatersJ9 $3A/P15$ K30BAT 1 Primary PowerJ8 $3A/P15$ K31SSA Transmitter 1 and 2 Primary HeatersJ8 $5A$ K32SSA Transmitter 1 Overtemp ControlJ7 $10A$ K33COSTAR RIU Redundant PowerJ8 $3A/P15$ K34(Spare)J9 $3A$ K35RWA 3 Redundant HeaterJ9 $3A$ K36Computer Redundant HeatersJ9 $3A$ K37FHST 3 Redundant HeatersJ9 $3A$ K38WF/PC RIU Redundant PowerJ8 $3A/P15$ K38WF/PC RIU Redundant HeatersJ9 $3A$ K38WF/PC RIU Redundant HeatersJ9 $3A$ K40Battery 3 Pressure Transducer PowerJ6 $5A$ K41*DISABLE SDM Retrieval HeatersJ7 $3A$	K18	(Spare)	J6	10A/P16
K20 BAT 2 Primary Heaters J9 3A/P15 K21 SDM +Wing Retrieval Heaters J7 3A K22 RSU 2 Survival Heater Ch. 3 J9 5A/P15 K23 RSU 2 Operational Heaters Ch. 3 J9 5A/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater Ch. 1 J9 5A/P15 K26 FHST 3 Power J7 3A K27 FHST 3 Primary Heater J7 3A K28 SSR 1,3 Redundant Heaters J9 3A/P15 K30 BAT 1 Primary Heater J9 3A/P15 K31 SSA Transmitter 1 and 2 Primary Heaters J8 3A/P15 K31 SSA Transmitter 1 Overtemp Control J7 10A K33 COSTAR RIU Redundant Power J8 3A/P15 K34 (Spare) J9 3A K35 RWA 3 Redundant Heater J9 3A K36 Computer Redundant Heater J9 3A K35 RWA 3 Redundant Heater J9 3A K36 Computer Redundant Heater	K19	SADE 1 Power	J6	10A
K21 SDM +Wing Retrieval Heaters J7 3A K22 RSU 2 Survival Heater Ch. 3 J9 5A/P15 K23 RSU 2 Operational Heaters Ch. 3 J9 5A/P15 K24 MA Transmitter 2 Power J7 3A K25 RSU 3 Survival Heater Ch. 1 J9 5A/P15 K26 FHST 3 Power J7 3A K27 FHST 3 Primary Heater J7 3A K28 SSR 1,3 Redundant Heaters J9 3A/P15 K30 BAT 1 Primary Heater J9 3A/P15 K31 SSA Transmitter 1 and 2 Primary Heaters J8 SA/P15 K32 SSA Transmitter 1 Overtemp Control J7 10A K33 COSTAR RIU Redundant Power J8 3A/P15 K34 (Spare) J9 3A K35 RWA 3 Redundant Heater J9 3A K36 Computer Redundant Heaters J9 3A K36 Computer Redundant Heaters J9 3A K38 WF/PC RIU Redundant Heaters J8 SA/P15 K39 SADM/PDM Redun	K20	BAT 2 Primary Heaters	J9	3A/P15
K22RSU 2 Survival Heater Ch. 3J95A/P15K23RSU 2 Operational Heaters Ch. 3J95A/P15K24MA Transmitter 2 PowerJ73AK25RSU 3 Survival Heater Ch. 1J95A/P15K26FHST 3 PowerJ73AK27FHST 3 Primary HeaterJ73AK28SSR 1,3 Redundant HeatersJ93AK29NICMOS RIU Primary PowerJ83A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 and 2 Primary HeatersJ85AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeaterJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ8SAK40Battery 3 Pressure Transducer PowerJ6SAK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K21	SDM +Wing Retrieval Heaters	J7	3A
K23RSU 2 Operational Heaters Ch. 3J95A/P15K24MA Transmitter 2 PowerJ73AK25RSU 3 Survival Heater Ch. 1J95A/P15K26FHST 3 PowerJ73AK27FHST 3 Primary HeaterJ73AK28SSR 1,3 Redundant HeatersJ93AK29NICMOS RIU Primary PowerJ83A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 and 2 Primary HeatersJ85AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeatersJ93AK36Computer Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ93AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K22	RSU 2 Survival Heater Ch. 3	J9	5A/P15
K24MA Transmitter 2 PowerJ73AK25RSU 3 Survival Heater Ch. 1J95A/P15K26FHST 3 PowerJ73AK27FHST 3 Primary HeaterJ73AK28SSR 1,3 Redundant HeatersJ93AK29NICMOS RIU Primary PowerJ83A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 and 2 Primary HeatersJ85AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeatersJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K23	RSU 2 Operational Heaters Ch. 3	J9	5A/P15
K25RSU 3 Survival Heater Ch. 1J95A/P15K26FHST 3 PowerJ73AK27FHST 3 Primary HeaterJ73AK28SSR 1,3 Redundant HeatersJ93AK29NICMOS RIU Primary PowerJ83A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 and 2 Primary HeatersJ85AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K24	MA Transmitter 2 Power	J7	3A
K26FHST 3 PowerJ73AK27FHST 3 Primary HeaterJ73AK28SSR 1,3 Redundant HeatersJ93AK29NICMOS RIU Primary PowerJ83A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 and 2 Primary HeatersJ85AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K25	RSU 3 Survival Heater Ch. 1	J9	5A/P15
K27FHST 3 Primary HeaterJ73AK28SSR 1,3 Redundant HeatersJ93AK29NICMOS RIU Primary PowerJ83A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 and 2 Primary HeatersJ85AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K26	FHST 3 Power	J7	3A
K28SSR 1,3 Redundant HeatersJ93AK29NICMOS RIU Primary PowerJ83A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 and 2 Primary HeatersJ85AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K27	FHST 3 Primary Heater	J7	3A
K29NICMOS RIU Primary PowerJ83A/P15K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 and 2 Primary HeatersJ85AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant HeatersJ93AK39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K28	SSR 1.3 Redundant Heaters	J9	3A
K30BAT 1 Primary HeaterJ93A/P15K31SSA Transmitter 1 and 2 Primary HeatersJ85AK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K29	NICMOS RIU Primary Power	J8	3A/P15
K31SKA Transmitter 1 and 2 Primary HeatersJ8SAK32SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant HeatersJ93AK39SADM/PDM Redundant HeatersJ8SA/P15K39SADM/PDM Redundant HeatersJ8SAK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K30	BAT 1 Primary Heater	.19	3A/P15
K31SSA Transmitter 1 Overtemp ControlJ710AK33COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant HeatersJ93AK39SADM/PDM Redundant HeatersJ8SAK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K31	SSA Transmitter 1 and 2 Primary Heaters	.18	5A
K32COSTAR RIU Redundant PowerJ83A/P15K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant HeatersJ93AK39SADM/PDM Redundant HeatersJ8SAK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K32	SSA Transmitter 1 Overtemp Control	.17	10A
K34(Spare)J93AK35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant HeatersJ93AK39SADM/PDM Redundant HeatersJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K33	COSTAR RILL Redundant Power	18	3Δ/P15
K35RWA 3 Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K3/	(Spare)	10	30
K36Computer Redundant HeaterJ93AK36Computer Redundant HeaterJ93AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K34 K35	N/A 3 Pedundant Heater	10	37
K30Computer Redundant Heater393AK37FHST 3 Redundant HeatersJ93AK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K36	Computer Redundant Heater	10	34
K37FIRST 3 Redundant HeatersJ9SAK38WF/PC RIU Redundant PowerJ83A/P15K39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K30 K27	EUST 2 Dedundent Leaters	10	24
K38WF/PC RID Redundant PowerJ6SA/PTSK39SADM/PDM Redundant HeatersJ85AK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A		ME/DC DILL Dedundent Dewer	J9 10	3A 3A/D1E
K39SADiv/PDM Redundant HeatersJoSAK40Battery 3 Pressure Transducer PowerJ65AK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	N30	WF/PC RIU Redundant Hostoro		3A/P13
K40Battery 3 Pressure Transducer PowerJ6SAK41*DISABLE SDM Retrieval HeatersJ73AK42(Spare)J610A	K39	SADIM/PDM Redundant Heaters	JO	
K41 ^a DISABLE SDM Retrieval Heaters J7 3A K42 (Spare) J6 10A	K40	Battery 3 Pressure Transducer Power	JD	5A
K42 (Spare) J6 10A	K41 [°]	DISABLE SDIM Retrieval Heaters	J/	3A
	K42	(Spare)	J6	10A
K43 (Spare) J9 3A	K43	(Spare)	19	3A
K44 FHST 3 Shutter Power J9 3A	K44	FHST 3 Shutter Power	J9	3A
(H) Data Management Unit B Power J6 10A	(H)	Data Management Unit B Power	J6	10A
(H) DIU 1 Element B Power Primary J7 3A	(H)	DIU 1 Element B Power Primary	J7	3A
(H) DIU 2 Element B Power Redundant J7 3A	(H)	DIU 2 Element B Power Redundant	J7	3A
(H) DIU 4 Element B Power Redundant J7 3A	(H)	DIU 4 Element B Power Redundant	J7	3A
(H) DIU 5 Element B Power Primary J7 3A	(H)	DIU 5 Element B Power Primary	J7	3A
(H) MTE 2B Power Redundant J7 3A	(H)	MTE 2B Power Redundant	J7	3A
(H) MTE 4B Power Redundant J7 3A	(H)	MTE 4B Power Redundant	J7	3A
(H) RGA 2 Electronics Control Unit Ch. 3 J8 5A/P15	(H)	RGA 2 Electronics Control Unit Ch. 3	J8	5A/P15
(H) MCU A Control Power Supply Power Primary J8 3A	(H)	MCU A Control Power Supply Power Primary	J8	3A
(H) MCU A Link Power Supply J7 3A/P15	<u>(H)</u>	MCU A Link Power Supply	J7	3A/P15

Table 2.4-3. PDU circuit assignments (continued)

H = Hardwire. * = Nonlatching.

PDU 4 relay	Item controlled or function	PDU 4 conn.	CKT rating
K1	NICMOS Redundant Power	J4	20A/P16
K2	486 A Power	J4	20A/P17
K3	WF/PC Primary Power	J4	20A/P16
K4	COSTAR Primary Power	J5	20A/P16
K5	(Spare)	J8	20A
K6	RWA 4 Main Bus A or B Select	J10	30A/P17
K7	RWA 4 Main Bus C Select	J10	30A/P17
K8	FP/TCF Bus 4 Primary Power	.15	20A
KQ	EP/TCE Bus 1 Redundant Power	15	204
K10	EP/TCE Bus 7 Redundant Power	15	10A/P16
K10	Battery 6 Pressure Transducer Power	15	1041 10
K12	(Spare)	17	104
K12	(Spare)	18	30
K1/	SSA Transmitter 2 Power	17	104
K15	(Spare)	16	10A 10A/D16
K1G	(Spare)	10	20/D15
	NIN-4 Disable DMA 4 Drimony Hostor	10	3A/F 13 2A
	(Spore)	JO	3A 101/D16
	(Spare)	10	10A/P10
K19	SADE 2 Power	J6	
KZU	VIK-2 Disable	19	3A/P15
KZ1	SDIVI - Wing Retrieval Heaters	J	3A 5 A (D 4 5
K22	RSU 2 Survival Heater Ch. 4	19	5A/P15
K23	RSU 2 Operational Heater Ch. 4	<u>1</u> 9	5A/P15
K24	(Spare)	J7	3A
K25	RSU 1 Survival Heater Ch. 2	J9	5A/P15
K26	RMGA Power	J7	3A
K27	(Spare)	J7	3A
K28	ESTR 2, BAY 7, 8 Redundant Heaters	J9	3A
K29	NICMOS RIU Redundant Power	J8	3A/P15
K30	VIK-1 Disable	J9	3A/P15
K31	SSA Transmitter 1 and 2 Redundant Heaters	J8	5A
K32	SSA Transmitter 2 Overtemp Control	J7	10A
K33	COSTAR RIU Primary Power	J8	3A/P15
K34	(Spare)	J9	3A
K35	RWA 4 Redundant Heater	J9	3A
K36	Computer Primary Heater	J9	3A
K37	(Spare)	J9	3A
K38	WF/PC RIU Primary Power	J8	3A/P15
K39	Diode Box Redundant Heaters	J8	5A
K40	Battery 4 Pressure Transducer Power	J6	10A
K41*	DISABLE SDM Retrieval Heaters	J7	3A
K42	(Spare)	J6	10A
K43	(Spare)	J9	3A
K44	(Spare)	J9	3A
(H)	Data Management Unit B Power	J6	10A
λΗ Η	DIU 1 Flement B Power	J7	3A
(H)	DIU 2 Element B Power		3A
(H)	DILL 4 Element B Power	.17	34
(H)	DILLS Element B Power	.17	34
λήζ	MTF 2A Power Primary	.17	3A
λΗ	MTE 4A Power Primary	.17	34
λ H	RGA 2 Electronics Control Unit Ch A	18	54/P15
	MCLLB Control Power Supply Power Peduadant	18	30
ፚ	MCLER Link Power Supply	17	3Δ/D15
<u> (11)</u>		51	JAVE IJ

Table 2.4-3. PDU circuit assignments (concluded)

H = Hardwire. * = Nonlatching.

2.4.2 STS Interface Description

Regulated orbiter power and SSP control and feedback are provided to the telescope from the SIP. Reference the SSE CSM (*JSC-TBD*) for the specific orbiter-to-SSE interface.

2.4.2.1 HST Bus Configuration Control

While the telescope is berthed to the orbiter, the diode buses are connected to the main buses and essential buses. Orbiter power supplies the HST main buses during attached operations until the transfer to HST internal power just prior to umbilical disconnect. Bus control from the orbiter AFD is enabled by placing the ESS/MAIN SW ENABLE switch on the SSP to the ON position.

External Power

As shown in Drawing 2.4 sheets 3 and 4, the external (orbiter) power is supplied to the PCU from the *Interface Power Control Unit* (IPCU) via the FSS umbilical. The application of external power to the HST main and essential buses is performed by the orbiter crew from the AFD SSP. When *the correct* SSP switches *are thrown (refer to the Payload Operations Checklist)*, four transfer switches in the IPCU close to apply orbiter power directly to the HST main buses. This function can also be performed by an EVA crewmember from the ACP located in the +V2 trunnion bay on the HST. The four transfer switches are cross-strapped so that malfunction of any one of them will not cause loss of external power to the telescope.

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Three separate SSP talkbacks are provided for verification of external main bus status. When the four Interface Power Control Unit (IPCU) transfer switches close, they provide voltage feedback to SSP talkback DS20 (IPCU RELAYS CLOSED). The SSP talkback DS21 (MAIN BUS EXT PWR) provides direct indication that STS power has been applied to the HST PCU through the umbilical. Finally, the SSP talkback DS17 (MAIN BUS ON) provides direct indication that the telescope main bus is powered. This last talkback shows gray regardless of the source of the power (i.e., internal or external).

The external essential bus power is applied by placing SSP switch S13 in the ON position, also while S18 is ON (DS18-gray). This action commands closed the K6 and K7 relays of the IPCU, shown in Drawing 2.4 sheet 4. SSP talkback DS13 (ESS BUS EXT PWR) indicates the application of external power to the essential bus circuits of the PCU. The orbiter provides continuous power to the essential bus until the umbilical is demated.

Internal Power

Control of internal main power is provided to the orbiter crew by using the AFD SSP switch S17 while S18 is ON (DS18-gray), which operates PCU transfer switches S1 and S2 through IPCU pilot relays K10 and K11 (Drawing 2.4 sheet 3). The only direct verification is provided by telemetry from DIU-2 and consists of indications that the transfer switches have closed. The transfer switches can also be controlled by the STOCC using CDI special discrete commands. Internal main power ON also closes the diode bus interconnect switches and opens the S3 and S4 transfer switches for main bus C.

Control of internal essential bus power is provided using SSP switch S14 also while S18 is ON (DS18-gray). Acting through two IPCU pilot relays, K12 and K13, S14 operates three relays in the PCU (K7, K8, and K9) that connect/disconnect the diode buses to the HST essential bus (Drawing 2.4 sheet 4). Verification of internal essential bus status is provided by SSP talkback DS14 (ESS BUS INT PWR) and by telemetry monitors wired to DIU-2. No STOCC command capability exists for this function. If, for any reason, these PCU relays are not closed at umbilical disconnect, they are automatically pulsed closed by physical removal of external (orbiter) power, which causes a capacitive discharge to each of the three relay coils.

2.4.2.2 SSM Worklights

The SSM worklight circuit is a distribution network in the SSM AS that provides orbiter dc power internal lighting outlets for performance of EVA tasks. Also powered through this circuitry are the axial and radial SI indicator lights, as shown in Drawing 2.4 sheet 5. Orbiter power is applied to the SSM worklight circuit through nonlatching IPCU relays K18 and K19 on command from SSP switch S24.

2.4.3 EPS Operational Limitations

Essential Bus Interruption

For SM-1 and SM-2, because of safety and mission success concerns, the HST required continuous dc power from the orbiter primary payload bus to the HST external essential bus during HST berthed operations. Loss or interruption (i.e., bus voltage below 10 V for 3 ms or more) of the HST essential bus would have caused an automatic failover to internal essential bus power, as described in Section 2.4.2. This would have caused the HST transmitter to turn off and place the essential bus load on the HST batteries, which could have impacted the battery SOC. Should an interruption in service have occurred, the HST transmitters would have been powered back on and reconfigured from internal essential bus power, orbiter power would have been restored to the external essential bus as soon as possible, and internal essential bus power would have been disabled to minimize the effect on the HST batteries.

Battery State-of-Charge

The HST battery capacity is severely limited during those flight phases when the SAs are stowed. Although a dual failure involving both SA wings is unlikely, the malfunction of a single wing is possible. Figure 2.4-17 depicts the battery SOC as a function of time for the one-wing-deployed case. This chart assumes an HST load of 1545 watts and a battery EOL of 15 percent SOC. It also uses 2265 watts as the output of a single SA wing (BOL) and assumes optimum orientation in full Sun for the complete daylight portion of the orbit. Note that over a period of 10 orbits, spanning 16 hours, the single SA wing cannot meet the electrical power levels HST requires to be power positive. Therefore, an unscheduled EVA is necessary if either SA wing fails to fully deploy.

Battery SOC predictions for a planned servicing mission will depend on how long the batteries have been in service as well as their performance during the months leading up to servicing. The battery SOC will become critical within a short time, making retrieval and berthing operations also time-critical events. Once berthed to the FSS, the maintenance umbilical will be remotely connected from the AFD, placing the HST load on orbiter power. The SAs will be stowed after successful transfer of the HST to external power.



Figure 2.4-17. Battery state-of-charge

External/Internal Power Transfer

Servicing mission operating procedures have been specifically designed to provide a make-before-break in transferring the HST load between orbiter (external) and HST battery (internal) power. Specifically, during the redeployment scenario with the HST load initially connected to orbiter power, the flight crew will take the internal main bus power switch to ON and receive verification from the ground that the HST diode-bus-to-main-bus transfer switches have closed prior to removing orbiter power from the HST interface.

Note: No direct talkback to the flight crew is provided for verification of diode-bus-tomain-bus transfer switch status.

A break-before-make procedure would have an adverse effect on the configuration of many HST components, causing most to shut down and requiring reconfiguration by STOCC prior to proceeding with the redeployment operations.

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2.4.4 Power Management

This section provides a description of the battery capacity estimating method, battery reconditioning, and battery charge control by autonomous operation as well as by ground command. The method of onboard power management of the ST, including the use of flight software, is presented.

The normal operational capability of the HST EPS is managed by working with the following variables:

- a. Selecting CCC operation level for the V/T settings
- b. Switching SPAs on line and off line using software routines located in the flight 486 computer that applies the software step-to-trickle method of charge control

In the event of abnormal power consumptions, limited options are available in load reduction. Power usage may be curtailed by controlling the number of SIs operating at any one time. All other onboard hardware are maintained in a standby redundant state and are placed on line only when the selected primary unit either has failed or is performing marginally. Therefore, they cannot be used to help in the management of the EPS.

The major power change during normal mission operations is caused by the slewing of the spacecraft to position it for target viewing. During this period, additional power is consumed and the array wings must be reoriented to ensure that adequate power output is generated for charging the batteries as well as for supporting the load. One of the power consumption variables is the Thermal Control System (TCS). Because the TCS elements are resistive, they represent a load that varies as a function of bus voltage. However, it is anticipated that this effect will be compensated for and minimized by the regulated loads.

2.4.4.1 Estimating Battery Capacity

Each battery has two pressure monitors (one on DIU A and one on DIU B) that may be used for estimating the battery SOC. Each pressure monitor consists of a strain gauge attached to the dome of a cell whose output represents pressure vessel deflection that relates to internal cell pressure. The output of the strain gauge is routed to an electronics card where it is amplified and conditioned at 0 to 5 V dc (representing 0 - 1500 psi) for telemetering and downlinking to the ground station. The pressure values between 0 and 1 V dc (0 to 300 psi) are not used for estimating capacity because of the nonlinearity of the conditioning amplifier.

2.4.4.2 Battery Reconditioning

Reconditioning is accomplished by discharging the selected battery through the high discharge (5.1 ohm) resistor until the battery terminal voltage is approximately

<26.5 volts. The following battery parameters must be recorded for proper capacity evaluation and reconditioning:

- a. Voltage
- b. Current
- c. Pressure
- d. Temperature

The battery capacity is determined by integrating the battery current over the time of discharge to 26.5 volts (1.2 V/cell). A capacity versus pressure curve is generated by comparing the pressure against capacity for the entire discharge period, which is expected to be 16 to 17 hours for the reconditioning test (it actually took 17 to 18 hours). Since the discharge rate through the 5.1-ohm resistor is \approx 5 to 7 amps and the normal operation discharge rate, on orbit, during eclipse is \approx 10 to 13 amps, a correction factor must be applied to the acquired data to adjust for this difference. The adjustment is required since the obtainable capacity at the higher rate of discharge is less because of the higher internal losses in the battery during the discharge.

All six batteries should be reconditioned within the same timeframe in order to ensure proper system balance. Each battery is taken off line in sequence, discharged, and reconnected to the system as follows:

- 1. Send command to connect SPAs to the diode bus.
- 2. Send command to connect high discharge rate.
- 3. Monitor until battery reaches \approx 18-22 volts.
- 4. Remove battery from high discharge rate (removal of discharge while the battery voltage is above 18 but below 26.5 volts would be desirable, but below 22 volts is also acceptable).
- 5. Reconnect SPAs to battery only during eclipse. The high rate of charge of 21 to 23 amps causes no problem.
- 6. Allow battery to recharge until approximately 50 A-h is obtained before reconditioning the next battery.

Other constraints, requirements, and recommended guidelines are

- a. No restrictions or limitations to science mission operations.
- b. Only one battery will be in reconditioning at any time.
- c. As a guide, for each orbit during the discharge period, a forward link of at least 15 minutes shall be available for contingency purposes.

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Batteries should be reconditioned in the following order to minimize thermal effects: battery 1, battery 4, battery 2, battery 5, battery 3, battery 6. Figure 2.4-18 shows a typical battery reconditioning scenario.

2.4.4.3 Battery Recharge Ratio

Battery Recharge Ratio (R/R) is used for evaluating the battery efficiency during the charge (sunlight) and discharge (eclipse) periods. It is defined to be Ah into the battery divided by Ah out of the battery. Nominal R/R will be in the 1.01 to 1.12 range, indicative of a battery efficiency of 88 to 99 percent, *respectively*. Efficiency determines the amount of energy required to recharge the battery to full SOC following the eclipse discharge period. A ratio of less than 1.00 would be indicative of a possible battery rundown condition (more energy taken out during eclipse (discharge) than put back in during sunlight (charge)). Once a nominal R/R has been established for each battery, a trend analysis should follow to monitor the health of each battery for future performance predictions. Battery temperature and efficiency are inversely related. As battery temperature increases, battery efficiency decreases (Figure 2.4-19).

2.4.4.4 Battery Load Sharing

It is desirable to have each of the six NiH_2 batteries share equally during the discharge and the charge periods. This is an ideal condition and probably will not be achieved. However, small variations (± 2 percent) on battery sharing are not uncommon and are considered operationally acceptable.

2.4.4.5 Battery Charge Current

As far as the battery charging is concerned, the sunlit portion of the orbit is divided into a full charge period and a trickle charge period. During the full charge period, all 20 SPAs are connected to batteries. The amount of charge current for the batteries, however, would be the total SPA output minus the load current. The load always has the preference in satisfying its demand for the current because the batteries are the sources that provide the necessary voltage regulation for the system. The amount of charge current received by each battery is determined by a number of factors such as temperature, efficiency, and initial SOC. If a large disparity of these conditions exists between batteries, one or two may reach full SOC and activate trickle charge before the remaining ones have reached their charge cutoff level. This could result in some batteries not being fully charged. However, these same batteries give up less capacity during the discharge period, somewhat equalizing the system for one orbit. Therefore, it is imperative that a history of battery charging/discharging be recorded for trending analysis. Figure 2.4-20 shows a typical trickle-charging scenario.



Figure 2.4-18. Battery reconditioning/capacity test



Figure 2.4-19. NiH₂ battery charge efficiency vs. temperature





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2.4.4.6 Battery Temperature Control

Each battery contains a primary and a backup heater system. Each system consists of a heater, a sensor, and a controller to maintain battery temperature at 32° F (primary) and 26.6° F (backup). In addition, a thermostat is connected in series with the heater control to remove heater power if the temperature exceeds 50° F. Each cell contains a primary and a redundant heater strip between the cell sleeve and the cell wall. All primary heaters are in series and are controlled by a heater controller that can be enabled or disabled by command. The redundant heaters are also connected in series and have their own heater controller that maintains the batteries above 26.6° F. These heaters can also be enabled or disabled by command. The steady state heater power for each circuit is 40 ± 2 watts at 28 V dc, with a turn-on transient of approximately 2.0 amps.

2.4.4.7 Onboard 486 Computer Power Management

The onboard computer (486) is programmed to monitor the state of HST electrical power consumption and perform management of EPS hardware. EPS management includes control of normal operations and data calculations. In addition, certain functions are monitored for the existence of anomalous conditions. A limited number of corrective actions are programmed.

The software used for EPS power management and trend analysis is divided into two distinct parts:

- a. Onboard computations within the 486 for safemode determinations with some data transmitted real time via telemetry.
- b. Ground computations relying on real-time telemetry data through the TDRSS and delayed data from the onboard tape recorder.

The battery Charge Ampere-Hours (CAH) and Discharge Ampere-Hours (DAH) are computed at a 1-Hz rate for each of the six batteries. Battery Total Ampere-Hours (TAH) is computed at a 1-Hz rate as the total SOC of all batteries. Battery R/R is computed once each cycle (orbit) just prior to reinitialization of the aforementioned parameters. Current R/R is stored for telemetry, along with the vehicle time that the computation occurred, until the next R/R computation. CAH, DAH, and TAH are updated each second, and the current value is made available for telemetry. The rate of transmission of these parameters (four parameters for each of the six batteries) is dependent on the telemetry data rate (0.1 Hz for 4 kbps and 1.0 Hz for 32 kbps).

For all planned data link interruptions (i.e., TDRSS not available for Zone of Exclusion (ZOE) passage), the Engineering Tape Recorder (ETR) will record the battery computation parameters for later transmission to the ground. An exception to this is TDRSS handover (i.e., TDRSS east to TDRSS west), during which operation of the ETR is not planned and 30 seconds to 1 minute of data loss can be expected. All unplanned data link interruptions will result in lost data.

Autonomous onboard failure mode control in the EPS is exercised by the 486 software. If the battery SOC falls below 225 amp hours, the 486 causes the HST to be driven to the software Sun point mode attitude (+V3 or -V1 axis, whichever is closer, and solar array active surfaces toward the Sun). Further decline to 180 amp hours SOC results in the loss of "keep alives," and HST control is switched to the PSEA that establishes the hardware Sun point mode (+V3 or -V1 axis to the Sun). In the hardware Sun point mode (+V3 or -V1 axis to the Sun). In the hardware Sun point mode (+V3 or -V1 axis to the Sun). In the hardware Sun point mode, HST power can be reduced by removing noncritical electrical loads (depending on PSEA timer settings).

Other EPS configurations accomplished by the 486 are excessive load current and structure current autonomously isolating the buses and bringing bus C on line. A bus voltage below safemode limits will cause entry into hardware safemode and then autonomously close the SA trim relays. An SADE failure flag will autonomously result in a switchover to the redundant SADE.

Autonomous execution capability is required in view of the 10 to 15 percent of time that the STOCC is not accessible in real time and the possibility that STOCC may be unattended for periods up to 72 hours. Most of these actions constitute a temporary hold requiring diagnosis and appropriate circuit reconfiguration, or other corrective action as determined by STOCC when it returns on line.

The battery charge control directed by the 486 provides the capability to open circuit excess portions of the solar arrays when the batteries have reached a selected charge level as indicated by the KX1/KX2 relays opened. The solar array circuits are subsequently closed during eclipse as indicated by closure of KX1/KX2 relays.

2.4.4.8 Flight Software - Electrical Power System Processing

The processing software integrates the currents into and out of each battery to provide an estimate of the current SOC. The EPS processing is primarily divided into battery SOC calculations and battery trickle charge control.

2.5 DATA MANAGEMENT SUBSYSTEM

The HST DMS performs the acquisition, processing, storage, and distribution of all data between the Instrumentation and Communication (I&C) subsystem and the other subsystems on the spacecraft. The major DMS elements are a 486 computer, the DMU, four DIUs, one Engineering/Science Tape Recorder (ESTR), two Solid State Recorders (SSRs), and a redundant Oven-Controlled Crystal Oscillator (OCXO). The computer performs Pointing Control Subsystem (PCS) computations, real-time and stored command processing, TLM format control, and SA and HGA pointing. The DMU performs the central DMS functions, which include interfacing and interconnecting the following units:

- a. 486 flight computer
- b. SSA transmitters
- c. S-band MA transponders
- d. ESTR
- e. SSRs
- f. FHSTs
- g. Rate Gyro Assemblies (RGAs)
- h. OCXO
- i. DIUs three for SSM, one for OTA
- j. PSEA
- k. GSE
- I. PDUs of the EPS
- m. SIC&DH
- n. FGSs
- o. MCU
- p. SA Electronics Control Assembly (ECA)
- q. OTA

The DMS architecture has the DMU as the central controlling node or hub with all other units making up the outer nodes or spokes, as shown in Drawing 2.5 sheet 1. Paths in

2.5-1

the network pass through the DMU (i.e., there are no direct connections between the outer nodes except between the OCXO and the WF/PC). (This structure is different from a data bus network, such as that employed by the orbiter General Purpose Computers (GPCs), where there is a closed loop, and data placed on the bus are available to all units. Each element of a data bus network is required to recognize and read its data from the bus.)

All the external units that interface with the DMU have functional redundancy and most are cross-strapped. Data are transferred serially between the DMU and the external units, with the exception of the DMU/486 computer interface, which is a parallel data interface. Data rates and command rates between the DMU and an external device depend upon the unit and its function.



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5

2.5.1 DMS Hardware

2.5.1.1 486 Flight Computer

The 486 is a functional replacement for the DF-224/Coprocessor flight computer and is used for onboard command, control, and monitoring of the HST. The major functions of the flight computer are PCS computations, safe-mode routines, stored command processing and execution, TLM format control, SA and HGA pointing, and power system monitoring. The computer interfaces with the PDU for input power, the DMU for timing, control and data transfer, the DIU for bilevel telemetry of its own status, and the ICU for temperature output.

The 486 is configured as a single-string computer with multiple backup units for reliability. Reconfiguration is by ground command only, via the DMU CMD module. The basic configuration consists of one of three Central Processing Modules (CPMs) and one of three Power Converter Modules (PCMs). Up to three computer strings are possible; however, only one string is needed to meet mission functional requirements.

Physical Description

The 486 box, shown in Figure 2.5-1, measures approximately 18.8 by 18.0 by 13.0 inches and weighs approximately 69 pounds. Six ORU bolts and four shear pins are used to mount the 486 to the inner barrel of the SSM ES bay 1.

Electrical Power

The primary power source for the 486 is PDU-3, and the redundant power source is PDU-4. The PDUs provide fuse protection and switching of 486 power, controlled by special discrete commands from the CDIs in the DMU. Power consumption under a single string is approximately 35 watts.



Figure 2.5-1. The 486 flight computer

Thermal Control

The primary heat dissipation of the computer is through radiation to SSM bay 1; it is designed for passive cooling by providing low thermal resistance paths to the chassis. The operational temperature range is -6° to 25° C and the survival temperature range is -40° to 60° C. Redundant, thermostatically controlled heaters are mounted adjacent to the computer in SSM ES bay 1.

There are 13 connectors on the 486. The function for each connector is shown in Table 2.5-1.

Conn.	Function
J1	Main power A from PDU-3
J2	Main power B from PDU-4
J3	Bilevel telemetry directed to DIU-5 and ICU
J4	External Data Bus A carries 24-bit data input, 12-bit data output, and control signals to the DMU
J5	External Data Bus B carries 24-bit data input, 12-bit data output, and control signals to the DMU
J6	Configuration control from DMU
J7	External interrupts from DMU
J8	Voltage monitors
J9	1553 interface
J10	Heater power A from PDU-3, heater power B from PDU-4
J11	Fuse plug
J12	Auxiliary power out
J13	Heater power transfer

Table 2.5-1. The 486 connector descriptions

Functional Description

The computer consists of three CPMs and three PCMs, which are connected through a passive backplane (Figure 2.5-2).

Each CPM consists of an Intel 80486DX2-66 microprocessor, 2 Mbytes of RAM, two 512-kilobyte EEPROM banks, an interrupt controller, a watchdog timer, a DMU interface, and a 1553 interface. A CPM can be connected to one of two external data buses; however, only one CPM at a time should be connected to an external data bus to prevent bus contention or sending false signals to the DMU.

The PCMs provide undervoltage and overvoltage protection, temperature monitoring, and RAM backup power (from heater power) if main power to the 486 is lost. Each PCM is redundantly powered from the HST main bus and can power any of the CPMs. However, only one CPM at a time should be powered from a PCM to preclude overloading the PCM.



Figure 2.5-2. 486 block diagram

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There are six power configuration states for the 486, shown in Figure 2.5-3. The OFF state occurs when no power is being provided to the 486 box. The STBY state occurs when main and heater power are being provided to the 486, and all PCMs and CPMs are powered OFF. In the PCM ON state, power is applied to the power converters, and all secondary outputs are ON, except the CPM input, which is inhibited. Temperature measurement is functional in the PCM ON state. The CPM ON state occurs when the PCM enables the CPM input converter. The RAM Backup state occurs when main power is lost, and the RAM contents are maintained from heater power.



Figure 2.5-3. 486 power configuration states

2.5.1.2 Data Management Unit

The DMU provides the interface between the 486 computer and all other subsystems and HST modules. It combines the following DMS functions into a single unit:

- Error-check forward link messages
- Decode and distribute forward link messages
- Format and encode return link TLM

- Provide dedicated interfaces for selected units on the HST (i.e., FHSTs, RGAs, MA transponders, SSA transmitters)
- Provide interface circuits and logical operations as required to all units of the DMS
- Provide regulated power to the OCXO
- Distribute central vehicle timing source

Physical Description

The DMU consists of a single enclosure that houses the primary (A) and redundant (B) functional modules, dc/dc converters, and 38 connectors. The DMU is mounted on the door of the SSM ES bay 1; it weighs approximately 94.7 pounds and measures 25.5 by 29.2 by 7.9 inches. The unit is mounted using 20 noncaptive 0.25-inches hex head bolts with washers, torqued to 90 to 110 in-lb. Figure 2.5-4 shows the DMU installed on the SSM bay 1 door. Figure 2.5-5 depicts the unit construction and layout of the DMU modules. These functional modules are each composed of one or more 5- by 7-inch plug-in printed circuit boards. Each is fully redundant and connected through a backplane to each other and by wire-wrap to external connectors. The DMU is classified as a block II ORU.

Electrical Power

The two CDI dc/dc converters are totally enclosed and plug into the backplane of the DMU. These receive continuous power from the HST essential bus through redundant cabling. Both the CDI-A and CDI-B input power control relays are nominally in the CDI power-on state. However, in the case of a CDI malfunction, the failed CDI may be powered off via the remaining CDI by STOCC command.

The two DMU dc/dc converters that power the rest of the DMU are totally enclosed. The converters are mounted on the inside end wall of the unit for good heat transfer. Primary and redundant fused power is supplied to the A side of the DMU dc/dc converters from PDU-1 (PRI) and PDU-2 (RED) and to the B side from PDU-3 (PRI) and PDU-4 (RED). Selection of the primary or redundant source for each side is performed by STOCC via special CDI discrete commands.



Figure 2.5-4. DMU installed on SSM bay 1 door



Figure 2.5-5. DMU physical detail

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Command Data Interface Module

The two CDI modules accept 48-bit forward link messages from the MA transponder, as described in Section 2.5.4. The STOCC must send a sync word prior to the messages to initiate forward link message processing in the CDI. The messages are decoded to check for valid spacecraft address and polycode. Valid messages are processed as either CDI special commands or SIC&DH commands, or the messages are transferred to the DMU CMD module. Each CDI module is nominally powered continuously by the HST essential bus. The processing of forward link messages is fully described in Section 2.5.4.

Special commands are 16-bit serial digital commands or HLD commands that are directly executed by the CDI. The CDI provides 256 HLDs, of which 18 are used within the DMU, and the remainder used for control of other critical HST subsystems. CDI special serial digital interfaces are provided to the MA transponder, the SA DCE, and PSEA. The SIC&DH messages are 27-bit serial digital words. All other valid messages are forwarded to the CMD module for further processing.

Command Module

The CMD module receives forward link messages from the CDI, decodes each message "opcode," and acts on all valid messages. The types of messages processed by the CMD module are Configuration Control (C/C) commands, commands for Tape Recorder Interface (TRI) module control, RTCs, and header and data words (command data blocks) to the 486. When the 486 is processing in the normal mode, RTCs are executed under computer control. When the DMS is in bypass mode, the CMD module transfers RTC messages directly to the DIU Interface (DIUI) module.

Telemetry Format Control Module

The TLM Format Control (TFC) module sequences HST data into a serial digital TLM bit stream and generates major and minor frame sync codes. This module uses either fixed formats stored in Read-Only Memories (ROMs) or preprogrammed formats stored in the 486, as commanded from the STOCC operationally or by the PSEA in the case of a safe-mode condition. The PSEA is capable of selecting only the fixed ROM 4-kbps format. The TLM function is fully described in Section 2.5.5.

Communications Module

The Communications Module (COM) buffers and performs convolution encoding on the selected input. The data are placed on the selected output. COM receives input from the TFC module for normal and backup HST engineering TLM, from the Computer Interface (CIF) module for direct 486 memory dump, from the TRI module for ESTR and SSR playback, and from the SIC&DH for real-time science data and NSSC-I memory dump. Return-link processing is fully described in Section 2.5.5. The return-link data may be output to the MA transponder I and Q channels, the SSA transmitter, the SSRs, or the ESTR.

Computer Interface Module

The CIF interfaces with the 486 CPMs via the EDB by means of 8 address, 24 data, 1 data strobe, and 2 hardware load control signals. All data and commands are transferred between the computer and the DMU via this interface. Each CIF provides EDB output already connected to each of the three CPMs. Each CPM has two EDB ports that independently interface with the two CIFs (CIF-A and CIF-B) and logically OR the inputs following the receiver circuitry.

DIU Interface Module

The DIUI module controls and buffers communications between the DIUs and the DMU/486. The DIUI is capable of separately communicating with each element (A or B) of any DIU by means of three signal lines (a request line, a clock line, and a reply line). This module contains a request First-In-First-Out (FIFO) register that can store up to 32 DIU request messages (command or data requests) from the STOCC or the 486. DIUI also contains a data FIFO for receiving and storing up to 32 replies from the DIUs. Only one DIUI module may be powered at any one time, but the DIUs require no information as to which DIUI module is on.

Under nominal operating conditions, the DIUI accepts DIU commands and data requests from the 486 via the CIF. When the DMS is placed in the bypass (486 off-line) or diagnostic (486/DIU data request only) mode, forward link messages (RTCs) are routed to the DIUI module directly from the CMD module. Also, when the TFC module is processing a fixed format (backup TLM), the DIUI accepts DIU data requests directly from the TFC module.

FHST Interface Module

The FHST Interface (FHSTI) module provides a dedicated interface between the three FHSTs and the DMU. The FHSTs operate asynchronously with the DMU and can be controlled only when the 486 is on line (not in diagnostic or bypass mode).

RGA Interface Module

The RGA Interface (RGAI) module provides a dedicated interface between the three RSU/Electronics Control Unit (ECU) combinations and the DMU. Each RGAI contains six independent interface channels for the six gyros in the system (two per RSU). Each RGA channel outputs a pulse-width-modulated rate gyro signal and clock to the RGAI. The RGAI can be controlled only when the 486 is on line (not in diagnostic or bypass mode).

Tape Recorder Interface Module

The TRI module provides the data interface to the two SSRs and two tape recorders. Command and control of the SSRs and ESTR is accomplished via the DIUs, with the exception of the safe-mode tape recorder. Operation of the safe-mode ESTR is initiated

by a signal from the CMD module when it receives a safe-mode initiate signal from the PSEA. The TRI module accepts and routes data from the CIF module (computer dump), the TFC module (engineering data), and the COM (SIC&DH/science data). The TRI module does this per routing instructions that it receives from the CIF or CMD modules. This module also outputs SSR and ESTR playback data to the COM for transmission via the SSA transmitter to the STOCC.

Filter/Shaper Module

The Filter/Shaper (F/S) module receives a 6.144-MHz signal from the OCXO and performs filtering and shaping on the DIUI to provide a square wave signal to the Timing Interface Module (TIM). (F/S-A is connected to OCXO-A, and F/S-B is connected to OCXO-B.) The F/S module outputs of the A side and B side are provided to both TIMs.

Timing Interface Module

The TIM divides down the 6.144-MHz square wave output of the F/S module and distributes several different derived clocks to various HST subsystems. These are summarized in Table 2.5-2. Both F/S modules are cross-strapped to both TIMs, but only one TIM and one OCXO are nominally powered on at a time.

Table 2.5-2. DMS timing signals

DMU Timing Signals

TFC module:	1.024	MHz, 1 kHz sync
CMD module:	1.024	MHz, 256 kHz
TRI module:	2.048	MHz
COM module:	3.072	MHz, 1.024 MHz, 2.048 MHz, 8 kHz, kHz
RGAI module:	1 kHz	and 40 Hz sync
FHSTI module:	2.048	MHz, 1 kHz
DIUI module:	1.024	MHz, 2.048 MHz, 1 kHz sync
CIF module:	2.048	MHz
486 Timing Signal		
1 kHz sync, 40 Hz sync		
SSR and ESTR Timing	Signal	
2.048 MHz		
External Users		
OTA:	(5)	1.024 MHz (from DIUI)
	(3)	40 Hz sync
	(6)	Minor frame sync (from TFC)
	(1)	Major frame sync (from TFC)
WF/PC:	(1)	6.144 MHz (sine wave from OCXO)
SIC&DH:	(1)	1.024 MHz
	(1)	Master Timing Pulse (MTP) (from TFC)
PSEA:	(2)	Major frame sync (from TFC)
MCU:	(2)	Major frame sync (from TFC)
GSE:	(1)	Minor frame sync (from TFC)
	(1)	40 Hz sync (from TIM)
SPARE	(1)	Minor frame sync (from TFC)
SPARE	(1)	40 Hz sync
SPARE	(1)	1.024 MHz

2.5.1.3 Data Interface Unit

The HST DIU is used to provide a common command and data interface between the DMS and other telescope subsystems. The DIU receives instructions from the DMU in the form of serial messages that define the operation to be performed. The DIU performs the operation requested and returns data and status information to the DMU. Each DIU contains two complete elements, either of which is capable of handling all the required functions. The basic design provides for issuing HLD commands, Low-Level Discrete (LLD) commands, or serial digital command messages to the end items. The design also provides for accepting analog data, bilevel data, and serial data messages from the end items. (Figure 2.5-6 is a photo of the DIU.)

Physical Description

Four flight DIUs are used on the HST although the DMU is capable of servicing six flight DIUs. DIU-1 provides full capability (both HLDs and LLDs) for interfacing to the OTA subsystems and resides in bay B of the OTA ES. DIU-2, DIU-4, and DIU-5 have HLD capability only (no LLD service required) for servicing the SSM subsystems. These DIUs reside in SSM ES bays 3, 7, and 10, respectively, and all mount to the ES tunnel. All units are mounted through six noncaptive 0.25-inch hex-head bolts with washers, which are torqued to 90 - 110 in-lb. The DIUs are classified as block II ORUs.

Each DIU is housed in a box that measures 15.3 by 13.4 by 6.6 inches and weighs 34.5 pounds. The OTA DIU (DIU-1) contains 19 printed circuit cards, and the remaining units contain 17 cards since LLDs are not required. The command and data cards contain the logic for both the A and B elements. The logic associated with each element is independently tied to the appropriate dc/dc converter and control logic module. The same card has both A and B circuits, which allows cross-strapping on the card as needed. The cards are wire-wrapped to 20 external connectors.

Electrical Power

The A element of each DIU is provided with 28 V dc power from both PDU-1 (PRI) and PDU-2 (RED) whenever the HST main bus is powered. Similarly, the B element of each unit is provided with redundant power from PDU-3 (PRI) and PDU-4 (RED). Normal operating voltage range is 24 to 32 V dc, but the DIU is designed to survive power at 21 to 24 V dc or 32 to 35 V dc indefinitely. An element consumes about 4.5 watts.

Each DIU element has three states: off, standby, and active. STOCC commands (CDI special discretes) are used to apply or remove power from individual DIU elements. A power-on command brings the selected element into the standby mode and into a fully operational condition within 2 seconds. Receipt of clock pulses from the DMU DIUI causes the DIU element to switch to the active mode. The absence of three consecutive clock pulses causes it to revert to the standby mode.



Figure 2.5-6. DIU

2.5-18

Nominally, only one element of a DIU is powered at a time. However, the DIU hardware provides the circuit capability to have both elements on at one time. This operating mode can be used if partial failure occurs in both elements of any DIU. As long as the failures are different, full DIU capability may be achieved by simultaneously using a portion of both elements in the unit. However, DIU 1 has thermal restrictions associated with powering on both sides simultaneously.

Thermal Control

There are no provisions in the DIUs for heaters or other active thermal control devices. The unit is passively cooled by the transfer of heat into the box walls and covers, from which it can be radiated. Temperature sensors are provided on the SSM DIUs for ground monitoring.

Command and Data Handling Capabilities

A DIU element is capable of issuing 196 HLD commands (also referred to as power pulse commands) and supports 20 serial digital command channels. An HLD command is a +28 volt nominal, 22-ms pulse, while serial command messages are comprised of 16-bit data words. In addition, DIU-1 can issue 44 LLD commands in each element, which are 4-µs, logic-level pulses, to satisfy the unique command requirements of the OTA electronics.

The two elements in the DIU can also obtain data from subsystem interfaces and transmit the data to the DMU for access by the 486 and the TFC module. Each element of the DIU can acquire the following types of data:

- a. Analog data (8 bits, 0 to +5 volts)
- b. Bilevel data (8 bits, logic level)
- c. Serial digital messages (8, 16 bits per message)

The data inputs are independently wired (noncross-strapped), except for a portion of the bilevel and analog inputs that are brought in on one wire and are internally routed to both elements (cross-strapped). Table 2.5-3 lists the command and data channels for each type and shows which are cross-strapped in the DIU.

	Number of signals							
Signal description	Cross-strapped	Noncross-strapped						
	(A/B)	A	В					
Commands								
High-level power pulse discrete	144	48	48					
Low-level discrete	-	44	44					
Serial channels	-	20	20					
Data								
Analog channels	185	30	30					
Bilevel bits	120 (15 words)	44 (5.5 words)	44 (5.5 words)					
Serial channels	-	16	16					

Table 2.5-3. DIU cross-strapping

2.5.1.4 Engineering/Science Tape Recorder

One tape recorders is used in the DMS to store HST engineering data. One ESTR is assigned as the safe-mode recorder or used as a backup to the SSRs. The two SSRs, described in Section 2.5.1.5, supplement the ESTR by recording either 4- or 32-kbps science data. The safe-mode recorder is used to record fixed-format 4-kbps engineering data or 4-kbps programmable engineering data when software sunpoint is entered in the event the PSEA takes command of the vehicle. The unit records and plays back while running in either direction. It records in forward direction on track 1 and in reverse direction on track 2. The tape direction is opposite to the record mode direction in order to reproduce. Noncontact redundant end-of-tape and beginning-of-tape sensors are provided to prevent tape breakage. Table 2.5-4 provides the ESTR design features.

Table 2.5-4. ESTR design features

Tape type	Ampex 797, 1/4 inches wide, 1.0 mil thick
Tape length	2068 feet of oxide, 41 feet of leader
Tape tracks	Two, sequentially switched
Data packing density	25 kbpi
Recording code	Modified delay modulation
Capstan drive	Odetics three-capstan delta drive, three belts
Motor	Brushless dc motor with optical commutation
Tachometer	Optical 2884 lines
Reeling system	Coaxial, counter-rotating
Tape tensioning	Negator spring
Heads	Two record heads, two tracks each, alternating in erase and data recording function. Two single-track reproduce heads

Physical Description

The recorder is housed in a hermetically sealed case approximately 12.78 by 9.0 by 7.0 inches and is pressurized to 18 psia. A pressure transducer provides monitoring of the vessel integrity. Figure 2.5-7 shows the internal packing of the unit. The ESTR weighs approximately 21.7 pounds and is installed in the SSM ES on the tunnel of bay 8. The unit uses four ORU bolts to mount to the unit support brackets. The ESTR is a block II ORU.



Figure 2.5-7. ESTR internal packing

Electrical Power

Power is supplied from the SSM main bus through PDU 2 for ESTR 2. Power is available at the ESTR interface when the HST main bus is powered, but the unit is turned on by DIU HLD. The power drawn from the bus varies with the mode. The approximate average power drawn by the unit is as follows:

Mode	Avg PWR/worst case (watts)						
Record, low rate	15/17.5						
Record, high rate	20/26						
Reproduce	27/32						
Fast (FWD/RVS)	21/30						
Standby	3.5/4.5						

The ESTR power buses are fused at 10 amp, and power is provided to the recorder from main buses A, B, and C.

Thermal Control

Thin Kapton heater elements are installed near the tape recorder and are controlled by two series thermostats to maintain temperatures above -12° C. The heater circuits are enabled by PDU relays and commanded by DIU HLD. Two temperature monitors are installed in the recorder to monitor tape temperature. Tape temperature is limited to a maximum of 45° C. One sensor is conditioned by the recorder and the other by the Instrumentation Control Unit (ICU) so that tape temperature can be monitored while the recorder is off. Two additional monitors are installed on the mounting structure of the ESTR. ESTR-2 installed in SSM ES bay 8 is shown in Figure 2.5-8.



sfoc_fl2705_249.tif

Figure 2.5-8. ESTR-2 installed in SSM ES bay 8

2.5.1.5 Solid State Recorder

The SSR is a GSFC-developed functional replacement for the ESTRs. During SM-2, ESTR-1 was replaced with SSR-1, and ESTR-3 was replaced with SSR-3 during SM 3A. Figure 2.5-9 is a photo of SSR-1 and SSR-3 in bay 5. Figure 2.5-10 shows the current SSR-1/SSR-3 configuration.



sfoc_fl2705_053.tif

Figure 2.5-9. SSRs 1 and 3 installed in SSM ES bay 5

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Figure 2.5-10. SSR-1/SSR-3 configuration

The SSR has several design improvements over the ESTR. The storage capacity has been substantially increased from 1.2 Gb to 11.5 Gb. Because of the solid state design, the recording/playback can easily start anywhere in storage in the SSR, as opposed to requiring rewinds/fast forwards for the ESTR. The solid state design also eliminates the head to tape mechanical interface, which should improve reliability. The SSR has the capability to simultaneously playback and record, while the ESTR can only do one at a time. The science and engineering data can be partitioned on the SSR instead of having mixed data on the ESTR.

Physical Description

The SSR is housed in a box that measures 12.0 by 9.0 by 7.0 inches and weighs approximately 26 pounds. The connectors J1, J2, and J3 are identical to the ESTR, and the SSR has an additional connector J4. This connector provides simultaneous record and playback capability when connected to another ESTR or SSR. SSR-1 and SSR-3 are installed in the SSM ES bay 5, each using four ORU bolts to mount to the unit support brackets.

Electrical Power

Power is supplied to SSR-1 and SSR-3 from the SSM main bus through PDU-1 and PDU-3, respectively. Two redundant relay-selected power supplies feed each memory array group. The power supply select relay receives 28 V input power from a single bus, and the input to each power supply is fused at 2 amps. The SSR power draw

during continuous record and playback mode is 25 watts and the power draw during standby mode is 20 watts.

Functional Description

The data is stored in 234 byte Reed Solomon encoded Error Detection and Correction (EDAC) blocks, which consist of 224 user bytes and 10 check bytes. The RS encoding corrects up to 5 bytes per block due to a failed memory chip or Single Event Upset (SEU). The memory pointers increment in steps of 28k logical blocks, which are made up of 120 EDAC blocks. For each Storage Area (SA), the SSR software provides three pointers to keep track of the data that have been recorded and the data that is ready to be played back. These three pointers are the Record Pointer, Playback Pointer, and Playback Stop Pointer. The Record Pointer indicates the current block to which the data is being recorded or was last recorded and points to the beginning of the block. The Playback Pointer indicates the current block from which the data are being played back or was last played back and points to the end of the block. The Playback Stop Pointer indicates the block where the last record stop command was executed, denotes record "sessions" or complete data sets, and points to the beginning of the block. The Retransmit Pointer is a generic pointer used to play back any block range and points to the beginning of the block. This pointer does not affect the record or playback pointers, and it is not affected by the constraints on the other pointers.

The SA consists of 16 Megabit DRAM die (chips) arranged in successive layers, as shown in Figure 2.5-11. The capability exists to disable specific memory areas, which allows a graceful degradation of the SSR when memory die fail.

Recording operations occur at 4, 32, and 1024 kbps, just as in ESTR operations. Because of the solid state design of the SSR, there is no tape motion telemetry associated with it. However, the record pointer does update. All playbacks occur at 1024 kbps. SPCs under 486 control initiate records as in the ESTR operations either via a direct H/W serial digital command or an SSR management command.



(b) SSR memory array group



2.5.1.6 Oven-Controlled Crystal Oscillator

The master oscillator consists of redundant OCXOs that provide a central time source for the HST. Only one of the two OCXOs may be powered on at a time. Each OCXO is capable of providing multiple, highly stable 6.144-MHz sine wave outputs. These stable frequencies are used within the DMU to generate HST vehicle time and to generate the timing signals required for the DMS, OTA, and SIC&DH. The OCXO also provides an output directly to the WF/PC. The outputs are interfaced to the radial SI and DMU F/S module using coaxial cables. The F/S module outputs are cross-strapped to the TIM A and B.

Physical Description

The oscillator (Figure 2.5-12) is enclosed in a cylindrical housing that is 4 by 9 inches. The oscillator has two coaxial output connectors and one power/instrumentation connector mounted on one end. Both OCXOs are mounted in SSM ES bay 2 on the tunnel wall. Each OCXO weighs 2.8 pounds.

Electrical Power

Only one OCXO is on at a time, although the OCXO meets most of the specified parameters after a warmup of only 30 seconds. The OCXO takes approximately 24 hours after power is applied to reach frequency accuracy and long-term stability specifications.

An OCXO is selected by applying power to it and its associated bandpass filter. The OCXOs and filters derive their power from the DMU dc/dc converters. Whenever the DMU is powered on, one of the OCXOs is also on. The selection of the OCXO and F/S (A or B) is controlled by special HLD commands from the CDI and modules. These HLD commands are wired internal to the DMU and are not brought to external connectors.



Figure 2.5-12. Oven-controlled crystal oscillator

2.5.2 DMS Flight Software

The HST flight software consists of two executable images that are loaded concurrently but are mutually exclusive in the functions they perform. These are the Vehicle Support Software (VSS) module and the Safe-Mode Utility (SMU) module. The SMU receives processing control when the CPU first begins processing (CPU RESET OFF), and transitions control to the VSS on command from the STOCC. This is performed only after the VSS variable database has been reloaded via uplink and verified. The functions contained in each module are described below.

Vehicle Support Software

The VSS module of the DMS flight software performs the following functions:

- a. Executive processing to control the initiation and frequency of all other processes, according to the 40 Hz interrupt, and to provide the needed hardware/software data exchanges, driven by the kHz interrupt.
- b. TLM including 0.5k, 4k, and 32k formats, as well as a memory dump.
- c. Command handling including stored commands and RTCs for either software or hardware.
- d. SIC&DH interface to control the Processor Interface Table (PIT) interface.
- e. Gyro signal processing using gyro data to support control law processing.
- f. Control law processing to develop an error acceleration control system vector based on position, rate, integral, and observer components.
- g. Torque computation to develop and combine the command torque (based on feed forward and error accelerations), the disturbance or kinematic torque, the Gravity Gradient (GG) torque, and the MT.
- h. Steering law processing to allocate the torque sum to the reaction wheels (three or four) and add distribution law torque components. The distribution law adds torque increments in null space to the commands in order to minimize maximum wheel speeds.
- i. Observer source control to activate and deactivate FGSs and FHSTs as attitude observer input sources to the control law.
- j. FGS to vehicle transform evaluations to support observer signal processing.
- k. FGS attitude update determinations to process fine-lock or coarse data from one or two FGSs to support observer updates.
- I. Acquisition processing to provide processing elements in support of FGS fine-lock acquisition.

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- m. FHST attitude update processes data from one or two FHSTs to provide an attitude error determination to the control law observer.
- n. Vehicle command generator processing to develop driving functions for vehicle maneuvers. The driving functions are the incremental attitude change quaternion and a feed-forward angular acceleration vector. The maneuvers include simple eigenvector slews as well as continuous (serpentine) and dwell scans. Peak-up maneuvers to time-correlated scanning pointings are provided. Step maneuvers superimposed on the constant rate intervals of primary maneuvers are also provided. Other maneuvers are developed to null the offsets encountered by SIs. All these vehicle maneuvers are developed according to smoothed or shaped jerk profiles in order to avoid vehicle excitation.
- FGS command generator processing to slew the star selector angles from their starting positions to commanded destinations using the smoothing techniques of the vehicle command generator or to generate the star selector command values that would hold the star fixed in the FGS Field of View (FOV).
- p. FGS control law processing to close the star selector servoloop on the FGSs. Position, rate, and integral control paths are employed. The command rate is based on successive differences of command positions.
- q. Velocity aberration and parallax processing to remove the drift that guide stars or target stars would otherwise experience because of these effects. Corrections are developed for the vehicle as an additional command frame increment. The FGSs must then be compensated for this vehicle motion as well as their own unique corrections for the guide stars. In a contrary example, an astrometry FGS may require corrections for a target while the vehicle remains inertially held.
- r. HGA processing to command antenna gimbal angles through shaped slews from their prevailing values to destination values. The shaping profile is ground-command selectable in order to sustain the pointing accuracy required over the duration of the slew. The command generator develops position commands at 1 Hz and command acceleration at 10 Hz. The processing closes the gimbal servoloop at a 1-Hz rate.
- s. Minimum-energy law momentum management processing to prevent excess reaction wheel angular momentum by computing appropriate commands to the magnetic torquers. The constant component of the external GG torque if left uncompensated would otherwise drive the reaction wheels to saturation. The MT commands are based on an evaluation of the nominal system momentum expected to prevail over a desaturation interval. Maneuvers will be identified to momentum management processing which are adequately representative of actual vehicle maneuvers whenever vehicle attitude changes are large enough to impact the nominal momentum computations.

- t. Cross-product law momentum management to provide MT commands during certain degraded modes of operation under software control. In this case, the magnetic field is obtained from the Magnetic Sensor System (MSS).
- u. SA commanding, based on uplinked command packets, to develop commands to control the SA orientations with respect to the vehicle.
- v. Sun-pointing processing to provide graceful degradation to Sun-pointing attitudes given certain anomalous conditions. The Sun-pointing control law orients the vehicle +V3 or -V1 axis toward the Sun, using Sun-sensor and RGA data.
- w. Safe-mode processing to incorporate all the vehicle protective measures, as well as the final reversion to backup hardware control, if required.

Safe-Mode Utility Software

The SMU module of the DMS flight software will execute at 486 initialization or if VSS processing is no longer possible. SMU processing performs certain safe-mode recovery functions including reporting the last VSS state, 500 bps memory dump, diagnostic TLM format, and software block load capability.

2.5.3 Description of DMS Modes

Hardware Load Mode

The Hardware Load (HWL) mode allows the ground to load a block of 1 to 255 words directly into computer memory by direct access without the computer software running. During an HWL, the computer CPU is held in reset, which prohibits the software from running. In CPU reset, the computer cannot output the Keep-Alive (KA) words to the PSEA. The HWL mode is activated or deactivated only by C/C commands from the STOCC (via the DMU CMD module).

Normal Command Mode

This mode refers to DMS command processing during HST on-orbit operation. The DMU, DIUs, OCXO, and 486 computer are in operation. Forward link messages are decoded in the DMU to determine the type of command. Special commands and SIC&DH messages can be output by the CDI module in all DMS modes. Configuration control, TRI control, and TFC/COM control commands can be executed by the CMD module in all modes except DMS poweroff mode. In the normal command mode, RTC, headers, and data Stored Program Command (SPC) words are sent to the computer. Hardware RTCs and SPCs are DIU commands that are routed through the 486 and output via the DMU to the DIUs for execution. The computer also issues commands to the TRI, TFC, COM, and DIUI modules. The normal command mode is activated or deactivated by transmitting a small block of data words to the computer to enable or disable the command software.

Bypass Mode

This mode bypasses the computer and routes hardware RTCs (DIU commands) directly from the CMD module to the DIUI module for transmission to, and execution by, the DIUs. Software RTCs, headers, and data words are still input to the computer. In addition, the DMU inhibits execution of computer-issued DIU, FHST, RGA, and DMU module control commands. Special, SIC&DH, C/C, TRI control, and TFC/COM control commands can be executed directly in the DMU. The 486 may obtain data from the DIUs and DMU. The 486 may control a programmable TLM format. The bypass mode is initiated either by the safe-mode initiate command from the PSEA or by STOCC command. Bypass can be disabled only by STOCC C/C command.

Diagnostic Mode

This mode is intended to be used to perform fault isolation while in safe mode, so that the HST is protected from undesired DIU commands. The diagnostic mode is initiated and disabled by STOCC C/C command only. In this mode, software RTCs, header, and data words are routed to the computer, while hardware RTCs (to the DIU) are routed directly to the DIUI. All DIU commands are output by the DIUI with insufficient clocking signals for command execution. Therefore, no DIU commands can be executed in the diagnostic mode. The DMU also inhibits computer-issued FHST, RGA, and DMU module control commands. The computer may obtain data from the DIUs and DMU and may control a programmable TLM format. The DMU can execute special, SIC&DH, C/C, TRI control, and TFC/COM control commands.

Programmable (Normal) Telemetry Mode

This mode refers to any programmable TLM format that is run under computer control (except for direct memory dump). In this mode, TLM can be output in one of several formats: 500-bps, basic 4-kbps, or 32-kbps diagnostic formats. The DMU also has the capability to perform full frame and multiplexed memory dump in normal TLM mode in place of these TLM formats. The computer issues major frame, minor frame, and SIC&DH sync enables. The SSRs and ESTR are available for recording or reproducing ED or SD. This mode is initiated or disabled by software RTCs or SPCs.

Backup Telemetry Mode

In this mode, TLM is output in a single, fixed format at either 4 kbps or 500 bps and is controlled by PROMs in the TFC module. Backup TLM cannot be operated simultaneously with the normal TLM mode, because the DMU is designed to handle only one engineering TLM stream (except for direct memory dump). In backup TLM mode, computer-issued major frame, minor frame, and SIC&DH sync enables are inhibited in the DMU, and sync signal enables are generated directly by the TFC module. The 4-kbps fixed format is activated by the safe-moding initiate command from the PSEA, by computer-issued command, or the TFC/COM control command from the STOCC. The 500-bps fixed format is activated only by the TFC/COM control command or by the TFC/COM control command or by the TFC/COM control command.

Direct Memory Dump Mode

This mode bypasses the DIUI and TFC modules and dumps data from the 486 directly to the COM module for output via the CIF. This allows simultaneous operation of direct memory dump and either programmable or backup TLM modes. Memory dump is limited to a 4-kbps rate in this mode. The direct memory dump mode is initiated and disabled by computer-issued commands.

TRI/ESTR Safe Mode

This mode is activated by entry into safe mode. The SAFEMODE INITIATE command from the PSEA causes the ESTR designated as the safe-mode recorder to be turned on and the other two recorders to be turned off. The TRI module routes the 4-kbps fixed format TLM to the safe-mode recorder. This mode is deactivated by a SAFEMODE OFF HLD command from the CDI module or PSEA to the safe-mode recorder and a configuration control command to remove the DMU SAFEMODE from active interrupt.

DMS Power-On Mode

This mode refers to the initial state of the DMS units when they are turned on. All units are off, except the operating CDI modules in the DMU, which are powered by the HST essential bus. The DIUs are placed in standby mode when they are powered on. A complete set of functional modules and an OCXO are turned on when power is applied to the DMU. Bypass mode and 500 bps backup TLM are activated in the DMU, and the SSRs and ESTR are in standby. The 486 must be configured by ground command after powerup.

DMS Power-Off Mode

This is a nonoperating mode where all DMS units, except the operating CDI modules in the DMU, are off. This mode provides only processing of forward link messages from the ground and outputting special commands and SIC&DH messages. This capability is present in all DMS modes. The DMS is placed in this mode by the PSEA after the PSEA has been in safe mode for 2 hours or if the PSEA is in contingency Gravity Gradient Mode (GGM). The DMS is also in this nonoperating mode during the initial 4 hours of the deployment mission.

2.5.4 Forward Link Commanding

Forward link command messages are received by two NASA near-Earth MA transponders, each of which contains a Command Detector Unit (CDU) that performs bit synchronization and digital regeneration of the ground command signals. These signals are provided to the CDI, a functional module of the DMS physically located in the DMU. The CDI extracts and validates forward link message, outputs these messages to the DMU CMD module, or SIC&DH, or decodes these messages to output special commands. The CMD module decodes the operations code of the message it receives and executes C/C commands, transfers messages to other DMU modules, or routes RTC, header and data messages to the computer or DIUI (in bypass mode) for execution.

The CMD module is described in 2.5.1.2. An overview of the HST forward link is provided in Figure 2.5-13.



Figure 2.5-13. STS/HST forward link configuration

Command Data Interface

The two CDI modules accept 48-bit forward link messages from the MA transponder in the format shown in Drawing 2.5, sheet 2. To initiate forward link message processing in the CDI, the ground must send a sync word prior to the messages. Each message is decoded to check for valid spacecraft address and Hamming code. Valid messages are processed as either special commands, SIC&DH messages, or transferred to the CMD module. Special commands are 16-bit serial digital commands or 8-bit HLD commands that are directly executed by the CDI. SIC&DH messages are 27-bit serial digital words. All other valid messages are transferred to the CMD module.

Command Module

The CMD module receives valid messages from the CDI and decodes each message operations code to determine the type of message. The message types are C/C commands, control commands for the TRI, TFC, and COM modules; RTCs; and header and data words to the computer. The C/C commands are executed by the CMD module to configure the computer and select DMU modules and modes. The TRI and TFC/COM control commands allow the ground to set up these modules when the computer software is not running. The RTC, header, and data words are passed on from the CMD module to the CIF module for input to the computer in the normal mode of operation.

The CMD module also controls the HWL, computer bypass, and diagnostic modes. The HWL capability allows the ground to directly load a block of data (up to 255 words) into any portion of computer memory. The DMU bypass and diagnostic mode routes DIU RTCs directly to the DIUI module for execution, bypassing the computer. In the DMU diagnostic mode, DIU SPCs are input to the computer as in the normal mode and all DIU commands are output by the computer, but these commands are transferred to the DIUs with insufficient clock pulses for command execution. DIU fault indications are provided to the software for diagnostic purposes.

The CMD module accepts a safe-mode initiate signal from the PSEA. This signal causes entry into DMU safe-mode. The CMD module transfers this signal to other DMU modules and uses it to go to the CMD-bypass mode and output a safe-mode active interrupt to the computer.

Drawing 2.5 sheet 2 shows a figure of the CMD module interfaces with other DMU modules and external units. For clarity, module cross-strapping and status signal interfaces used for TLM mode are omitted. The primary inputs to the CMD are the 33-bit data messages from the CDI and the safe-mode initiate signal indicating entry into safe-mode from the PSEA (right side of CMD module figure). The primary CMD output is the 24-bit parallel data to the computer via the CIF module (left side of CMD module figure). Also shown are the interfaces to the computer for HWL and C/C. The CMD outputs to the DIUI are used to transfer DIU RTCs in the bypass mode and load the DIUI C and D registers (upper left of CMD module figure). The CMD interfaces to the TFC/COM and TRI modules are used to transfer control commands from the

ground or safe-mode from PSEA. The TIM module provides 1.024-MHz and 25-kHz timing signals to the CMD. The CMD outputs control signals to the various DMU modules to inhibit response to computer commands in bypass and diagnostic modes or to initiate module safe-mode.

The CMD module is redundant, and all signals to and from other modules are crossstrapped to both CMDs. Only one CMD module is on at a time. Two special HLD commands from the CDI are used to select the CMD modules: One selects module A and turns off module B, the other selects module B and turns off module A. With the exceptions of the OCXO, TIM, and CMD, which are selected by special HLD commands, the remaining DMU modules are selected by a DMS C/C command (Drawing 2.5 sheet 2). The CMD module power-on state is as follows:

- a. HWL off
- b. Bypass mode on
- c. DMU status register 1 reset by fixed-format TLM mode ROM
- d. Safe-mode active interrupt off
- e. DIUI clock is inhibited if a bad echo is detected for a DIU command (i.e., echo checks is enabled)
- f. Diagnostic mode off
- g. CPU resets off



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				GLOSSARY						DATE: 01/14/	08 NAME: L	АМ	VERSION: REL	3	
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				AGC ALU	· /	UTOMATIC GA MPLIFICATIO RITHMATIC L	IN CONTROL. IN OVER GODB OGIC UNIT. I	PROVIDES AD DYNAMIC RAN PERFORMS ALL	JUSTABLE IGE WITH 10 . ARITHMAT	D BIT RESOLUTIONS)N			8	
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				BRF CDU	: E	AND REJECT	FILTER CTOR UNIT								
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)				DATA ROUT DBI	• •	ATA ROUTING	ERFACE. CON	S THE ADC TO TROLS READ/W	THE DATA	BUS RUCTIONS TO TH	:				
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				OUT BUFF PLL	• C • F	NUTPUT BUFFE	R. INTERFAC	ES COMMAND D	ETECTOR I	NPUT AND OUTPU	T SIGNAL				
				PRE SELECT	1 F	RESELECTION	ANTENNA POR	SO, ATTENUAT T	ES LOCAL (DSC SIGNAL	DOCRAM				
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4				RF SW	• F	ADIO FREQUE	NCY SWITCH							6	
0				ROM	• F	EAD ONLY ME	MORY				IME				
				S-BAND MOD	• •	CCEPTS RNG	AND/OR TM FI	ROM TH COND	PHASE MODI	ULATES 30 F1					
				TM COND	• 1	ELEMETRY/SI ROVIDED COM	GNAL CONDIT IMANDS	IONER. CONTA	INS RELAYS	S FOR OPTIONAL	USER				
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	\$		8	THIS DETECTO	DR CO	NTAINS FOUR	SEPARATE PI	HASE DETECTO	IRS WHICH F	FUNCTION AS TH	:				
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SM GPC				C) TRA	ANSFE J (OF	CODES 1001	ARE COMMAND: 00, 100001,	S (RTC'S AND OR 100111 A	ND 100011	AT APPROPRIATE	TIME) TO THE				
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SHEET 2 OF 3

2.5.5 Telemetry Processing

Fourteen basic TLM formats are being provided to obtain HST information. These are listed below, along with a brief description of when/why each format is used.

- AN 4 kbps basic programmable (normal TLM) The original HST operational telemetry format, which contains the majority of the telemetry points available from HST.
- 2. FN 32 kbps programmable Used to provide high update rate telemetry on the FGSs at the expense of other telemetry.
- 3. XN 32 kbps programmable Used to provide diagnostic readout of FGS1, not generally used operationally.
- 4. YN 32 kbps programmable Used to provide diagnostic readout of FGS2, not generally used operationally.
- 5. ZN 32 kbps programmable Used to provide diagnostic readout of FGS3, not generally used operationally.
- 6. PN 32 kbps programmable Used to provide high update rate telemetry on the PCS telemetry at the expense of other telemetry.
- HN 32 kbps programmable The main operational telemetry format in use. Used over 95 percent of the time. Combines the high update rates for FGS and PCS telemetry while providing the majority of the other telemetry points as well.
- 8. TN 32 kbps programmable Originally designed to be used for testing purposes, but later reworked to support SM activities. It is currently the main telemetry format use during the SM and contains all of the mechanism telemetry to support the SM.
- 9. D 4 kbps fixed The telemetry format that is autonomously activated upon entry in hardware safe-mode. The data requests are nonchangeable (fixed) in HW.
- 10. E 500 bps fixed The same data requests as D format, but slower. This is the format that is active upon HST powerup if a power interruption occurs.
- 11. C 4 kbps programmable The telemetry format that was used on deploy. At that time it was not believed that TDRSS could support 32-kbps telemetry from HST. This format has not been used since HST was deployed.
- S 500 bps programmable Used to read out the flight computer memory as a telemetry format. This format was used only during STS-61 in support of the CoProcessor changeout; it is not likely to be used again.
- 13. M 4 kbps Flight Computer memory dump format.
- 14. U 500 bps Same as M format, but at a slower rate.

2.5-39

Format refers to the organization or structure of the TLM pattern in terms of minor frame length and the number and sizes of the subframes. The fixed formats are PROM formats, and neither their structure nor data content can be changed without hardware modification (except for slots held open for computer inputs). The formats are identical and only differ in their bit rate.

The programmable formats are unique in that the data content can be changed in flight by ground commands. For example, the location of measurements can be changed from one subframe to another, or measurements can be removed from the format entirely and others added. The nature of the programmable formats allows some or all words in the format to be the contents of computer memory. The only method of performing a memory dump is called direct memory dump and is described in Section 2.5.3.

Drawing 2.5 sheet 3 provides an overview of the HST telemetry processing flow for engineering data and science data. When the computer software is controlling a programmable TLM format (normal TLM), the software provides TLM bit-rate control and major and minor frame-synchronizing pulse enables; the software also controls the buffering of DIU and computer data to the TFC module in the DMU. (TLM data refers to engineering data unless science data is specified.)

The TFC module formats the TLM data at the proper bit rate and outputs the data and necessary clocks to the TRI module for recording and to the COM for transmission to the ground. The TFC module can support only one engineering TLM format at a time. Sync-pulse enables received from the computer are used to generate the major and minor frame syncs on the next 1-kHz sync pulse from the TIM. The major and minor frame syncs are output to various users as listed in Table 2.5-2. The TFC also issues the MTP to the SIC&DH in the same manner as the frame syncs. The TFC module provides the status of the various DMS functions in four 24-bit DMU status registers (DMU status registers 2, 3, 4, and 5 refer to Drawing 2.5 sheet 3). The TFC module provides the control for the 500-bps and 4-kbps fixed TLM formats (backup TLM). Ground control selects the desired TLM formats and rates using forward link commands. However, under safe-mode conditions, the PSEA outputs a safe-mode initiate signal to the CMD module in the DMU to put the DMS into hardware safe-mode operation. The CMD provides this signal to the TFC module to switch from normal TLM format to 4-kbps backup TLM fixed format. Either ground control or computer software can direct the use of 4-kbps fixed format, and only the ground can direct use of 500-bps fixed format in the bypass mode. The computer cannot, either by RTC or SPC, direct the use of 500 bps fixed format. When operating in either of the two backup TLM fixed formats, the TFC module controls data requests to the active DIUI module to acquire DIU data for TLM. Computer output words are sent to the TFC module via an 8-bit buffer register in the DIUI module when operating in the backup TLM mode.

The TFC module interfaces with the CIF module for receiving TLM rate and format information from the computer and sending DMU status register data to the computer; the DIUI module for receiving acquired data for TLM from the computer and DIUs and for providing DIU requests and reading DMU status registers 0 and 1 when in the

backup TLM mode; the COM for real-time transmission of engineering TLM; the TRI module for recording engineering TLM; the TIM for obtaining the necessary timing signals; and the CMD module for the serial TFC/COM control command from the ground and for the safe-mode signal derived from the safe-mode initiate signal from the PSEA.

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2.6 INSTRUMENTATION AND COMMUNICATION SUBSYSTEM

The I&C subsystem provides the equipment required by HST for transmission of engineering data and science data, reception of command messages, provisions for tracking via the TDRSS or GSTDN, and the conditioning of temperature sensors within the SSM. The I&C includes the ICU, the SSA transmitters, the MA transponders and diplexers, RF multiplexers, RF switches, RF transfer switches, a circulator switch, forward and aft LGAs, \pm V3 HGAs, coax cables, and waveguides. During nominal operations, all transmissions to and from HST are normally routed via the TDRSS, with backup capability via GSTDN. Drawing 2.6 sheet 1 shows an overview of the I&C subsystem and its interface with the orbiter payload communication subsystem. A photo of the SSM ES bay 5 door, showing the HST communications system, is provided in Figure 2.6-1.

Fully redundant S-band MA transponders and SSA transmitters provide nominal communications with the STOCC via the TDRSS. In the event that service through the TDRSS is not available, the MA transponder may also communicate with the STOCC via an S-band compatible GSTDN site. Command (forward link) service (125 bps or 1000 bps) to the HST is provided using the HST LGA and the receiver portion of the MA transponder. TLM (return link) service from the HST is accomplished using the transmitter portion of the MA transponders for transmission of engineering data and ESTR playback data and using the SSA transmitters for transmission of science data, as provided below. The MA transmitters may be switched by ground command between the LGA and HGA. The SSA transmitter is hardwired to the HGA only. The transmission and reception capabilities of the HST I&C subsystem are summarized in Figure 2.6-2.

SSA transmitters

TDRSS only: 1-mb real-time or playback science engineering playback

MA transponders

 TDRSS mode: Engineering data 4-kbps real-time science data or 4-kbps 486 memory dump data
 GN mode: Engineering data/memory dump/ESTR through GSTDN port



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Figure 2.6-1. SSM bay 5 door

2.6-2


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Figure 2.6-1. SSM bay 5 door



Figure 2.6-2. HST communications summary

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6 HST COMM O/V

NOTES	$\left \right\rangle$	THE	нsт	HGA	SYSTEM	AND	SSA	XMTRS	ARE	NOT	INTENDED	FOR	USE	DURING	JOINT
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- THE HST ESS BUS POWER MAINTAINS THE TRANSPONDER CONFIGURATION CONTROL LOGIC DURING ALL MISSION PHASES
- A BYPASS OF THE PSP PROVIDES THE CAPABILITY FOR ORBITER TO RELAY HST TM AT 32 KBPS VIA THE PDI AND ORBITER OPERATIONAL DOWNLINK (OD)
- STOCC COMMANDS TO HST WILL BE ROUTED THROUGH THE ORBITER OPERATIONAL UPLINK AND ORBITER PI
- SWITCH SETTING
- PSP POWER-ON COMMANDS ARE FROM PANEL AIL SWITCHES OR FROM GCIL. REFERENCE SSSH 16.15-1 FOR DETAILS
- CONFIGURATION IS SHOWN WITH PSP BYPASS ENABLED
- B HPGSCA PERFORMS DIGITAL BIT SYNCHRONIZATION BETWEEN THE OUTPUT OF THE ORBITER PI AND THE INPUT OF THE ORBITER PDI TO REGENERATE THE DEGRADED WAVE FORM, WHEN THE HPGSCA IS NOT CONNECTED TO THE PDIP, A JUMPER IS INSTALLED TO MAINTAIN THE PSP BYPASS
- 9 TURNAROUND PLUG IS INTERCHANGEABLE WITH HPGSCA

SSA XMITTER IS CURRENTLY FAILED AND WILL BE REPLACED ON SM-3A

В

DATE: 04/23/99 NAME: VSH VERSION: REL1 DATE: 05/03/01 NAME: MDU VERSION: REL1 DATE: 01/14/08 NAME: LAM VERSION: REL2

HST Communications during Operations with Shuttle

During joint operations, the HST is capable of receiving commands relayed by or originated by the orbiter via the Payload Signal Processor/Payload Interrogator (PSP/PI). The forward link through the orbiter is identical to that of a GSTDN site. In addition, HST TLM may be relayed by the orbiter through the PI/Ku-Band Signal Processor (PI/KUSP) in a mode known as the Ku-band bent pipe. In this mode, the HST RF TLM signal is demodulated by the PI and remodulated onto a channel of the orbiter Ku-band return link. The orbiter Ku-band system is single string and is utilized when available. A GPC software mask is employed to protect the HST from direct exposure to Ku-band radiation levels higher than 68 V/m while the HST is berthed in the orbiter payload bay. While HST is on the RMS, Ku-band operations may be further constrained and managed manually to protect the vehicle.

2.6.1 *Bit Synchronizer Assembly* Description

During the HST deploy and SM-1 missions, HST telemetry was downlinked to the ground via the orbiter Ku-band and S-band systems. The S-band system downlinked the 32-kbps "PSP-bypass" mode telemetry only, while the Ku-band system downlinked either the 4-, 32-, or 1024-kbps telemetry in parallel. There were several known HST-related limitations with both of these modes.

The normal path for a shuttle payload requiring RF processing is Payload Interrogator (PI) \rightarrow Payload Data Interleaver (PDI) and subsequent S-band downlink. However, HST does not utilize a PSK-modulated subcarrier scheme. In order to accommodate this hardware limitation, JSC devised the PSP-bypass mode concept. A mission-unique cable was built to route the PI baseband output around the PSP, directly to the PDI. The PSP is bypassed because it requires the use of a PSK-modulated subcarrier input. During proof of concept testing prior to the deploy mission, it was determined that PSP-bypass mode was feasible at the 32-kbps data rate only. For the HST deploy and SM-1 flights, this limitation was acceptable for nominal on-orbit operations.

The limitation of the Ku-band system was also known prior to the HST deploy and HST SM-1 missions. The low-end bandpass for the orbiter Ku-band system is 16 kbps for channel 2. HST 32- and 1024-kbps telemetry is passed within specification. However, HST 0.5- and 4-kbps telemetry was downlinked on a "best effort" basis. Additionally, Ku-band communications are dependent on maintaining direct line of sight between TDRS and the orbiter Ku-band antenna. Significant periods of interruption can be expected, thus effecting the requirement to provide near-continuous downlink to the STOCC. For the HST deploy and SM-1 flights, these limitations were acceptable for nominal on-orbit operations.

During the HST SM-1 mission a memory dump was required after EVA operations to install the DF-224 Coprocessor ORU. The dumps were performed in a 4-kbps telemetry mode. The 4 kbps was downlinked via the Ku-band system to the STOCC. The data that the STOCC received and attempted to process was corrupted, which led to a series of troubleshooting operations. After evaluating the somewhat suspect data, it was

surmised that the DF-224 Coprocessor was in fact operating nominally and that the corruption of the downlink telemetry was a consequence of violating the specified bandpass requirements of the orbiter, and ground station Ku-band processing systems.

For SM2 and subsequent missions, GSFC chose to insert a digital bit synchronizer between the output of the orbiter PI and PDI to regenerate the degraded wave form witnessed during test at the 4-kbps data rate. JSC recommended hosting the bit synchronizer boards in a Payload General Support Computer Assembly (PGSCA). An evaluation was performed of several vendors' digital bit synchronizer boards. Aydin Corporation of Horsham, Pennsylvania was selected to design and fabricate these boards. The boards are designated PC335 digital bit synchronizer boards.

The PGSC was a modified version of the IBM 760 Thinkpad laptop computer with Windows 98 installed on the hard drive. The HST program used the PGSC in its expansion chassis configuration. (PGSCA refers to the PGSC/Expansion Chassis configuration as an integrated system). The HST program used the PCSCA for SM-2 and through SM-3B.

For SM-4 a new way needed to be developed to perform the bit synchronization due to the retirement of the 760XD based PGSCA onboard the *shuttle*. It was determined that a new standalone Bit Synchronizer Assembly (BSA) needed to be developed to perform this function.

BSA (GSFC Provided)

At the time of publication, the specifications for the BSA were under development, and details are to be provided at a later date.

Y-Cable (GSFC Provided)

The Y-cable provides the interface between the orbiter Payload Data Interface Panel (PDIP) and the BSA. The cable is connected to PDIP input port 1 (J101) if PI 1 is to be used, or port 2 (J103) if PI 2 is to be used. The Y-cable PDIP connector is a NLS6GT12-35P. The "Y" end of the cable consists of the DB9 PC335 input connector, and DB25 PC335 output connector. Each leg of the cable consists of a 22 AWG twisted shielded pair. The cable length is 14 ft ±6 inches. This allows the BSA to be staged on the middeck while the PDIP panel is located on the AFD. For flight, the cable is routed in a plastic cable tray or taped to the front face of the panels that run between the PDIP panel on the AFD, and the access way ladder to the middeck. The cable length provides a 3- to 3.5-foot arc around the top of the ladder for possible staging locations to the middeck ceiling.

28 V dc Power Cable (JSC Provided)

The BSA receives 28 V dc power from an orbiter 5-amp breakered source on the middeck. The particular power outlet is determined by JSC crew systems during development of the plug-in plan.

Flight Configuration/Operation (General)

The BSA mode of communication is actually an enhanced mode of the PSP-bypass configuration. In order to support the PSP-bypass mode of communication a mission-unique cable was built to route the data from the PI to the PDI via the Payload Station Distribution Panel (PSDP). As previously mentioned, this mode of communication was successful at the HST 32-kbps rate only. The output waveform of the PI at the 0.5- and 4-kbps rates was degraded to the point where the non-reconfigurable PDI bit syncs could not process the "triangular" shaped waveform. To regenerate the degraded signal, a BSA configured with a digital bit sync board is inserted into the string.

Two separate mission-unique cables are installed between the PSDP and the PDIP located on the port side of the AFD. One cable provides the interface from orbiter PI 1 to PDIP port 1), and the second from PI 2 to PDIP port 2. To support BSA enhanced mode, PI telemetry is routed to the BSA staged on the middeck via the Y-cable. If normal PSP-bypass mode is required, a turnaround plug is installed into the applicable PDIP input port.

The two HST dedicated BSAs are stored in middeck lockers packed in foam for launch and landing. One Y-cable and both power cables are also stored in a middeck locker. The primary Y-cable is installed preflight. It is restrained in a locking plastic cable tray on the AFD and tethered to the ceiling with Velcro loops on the middeck. The Y-cable PDIP connector is either connected for launch or taped to the forward face of the PDIP. This determination is made as part of the plug-in plan development. The PSP-bypass turnaround plugs are connected in two of the three remaining PDIP input ports for storage purposes.

The BSA is staged somewhere in the proximity of the ladder between the middeck and AFD, probably on the middeck ceiling. Staging location is finalized during development of the crew plug-in plan. The BSA is connected to an orbiter 28 V dc power feed by a JSC provided power cable. The Y-cable is then mated to the applicable PDIP panel input port and subsequently connected to the BSA. The BSA interface consists of two "D" type connectors that provide the input and output interface to the PC335 bit sync card.

2.6.2 I&C Hardware

2.6.2.1 S-Band Single Access Transmitter

The SSA transmitter transmits data from the SIs and recorders on the HST at a return link frequency of 2255.5 MHz through the TDRSS and the NASA Communications (NASCOM) ground network to the STOCC. The unit interfaces to the SIs and recorders via the COM of the DMU. Two identical transmitters are used by the HST, and either can be used during nominal operations.

Physical Description

Each transmitter measures 14.0 by 8.0 by 1.65 inches and weighs 7.0 pounds. Figure 2.6-3 shows the SSA transmitter physical details. The unit is a 19-watt, +2/0-dB-output, Biphase-Shift-Keyed (BPSK) transmitter in a machined-aluminum, one-piece housing. The power converter, defined as part of the transmitter, is in a separate aluminum housing fastened to the main transmitter structure. The whole assembly is bolted to a heat sink for passive thermal control. The SSA transmitters are located on the SSM ES bay 5 door, retained by eight noncaptive 3/16-inch torque set hex bolts, with washers, and torqued to 45 in-lb. Each unit includes three coaxial and two cannon-type connectors and is classified as a block II ORU.

Electrical Power

SSA transmitters 1 and 2 receive +28 V dc power from the HST main buses through relay K14 of PDUs 3 and 4, respectively. These same PDUs also provide power to the SSA transmitter heater and overtemp control circuit.



sfoc_fl2705_059.tif



Figure 2.6-3. SSA transmitter physical details

2.6-11

Functional Description

Real-time science data, playback science and engineering data, and SIC&DH memory dump are transmitted as high rate (1024 kbps) data to the TDRS. This playback uses the SSA transmitter/HGA. Science data from multiple simultaneous sources are interleaved by the Control Unit Science Data Formatter (CU/SDF) in the SIC&DH module. The CU/SDF is optionally capable of Reed-Solomon encoding/interleaving and Pseudorandom Noise (PN) encoding science data prior to outputting it to the COM of the DMU.

All 1024-kbps return link data are differentially encoded (NRZ-S), rate 1/3 convolutionally encoded, and optionally periodic convolutionally interleaved in the DMU COM. The Reed-Solomon code, interleaving, 1/3 convolution code, etc., are used to enhance a low bit-error rate during periods of significant Radio Frequency Interference (RFI). The science data are sent from the DMS to the SSA transmitter, which employs direct modulation of the carrier for transmission to the TDRSS station at White Sands, New Mexico.

The White Sands Complex (WSC) performs bit synchronization on the received return-link signal and de-interleaves and decodes the data, which are then transmitted to the STOCC at GSFC in 4800-bit blocks at 1.544 Mbps.

2.6.2.2 MA Transponder

The NASA TDRSS MA transponder provides an S-band receiver, command detector, and 5-watt transmitter. The MA transponder is capable of supporting both GSTDN (orbiter PI-compatible) and TDRSS services for receiving commands, transmitting data, and providing ranging service.

The transponder receiver receives the command data from the HST LGA at a frequency of 2106.4 MHz (forward link) and decodes the commands for further processing by the DMU CDI module (Section 2.5). The HST command data rates are 125 bps or 1 kbps, and ranging code coherence is provided by the transponder when selected by the STOCC.

The transponder transmitter uses spread spectrum techniques on two quadriphase channels (I and Q) to broadcast HST TLM data. This broadcasting is done by the HST HGA system or the LGAs to the TDRSS or GSTDN. In the TDRSS mode, the I and Q channels transmit engineering data at 0.5, 4, or 32 kbps; and the I channel also transmits real-time science data or 486 memory dump at 4 kbps. In the GSTDN mode, all data rates including 1-Mbps SSR dump data, and SD can be accommodated. The MA return-link frequency is 2287.5 MHz.

Physical Description

Figure 2.6-4 provides a physical overview of the MA transponders. The transponders are installed on the door of SSM ES bay 5 by nine noncaptive, 3/16-inch, torque-set hex bolts and washers, which are torqued to 45 in-lb. Each unit measures 13.9 by 9.9 by 4.3 inches and weighs 13.6 pounds. The MA transponders are block II ORUs.

Electrical Power

The MA command receivers are continuously powered from the HST essential bus, as are the CDI modules in the DMU (Section 2.5). The essential bus power also maintains the transponder configuration logic. Any loss of essential bus power causes the unit to reset itself (Section 2.6.2.2.4). The MA transmitters receive power from the HST main buses via PDU-1 and PDU-3 for MA transponders 1 and 2, respectively.

2.6.2.2.1 Receiver characteristics

The transponder receiver section receives the 2106.4-MHz signal from the LGA. The transponder diplexer provides separation between the high-level transmitter signals and the sensitive receiver circuits. The RF circulator switch is installed between the diplexer and the receiver to permit signal switching between the two transponder receivers. The receiver is used for forward link signal acquisition and may receive command or ranging signals. The HST forward link is described in Section 2.5.4.

The receiver simultaneously attempts to acquire two S-band signals that have markedly different characteristics on the same assigned frequency. The forward link from the TDRS is a suppressed-carrier, spread-spectrum, quadriphase signal with the command data modulo-2 added asynchronously to a short PN spreading code to obtain the In-phase (I-channel) signal. The Quadrature phase (Q-channel) consists of a longer PN code suitable for ranging. The forward link from a GSTDN site, which may also be present, is a residual carrier signal with the command data carried on a 16-kHz subcarrier that is phase modulated on the carrier.

The TDRS signal is typically frequency compensated, and the receiver must provide acquisition circuits that accommodate only the unconverted Doppler frequency offset and error caused by the receiver internal frequency stability. The GSTDN signal is swept in frequency about the nominal frequency by the ground station (or the orbiter) during the acquisition phase. The receiver in the transponder acquires the command and ranging signal as it sweeps through the natural rest frequency of the receiver. Once the receiver has acquired the signal, the receiver tracks frequency offsets with a phase lock loop tracking circuit for both cases. When the TDRS signal is being received from a relay satellite, especially when over water, a reflection path may provide a delayed signal of sufficient amplitude for the receiver to falsely lock on the multipath signal. The transponder automatically corrects for this possibility.



Figure 2.6-4. MA transponder physical overview

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Forward Link Acquisition

Either a sweeping GSTDN signal or a PN code TDRSS signal may attempt to contact the telescope. The receiver simultaneously tries to lock onto a GSTDN signal and attempts to establish correlations with a PN code sequence stored in the transponder. One of the multiple detector channels uses a power detector for the GSTDN signal, while four other correlator channels search for different portions of the PN code. Simultaneous search of code segments is required to reduce acquisition time to an operationally acceptable value. A Temperature Controlled Crystal Oscillator (TCXO) provides the required frequency stability during acquisition.

If a GSTDN signal is detected before a code sequence, the transponder is automatically configured as a phase lock loop receiver. The receiver resumes searching for the PN code upon loss of the GSTDN signal. Therefore, receipt of a spurious GSTDN-like signal does not lock the receiver in the phase lock loop mode.

If a GSTDN signal is not detected, the code acquisition sequence continues in the four code-staggered channels, each channel stepping in half-bit intervals of code until one of the four channels detects code correlation. Full code correlation is then established, at which time the transponder is automatically configured for TDRSS spread-spectrum operations.

Once a signal (GSTDN or TDRSS) is detected, carrier acquisition is initiated. Once acquired, the tracking loop will maintain phase coherence with the received signal. A Costas type detector reconstitutes the suppressed carrier of the TDRS PN signal. The search logic looks for a multipath spectrum code correlation to avoid the possibility that the transponder has locked to a delayed multipath TDRS signal, which results in reduced forward link performance. If none is found, then a normal direct path correlation is assumed.

Command Data Detection

In the TDRSS mode, the command data are recovered from the forward link spreading code and chopped with a 16-kHz square wave prior to delivery to the CDU within the transponder. In the GSTDN (or orbiter PI-compatible) mode, the 16-kHz subcarrier containing the command data is delivered to the CDU. Based on the mode, the CDU is automatically configured by the transponder internal digital subsystem to accept either the TDRSS digital bit stream or the GSTDN (orbiter PI) 16-kHz biphase modulated subcarrier. The CDU delivers to the CDI in the DMU the original command bit stream, a clock timing signal, and an in-lock signal.

2.6.2.2.2 Transmitter characteristics

The transponder transmitter is used to return ranging signals or to send TLM data. Engineering data at 0.5, 4, or 32 kbps can be routed to the I or Q channels separately or simultaneously. Science data transmitted at 4 kbps can be routed only to the I-channel. Memory dump data from the 486 flight computer can also be transmitted at 4 kbps via any combination of I- and Q-channels. In GSTDN (or orbiter PI-compatible) mode, 1.024-Mbps data from the SSRs or SIs can be transmitted. Section 2.5.5 details the HST return link.

Transmit Return Link Telemetry

The transmitter in the TDRSS user transponder has inputs for two digital TLM data bit streams in the TDRSS mode and two digital TLM signals in the GSTDN mode. The two TDRSS data bit streams are modulo-2 added to internally generated PN codes and transmitted independently on the I-channel and the Q-channel as Staggered Quadriphase Shift Keyed (SQPSK) modulation on the S-band return link. The relative distribution of modulation energy between the Q channel and the I channel is set at 1:1. An external programming connector with selective pins shorted to ground permits mission selection of the command data rate. The TDRSS and GSTDN TLM channels, as well as the GSTDN tone ranging channel, can be turned on or off by STOCC command, which is linearly summed with the two GSTDN TLM inputs.

The configuration of the transmitter and the selection of modulation sources and levels for TDRSS or GSTDN operation are automatically accomplished by the transponder digital subsystem, based on the forward link mode, external commands, and fixed programming.

Following modulation, the S-band signal is amplified to the desired RF output level. The output filter/isolator, which protects the transmitter from shorts or opens on the antenna line, also contains power monitors for telemetering the forward and reflected signal power. A band reject filter protects the receiver.

Turnaround Ranging

The turnaround ranging function in the transponder is available only when the receiver is locked to a forward link signal. In the TDRSS mode, turnaround ranging is obtained by synchronizing the all-ones block of bits in the locally generated PN code with the all-ones block of bits in the received forward PN code. The code itself is not turned around. With the transponder operating in the coherent carrier turnaround mode, this function is available in the MA and SSA services except when transmitting user-supplied high-rate TLM signals.

The GSTDN tone ranging subsystem relies on the spacecraft transponder to detect sine wave tones in the range of 4 to 500 kHz, which have been phase modulated on the forward link carrier by the ground station, and to remodulate them at a predetermined phase modulation index on the return link carrier. In the phase lock receiver in the

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transponder, a separate intermediate frequency amplifier/phase detector channel is provided for the range tone signals. The amplitude of the output tones is maintained constant by an automatic gain control system. A commandable switch is provided in the ranging channel so that it can be commanded on or off by the forward link commands. The HST does not use subcarriers for TLM data. Therefore, ranging and data transmission cannot be performed simultaneously. The TLM data are baseband, modulated directly onto the return link carrier.

Coherent Carrier Turnaround

The ground Doppler range rate tracking subsystem relies on the spacecraft transponder to turn around the forward link carrier frequency in the precise ratio of 240/221 to generate the return link carrier. This is called the coherent turnaround mode.

In the suppressed carrier TDRSS mode, the carrier that is turned around is reconstructed in the receiver detector circuits from the sidebands of the forward link spread spectrum signal. The transmitter drive frequency is obtained from the phase lock tracking loop in the receiver, which is phase locked to the forward link residual carrier signal in the Space Flight Tracking and Data Network (STDN) mode. In the TDRSS and GSTDN modes, the phase lock receiver is a double conversion superheterodyne type in which the effect of the down conversion is to cause the receiver Voltage Controlled Oscillator (VCO) to run at precisely 240/221 times the received S-band forward link frequency.

If the receiver is not locked to a forward link signal, an RF switch in the transmitter automatically selects the transmitter drive frequency from the TCXO. This is known as the noncoherent mode.

2.6.2.2.3 Operational states

Numerous states of the transponder are automatically reconfigured by its self-contained digital subsystem. Representative examples include the GSTDN and TDRSS mode configurations, TDRSS mode acquisition states, and coherent/noncoherent return link carrier modes.

As mentioned earlier, various functions of the MA transponders can be controlled on or off, or values of parameters can be set by external command. Commands are contained in a 16-bit serial command word that is read into the transponder at a 256-kbps rate from the DMU COM.

2.6.2.2.4 Transponder reset configuration

Table 2.6-1 describes the reset configuration of the MA transponder after a loss of input power from the HST essential bus.

Parameter	Status
Receive command rate	125 bps
Transmit	Initially off
Frequency	TCXO until receiver acquires; then receiver VCO
Mode	Dual
GSTDN modulation index	N/A (high and low are identical)
GSTDN telemetry channels	Both off
Ranging channel	Disabled

Table 2.6-1. Powerup (reset) RF parameter configuration

2.6.2.3 Low-Gain Antennas

Two LGAs are installed at opposite ends of the HST (one forward, one aft), and together provide 95 percent spherical coverage. However, spherical coverage is lost if either of the two LGAs fails.

Physical Description

The antenna is an omnidirectional, conical spiral with radiator elements of copper on a Mylar substrate, bonded to a fiberglass cone, and mounted on a fiberglass base. The antenna is approximately 8 inches long and 3 inches in diameter at the base and is covered with Kevlar to provide thermal protection and reduce the amount of reflected light. Each antenna is supported by a rectangular waveguide originating from the SSM ES. The forward LGA waveguide is routed along the exterior of the FS and LS and weighs approximately 54.3 pounds. The aft LGA waveguide is routed along the interior of the AS and weighs approximately 28.1 pounds.

Each antenna has been modified to allow in-orbit changeout. The LGA ORU design is shown in Figure 2.6-5; it features three ORU bolts that secure the antenna cone through a keyed flange to an adapter plate fixed to the antenna mount. A coaxial cable provides the RF connection between the rectangular waveguide and the antenna cone. Adequate cable length is provided to allow the cone to separate from the antenna mount up to 5 inches to provide access to the cone coax connector by the gloved hand of an EVA crewmember.

JSC-29029 Basic, Rev. A



Figure 2.6-5. LGA ORU design

Antenna Characteristics

The LGA is manufactured by LMSC, and key performance features are provided in Table 2.6-2.

Figure 2.6-6 provides the LGA RF plots for each antenna, showing regions of the LGA FOV, in which the actual antenna output falls below -1 dB, referenced to the equivalent isotropic antenna output. Figure 2.6-7 also provides the LGA and HGA FOVs referenced to the HST axes.

Table 2.6-2. LGA characteristics

•	RF power level	20 W (CW) minimum
•	Operating frequency	Receive = 2106.4 ± 6 MHz Transmit = 2255.5 ± 6 MHz (not used) Transmit = 2287.5 ± 5 MHz
•	Polarization	Left-hand circular
•	Axial ratio	3.0 dB maximum
•	Gain	-1.0 dBi minimum
•	Voltage Standing Wave Radio (VSWR)	1.5:1 maximum
•	Impedence	50 ohm
•	Coverage	95 percent spherical (two antennas)

EVA Operations

An LGA cover guard can be installed on the aft (-V1) LGA during EVA operations. The cover guard is equipped with a standard socket that provides the capability to be installed in the FSS. The LGA protective cover is swung down into position and left in place until the final EVA day (Figure 2.6-8).



FWD LGA RF plot



JSC-29029 Basic, Rev. A





Figure 2.6-6. Concluded

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JSC-29029 Basic, Rev. A



Figure 2.6-7. HST and LGA fields of view

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Figure 2.6-8. EVA LGA protective cover

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2.6.2.4 High-Gain Antenna

The HGA is the primary downlink antenna used to transmit science data and engineering data to the TDRSS. The HGA is combined with an APS to provide precision pointing of the HGA under 486 control. The APS is described in Section 2.3.2. Antenna pointing control will not be active during joint HST/shuttle operations.

Physical Description

The HGAs are parabolic dish antennas approximately 51 inches in diameter with cutouts on two edges to provide clearance for the payload dynamic envelope in the orbiter payload bay. An HGA is comprised of three basic subassemblies: the feed assembly, the reflector, and the interface cable. The aluminum feed assembly is a slightly modified version of a previously tested design with added thermal protection coating. The reflector features honeycomb construction with graphite-epoxy face sheets, three aluminum honeycomb core sections for an overall tapered cross-section, individual titanium interface inserts, and a center-feed support fitting. The center-fed turnstile feed is vertex-mounted, and the crossed dipoles are located at the reflector focus.

The antenna dish mounts through a gimbal system to a 14-foot-long deployable mast, which provides a 20° boresight clearance to any HST structure, excluding the AD and SAs for any orientation of the antenna dish. The masts are rectangular cross-section assemblies, fabricated of graphite/aluminum matrix that serves as both a mast and an RF waveguide. Each mast and dish is deployed/stowed through a latch and hinge arrangement as shown in Drawing 2.3 sheet 2.

Antenna Characteristics

The HGA is manufactured by General Electric, and key performance features are listed in Table 2.6-3.

Figure 2.6-9 shows the HGA RF antenna pattern sidelobes within specification.

Table 2.6-3. HGA characteristics

RF power level	15 W (CW) minimum
Operating frequency	2255.5 MHz (nom) (SSA)
	2287.5 MHz (nom) (MA)
Polarization	Left-hand circular
Axial ratio	3 dB maximum within 1° of RF boresight
Gain	+26.4 dB
RF boresight alignment	Within 0.25° of mechanical boresight
VSWR	1.7:1 maximum
Impendence	50 ohm
Diameter	50.86 in. maximum
Focal length	12.86 in.
Weight	S/N 01 = 6.87 lb
-	S/N 02 = 6.75 lb

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Figure 2.6-9. HGA RF antenna pattern showing sidelobes

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2.6.2.5 RF Switching

A system of RF switching is included in the I&C to allow for multiple antenna configurations, providing STOCC additional operating flexibility in communicating with the HST. The switching system includes a circulator switch that allows either LGA to be connected to either MA receiver, and RF and transfer switches that allow either MA transmitter to radiate through any HST antenna and either SSA transmitter to radiate through either HGA. The I&C antenna switching is summarized in Figure 2.6-10. All RF switching is included with the other I&C elements on the SSM bay 5 door.

Circulator Switch

The circulator switch consists of two interconnected, three-port ferrite stripline circulators used to accomplish switching of MA receiver inputs to either of the LGAs. The unit operates at 2106.4 \pm 10 MHz with an insertion loss of less than 0.48 dB single channel (0.085 dB dual channel). Latch windings for the two units are connected in series so that a single command pulse switches both circulators. A separate current transformer provides sensing of switching status, which is provided as bilevel TLM monitors. Figure 2.6-10 depicts the circulator switch within the antenna switching system.

RF Transfer Switches

RF transfer switches 1 and 2 are break-before-make, latching coaxial devices. They are used to switch the MA transmitter outputs to the combination of either FWD LGA /+V3 HGA or AFT LGA /-V3 HGA and the SSA transmitter outputs to either the +V3 or -V3, as shown in Figure 2.6-10. The units operate at 2255.5 \pm 20 MHz and 2287.5 \pm 5 MHz with an insertion loss of less than 0.2 dB.

RF Switches

RF switches 1 and 2 are also break-before-make, latching coaxial devices. These are used to switch the outputs of the MA transmitter selected by RF transfer switch no. 1 to either LGA or HGA, as shown in Figure 2.6-10. The units operate at 2287.5 \pm 5 MHz with an insertion loss of less than 0.2 dB.



Figure 2.6-10. I&C subsystem block diagram

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2.6.2.6 Instrumentation Control Unit

The ICU supplies the electrical excitation and signal conditioning for SSM instrumentation. The ICU services both instrumentation not incorporated in equipment units as internal monitors and some internal instrumentation that does not have internal conditioning. The instrumentation supported by the ICU includes SSM temperature monitors and switch position indicators.

The ICU is a fully redundant unit in a single enclosure with unregulated power inputs, regulated supplies, and conditioning (biasing) circuits (all redundant). There is full separation of these circuits to the point of exit to the transducer. The transducers are not directly redundant, but the large number of structural points monitored ensures a degree of redundancy through measurements in close proximity.

Physical Description

The ICU assembly consists of 11 printed circuit boards mounted in a dip-brazed, machined aluminum, two-porch enclosure, as shown in Figure 2.6-11. The unit size measures 16.25 by 15.25 by 6.65 inches and weighs approximately 25 pounds. The ICU is installed on the door of SSM ES bay 8 and is currently a non-ORU item. Exterior interface connectors are installed on the two porches and are wire wrapped to connectors on the boards. PDU-1 and PDU-2 provide unregulated 28 V dc power to the ICU.

Functional Description

The ICU interfaces with all other SSM subsystems and structures. Table 2.6-4 provides a summary of the ICU functions and capabilities. A block diagram of the unit is provided in Figure 2.6-12.



Figure 2.6-11. ICU physical overview

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Figure 2.6-12. ICU block diagram

Table 2.6-4.	ICU	functional	summary
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Function	Performance capability
Condition positive temperature coefficient thermistors	96 circuits, range: -100° to +70° C
Condition negative temperature coefficient thermistors	88 circuits, ranges: -50° to +80° C 0° to +80° C -20° to +60° C
Condition position monitor switches in RF and RF transfer switches	One bilevel TLM circuit for each switch (four)
Operate from HST unregulated power bus	Includes voltage-regulating power supplies
Accept DIU commands to control power supplied on and off	Eight latching relays permit individual power supply on control and ganged off controls
Accept safing system (PSEA) commands to turn power off	Diode isolated input permits
Provide internal health and configuration status through bilevel and analog monitors	Relay position bilevel and regulated voltage analog monitors provided

2.6.3 STS Interface Description

The orbiter provides the capability to relay commands to and TLM from the HST. During both the deploy and maintenance missions, STOCC command messages are transmitted to the JSC Mission Control Center (MCC), which interleaves the message into the orbiter S-band or Ku-band operational forward link. Once received by the orbiter system, the command message is processed through the systems management GPC and the orbiter payload communications system and is retransmitted to the HST via its LGAs in a manner resembling the GSTDN uplink mode of the HST MA transponders. The command message is then processed further, as described in Section 2.5.4.

2.6.3.1 Orbiter Processing of HST Commanding

The orbiter systems management GPC issues each serial digital command message through the PSP/PI. A PSP transaction may contain up to sixty-four 16-bit command words per message, and the GPC supersedes each message with a PSP configuration message of five 16-bit words in length. This configuration message programs the PSP to output the subsequent HST command data words in the transaction in a specified manner via a specified route. For HST, the PSP output is RF only and has the following characteristics per the PSP configuration message:

PSP command rate	1 kbps
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PSP command data type NRZ-L

PSP subcarrier idle pattern Yes

Once the HST MA receiver has acquired the forward link carrier from the PI, the PSP subcarrier idle pattern will be enabled to allow the HST to remain locked onto the PI forward link. The PI will be configured to transmit at a nominal carrier frequency of 2106.4 MHz (channel 701).

2.6.3.2 Orbiter Relay of HST Telemetry

The HST return link is acquired by the orbiter PI, which receives the HST return link on a nominal carrier frequency of 2287.5 Mhz (channel 701). From the PI, HST telemetry is transmitted to the STOCC via orbiter S-band or Ku-band communication systems. Data relayed via the S-band system are routed around the PSP (PSP-bypass configuration) to the HST PGSC, then to the PDI for supporting 0.5, 4, and 32k telemetry. See Figure 2.6-13 for PSP-bypass configuration. The PDI synchronizes data to the data stream and buffer to allow the orbiter system to interleave the HST TLM into the orbiter operational downlink. Once received at the MCC, the HST stream is stripped from the orbiter data and redistributed to the STOCC over the NASCOM network.



Figure 2.6-13. PSP bypass for HST

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When available, the Ku-band wideband data system may be configured to relay the HST return link to the TDRSS through the orbiter. The Ku-band system is used primarily for downlinking 1024k HST telemetry and as a backup to the S-band communication system for 0.5, 4, and 32k telemetry.

The nominal HST forward and return link carrier frequencies coincide with those of the orbiter high-frequency mode. For this reason, the orbiter communications system must be configured to low frequency during servicing missions.

2.6.4 Safety Considerations

Nominal or inadvertent activation of an HST LGA or HGA while the HST is in the orbiter bay or during retrieval/deployment does not result in damage to the orbiter, RMS, payload subsystems, or EVA crewmember.

There is no limit to radiation at station levels higher than Zo 410. The HST HGAs and LGAs are positioned well above the orbiter Zo 410 station.

RF radiation does not exceed payload-to-payload 25 *Volts per Meter* (V/m) criteria or the payload-to-RMS 126 V/m criteria at nominal operational distances, including the 45° pivoted reboost position, from the LGAs or the HGAs.

The SSA transmitters are not used during the servicing mission. They are placed in a three-inhibit condition prior to HST retrieval. Drawing 2.6 sheet 1 depicts the physical inhibits for the MA and SSA transmitters. These three inhibits will be managed as hazardous commands from the STOCC. For a nominal mission profile, RF ICD limits are met constraining HGA dish pointing to ensure an HGA dish never points toward the orbiter in a configuration that could exceed ICD limits. *Though the three inhibits are implemented to provide the RF safety inhibits, they are not to NASA Standards with regards to rotational HGA. The inhibits do not meet the NASA Standards due to no Hard-line, Software Encryption, or Return Power inhibits.*

From an EVA standpoint, hazardous RF radiation exists only in a small area inside the HGA dish and behind the HGA feed dish. It is not possible to place the EMU into this area because of space limitations. The field strength and safe distances for both the MA and SA transmitters are provided in Table 2.6-5. The HGA and LGA field strengths as a function of region and distance are provided in Table 2.6-6. At the beginning of the mission, the aft LGA is covered with the LGA protective cover, which provides a standoff distance and maintains EMU exposure to less than 106 V/m.

Likewise from an RMS standpoint, hazardous RF radiation only exists in a small area inside the HGA dish and the physical dimension of the dish prevent RMS access to this area.

			Near field region Near field region		Near field region		eld region
Transmitter	Max. output power (W)	Cable loss (dB)	Transmit pwr at HGA (W)	Max. power density (mW/cm ²)	Max. field strength (V/m)	Distance for 106 V/m (cm)	Distance for 10 mW/ cm ² (cm)
MA	5.3	4.9	1.72	0.52	44.5	12.0	6.4
SA	21.4	4.6	7.42	2.27	92.5	25.1	13.7

Table 2.6-5. Field strength and safe distances for MA and SA transmitters

Table 2.6-6. HST antenn	a field strengths as a f	function of region and distance
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	MA transmitter through HGA							
Field strength (V/m)	Power density (mW/cm ²)	Region	Dist	ance				
			(cm)	(in.)				
195.00	10.09	Feed	6.52	2.57				
126.00	4.21	Feed	10.10	3.97				
106.00	2.98	Feed	12.00	4.72				
44.50	0.53	NF	498.00	196.06				
25.00	0.17	xtion	572.41	225.36				
20.00	0.11	xtion	715.52	281.70				
5.88	.92	FF	2550.00	1003.2				
	SA transmitter thr	ough HGA						
Field strength (V/m)	Power density (mW/cm ²)	Region	Dist	ance				
			(cm)	(in.)				
195.00	10.09	Feed	13.64	5.37				
126.00	4.21	Feed	21.12	8.31				
106.00	2.98	Feed	25.10	9.88				
92.50	2.27	NF	491.00	193.31				
25.00	0.17	xtion	1190.85	468.84				
20.00	0.11	xtion	1488.57	586.05				
12.41	.041	FF	2510.00	988.8				
	MA transmitter the	rough LGA						
Field strength (V/m)	Power density (mW/cm ²)	Region	Dist	ance				
			(cm)	(in.)				
101.00	2.98	FF	7.6	3.00				
25.00	0.17	FF	30.5	12.0				
20.00	0.11	FF	37.9	14.9				

NF - Near field

FF - Far field

2.7 POINTING CONTROL SUBSYSTEM

The HST PCS provides the following functions for the vehicle: an attitude reference, attitude control and stabilization, and the ability to maneuver in support of HST orbital operation (Drawing 2.7 sheet 1). In addition to these primary functions, the PCS also supports vehicle momentum management, instrument/subsystem calibration, and fault detection and isolation within the PCS itself. Functional control of the system resides within PCS software modules executed by the 486 flight computer. Ground support is required for uplink of database updates and vehicle pointing requirements.

The PCS operates in a number of different mission modes to provide pointing (coarse and fine) and a stable platform for scientific operations. These include the capability to reorient the spacecraft and to track a target, as well as a raster scan capability. In the event of two major PCS failures, a backup system provides HST coarse +V3 Sun-point attitude control and stabilization to allow for HST survival until a subsequent revisit by the orbiter.

Attitude Reference

The PCS utilizes Sun sensors to provide initial attitude reference data. RGAs and the Sun sensors are used as position error sensors to orient the HST +V3 axis toward the Sun upon release from the orbiter RMS. Coarse attitude data may be obtained from the HST MSS magnetometers, which sense the vehicle orientation with respect to the Earth's magnetic field lines (which generally run from north to south). Once clear from the orbiter, the FHSTs are used to determine the fine vehicle reference axes alignment to the inertial frame, allowing the identification of preselected guide stars within the FOV of the FGSs mounted within the OTA. Final alignments of the SIs to the targets are accomplished using references within the instruments and STOCC commands.

Attitude Control and Stabilization

The control and stabilization of the HST is provided using attitude reference sensors (FHST is for coarse pointing, FGSs for fine pointing), RGAs to provide the sensing of vehicle rates, and RWAs to provide vehicle control torques. The external torques that tend to displace the HST from its inertial attitude are counteracted by the reaction wheels through a method of momentum storage controlled by the 486 flight computer. Saturation of the reaction wheels is prevented through continuous desaturation torques provided by the Magnetic Torquing System (MTS) under control of the 486 PCS software.

In the event of a major PCS or DMS failure, the PSEA will take command of the vehicle and issue the required commands to place the vehicle in a +V3 Sun-point attitude and hold it there until commanded otherwise. The PSEA has the ability to use the primary PCS sensors (excluding the FGS system and FHSTs), plus a contingency rate sensing capability provided by a Retrieval Mode Gyro Assembly (RMGA) if there is a failure associated with the RGAs. The PSEA maneuvers the vehicle and control attitude using

the RWAs, if available, or by issuing magnetic moment commands to the MTS when in the GG mode.

Maneuvering

The reorientation, target tracking, and raster scanning maneuvers of the HST are performed on command from the 486 flight computer per STOCC command loads. The desired attitude is uplinked by the STOCC and processed by the PCS software modules, which generate the prescribed torquing commands to the RWA, to slew the vehicle. The maximum HST maneuvering slew rate is approximately 90° in 15 minutes (6 deg/min).

Reaction Wheel Desaturation

Desaturation of the stored momentum in the HST reaction wheels is continuously managed by momentum management software in the 486 flight computer. The MSS magnetometer outputs, which are proportional to the measured field strength of the magnetic field of the Earth, are used by the software to calculate the required torquing currents to be output to the MT bars of the MTS. If the MSS is not usable, the software employs a model of the magnetic field of the Earth.

Instrument/Subsystem Calibration

The FGSs provided within the OTA are used for attitude control in the fine pointing mode. The FGSs are also used to calibrate the misalignment of both the FHSTs and the RGAs. Additionally, by using vehicle maneuvers over known slew angles determined through FHST and/or FGS star identification with precise time measurements, the RGA scale factors and biases can be calculated. Generation of commands, as well as the data analysis required for calibration activities, is provided by the STOCC.

Fault Detection and Isolation

Sufficient telemetry monitor points are provided within the PCS to assure unambiguous and timely failure isolation and vehicle health statusing at the STOCC. Software failure detection flags are used by the onboard system to switch to preprogrammed safe-mode configurations and attitudes. The 486 software provides for the autonomous handling of minor SSM subsystem faults to minimize the impact to science operations. The occurrence of significant HST subsystem failures causes an autonomous handover of vehicle control to the PSEA, which performs load shedding and orients the vehicle in a predefined Sun-point attitude until the STOCC intervenes. The SSM safing system is described in Section 2.9.


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2.7.1 PCS Sensors

2.7.1.1 Fine Guidance Sensor

The FGS is the attitude sensor used in conjunction with the rate gyros to provide precision attitude reference during science operations through search, acquisition, and tracking of target stars. The two major subassemblies of an FGS system are the FGS Radial Bay Module (RBM) and the Fine Guidance Electronics (FGE). The FGS RBM is composed of the optical bench assembly, an interferometer assembly, and star selector servo assemblies.

Physical Description

The three FGS RBMs are mounted radially in the AS, just aft of the SSM ES, within the Focal Plane Deck Assembly (FPDA) of the OTA focal plane structure. Each FGS RBM measures approximately 54 by 47 by 22 inches and weighs 494 pounds. Figure 2.7-1 shows an FGS RBM during installation and provides a cutaway view of the FGS internal components.

Each FGS RBM is supported electronically by a corresponding FGE. The FGEs are installed on the doors of OTA ES bays D, F, and G, as shown in Figure 2.7-2. The FGE electronics are housed in a machined aluminum enclosure and are packaged in modular subassemblies that can be individually removed without disturbing other modules. Logic modules are plug-in connector units, whereas the power supply and EMI filter contain soldered-in, feedthrough terminals. The unit weighs 29.2 pounds and measures 23 by 11 by 10 inches. The FGE is a block I ORU; retained to its corresponding OTA ES door by 10 ORU bolts. An additional ORU bolt is used at the connector interface to allow simultaneous disconnect of all connectors.

Electrical Power/Thermal Control

The FGS RBMs receive power from their respective FGEs. Each FGE receives +28 V input power from the OTA Electrical Power/Thermal Control Electronics (EP/TCE). The FGS RBM consumes 20 watts during operation, and an additional 3 to 22 watts for heater power during mission operations. Each FGE consumes 64 watts. Active thermal control of the FGS RBMs are maintained by over 30 primary and redundant heater circuits distributed throughout the FGS. The FGS RBM optical bench components operational temperature range is from +60.8° to +73.4° F, and the survival temperature range is from -0.4° to +89.6° F. Thermal control of each FGE is provided by primary and redundant heater circuits on the door of each corresponding OTA ES bay. The FGE operational temperature range is -9.4° to +86.0° F, and the survival temperature range is -49.0° to +111.2° F.

Command and Data Interfaces

Each FGS RBM interfaces to the SSM DMS through its corresponding FGE. Each FGE receives a 40 Hz timing signal, a 1.024 MHz clock signal, and a minor frame sync signal directly from the DMU. Digital commands (serial digital and LLDs) and HLD commands are provided from DIU-1. The FGE provides serial digital and analog inputs to DIU-1, allowing SSM polling of FGS data for performance of FGS-related PCS functions.



Figure 2.7-1. Fine guidance sensor radial bay module



OTA ES bays D, F, G



FGE interfaces

Figure 2.7-2. Fine guidance electronics

2.7-7

FGS Operational Modes

This section briefly describes the operation of the FGS. Further discussion of the FGS optics is deferred to Section 3 because the FGS optical system uses the OTA as an image source. The FGS operational modes are summarized in and referenced in the following material.

After power turn-on and initialization is accomplished (START UP), the STOCC provides data on the guide star to be acquired by the FGS (SSM CONTROL). The acquisition process is then initiated by the SEARCH TRACK ON software-generated command. The FGS autonomously begins a spiral scan outward (search mode) until a star of the designated brightness is encountered. If a guide star of the appropriate brightness is not encountered during the search, the FGS goes to the stop mode, and the "star presence" flag is not set. If an appropriate star is found, the FGS autono-mously enters coarse track mode and the sensor moves its 3.5-arcsecond square aperture in a circular pattern around the star position. If the coarse track acquisition is not successful, the FGS goes to the stop mode and resets the star presence flag. If the acquisition is a success, and the data valid HOLD software flags are set, the FGS remains in coarse track mode until the HOLD command is released by the PCS software. Once the HOLD is released and the coarse track "data valid" flag is reset, the FGS enters fine lock mode and attempts to acquire the guide star using the estimate of star position determined in coarse track. If the acquisition is not successful, the sensor goes to the stop mode and resets the star presence flag. The FGS automatically reverts back to the search mode and continues the spiral scan until the star is found or the commanded search radius is reached. If the transition to fine lock is successful, the data valid flag is set and the software configures the PCS for science operation based on the STOCC-defined configuration.

Figure 2.7-3 illustrates the FGS operational modes.

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START UP

- SELECTION OF POWER BUS
- POWER TURN ON
- INITIALIZATION

SSM CONTROL

- RECEIVE DATA FROM DIU
- FORMAT DATA FOR OPERATION
- TRANSMIT DATA TO PROCESSING
- DATUM ACCURACY: 30 ARC SECONDS

SEARCH

- SPIRAL SCAN
- STAR COUNT BETWEEN I_H AND IL
- DYNAMIC RANGE: 90 ARC SECONDS
- ACQUISITION: 1.6 ARC SECONDS



COARSE TRACK

- NUTATING SCAN
- DYNAMIC RANGE: 3 ARC SECONDS
- ACQUISITION ACCURACY: 0.1 ARC SECOND



TRACK

FINE LOCK

- SLIDING SCAN
- FINAL ACCURACY, LOCKED
- DYNAMIC RANGE: 0.1 ARC SECOND
- ACQUISITION ACCURACY: 0.007 ARC SECOND



LOS

- ORTHOGONAL GRID PATTERN
- ENTERS FROM AND RETURNS TO COARSE
- OPTICAL ALIGNMENT MODE



RESUME SEARCH

- RETURN TO SEARCH
- ONLY FROM COARSE TRACK OR FINE LOCK

SHUT DOWN

- STAR SELECTOR SERVO TO DEFAULT POSITION
- HIGH VOLTAGE LOW TO INTERFEROMETER
- POWER DOWN

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Figure 2.7-3. FGS operational modes

2.7-9

2.7.1.2 Fixed Head Star Trackers

The FHST is a sensitive electro-optical detector, which has the capability of locating and tracking a target star within its FOV. As part of the PCS, the FHSTs assist in coarse attitude determination after deployment and slew operations. The FHSTs also provide attitude data to supplement FGS and RGA data during fine pointing. The HST is equipped with Three FHSTs mounted to the SSM ES within the HST AS. Each one is provided viewing access to space by corresponding viewing ports through the AS bulkhead. Motorized shutters, operated on command, protect the star tracker optics from exposure to bright sources (Earth, Moon, and Sun). The units are precisely aligned with the RGA RSUs and the OTA optical axes.

Physical Description

Each FHST measures 6.6 by 7.1 by 12.2 inches and weighs 16.9 pounds. Each one is equipped with a conical lightshade and mounts to the SSM ES through a delta plate, which provides the precise alignment of the tracker. The lightshade reduces stray light that could degrade performance and performs a secondary role as a heat sink to dissipate heat generated by the tracker electronics. The lightshade assembly may be removed from the tracker by an EVA crewmember to allow access to the RSUs mounted to the SSM ES platform beneath. The FHSTs currently are not classified as ORUs.

The FHST mounting arrangement is shown in Figure 2.7-4 and Figure 2.7-5. FHST 1 is oriented directly along the -V3 axis. FHST 2 and FHST 3 are both aligned 45° toward the -V1 off the V2-V3 plane, and each 30° off the V1-V3 plane toward the -V2 and +V2, respectively. Each has a maximum FOV diameter about this orientation of 11.3° (full angle).

A bright object protection system, consisting of the shutter assembly and a target suppression circuit, protects the tube of the FHST from damage if a bright source approaches or enters its FOV. The shutter is positioned between the lightshade and the FHST aperture window. The output current level of the image dissector tube (the active optical sensor of the FHST) is monitored by the target suppression circuit. If it exceeds the maximum level expected for star brightness, a TARGET SUPPRESS signal is generated closing the shutter. Once triggered in this fashion, the shutter remains closed until reset by STOCC command. The shutter may also be powered shut by STOCC command to the K44 relay of the corresponding PDU or by the orbiter flight crew from the SSP.

Electrical Power/Thermal Control

Switched electrical power is supplied to FHSTs 1, 2, and 3 through the K26 relay of PDUs 1, 2, and 3, respectively (Drawing 2.7 sheet 2). Primary and redundant heater circuits installed on the delta plate of each FHST are supplied through separate relays of their corresponding PDUs (K20-PRI and K27-RED). Heater cycling is controlled by redundant series connected thermostats, with setpoints of 28.4° F for the primary elements and 24.8° F for the redundant elements. The FHST requires about a 15-minute warmup after power is applied before it attains its specified performance.



Figure 2.7-4. FHST installation



Figure 2.7-5. FHST installation in AS

Each FHST consumes 18 watts of electrical power, and its shutter consumes an additional 3 watts when commanded closed. Although it is possible to power all three trackers simultaneously, only one or two will ever be used at the same time. Because they are normally used only in coarse attitude determination operations, they can be powered off during longer observations to conserve HST battery power.

FHST Shutter Control

The PDU circuits that close the FHST shutters can be alternately powered externally by the orbiter via SSP-commandable relays in the IPCU. Power for these units is then fed through the main umbilical to terminal board 101 on the HST AS aft bulkhead, which distributes it directly to the corresponding PDU circuits. The FHST shutters are externally closed by SSP switch S15 command to IPCU relay K5. Closure of the commanded IPCU relay by this switch is indicated by SSP talkback DS15. This function is shown schematically in Drawing 2.7 sheet 2.

Command and Data Interfaces

The FHST interfaces with the DMU for all serial digital commands from, and data to, the 486 flight computer. Temperature and star magnitudes, the only analog measure-ments from the trackers, are monitored through DIU-4. The DMU provides a dedicated redundant interface to the FHST through the FHSTI modules (A and B) described in Section 2.5. The FHSTI buffers star tracker data for subsequent processing by the 486, and all FHST commanding must pass through the computer prior to being distributed to the units. FHST digital data are, therefore, not available to the STOCC in DMS modes not supported by the 486.

FHST Pointing Restrictions (Shutters Open)

The star trackers are restricted from viewing any object (Sun, Earth, Moon) brighter than the brightest viewable star. Orientation of the HST during on-orbit operations with the space shuttle are therefore restricted, regardless of the status of the FHST shutters. In the event that one or more of the FHSTs are powered off with the shutters open, the minimum angle between an FHST boresight and the Sun vector is 24.5° . However, if any FHSTs are powered on with the shutters open, the following restrictions apply to the telescope attitude:

- a. Earth
 - 1. The minimum angle between an FHST boresight and the bright Earth limb is 32°.
 - 2. The minimum angle between an FHST boresight and the dark Earth limb is 16°.
- b. Moon The minimum angle between an FHST boresight and the bright Moon limb is 15°.
- c. Sun The minimum angle between an FHST boresight and the Sun vector is 40°.

In addition to these restrictions, use of the payload bay floodlights may be constrained during maintenance mission berthing operations if an FHST shutter fails.

FHST Pointing Restrictions (Shutters Closed)

The FHST shutters provide bright object protection of the tracker optics and electronics. However, direct exposure of the shutters to sunlight for more than 45 minutes in any 60-minute period could damage any powered-on FHST and result in loss of that FHST. Therefore, orientations of the HST with respect to the Sun during maintenance mission operations are constrained to keep the Sun vector outside of a 25° cone about each star tracker boresight. There are no pointing constraints if the FHSTs are powered off and the shutters are closed. This page intentionally blank



2.7.1.3 Coarse Sun Sensors

The Coarse Sun Sensors (CSSs) are small electro-optical devices that use silicon diode detectors to sense the presence of solar radiation. They provide Sun acquisition control signals during the initial deployment phase, produce APERTURE DOOR CLOSE commands from the PSEA when the Sun vector is within 35° of the HST boresight, and provide Sun direction signals during safe-mode operations to orient the vehicle for maximum illumination of the solar arrays to prevent battery depletion. A total of five individual CSS modules of two types (-501 and -503) make up the CSS system. Two type -503 modules (CSS no. 1 and no. 2) are mounted on the HST LS along the V2 axis (i.e., each side of the AD) and aligned to view along the +V1 axis. These sensors primarily provide the Monitor and Control Electronics (MCE) with Sun position information for bright object protection of the OTA optics. Three type -501 modules are mounted together on the AS aft bulkhead, with one aligned along the +V3 (CSS no. 3), the second aligned at 45° between -V3 and -V1 (CSS no. 5), and the third aligned along the -V1 (CSS no. 4). These sensors are used for capturing the Sun during initial deployment and safe-mode operations. Figure 2.7-6 depicts a CSS module and shows its locations and FOV orientations on the SSM. A photo of the HST (Figure 2.7-7) shows the forward CSS installation on the SSM LS.

Each CSS includes three monitor sensors and two control sensors. The control sensor detectors of all five units in the CSS system provide spherical coverage about the HST for coarse determination of vehicle orientation with respect to the Sun. Each sensor provides an angular analog Sun position error signal within its FOV. This position can be combined with the output of the other four CSS control sensors to determine the Sun position for maneuvering the HST during Sun-pointing operations.

The monitor sensor FOV is restricted by a fused silica window with a specific aperture corresponding to either $\pm 35^{\circ}$ for the -503 or $\pm 41^{\circ}$ for -501. The FOV of the type -503 monitor sensors corresponds to the minimum angle allowed between the Sun vector and the HST optical boresight (+V1 axis). The -501 monitor sensors resolve ambiguity in the control sensor output and provide input signals upon which the final determination of Sun capture is based. The control sensors of the -501 units are equipped with a small shade to eliminate errors caused by spacecraft reflections.

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Figure 2.7-6. Coarse Sun sensors

2.7-20



Figure 2.7-7. LS/FS (MT bars, magnetometers, CSS)

Physical Description

The CSS measures approximately 2 by 1.5 inches, weighs 0.3 pounds, and requires no input power. Each module consists of three identical monitor sensor subassemblies and two identical and independent control sensors. Each monitor subassembly contains a photodiode detector under a defining aperture and a fused silica window. Each control sensor contains four photodiode detectors with both sets of detectors mounted on the inclined faces of a single ceramic substrate pyramid. The axis of the pyramid defines the sensitive axis of the control sensor. The entire control sensor subassembly is covered by a fused silica dome.

Command and Data Interfaces

The silicon photodiode detectors in the CSS generate a small current, on the order of 680 microamps, when exposed to a full one-Sun level of illumination. As the angle of the incident radiation increases with respect to the normal chip surface, the detector generates correspondingly less current. These output currents are monitored by the MCE portion of the PSEA, which conditions the signals for use by the PCS and monitors the forward CSSs (CSS no. 1 and no. 2) for bright object detection. The PSEA is discussed further in Section 2.7.3.2. In its capacity as a simple solar radiation detector requiring no external power, the CSS has no direct command interface.

Bright Object Detection

The MCE includes a voter circuit for each -503 CSS (nos. 1 and 2), as shown in Figure 2.7-8. This circuit provides threshold detection logic to sense a CSS monitor sensor output corresponding to a Sun orientation within 35° of the HST optical boresight. If the BOD is enabled under such conditions or if the Safe-Mode Computer (SMC) issues a GGM initiate command, the MCE will issue aperture door close commands to MCU-A and -B.

2.7.1.4 Magnetic Sensing System

The HST MSS comprises two three-axis magnetometers to sense the magnetic field of the Earth along each of three orthogonal axes. Only one magnetometer is required for all operations; the second provides for redundancy. The MSS provides vehicle orientation data with respect to the Earth's magnetic field lines, which generally run from north to south. This information is used for coarse vehicle attitude determination (such as during deployment) and as input to the momentum management control law (in the flight software) for calculating the required torquing currents to the MTS for RWA desaturation. In addition, the MSS can provide input on magnetic field effects to the summing point within the flight software or PSEA firmware that produces RWA torque commands. A 486 software model of the magnetic field of the Earth provides an alternate source for these data within the PCS and may be used interchangeably during normal operations.



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Figure 2.7-8. BOD operation (typical)

Physical Description

Each magnetometer consists of a sensor unit and an Electronics Unit (EU). The sensor unit consists of three sensing coils (one per axis), while the EU contains the oscillator for sensor excitation and phase detectors with associated amplifiers to drive the magnetometer output. The sensor unit measures 4.6 by 2.75 by 2.76 inches and weighs approximately 0.8 pounds. The EU measures 5.63 by 5.56 by 1.9 inches and weighs approximately 1.6 pounds. They are matched sets that are not interchangeable. The units are mounted to a common bracket, which provides alignment of the sensor unit with the vehicle axes when installed on the HST. Both magnetometer systems are mounted to the SSM LS and are aligned with the HST vehicle axes as shown in Figure 2.7-9. The photo in Figure 2.7-7 shows the MSS installed on the LS of the flight vehicle.

During the SM-1 mission, the original magnetometers were replaced with new identical units. The replacement units were installed over the existing units via the mounting bracket and make use of the MSS heaters for thermal control as shown in Figure 2.7-10.

Electrical Power/Thermal Control

The magnetometer power interface is with PDU-1 and -2 for MSS-1 and -2, respectively, as shown in Drawing 2.7 sheet 3. Each magnetometer consumes approximately 1.4 watts, and it is expected that one unit will be powered continuously during orbital operations. The EU of each MSS is thermally conditioned by two 3.75-watt strip heaters and thermostats mounted on its case, and both are covered with MLI blankets. The thermostats have set points of -15° F for the primary heaters and -25° F for the redundant heaters. The heaters are powered as part of the LS/FS primary and redundant heater circuits, which are controlled by PDU relays in PDU-1 and -2. The sensor unit of each has neither MLI covering nor heaters.

Command and Data Interfaces

Each MSS provides two analog output signals per axis. One signal in each MSS is redundantly routed to each half of DIU-4, and the other is routed directly to the SMC (SMC-A and -B) of the PSEA.

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Figure 2.7-9. MSS mounting and orientation

2.7-25



Figure 2.7-10. SM MSS installation hardware



2.7.1.5 Rate Gyro Assembly

The RGA is a strapdown reference gyro package that senses vehicle motion using two modified ultra-low noise Rate Integrating Gyros (RIGs) to provide two-channel digital attitude and analog rate information. The resulting short-term attitude and attitude rate data are used during normal operations for fine pointing and spacecraft maneuvers. In addition, the RGA system is used by the SSM safing system for attitude and rate information during safe-mode operations if the RGAs are not the cause of safe-mode entry.

Physical Description

A single RGA is composed of an RSU, which houses the RIGs and an ECU. Each RSU/ECU set provides redundant electronic channels (channels 1 and 2) and RIGs. Each RSU measures 12.0 by 10.0 by 4.8 inches and weighs 23.7 pounds. The RSU enclosure is equipped with an EVA handle and tether loops for on-orbit changeout. The enclosure mounts to the SSM ES within the AS along with the FHSTs as shown in Figure 2.7-11. Three ORU bolts retain the RSU to the ES, aligned with guide pins in the base of the unit. The RSU is shown in Figure 2.7-11 prior to installation into the flight vehicle. This figure also provides an overview of the RSU enclosure and internal arrangement. RSUs 2 and 3 were replaced during the SM-1 mission. The replacement RSUs were identical to the old RSUs except for the hybrid circuit systems. The Bendix pulse rebalance loops were replaced by Teledyne hybrid circuits. All three RSUs were replaced with new units during the SM-3A mission

The ECUs are mounted to the tunnel of the SSM ES bay 10 by four ORU bolts per unit. Each ECU enclosure measures 11.0 by 9.0 by 7.5 inches, and the unit weighs 17.4 pounds. EVA tether loops are provided for retention during on-orbit changeout. The ECU contains two independent sets of control electronics on a total of eight printed circuit boards (four per channel). These boards wire wrap to six connectors (three per channel) on the exterior of the unit. In bay 10, the mounting provisions for each unit include ORU connector plate interfaces, allowing automatic alignment and interface of all connectors through a single ORU bolt. The ECUs are shown installed in bay 10 in Figure 2.7-12, which also gives an overview of the ECU enclosure and internal arrangement. ECUs 1 and 3 were replaced during the SM-1 mission.

The RGA system is classified as block I ORU.

Electrical Power

The total rated steady state RGA power required, exclusive of heater power, is 18 watts per channel (36 watts per RSU) (Figure 2.7-13). Power is supplied to each ECU channel through hardwired PDU circuits protected by 5-A fuses in PDU fuse plug P15. RGA-1 and -3 receive their channel 1 power from PDU-1 and channel 2 power from PDU-2. RGA-2 receives channel 1 power from PDU-3 and channel 2 power from PDU-4. Power to the RSUs is controlled by relays in the dc power supply of the ECU, which control dc-to-dc converter output (Drawing 2.7 sheet 4).



Figure 2.7-11. RGA rate sensor unit (RSU locations and installation)









Figure 2.7-13. RGA electronic control unit



Thermal Control

Each RSU RIG includes an internal heater that maintains the gyro at 134.6° F during normal operation. The RSU also contains two survival heaters, bonded to the RSU housing, that maintain the gyro at or above its 39.92° F survival temperature during periods when the RSU is powered off. These heaters are powered from the orbiter during the deployment mission from prelaunch until main bus powerup. The orbiter also provides this service if the servicing mission flight plan should call for deactivation of the HST main bus during an EVA, unless the ORU being changed out is the RSU. During mission phases when the main buses are powered, each survival heater element may be separately switched on by command to its corresponding PDU relay. The external and internal power sources to the RSU survival heaters are described in Section 2.4.2.

The RSUs are the most thermally time-critical items on the HST vehicle, in that the unit cools down quickly if not provided with operating or survival heater power. During RSU changeout operations on a servicing mission, the RSUs are without power for an extended period. RSU survival heater power should, therefore, be restored to the unit as soon as practical to avoid damage to the RIGs.

RSU Survival Heater Control

The PDU circuits, which power the RSU survival heaters, can be alternately powered externally by the orbiter via SSP-commandable relays in the IPCU. Power for these units is then fed through the deployment umbilical to terminal board no. 101 on the HST AS aft bulkhead, which distributes it directly to the corresponding PDU circuits. The RSU survival heaters are externally powered by SSP switch S16 command to IPCU relays K16 and K17. Closure of the commanded IPCU relays by this switch is indicated by SSP talkback (tb) DS16. This function is shown schematically in Drawing 2.7 sheet 2.

Command and Data Interface

Commands from the DMS and safing system are all routed to the ECU for processing and distribution to the RIGs in the RSU. The ECUs provide vehicle body rate data to the RGAI of the DMU, which contains six independent interface channels for the six RIGs (two per RGA). The DMU provides redundant RGAI modules. Each RGA channel outputs a pulse-width modulated rate gyro signal and clock on its corresponding RGAI channel, for subsequent processing by the 486 flight computer. The computer also issues the RGA high-/low-rate mode commands to each RGA via this interface. All other functions interface directly with a DIU or the PSEA.

EVA RSU Changeout

The RSU changeout operation is identical for each RSU.

Before EVA changeout, the FHST shutters should be verified closed from SSP tb DS15 and the RSU deactivated by commanding OFF the RSU control and survival heaters as well as the rate sensor and support electronics. Commands are sent from the DIU and DMU and may be sent from the ground or the crew, if the crew is given the command and monitoring capability on the crew spec. The RSU survival heater power can be commanded ON/OFF from SSP sw S16 and power status can be verified from SSP tb DS16.

At the -V3 aft shroud worksite, FHST 2 and 3 shade seals are retracted by driving the three fasteners 5-1/2 turns Clockwise (CW), as shown in Figure 2.7-14. The latches on the -V3 aft shroud doors are disengaged by backing out each door latch and pivoting the latch back out of the way (Figure 2.7-15). The handle latch is disengaged and the doors are opened and tethered, if required, at the padded stop position. An overview of the RSU 1 location is provided in Figure 2.7-16.

The portable worklight assembly can be installed and powered from SSP sw 24, ifnecessary. A PFR can be installed inside the aft shroud at FR29 to assist the EVA crew if necessary.

Once the power connectors are deadfaced, the two wingtabbed connectors are disconnected, the three captive bolts are disengaged, and the RSU is removed. The replacement RSU is tilted slightly for easier installation. Two alignment corner brackets and a Velcro strip aid in seating the RSU properly (Figure 2.7-17). The three captive bolts, as shown in Figure 2.7-18, are re-engaged, and the two power connectors reconnected. The three bolts require 9 pounds of push to engage the threads and are torqued to between 4.6 and 6.3 ft-lb each.

If used, the PFR is removed, the worklight powered off and removed, and the doors closed. The left door (-V3/-V2) is fully closed, and the right door (-V3/+V2) is closed so that the handle latch engages. Once the handle latch is latched, the remaining latches are pivoted back into position, relatched, and torqued appropriately.

The changeout procedure concludes with re-engaging the FHST 2 and 3 shade seals by driving all three fasteners 5-1/2 turns Counterclockwise (CCW).

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Figure 2.7-14. FHST 2 light shade fasteners



Figure 2.7-15. AS (-V3) doors and latches



Figure 2.7-16. RSU 1 location overview



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Figure 2.7-17. RSU mounting plate



Figure 2.7-18. RSU, bottom view

EVA ECU Changeout

The ECU changeout operation is identical for each ECU.

Prior to the beginning of the EVA changeout, the ECU is deactivated by commanding OFF the RSU control and survival heaters and the rate sensor and support electronics. Commands are sent from the DIU and DMU and may be sent from the ground or the crew if the crew is given the command and monitoring capability on the crew spec. The crew can verify heater status from SSP tb DS16.

At the -V2 SSM ES bay 10 worksite, the six bay 10 door J-hook bolts are released, and the J-hooks disengaged. The bay 10 doors are opened to the full open position (Figure 2.7-19). The bay 10 door has an integral door stay that opens into a detent position when the door is in the full open position.

The ECU is secured by four keyway slot bolts (Figure 2.7-20). Removal of the ECU is accomplished by backing off these bolts five or six turns, disconnecting the connector drive mechanism (Figure 2.7-21), and sliding the ECU through the keyway slot to the bolt hole opening for removal. The empty mounting bracket and connectors are shown in Figure 2.7-22.

The replacement ECU is installed into its mounting position in the keyway slots, the connector engaged, and the connector drive mechanism torqued. The four keyway bolts are then tightened and torqued to approximately 5 ft-lb, and the power to the ECU and RSU heaters reapplied. The bay 10 door is closed, the six J-hooks engaged, and the J-hook bolts torqued appropriately.
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Figure 2.7-19. Bay 10 overview

JSC-29029 Basic, Rev. A



sfoc_fl2705_093.tif

Figure 2.7-20. ECU closeup view



Figure 2.7-21. ECU electrical connector EVA drive mechanism

2.7-45



SIOC_112705_094.

Figure 2.7-22. ECU mounting bracket

2.7.2 PCS Actuators

2.7.2.1 Reaction Wheel Assemblies

Variable speed reaction wheels function as momentum exchange devices in the HST PCS to damp initial orbiter RMS tipoff rates; to effect spacecraft attitude control during nominal HST operations, including inertial hold and Sun-point safe modes; to provide control torques to produce vehicle maneuvering (slewing) for change of target; and to compensate for external disturbance torques, including GG and aerodynamic influences, acting on the HST. RWA torque control is nominally exercised by the PCS software in the 486 flight computer in response to STOCC commands and PCS sensor data. This software is described in Section 2.7.3. During hardware safe-mode operations, the PSEA takes command of the RWA system (if operational) and provides analog torque commands in response to the remaining PCS sensor data. The safe-mode operation of the PSEA is described in Section 2.9.

All four reaction wheels are kept running at all times, even though any three are sufficient to provide full three-axis attitude control. The actual vehicle control torques are produced when the speed of one or more wheels is changed in response to either gyro control signals or ground commands and the control laws and transformation matrices contained in the 486 flight software. This change in speed results in a change in angular momentum of the wheel, producing a corresponding reaction torque on the vehicle that causes it to rotate and effect the required spacecraft attitude change.

External vehicle disturbances, such as aerodynamic drag, produce gradual vehicle torques that must be compensated for by the reaction wheels. This results in steadily increasing wheel speeds, requiring a periodic reduction in wheel angular momentum, or desaturation, accomplished through the generation of alternate vehicle torques using the MTS. The MTS interacts with the local magnetic field of the Earth to produce the desaturation torques under control of the 486 flight software using either a magnetic field model or output of the MSS.

Physical Description

The HST is equipped with four RWAs, mounted in a way that allows any three to provide full three-axis attitude control. Two wheels each are housed in SSM bays 6 and 9, as shown in Figure 2.7-23. Both bays are separated by 90° about the HST optical axis. The two wheels of a single SSM bay are canted off the V2-V3 plane by 20°, one toward the +V1 and one toward the -V1. They are fixed to their mounting hardware through three ORU bolts acting through corresponding keyway slots on the outer wheel diameter. SSM bays 6 and 9 are equipped with curved honeycomb doors for closeout.



Figure 2.7-23. RWA detail installation (SSM bay 6)

Each unit measures 21 inches axially and 24 inches in diameter and weighs approximately 103 pounds. A single RWA is depicted in Figure 2.7-24. The RWA design includes a large rotating flywheel, torquing motor, tachometer, rotor control electronics, and power control electronics. Nominal dynamic torque range is 0.003 to 0.605 ft-lb (0.004 to 0.7 N·m), with a maximum wheel speed of ± 3000 rpm. The RWA may be driven to 150 percent of the maximum wheel speed without damage.

Electrical Power

The input power required by an RWA is variable, depending on the degree to which the wheel speed is to be changed, up to a maximum of 400 watts. A nontorquing/ nonspinning wheel draws less than 15 watts of power. Each unit receives unregulated dc power from PDUs 1 through 4 for RWAs 1 through 4, respectively. Unlike most SSM systems, the RWA power source may be discretely chosen from any of the three main buses (A, B, or C). PDU relay K6 selects, by DIU command, either main bus A or B for nominal operations, during which two wheels are powered from bus A and two from bus B to equalize loading. (Most other SSM components receive power from buses A and B in parallel.) PDU relay K7 selects, by CDI special discrete, main bus C for its corresponding RWA in the event of a failover to that bus during contingency operations.

Thermal Control

With only a three-point mounting on each RWA, the thermal design provides for the dissipation of 75 percent of internal RWA heat by radiation. This is facilitated by the use of a black thermal exterior finish and by the use of the curved honeycomb door panels on their corresponding SSM bays. Thermostatically controlled heaters are provided to maintain the wheel bearings within operational temperature limits. These heater systems are redundant and enabled by thermostats on the RWA.

Command and Data Interface

During nominal operation, the RWAs exchange digital commands and data with the 486 flight computer through the DIUs. Specifically, RWAs 1 and 2 interface with DIU-4, and RWAs 3 and 4 interface with both DIU-4 and -5. Serial digital RWA torque commands are generated by the flight software control laws based on the RWA status and PCS sensor data. The four RWAs also interface with the PSEA for safe-mode operations and are controlled using analog torque commands and wheel speed measurements.





2.7-50

2.7.2.2 Magnetic Torquing System

The MTS consists of the Magnetic Torquer Electronics (MTE) and four fixed-position electromagnets or torquer bars. The MTS is used by the HST PCS for angular momentum desaturation of the reaction wheels and for coarse vehicle attitude control during contingency or safe-mode operations. Under 486 flight software or PSEA control, the current to the MTs produces a local magnetic field which reacts against the magnetic field of the Earth to counterbalance torques experienced by the HST that are caused by nonalignment of the vehicle inertia axis with respect to GG and aerodynamic forces. The MT currents are produced by 486 software commands based on total system momentum plus data from the MSS or from a software model of the magnetic field of the Earth.

Magnetic Torquers

Each of the four MT coils is wound on a cylindrical iron core measuring approximately 100 inches long by 3 inches in diameter. Each unit weighs approximately 95 pounds. At rated current, each unit produces a magnetic moment of ±2800 ampere-meter squared. The MT bars are mounted externally on the SSM FS, oriented at 45° to the V2-V3 axes, and 35.25° above the V2-V3 plane. MT bar installation is depicted in Figure 2.7-25.

Magnetic Torquer Electronics

The MTE provides control currents to the magnetic torquer bars in response to magnetic moment commands from the 486 flight computer or SMC specifying current amplitude and polarity. The 486 flight software issues digital command messages to the MTE during nominal operations, based on total system momentum (RWAs plus vehicle) and either MSS data or a software-resident model of the magnetic field of the Earth. During safe-mode operations, the MTE receives analog commands from the active SMC.

The MTE is a partitioned section of the PSEA, which is discussed in Section 2.9. Two identical MTE channels are provided for each of the four MTs (an A and B circuit for each torquer bar). Only a single MTE channel per MT is required at any one time, and selection is controlled by STOCC command.

Electrical Power

The operating power to the MTS is provided through the MTE and is divided among the four PDUs. PDU-1 provides service to channels 1A and 3A; PDU-2 powers channels 1B and 3B; PDU-3 supports 2B and 4B; and PDU-4 powers 2A and 4A. Each PDU circuit is a 3-amp, fused, hardwired feed. Each MTE/MT circuit dissipates a maximum of 16 watts.



22063X716.ART;1

Figure 2.7-25. Magnetic torquing system

Command and Data Interfaces

Serial digital magnetic moment commands generated by the 486 flight software during normal operations are routed to the MTE channels through DIU-4 and -5. DIU-4 interfaces with all MTE A channels, and DIU-5 commands the B channels. The STOCC selects A or B channels by means of DIU HLDs, which apply or remove power to individual MTE channels. A digital command from a DIU is sent through a 12-bit shift register to a Digital-to-Analog (D/A) converter. The resulting analog output of the D/A converter is amplified by a power amplifier to provide the MT current.

During hardware safe-modes, the active SMC powers on preselected MTE channels, as specified in the PSEA configuration memory, and enable analog magnetic moment command processing by the MTE. The PSEA configuration memory is described in Section 2.9. The analog command signals from the SMC are provided to the same power amplifier as the D/A converter for amplification to drive the MT bar load. The STOCC may disable the analog magnetic moment command processing by CDI special discretes to recover from safe-mode operations.

2.7.3 PCS Control Systems

2.7.3.1 486 Flight Software

The primary SSM element that processes the PCS sensor data and controls the PCS actuators is the PCS software modules, executed in the 486 flight computer of the DMS. These software modules provide for the calculation of the PCS control law, attitude updates, the momentum management control law, and the command generator.

Command Generator

The command generator is used for large and small angle maneuvers and to generate constant vehicle rate profiles. This software module shapes the acceleration and incremental angle commands to the control system to limit angular acceleration loads on delicate vehicle structures, such as the SAs, HGAs, and OTA elements. The command generator provides jerk (step function) pulse shaping comparable to a (1 - cos) shape profile. The shaped jerk pulse provides a smoothed acceleration input to the control system, which translates into torque commands to the RWAs shaped to minimize vehicle perturbations. The duration of the jerk pulse is sized to end when the desired constant vehicle angular velocity is achieved, at which time an opposite polarity jerk pulse is applied to null the wheel torque commands to zero. This cycle is repeated in similar fashion to stop the slew maneuver to achieve the desired HST attitude.

Control System

The command generator acceleration and incremental position commands are provided to the control system software together with the RGA incremental rate data. The control system uses position, rate, and integral compensation data and also a rate filter to stabilize high frequency bending modes. An estimate of external torque is used to

eliminate cross-coupling between axes, effectively decoupling V1, V2, and V3 control torques. The acceleration command is routed directly to the RWA torquer to apply slewing torques to the vehicle. A feedback loop provides an error calculation path to account for variations in parameters, such as the vehicle inertia estimate and RWA feed forward gain.

The control loop is a high-gain system and can saturate for unsmoothed external torques, attitude updates, or movement of FGS star selections. Therefore, all inputs to the control system must be smoothed by the command generator in order to prevent loop saturation and the resulting HST instability from initiating a transition to safe mode. Recovery from safe-mode can be a time-consuming process that reduces mission science time and available SA exposure time to sunlight.

Momentum Management

A cross-product law is provided for momentum management. The closed-loop characteristic of the cross-product law utilizes sensed tachometer and magnetometer outputs to generate magnetic commands for RWA desaturation. Cross-product law is used in inertial hold and software Sun-pointing modes, and a similar algorithm is incorporated into the PSEA firmware for use during hardware Sun-point safe-mode.

Attitude Updates

Attitude reference updates during normal PCS operating modes are provided by the FHST and FGS. If the attitude error determined from the FHST data is larger than some database value (nominally 30 arcseconds), the software autonomously maneuvers to correct the error. If the error is less than the database value, the error is corrected by means of an attitude observer algorithm.

Fault Detection and Isolation

The flight software tests a number of software, PCS, EPS, and DMS parameters and takes corrective action if any are not within specified limits. When a test determines that the monitored parameter exceeds the associated threshold a specified number of times or for a specified duration, a predetermined response and associated action, known as a safe-mode macro, is executed by the VSS portion of the flight software. For all tests, the operational limits, number of consecutive time periods out of limits, and response are VSS database items. The capability is provided for the STOCC to individually inhibit the effective action of each test based on STOCC-provided data. The software tests can be activated or deactivated as a block. In addition, the capability is provided for temporary block deactivations of the functions, which autonomously reactivate themselves, after a time period, in the absence of further STOCC inputs. The software safe-mode monitors and the EPS, DMS, and PCS parameters are discussed in Section 2.9.

2.7.3.2 Pointing and Safe Mode Electronics Assembly

The PSEA supports nominal PCS operations and provides complete vehicle control during hardware safe-mode. The unit consists of three functional elements in the same enclosure. Two of these elements, the MCE and the MTE, each operate as part of the nominal PCS system. The third element is the SMC, which is nominally passive. When significant vehicle failures occur that could threaten the spacecraft, the SMC is brought on line and takes action to place the HST in a safe configuration until the STOCC can intervene. The safing system functions of the PSEA, specifically the SMC, are discussed completely in Section 2.9.

Monitor and Control Electronics

The MCEs (A and B) are independent electronics sections, each of which has the capability to command the AD closed when the optical axis of the vehicle is in close proximity to the Sun, to process Sun angle data for use by the DMS and SMC, to monitor the health of the DMS and EPS as indicated by the presence of KA signals, and to store vehicle configuration data for use in hardware safe-mode to execute the load control strategy.

KA Monitoring

The 486 flight software generates KA signals that are output to the DMU CIF module every 250 ms. The CIF module decodes the signals and issues KA-1 and KA-2 pulses to the PSEA. These pulses are monitored by a dedicated circuit in both MCE-A and -B as shown in Figure 2.7-26. Loss of KA-1 or -2 for greater than 2.7 seconds causes activation of the SMC, which will autonomously take command of the vehicle.

KA monitoring may be enabled/disabled by STOCC command as required. Two separate levels of control for KA monitoring are provided, known as a HARD ENABLE/DISABLE and a SOFT DISABLE. Once the HARD ENABLE is issued from the STOCC, the SOFT DISABLE command may be issued to inhibit KA monitoring for 91 seconds. The SOFT DISABLE command may be reissued within the 91-second period to reinitialize the SOFT DISABLE timer circuit. During deployment operations, KA monitoring is disabled in this fashion until confidence in the HST DMS and PCS is established. This requires a continuous communications link to the STOCC during the initial postrelease phase.



Figure 2.7-26. Keep-alive signal processing

2.7.4 PCS Operating Modes

The PCS is capable of operating in three basic modes, each of which involves several submodes. These basic modes are normal, Sun-point, and drift modes (zero torque commands to RWAs and MTs). All modes are initiated from the STOCC. A summary of these PCS and attitude control modes, specifying the PCS sensors, actuators, and control systems utilized in each, is provided in Table 2.7-1. The submodes of interest during joint HST/STS operations are briefly summarized in the following paragraphs.

Mode	Purpose	Minimum H/W	Worst-case attitude error	Worst-case rate
Normal	Science operations	486, 3 RWA, 3 RGA, 3 MTE, 1 MSS	Arcseconds	Negligible
Drift	No active attitude control for short periods; used for grapple and release		Tipoff rate*time or error from previous mode	Tipoff rate
Software Sun-point Align +V3 axis with Sun vector under 486 control		486, 3 RWA, 3 RGA, 3 MTE, 1 MSS	6° (due to Earth albedo). Attitude about Sun line is arbitrary	0.007 deg/sec
Zero gyro software Align +V3 axis with Sun Sun-point vector under 486 control following loss of two gyros		486, 3 RWA, +V3 = 50° 0 RGA, 3 MTE, (at end of eclipse) 1 MSS		0.15 deg/sec
Spin stabilized	Stabilize HST by spinning about V1 axis under 486 control following loss of two RWAs	486, 0 RWA, 3 RGA, 3 MTE, 1 MSS	5° 20° 50° 60° 180°	0.8 deg/sec 0.5 deg/sec 0.3 deg/sec 0.2 deg/sec 0.? deg/sec
Zero gyro spin stabilized* Stabilize HST by spinning about V1 axis under 486 control following loss of two RWAs and two RGAs		486, 0 RWA, 0 RGA, 3 MTE, 1 MSS	TBD	TBD
Hardware Sun-point Align +V3 axis with Sun vector under PSEA control		PSEA, 3 RWA, 3 RGA or RMGA, 3 MTE, 1 MSS	6° (due to Earth albedo). Attitude about Sun line is arbitrary	0.007 deg/sec
Gravity gradient	Align V1 axis with local vertical for retrieval under PSEA control following loss of two RWAs and 486	PSEA, 0 RWA, 3 RGA or RMGA, 3 MTE, 1 MSS	18°	Orbit rate +0.02 deg/sec

Table 2.7-1.	Summary of PCS	S modes
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Note: Refer to Section 2.9 for description of each of the PCS operating modes.

Deployment/Redeployment and Initial Attitude Determination

The HST PCS is placed in drift mode for release from the orbiter RMS. Once a safe distance from the orbiter is achieved, ensuring no recontact, the PCS will be commanded by STOCC to +V3 Sun-point mode. In this mode, the system will damp any vehicle error rates generated by the RMS at release (0.1 deg/sec maximum in each axis), while aligning the +V3 axis to within 2° of the Sun vector.

Once Sun-point mode is verified, the system is capable of performing a $\pm 180^{\circ}$ roll about the V1 to acquire and lock on to a reference star for attitude hold using the FHSTs. The total attitude determination procedure includes the following three basic steps:

- 1. Achieve +V3 Sun-point mode
- 2. Acquire key star when the vehicle is oriented so that a selected star is within the FOV of FHST-1
- 3. Acquire a second star requiring a roll about the eigen vector of star number 1 to the orientation that places a second reference star in the FOV of another FHST

Key stars are those that can be isolated from similar neighboring stars by a minimum of 4.8° to allow the FHST to lock on to them. Once coarse and fine attitude determination is completed, the STOCC puts the HST through a series of test maneuvers to verify proper operation of the BOD system prior to opening/reopening the aperture door.

Maneuvering

In this mode, the HST is slewed from one orientation to the next on command from the STOCC (real-time or stored program command). The PCS actuators have a momentum exchange capability sized to perform a 90° slew in less than 18 minutes. The maneuvers are executed by the command generator module of the 486 flight computer flight software.

The command generator accepts time-tagged HST eigen axis commands that were previously uplinked and stored in 486 memory and issues the required PCS actuator commands, as described in Section 2.7.3.1. Target-to-target slew commands are stored in the flight computer memory to implement consecutive maneuvers, allowing the desired time in the fine-pointing mode between maneuvers. Large angle maneuvers are either velocity limited, acceleration limited, or jerk limited, depending on the maneuver angle. The reaction wheel maximum speed attained during a maneuver is determined from the initial vehicle momentum conditions and the maximum HST angular velocity of 0.22 deg/sec.

2.7.5 Safety Considerations

A failure in the HST PCS can result in the spinning up of the RWAs, providing the potential for a collision with the orbiter during rendezvous and proximity operations (retrieval, berthing, and deployment). The concern exists when the HST is still a free-floating satellite, as well as when the HST is grappled by the RMS. Analysis has shown that the maximum torque exerted by the HST in a worst-case multiple failure (all reaction wheels on, maximum torque from each wheel) is not capable of damaging or inducing movement into the RMS with the RMS in any mode or configuration. The maximum rotation rate of the HST, prior to grapple or shortly after release if all reaction wheels failed ON at maximum torque output (worst-case condition), does not result in collision/recontact with the orbiter.

Analysis shows 2.8 minutes are required to rotate HST 45° with solar arrays deployed. Recontact/collision would result if no orbiter rendezvous breakout occurred with greater than 45° of HST rotation. Based on the orbiter rendezvous breakout rate of 1.0 fps, no recontact occurs if maximum RWA torque is applied during proximity operations, given a 1-minute delay due to crew response time.

Analysis shows the applied torque computed in a worst-case scenario is 11.0 ft-lb, far below the minimum amount of torque (310 ft-lb) required to backdrive any RMS joint (motor backdrive torque).

2.8 THERMAL CONTROL SUBSYSTEM

The HST TCS is designed to control temperatures of SSM components and the structure that interfaces with the OTA and SIs during all mission phases. The SSM uses passive thermal control design consisting of insulation, component arrangements, and mounting configurations augmented with thermostatically controlled heaters. White paint or clear anodized aluminum surface finishes are used where possible to meet high-emittance requirements for all internal surfaces and components to enhance lighting for on-orbit maintenance. Bimetallic louver systems are used on the two-battery bays (bays 2 and 3) to reduce heater power requirements.

The baseline SSM thermal design is depicted in Figure 2.8-1.

2.8.1 TCS Elements

2.8.1.1 Multilayer Insulation/Surface Finishes

MLI blankets are used on over 80 percent of the external surface area of the HST. These blankets have an outer layer that is reflective to energy in the solar spectrum to minimize the effect of orientation on satellite temperature. The MLI blankets insulate the structure from the external thermal environment, thus conserving heater power and minimizing temperature gradients.

The baseline MLI design is 15 layers of embossed Double Aluminized Kapton (DAK) with an outer layer of 5-mil surface aluminized 1/3-mil Fluorinated Ethylene Propylene (FEP) Teflon and an inner layer of 1-mil single aluminized Kapton. The surface properties of this outer layer will control external blanket temperatures to 20° C or less. The expected degradation of thermal control surfaces (increase in solar absorptance) is given in Table 2.8-1 for typical materials used on the HST. See Figure 2.8-2 and Drawing 2.8 sheet 1 for MLI details.

Long-term degradation in orbit is caused by a combination of atomic oxygen erosion solar radiation damage and surface contamination. The thermal control materials may be degraded seriously when exposed to solar radiation consisting of UV radiation, lowenergy proton charged particles, low-energy electrons, and oxygen ions. Long-term contamination degradation is attributed to a gradual buildup of contaminants in orbit and/or a change in the optical properties of the contaminant caused by radiation exposure. The source of the contaminant is attributed to spacecraft outgassing products, decomposition of materials exposed to radiation damage, and proximity to the orbiter during the deployment and maintenance missions. The combined effect of these damaged components is difficult to predict in terms of increasing solar absorptance because of limited laboratory simulation data and lack of radiation environmental data in the range of interest. The most reliable predictions are based on a combination of available flight experience and laboratory experimental data and returned hardware measurements.

The effect of increased solar absorptance is a slight warming trend and an increased sensitivity to satellite orientation with life.



Figure 2.8-1. SSM baseline thermal design

Thermal control surface	At beginning of orbit	After 5-yr orbit and solar albedo exposure only	After 5-yr orbit and 11,718 ESH solar UV exposure (typical)	After 5-yr orbit and 19,207 ESH solar UV exposure
Silverized FEP Teflon film second surface mirror	0.05 - 0.09	0.10 0.14	0.12 0.18	0.15 0.22
Aluminized FEP Teflon film second surface mirror	0.10 0.14	0.15 0.19	0.19 0.23	0.20 0.26
Aluminized Kapton film second surface mirror (2 mil)	0.37 0.43	0.42 0.49	0.47 0.55	0.52 0.61
Black Chemglaze polyurethane paint	0.92	0.90	0.88	0.84
	0.98	0.96	0.94	0.92
Aluminum foil tape	0.12	0.12	0.11	0.11
adhesive-backed	0.21	0.22	0.22	0.22
Aluminized Kapton film first surface mirror	0.11	0.11	0.11	0.11
	0.21	0.22	0.22	0.22
White Chemglaze polyurethane paint	0.23	0.33	0.40	0.48
	0.29	0.39	0.49	0.59
White low-gloss Tedlar	0.23	0.33	0.40	0.48
film	0.29	0.39	0.49	0.59

Table 2.8-1. Expected solar absorptance α_s range





2.8-4



JSC-29029

DSGN	SIGNATURE	DATE 10/05/89	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION LYNDON B. JOHNSON SPACE CENTER HOUSTON, TEXAS			
DR ENGR ENGR APP	T. ARNOLD		MULTILAYER INSULATION BLANKETS SSM LIGHT SHIELD/FORWARD SHELL			
APP			HST CSM	DWG NO 28		
CADS			ALL	2.0		
LTR PCN			126.5 X 34		SHEET 1 OF 4	

8 MLI SSM LT SHLD/FWD SHELL

NOTES: DULTILAYER INSULATION BLANKET

- 2 MLI ELECTRICAL BOND STRAP
- 3. DASHED RECTANGLES SHOW VELCRO LOCATIONS
- 4. X SHOW VENT LOCATION
- 5. SEE FIGURE 9.2-14 FOR MLI DAMAGE

475.64

493.70

FOOT RESTRAINT SOCKET

Α
 DATE:
 08/06/99
 NAME:
 MDU
 VERSION:
 REL;
 Z

 DATE:
 05/03/01
 NAME:
 MDU
 VERSION:
 REL;
 1

 DATE:
 01/14/08
 NAME:
 LAM
 VERSION:
 REL;
 2



JSC-29029



BAY 8 15°



BAY 9 45°









BAY 3 195°

SIGNATURE	DATE	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION				
GN B. TEDDER, JR.	8/31/89	LYNDON B. JOHNSON SPACE CENTER HOUSTON, TEXAS				
		MULTU	AVED INC			
GR T. ARNOLD		DIANKETS SUPPORT SYSTE				
GR		MODULE FOLLOMENT SECTION				
P		LINDULE I		SECTION		
P		HST CSM	DWG NO	0		
DS		ALL	<u>۲</u>	. 0		
R PCN		126.5 X 34		SHEET 2 OF 4		

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DETAIL Q ROTATED 90° CCW





DETAIL L ROTATED 90° CCW

D

Е

С

SHEET 3 OF 4

8 MLI OTA ES 3

	\$1	GNATURE	DATE	NATIONAL AERONAUTICS AND SPACE ADMINISTRATIC				
N	s.	EGAN-WYER	6/ 7/ 89	LYNDON B. JOHNSON SPACE CENTER HOUSTON. TEXAS				
				MULTI	AVED INSULATION			
R	т.	ARNOLD		BLANKETS OTA				
R								
				EQUI	PMENT SECTION			
				HST CSM	DWG NO			
s				ALL	2.0			

5. SEE FIGURE 9.2-14 FOR MLI DAMAGE 4. X SHOW VENT LOCATION

3. DASHED LINES SHOW VELCRO LOCATIONS

2 MLI ELECTRICAL BOND STRAP

NOTES: D MULTILAYER INSULATION BLANKET

В

A DATE: 08/06/99 NAME: MDU VERSION: REL;2 DATE: 05/03/01 NAME: MDU VERSION: REL;1 DATE: 01/14/08 NAME: LAM VERSION: REL;2











D

С

В

s	IGNATL	IRE	DATE	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION			
DSGN B.	TEDD	ER	08/18/89	LYNDON B. JOHNSON SPACE CENTER HOUSTON, TEXAS			
DR				мш т н		INCI	
ENGR T.	ARNO	LD		1 MULTILATER INSULATION			
ENGR				BLANKEIS AFT SHRUUD			
APP							
APP				HST CSM	DWG NO		0
CADS				ALL	2.0		
LTR		PCN		126.5 X 34			SHEET 4 OF 4

8 AFT SHRD 4

NOTES: NULTILAYER INSULATION BLANKET - ALL BLANKETS HAVE MATTE BLACK SURFACES FOR STRAY LIGHT ABSORPTION

- 2 MLI ELECTRICAL BOND STRAP
- 3. DASHED RECTANGLES SHOW VELCRO LOCATIONS
- 4. X SHOW VENT LOCATIONS

5 LIFT FLAP TO EXPOSE GROUNDING CLIP





2.8.1.2 New Outer Blanket Layer

During SM2, a significant portion of the HST MLI was damaged, predominantly on the HST +V3 "Sun" side. To protect the most heavily damaged areas, MLI patches were fabricated and installed on equipment section bays 8 and 10. In addition, tie wraps were looped together to form a baling system on ES bay 7 to hold damaged MLI in place. A more permanent solution was planned for SM3; i.e., installing a New Outer Blanket Layer (NOBL) over the existing MLI on the SSM bay 1 and bays 5 through 10, to prevent further MLI deterioration, to eliminate the MLI as a potential of source contamination, and to reduce the risk of thermal degradation of the observatory. During SM-3A, a New Outer Blanket Layer (NOBL) was installed for bays 1, 9, and 10. *During SM-3B, a NOBL was added to bay 6*.

Physical Description

Each of the NOBL sheets is made of 0.004-inch-thick stainless steel with a special thermal coating, measures approximately 60 by 48 by 1.5 inches, and weighs approximately 5 pounds. A picture frame of 304 stainless steel stiffens the edges. Each NOBL sheet has been sized and cut to match the coverage area of the MLI, since the hinges, doorknob, and vent hole positions vary from door to door. Figure 2.8-3, Figure 2.8-4, and Figure 2.8-5 show an overview of NOBL sheets for bays 1, 9, and 10, respectively.

The NOBLs are attached to the doors with expanding plugs in each of the four existing vent holes, shown in Figure 2.8-6. Bays 6 and 9 NOBL covers are attached to the SSM doors using integrated latches interfacing with the bay 6 or 9 door handle and existing door hardware. A hole through each plug ensures continuous venting of the SSM bays. To operate the plug, the plug is inserted through the NOBL and into a door vent hole, and the handle is pivoted 90°. This operates a cam action to translate the central shaft. The shaft has a cone on the end that engages ramps on 16 titanium fingers, which elastically flex outward to make contact with the existing vent spool. To remove the plug, the handle is pivoted back to the original position, which reverses the action described above. A grounding ring, with a spring clip inside to ensure contact, is pushed over a J-bolt head to electrically ground the NOBL.



Figure 2.8-3. Bay 1 NOBL configuration



Figure 2.8-4. Bay 9 NOBL configuration



Figure 2.8-5. Bay 10 NOBL configuration[k1]



jsc29029_043r1.cvx

Figure 2.8-6. NOBL expanding plug

2.8-16

2.8.1.3 Active Heaters

Electrical heaters and temperature controllers are used to maintain minimum temperatures for components when operating and survival temperatures when not operating. Thin film Kapton heaters with primary and redundant elements in the same heater blanket are used for temperature control. Each heater circuit is wired to a fused circuit off the HST main buses. The heaters are constructed of laminated Kapton, encasing a chemically etched resistance element. The Kapton heaters are adhesively bonded directly onto the component or structure, as required.

Table 2.8-2 contains a list of the heaters and wattages used on the SSM.

The operation of the heater elements is controlled autonomously using bimetallic switches. Two thermostats wired in series control each heater element. The two primary element thermostats are nominally set 3° C above the desired control point, and the two redundant element thermostats are set 1° C above this point. The latch and drive heater thermostats are wired in parallel and do not have a redundant system because these heaters are only required for deployment and retrieval.

To conserve commands, the disable commands normally disable both the primary and secondary heater circuits. Therefore, commanding a heater circuit off may require two commands, one command, which disables both circuits, and a second command to enable the circuit to be left on.

The \pm SDM retrieval/survival heaters, when enabled, bypass the thermostats and are used to raise the SDM temperatures above 0° C for deployment.

The SSM heater network is completely redundant and contains no single-point failures. The redundant circuit protects against an open primary heater, thermostat, or wire. A thermostat that has failed closed in either the primary or redundant system is backed up by the second thermostat in series. It requires two independent failed closed thermostats for a heater circuit to be continuously on. Temperature monitors adjacent to the heaters would detect this failure. The failed heater circuit would have to be disabled to correct this situation and subsequently sequenced manually by STOCC command if its operation is required.

A list of the SSM thermostat locations and set points is contained in Table 2.8-3.

Table 2.8-2.	SSM heater	data
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Heater location	Total watts at 28 V	Each heater at 28 V	Comments
Bay 1 computer mounting structure	30.9	15.46	Two strips on mounting structure
Bay 2 and 3 batteries (6)	240	40	Internal by vendor
Bay 4 PWR dist door	20.6	5.16	Four strips on door
Recorder (1, 2, and 3)	46.4	7.73	Two strips on each mounting bracket
Bay 5 MA XPNDRs 1 and 2	61.9	15.46	Four strips adjacent to MA xpndrs
RWA (1, 2, 3, and 4)	123.7	15.46	Two strips on each RWA
Bay 7 drive elect door	20.6	10.32	Two strips on door
Bay 8 PSEA door	20.6	10.32	Two strips on door
Bay 10 SIC&DH tray	113.6	28.41	Four strips on SIC&DH tray
FHST delta plate	75	8.33	Nine heater strips on delta plate
Latches and drives	100	10	One for each motor
HGA gimbals	27.5	6.88	Internal by vendor
RSU operational	96	32	Internal by vendor
RSU survival	69	23	Internal by vendor
AD hinges	25	10/15	Active/passive sides
OCE	9.8	4.9	Two heater strips
FGE 1	99.2	49.6	Two heater strips
FGE 2	88.6	44.3	Two heater strips
FGE 3	71.2	35.6	Two heater strips
CSS	20.42	4.08	One heater strip per CSS (CSS 4 and CSS 5 share)
MSS	15.0	3.75	Two strips per MSS
PDM	13.1	3.27	Four heater strips near motors
SDM	17.7	8.84	Two strips near motors
SADM	13.3	6.64	Two heater strips on motors
SA diode tray	21	2.10	Ten heaters on diode plates

Note: Most heater strips have the secondary heater on the same strip. Latch and hinge motor heaters have only one heater per strip and only a single heater circuit. The aperture door hinge motor has secondary heaters on the same strip and a secondary circuit.

Location	Clos	e (° C)	Open (° C)
Bay 1 computers	Primary	-9 ± 1 12 ± 1	-9 to -6
Bay 4 PWR dist door	Secondary	-12 ± 1 -21 ± 3	-12 ± 3
Bay 5 (MA XPNDR, SSA XMTR)		-26 ± 3 -7 ± 1	-18 ± 3 -7 to -3
		-9 ± 1	-9 to -6
Recorders (bays 5 and 8)		-4 ± 1 -7 ± 1	-4 to -1 -7 to -3
RWAs (bays 6 and 9)		-12 ± 1 -15 ± 1	-12 to -8 -15 to -12
Bay 7 drive elect		-14 ± 1 -16 ± 1	-14 to -11 -16 to -13
Bay 8 PSEA door		-21 ± 3 -26 ± 3	-12 ± 3 -18 ± 3
Bay 10 SIC&DH		-7 ± 1 -7 ± 1	-7 to -3 -7 to -3
FHST		-1 ±1 -3 ±1	-1 to 3 -3 to 0
Latches/drives		-32 ± 3	-23 ± 3
AD drive		$\begin{array}{c} -26\pm3\\ -32\pm3\end{array}$	-18 ± 3 -23 ± 3
AD hinge		-51 ± 3 -58 ± 3	$\begin{array}{c} -42\pm3\\ -49\pm3\end{array}$
RSU		57.1	57.3
HGA gimbal		18.9 ± 2 16.1 ± 2	18.9 to 21.7 16.1 to 18.9
BATTS (bays 2 and 3)		-2 to 0 -5 to -3	-2 to 2 -4 to -1
OTA ES		-40 -40	-28.9 -28.9
CSS		-31.7 ± 2.8 -34.4 ± 2.8	$\begin{array}{c} -23.3 \pm 2.8 \\ -26.1 \pm 2.8 \end{array}$
MSS		-26.1 ± 2.8 -31.7 ± 2.8	-17.8 ± 2.8 -23.3 ± 2.8
PDM		-43 -43	-29 -29
SDM		-29 -29	-15 -15
SADM		-43 -43	-29 -29

Table 2.8-3. Thermostat requirements

2.8.2 HST Thermal Environments

2.8.2.1 Operational Environment

The HST TCS is designed to operate in a 28.5° inclined circular orbit at an altitude between 215 and 320 n. mi. The nominal HST attitude is with the Sun in the V1-V3 plane and on the +V3 side of the spacecraft. The HST permits viewing of targets to within 50° of the Sun, with no direct sunlight entering the forward baffle under conditions of up to \pm 5° of roll about the V1 axis from the normal attitude. Fifty orbits of normal roll (\pm 5°) attitude are required prior to initiating an off-normal roll maneuver to maintain proper EPS operation.

The initial orbit was chosen (in the absence of other overriding launch constraints) to put the Sun in the HST orbital plane at injection (i.e., zero orbit beta angle). The orbit beta angle (the angle between the Earth-Sun line and the projection of that line on the HST orbit plane) varies as the orbit plane rotates because of the oblateness of the Earth (through 360° in 53.7 days) and as the Sun changes declination over the year. Figure 2.8-7 shows this variation over the first year of the HST mission for a constant altitude of 270 n. mi. The beta angle varies between $\pm 52.2^{\circ}$. The eclipse duration depends upon the beta angle and orbit altitude. Figure 2.8-8 shows shadow time (eclipse) per revolution as a function of altitude for the extremes of a beta angle of 0° and 52.2°.

2.8.2.2 Safe-Mode Environment

The HST safing system safe modes autonomously reconfigure the onboard systems to ensure HST survival. The initial safe-mode attitude of inertial hold and various Sun-pointing modes result in orbit thermal environment extremes identical to on-orbit conditions described for the operational environment. During those attitudes when the HST is in a power-positive configuration, the telescope keeps all the normal mission mode heaters operational to maintain normal operating temperatures.

When Sun-pointing cannot be maintained, the HST is placed into a GG stabilized mode. The SA output is considerably less than during the normal Sun-pointing modes. Therefore, the telescope is power limited. The TCS maintains all essential components within operating limits, and all other components are maintained above their structural (survival) temperatures.



Figure 2.8-7. Variation of Sun-orbit beta angle and shadow time over 1 year



Figure 2.8-8. Variation in orbit shadow time with altitude

2.8-21

2.8.3 Safety Considerations

The HST structure is compatible with the SM on-orbit thermal environment. The HST servicing mission thermal analysis demonstrates that when normal power is available to the HST from any of the possible sources (i.e., orbiter power, batteries, or solar arrays), HST temperatures under all on-orbit thermal conditions, including orbiter contingency attitudes, remain well within the HST primary and secondary structural limits. In the contingency case of complete loss of power to the HST, the on-orbit thermal environment would be less severe than that which occurs in the HST free-flying gravity gradient mode. Thermal analysis of the gravity gradient mode has shown that the HST OTA structural limits are exceeded after 28 hours.

2.9 SSM SAFING SYSTEM

The SSM includes a Safing System (SS) that autonomously protects the HST from electrical power depletion during any 72-hour period when the STOCC may be out of contact. The system protects for single failures in any subsystem and permits the HST to survive until STOCC regains contact and control. The SS interfaces with all SSM subsystems and makes use of many of the PCS and DMS components. The only dedicated safe-mode hardware are the RMGA and the PSEA. The PSEA also performs some primary PCS functions, as discussed in Section 2.7. In addition, the SS makes use of rate gyros, reaction wheels, CSSs, magnetometers, MTs, and retrieval mode gyros (all PCS components). The DMU, DIUs, the 486 flight computer, and flight software are also essential elements of the SS.

2.9.1 SSM Safing System Hardware

2.9.1.1 Pointing and Safe-Mode Electronics Assembly

The PSEA is a part of the SSM SS, its primary purpose being to provide autonomous HST safing in the event of a failure in the DMS, PCS, and/or EPS subsystems. The PSEA issues appropriate commands to safe the vehicle. In addition to the purely safe-mode functions, the PSEA provides the electronics to drive the MTs during normal operations, commands the MCU to close the AD if the optical axis approaches the Sun line, processes Sun angle data for use by the DMS during normal operations, and nadir-points the vehicle in GG operation.

The PSEA is completely redundant, and each unit (A and B) is composed of three functional elements: the MTE, the SMC, and the MCE. The SMC and MCE comprise the safe-mode portion of the PSEA. The MTE and four fixed bar magnets comprise the MTS, which is used as part of the primary PCS, as well as for safe-mode operations (see Section 2.7).

The SMCs (A and B) are identical, independent electronics sections, each of which has the capability to Sun-point the vehicle in the Sun-point mode, orient the vehicle in the GGM, implement the load control strategy contained in the PSEA configuration memory, and control the SAs during safe-mode operations.

The MCEs (A and B) are also independent electronics sections. Each MCE has the capability to command the AD closed when the optical axis of the vehicle is in close proximity to the Sun, to process Sun angle data for use by the DMS and SMC, to monitor the health of the DMS and EPS as indicated by the presence of KA signals, and to store vehicle configuration data for use in hardware safe-mode to execute the load-control strategy.
The PSEA is partitioned internally to provide separate SMC, MCE, and MTE sections. Each SMC, MCE, and MTE has an independent dedicated power supply that derives all required bias voltages. In addition, each MCE power supply provides excitation to the configuration memory while the MCE is commanded off to keep the volatile memory active. However, it was necessary to load each configuration memory on initial deployment because the HST main buses were off for ascent.

Physical Description

The PSEA overall design features a completely redundant electronic unit enclosed in a 9.0- by 18.0- by 27.0-inch aluminum, dip-brazed housing and weighs approximately 81 pounds. Twenty-six interface connectors are mounted in the ends of the unit and are wired to 44 printed circuit board edge connectors. The unit mounts to the tunnel in SSM ES bay 8, and the top and bottom covers are removable for component access. The PSEA, shown in Figure 2.9-1 and Figure 2.9-2, is not classified *as an* ORU.

Electrical Power

There are two individual power circuits that service elements of the PSEA from the four PDUs. These consist of eight individual MTE channels (a PRI and RED channel for each torquer). In addition, there is a 5-amp fused circuit from PDU-1 to the PSEA for powering MCE A and SMC A, and a similar circuit from PDU-2 to SMC B and MCE B. These circuits are unswitched in the PDU so power is available at the PSEA when the HST main buses are energized. The PSEA consumes approximately 145 watts of power. The electrical power interfaces to the unit are shown in Drawing 2.9 sheet 1.

Command and Data

The PSEA can receive command inputs from the DMU CDI modules and DIUs 4 and 5, as shown in Drawing 2.9 sheet 1. DIU-4 interfaces primarily with PSEA-A and DIU-5 interfaces with PSEA-B. Digital magnetic moment commands from the 486 are routed to the MTE channels in use. DIU HLD commands provide controls for various SMC, MTE, and MCE functions.

The DMU provides both CDI special serial digital commands and CDI special discretes. The serial digital commands configure the configuration memory to establish the desired complement of RWAs, MTEs, RGAs, MSS, and SADE, as well as establishing rate biases, reasonableness check limits, and other functions (see Section 2.9.2). These data are stored for use when safe-mode is entered.

The PSEA transmits status data to the DIUs and issues the safe-mode initiate command to the DMU, which, in turn, distributes the signal to the rest of HST elements. The unit also sends analog torque commands to the RWAs and current commands to the MTs. The PSEA issues HLDs to the PDUs to perform load shedding, removing specific SSM and OTA components from the HST load per the desired configuration loaded into the PSEA configuration memory.



Figure 2.9-1. Pointing and safe-mode electronics assembly, exploded view

2.9-3



Figure 2.9-2. PSEA installed in SSM bay 8



Normal Operation Mode

During normal operations, four MTE channels and one MCE are active. The MTE receives digital magnetic moment commands from the DMS and provides controlled current to the torquer bars (Section 2.7). The selection of MTE A and B redundant channels is made by STOCC, and a mix of A and B units is possible.

The active MCE processes CSS data and supplies Sun position data to the DMS. It continuously monitors two DMS KA signals and, upon failure to receive either of these signals for 2.7 consecutive seconds, initiates safe-mode. The inactive MCE retains the content of its configuration memory.

Safe-Mode

During hardware safe-mode operation, one of the SMCs (SMC A or SMC B), four MTE channels, and one MCE (MCE-A or MCE-B) are active. The active SMCs are commanded into either the Sun-point mode or the GGM. During hardware safe-mode operations the active SMC controls SA orientation, provides commands according to a protocol appropriate to either the Sun-point or GG modes to safe the HST, implements EPS power management, and turns off the DMS.

The SMC firmware monitors for, detects, and compensates for a failed RGA, MSS, RWA, or MT. The GGM software section of the SMC firmware can be activated by the STOCC sending four HLDs in a predetermined order within 27 seconds. In GGM, vehicle rate damping is computed and supplied based on input from either the RGAs or the RMGA and the geomagnetic field data from the MSS. Sun data from the active MCE are employed to orient the SA, and the AD is commanded closed via the MCU (Section 2.3).

2.9.1.2 Retrieval Mode Gyro Assembly

The RMGA is an element of a backup system for attitude stabilization in the GG and hardware Sun-point modes. If fewer than three RGAs are available, the RMGA is used to provide vehicle rate signals for rate damping. It senses rate about three orthogonal axes and provides analog output signals proportional to that rate.

The RMGA is made up of three single degree-of-freedom gyroscopes in a single enclosure. It is a nonredundant, flight-proven design built by Northrop Corporation of El Monte, California, and modified by the substitution of long-life precision bearings. Performance requirements for this unit are much less precise than for the RGA. Although it substitutes in place of the RGAs, HST performance is limited to emergency survival when the RMGA is in use.

Physical Description

The RMGA is installed in SSM ES bay 8 on the tunnel structure, adjacent to the PSEA. The package measures 5.22 by 4.2 by 3.03 inches overall and weighs approximately 3.5 pounds. The RMGA, shown in Figure 2.9-2, is not classified as an ORU.

Electrical Power

The RMGA power interface is from PDU-4, which supplies fused, switched power to the RMGA from all three HST power buses. The circuit is protected by 3-amp fuses, and a latching relay provides the on/off control of the RMGA via commands from the STOCC or from the PSEA. The RMGA is powered only during orbital testing or when two RGAs fail the reasonableness checks performed by the PSEA. The RMGA draws 20 watts when operating.

2.9.2 SSM Safing System Mode Description

The HST SS is structured to operate without ground control for up to 72 hours. STOCC intervention is required to status the telescope configuration, determine the anomaly, and reconfigure and resume science operations. Figure 2.9-3 should be used to complement the subsequent paragraphs.

The primary threat to survivability of the HST is the loss of electrical power. The basic strategy behind the SS is to maintain pointing control and SA orientation for battery charging and, through use of power load shedding, to conserve electrical power usage. Progressive actions (modes) are employed to permit return to science operations as quickly as possible. The approach is to take the action for a failure that provides HST safety and yet permit the most rapid return to normal operations. If an unsafe condition persists, the SS progresses to the next mode that provides safety, though possibly at the cost of time to return to operations.

The SS detects system level anomalies and takes corrective action (safe-modes) independent of ground control to maintain the HST in a power-positive condition. A safe condition is defined as maintenance of all equipment in a thermally safe condition and in a configuration that, if ground control does intervene, allows reestablishment of normal HST science operations. The basic HST safing requirements used for structuring the SS are

- a. To autonomously protect HST from power depletion during any 72-hour period with a STOCC outage
- b. To not preclude retrieval by the orbiter because of the PCS failures



Figure 2.9-3. Safe-mode description, progression, and sequence

2.9.2.1 Mode Progression and Sequence

Progressive safing actions are employed to permit return to normal operations in as short a time as possible after fault correction. If a failure is detected that does not immediately threaten HST survival, an inertial hold mode permits maintaining existing attitude and merely suspends science operations. Ground action required to return to normal operations is minimal. If vehicle health deteriorates, the SS reorients SAs and the HST axes to point the active surface of the wings to the Sun. It turns off nonessential equipment and maintains a power-positive condition, if possible. Recovery from this point requires additional ground operations. If SS is unable to maintain this condition using primary PCS or DMS elements, the SS backup hardware takes over vehicle control and again attempts to maintain HST safety. Recovery from this condition requires progressively more operations and time.

The autonomous reconfiguration of the HST by the SS is limited to

a. Deactivation of failed hardware

With the exception of the PSEA and SADE, the SS does not autonomously switch to a redundant unit when a primary unit fails. Redundancy management is a STOCC function.

- b. Protection of the EPS and provision of thermal maintenance of critical telescope components
- c. Activation of preselected hardware for contingency operations

The STOCC can, as a last resort, enable transition to an additional nonoperational GGM, which is used only for contingency retrieval by the orbiter and return to Earth following multiple system failures. Return to normal operations, though possible (dependent on the duration in GG), is not required from this mode. If an inadvertent entry into GG is realized, STOCC must react within 2 hours after entry to reconfigure the HST for Sun-point survival power condition (minimum power condition) and Sun-point attitude to ensure a potential return to normal operation. This would be quickly accomplished by sending a PSEA reset command to allow entry into the hardware Sun-point mode, since no KAs is the condition at the time (the 486 flight computer is turned off in GG). In addition, to maintain the thermal safety of the OTA, the STOCC must command OTA bus no. 3 ON within 72 hours of entering hardware safe-mode.

2.9.2.2 Mode Descriptions

In addition to the routine monitor mode, three operational safe-modes can be entered autonomously. These modes (software inertial hold, software Sun-point, and hardware Sun-point) are described in subsequent paragraphs. GGM, which cannot be entered autonomously without prior enabling by the STOCC, is also discussed.

Software Monitor Mode

Critical SSM subsystem circuits and functions are monitored, and, upon detection of a failure, specified safing operations are performed. The monitor mode is active during normal mission operations of the HST. From this mode, transitions are allowed to software inertial hold, software Sun-point, and hardware Sun-point modes. Failure to acquire a guide star is not cause to exit the monitor mode. If guide stars are not acquired within a ground-specified time interval, the HST is held inertially stable on gyros. The processing and issuance of commands, as they are encountered in the 486 SPC file, are continued.

Software Inertial Hold Mode

This mode is entered upon detection of a failure during software monitor mode task. The capability to enter this mode by STOCC command is also provided. Entry into this mode is commanded by software upon completion of the last nonsafing command in the 486 command file.

Upon entry into this mode, the HST attitude is inertially held to either the resulting attitude after a maneuver or to an attitude required during science operations. Execution of SPCs is suspended; SAs and HGAs are allowed to complete their slews and then hold in that orientation. Selected nonessential equipment is turned off immediately, other equipment is turned off after 10 minutes, the safe-mode tape

2.9-10

recorder is turned on, and the SIC&DH module is notified to take appropriate SI safing action. Table 2.9-1 gives a detailed listing of HST equipment configuration in this and other modes.

The monitor-mode function continues during the inertial hold mode, and, upon detection of a different failure, the specified safing operation is performed. From the inertial hold mode, autonomous transitions are allowed to the software and hardware Sun-point modes only. The capability to inhibit entry into these modes can be commanded, and STOCC intervention is required to resume normal operations.

Software Sun-Point Mode

This software mode is entered if a failure specified in Figure 2.9-3 is detected. In this mode, the SS, operating under 486 software control and preselected by STOCC command, points the +V3 axis toward the Sun in a constant inertial orientation. The SA wings are oriented normally to the selected axis, with active surfaces pointing toward the Sun. The HGAs are not commanded from their last position. The hardware configuration remains basically the same as in inertial hold, except that SA trim relays are not closed (if they were open prior to entry into safe-mode). The MTS continues to support the RWAs in maintaining Sun-point orientation. The STOCC can command entry or inhibit entry into this mode, and the STOCC intervention is required to resume normal operations.

Software Zero Gyro Sunpoint Mode

This mode is entered if a failure specified in Table 2.9-3 is detected and if less than three RWAs are operational. In this mode, the SS, operating under 486 software control, points the +V3 axis toward the Sun in a constant inertial orientation. The SA wings are oriented with the active surfaces pointed normal to the Sun. The HGAs are not commanded from their last position. The hardware configuration remains basically the same as in inertial hold, except that solar array panel trim relays and the AD are closed (if open), and the RGA outputs are bypassed because fewer than three gyros are operating. The MTS continues to support the RWAs in maintaining Sun-point orientation. The STOCC can command entry or inhibit entry into this mode, and STOCC intervention is required to resume normal operations.

Software Spin Stablilized Sunpoint Mode

This mode is entered upon detection of a failure specified in Table 2.9-3 and if less than three RWAs are operational. In this mode, the SS, operating under 486 software control, points the +V3 axis toward the Sun in a constant inertial orientation. The SA wings are oriented with the active surfaces pointed normal to the Sun. The HGAs are not commanded from their last position. The hardware configuration remains basically the same as in inertial hold except that SA panel trim relays and the AD are closed (if open), and the RWAs are powered off. The MTS maintains Sun-point orientation by spinning the HST about the V1 axis. The STOCC can command entry or

inhibit entry into this mode, and STOCC intervention is required to resume normal operations.

Hardware Sun-Point Mode

This mode is controlled by the PSEA and is entered upon failure of the DMS to provide KA signals to the PSEA for greater than 2.7 seconds during monitor mode operation. The STOCC can inhibit or enter this mode by DMU CDI special discrete command. In this mode, the PSEA controls pointing of the HST +V3 axis toward the Sun in a constant inertial orientation. The SA wings are oriented normal to the +V3 axis with the active surfaces pointing toward the Sun. This is the same attitude as in software Sun-point mode.

Entry into the hardware Sun-point mode initiates the HST load shedding sequence, configuring the vehicle as shown in Table 2.9-1. The safe-mode initiate command is also sent to the SIC&DH, which places the SIs into their specific safing sequences. The EDs, which are critical for failure diagnosis, are recorded. The hardware Sun-point mode uses MTs and RWAs for providing control torques to maintain Sun-point control. Protection is provided for a failure of a full-on magnet by turning off its electronics autonomously, rendering the magnet inoperative. The PSEA can store vehicle configuration data uplinked from the STOCC using CDI special serial commands for performing its pointing control function when this mode is entered.

The PSEA configuration memory consists of the following:

- a. RMGA and RGA rate bias data for vehicle rate calculations
- b. SA position selection for GGM operation (function of beta angle)
- c. RGA status null/no null
- d. Enable/disable of MTE test, RGA reasonableness checks, MSS reasonableness checks, and routine self-tests
- e. Sun-point gain selection
- f. SADE power status
- g. Power configuration of MTE, RWA, RGA, and MSS
- h. Sun-point to GGM enable/disable
- i. Gyro hold mode initiate
- j. Sun-point and SA-axis selection for +V3 Sun-point operation
- k. Limits and thresholds for reasonableness checks and CSS data processing

2.9-12

- I. GGM control law gains, orbit rate bias values, and capture axis selection
- m. Telemetry diagnostic word addresses
- n. HLD timer selection
- o. AD control
- p. RWA bus switchover selection
- q. SI power ON configuration

The PSEA configuration memory contents are sampled in the normal spacecraft telemetry stream.

The PSEA receives vehicle angular rate data from the RGAs or RMGA (if two RGAs fail the reasonableness check) and Sun position data from the active CSS. RWA torque commands are computed and sent to position and hold the HST to a predetermined orientation with respect to the Sun. The PSEA computes and supplies the magnetic moment commands based on the momentum stored in the RWAs and the measured geomagnetic field provided by the MSS. Both MSS units are turned on by the PSEA, and the best data are used.

This safe-mode can also be used to back up deployment Sun capture and is ground selectable by setting the Sun-point axis selection bit of the PSEA configuration memory to +V3. This selection causes pointing of the telescope +V3 axis toward the Sun in a constant inertial orientation and positioning of the SA wings normal to the +V3 axis, with the active surfaces pointing toward the Sun under the same failure or command criteria as is used for +V3 axis Sun-point.

		Softwar	e	Hardy	ware		
Subsystem	Monitor	Inertial	Sun-	Sun-	Gravity		Remarks
unit/module		hold	point	point	gradient		
PCS						1	RMGA backs up RGA in Sun-point and gravity gradient modes
RWA	Х	X	X	X	O_{X^1}	2	Shutter held closed
RGA PSEA:	X	X	X	X	~	3	Both MSSs powered on
MTF	х	х	х	х	Х	4	STP and ETP on standby power unless commanded to be in
MCE	X	X	X	X	X		transport state; safe-mode T/R is powered off
SMC	0	0	0	Х	~	5	Recorder functions selected by STOCC. T/R records on track 1
Torquers	х	х	х	х	х		from BOT to EOT, then records on track 2 from EOT to BOT
RMGA	0	0	Ö	O ¹	O ¹	6	See Appendix A for selectable time definition
MSS	Х	X	X	X^3	X^3	6A	Safe-mode T/R off before DMU
FHST	X	0 ²	0 ²	0 ⁻	0² ×	7	PDUs in SSM
035	~	^	^	^	^	8	Active SADE preselected by STOCC
DMS				- 6.6A	~ 6A.13	9	SADEs off for 2 minutes after entry into Sun-point or GGM then
Data mgmt unit	X	X	X	O ^{s,a}	O ^m , e		switched on for array positioning
CDI	~	^	^	~	~	9A	SADEs off for 7 minutes after entry to Sun-point, then switched
Science tape rec ⁵	X^4	0	0	0	0		on for array positioning
Engineering tape rec ⁵	X		0 X	0 0 ⁶	0 0 ¹³		
DILLSSM	X	x	×	0 ⁶	0 ¹³		
486 computer	X	x	x	õ	õ		
EDS							
CCC (1)	х	х	х	х	х		
Power distribution unit $(4)^7$	X	X	X	X	X		
Power control unit	Х	Х	X ^{10A}	X ¹⁰	X ¹⁰		
Batteries (6)	Х	Х	X	x	x		
SADE no. 1°	X	X	X	X°	X°		
SADE no. 2°	0 0 ²³	0	0	0	0 0 ¹⁴		
	0	0	0	0	0 ¹⁴		
	Ĵ	Ĵ	3	3	9		

Table 2.9-1. HST equipment configuration in safe-mode

X = On O = Off

	Software		Hardware			
Subsystem	Monitor	Inertial	Sun-	Sun-	Gravity	Remarks
unit/module		hold	point	point	gradient	
MECHANISMS Auto point sys: GEA (2)	x	O ¹¹	O ¹¹	0	0	¹⁰ Separate diode and buses A and C, separate diode buses B and C. Activate bus C. Close SA panel relays and disable battery conditioning circuits
MCU: Aperture door/BOD	x	x	x	X ^{12A}	O ¹²	^{10A} Close SA panel relays only in software Sun-point
COMM: SSA TX no. 1 SSA TX no. 2 No. 1 MA XP RX No. 1 MA XP TX No. 2 MA XP TX No. 2 MA XP TX	X O X X X	0 0 X ¹¹ X x ¹¹	O O X X ¹¹ X x ¹¹	O O X O ⁶ X		 ¹² Aperture door closed automatically in GGM, MCU off after 3 minutes ^{12A} Aperture door closure automatic in hardware Sun-point selectable by PSEA configuration memory ¹³ Off after 2 minutes in GGM ¹⁴ Off after 3 minutes in GGM
Circ/RF SW	x	x	x	X	x	 ¹⁵ SIC&DH to initiate payload safing sequence ¹⁶ SIs subject to payload safing sequence
ICU	X	х	Х	O ⁶	O ¹³	 ¹⁷ Off after first 3 minutes in Sun-point mode ¹⁸ For autonomous turn-on see Appendix A
SIC&DH RIU SIs (5)	X X X	X ¹⁵ X X ¹⁶	X ¹⁵ X X ¹⁶	O ^{17,15} O ¹⁷ *	O ¹⁴ O ¹⁴ O ²⁰	 ¹⁹ Off upon entry into GGM. On 18 hours after entry into GGM ²⁰ 1 SI (FOC) on during transition to capture (9-hr worst case). Second, third, fourth, or fifth SI on after capture; STOCC selectable based on beta angle
OTA bus: 1 (PM,MR,FGS no. 1) 2 (SM. FGS no. 2 FGS no. 3, TCF)	X X	x x	X X	O ¹⁸ O ¹⁸	0 0	
3 (FPS no. 1, FPS no. 2, FPS no. 3)	Х	Х	Х	O ¹⁸	0	
4 (ACE, ACŚ, FGE no. 3, OCE, OCS)	Х	х	х	O ¹⁸	0	
5 (box door heaters) 6 (FGE no. 1) 7 (FGE no. 2) 8 (DIU) 9 (PM and MR lo heaters)	X X X X X	× × × ×	X X X X X	O ¹⁸ O ¹⁸ O ¹⁸ X O ¹⁸	O ¹⁸ O ¹⁸ O ¹⁸ X O ¹⁸	

Table 2.9-1. HST equipment configuration in safe-mode (continued)

X = On O = Off * = See Appendix A for autonomous turnoff and turning back on

		Software)	Hardware				
Subsystem unit/module	Monitor	Inertial hold	Sun- point	Sun- point	Gravity gradient	Remarks		
THERMAL (SSM) PCS heaters ²¹ RWA RGA (survival) FHST	X X X	X X X	X X X	x x x	O X O	 Heaters on are thermostatically controlled Also heats CCC The DCE is off during normal operations; it may be turned on periodically (every 30 days) for a short time 		
DMS heaters ²¹ Bay 5 Bay 7, 8 486	X X X	X X X	X X X	X X X	X O X			
EPS heaters ²¹ Bay 4 Battery PRI ²² Battery SEC ²²	X X X	X X X	X X X	X O X	X O X			
Communication heaters ²¹ SSA TX no.1 SSA TX no.2	x x	x x	X X	X X	x x			
SIC&DH tray heaters ²¹	Х	Х	Х	0	0			
Mechanism heaters ²¹ External CKT no. 1 External CKT no. 2	X O	X O	X O	X O	X O			
SA heaters 21 Diode survival SADM/PDM survival SDM survival SDM retrieval	X X X O	X X X O	X X X O	X X X O	X X X O			

Table 2.9-1. HST equipment configuration in safe-mode (concluded)

X = On O = Off

* = See Appendix A for autonomous turnoff and turning back on

Gravity Gradient Mode

The SSM has the capability to enter the GGM, which can be initiated by the STOCC if a failure requires contingency retrieval, or can be initiated autonomously if enabled by a series of prior STOCC commands when hardware Sun-pointing cannot be maintained. The HST is placed in a GG-stabilized attitude, and the loads are switched to a configuration that ensures a safe return to Earth. This mode maintains the telescope until orbiter retrieval, but low equipment temperatures encountered are expected to preclude return to science operations. The GGM technically is not a part of the SS since return to science operations is not guaranteed. GGM is not normally entered autonomously. It is covered here since it is controlled by the PSEA.

The GGM has the capability to capture the HST in a GG attitude through the use of the PSEA electronics with two MTs disabled and without use of the RWAs. Capture is accomplished with the +V1 or -V1 axis pointed toward Earth within $\pm 15^{\circ}$ of the local vertical and with the +V3 axis within $\pm 15^{\circ}$ of normal to the orbit plane.

Entry into GGM is limited to the following:

- a. Autonomous entry This is possible only from the hardware Sun-point mode and occurs only if autonomous entry into GGM has been previously enabled by the STOCC. Autonomous entry is initiated when the HST Sun-pointing cannot be maintained within $\pm 35^{\circ}$ of the Sun line.
- b. STOCC-commanded entry Four HLD STOCC commands issued in a known sequence and within a 30-second period are required to initiate GGM after entry into hardware Sun-point mode. There are four HLDs for each SMC.

At the time of entry into the GGM, the SA wings are automatically positioned perpendicular to the +V1 axis with the active surface of one wing facing the +V1 and the second facing the -V1. Following GG vector acquisition, the wings are autonomously maneuvered to a fixed position for maximum power generation dependent on final capture attitude and Sun beta angle. The SA position and the direction of the +V3 axis relative to its being perpendicular to the orbit plane (north or south) must be preselected by the STOCC and updated periodically.

Software Thresholds and Triggers

The 486 flight computer and the HST flight software are in control of the S/C when in the monitor, inertial hold, or software Sun-point modes. The flight software tests a number of software, PCS, EPS, and DMS parameters and takes corrective action if any are not within specified limits.

When a test determines that the monitored parameter exceeds the associated threshold a specified number of times or for a specified duration, a predetermined response is made and associated action is taken. For all tests, the operational limits, number of consecutive time periods out of limits, and response are software database items. The STOCC is capable of inhibiting the responses to certain safing test failures by disabling the selected safing or HLD macro. As an alternative capability, the safing test itself can be disabled, thus preventing the response from occurring. The software tests can be activated or deactivated as a block. In addition, the capability is provided for temporary block deactivations of the safe-mode test functions, which autonomously reactivate themselves after a time period, in the absence of further STOCC inputs. This feature is loosely termed "safe-mode test bypass."

The software safe-mode monitors and EPS, DMS, and PCS parameters are summarized in Table 2.9-2, as are their operational limits and noise filter specifications (number of times out of limits before corrective action is taken). Table 2.9-3 defines the nominal initial settings and time out of limits for the triggers as well as the corrective actions.

Table 2.9-2. Safe-mode trig	ggers telemetry monitors
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Test title	Counters limit	Comments
Safe-mode test bypass	60 sec	Bypasses safe-mode test
Body rate error in low mode	5 sec	Switch to RGA high mode
Body rate error in high mode	20 sec	Transition to inertial hold
First RWA momentum limit	30 sec	Remain at three wheels, power off the failed wheel, and enter S/W Sun-point
RWA speed limit	60 sec	Safe-mode only response to enter S/W Sun-point
2-gyro configuration	10 sec	 Gyro count disparity test should be performed only in normal mode Gyro H/W error failure test should be performed in all three modes (normal, Sun-point, spin stabilized) Gyro count saturation test should be performed in normal and Sun- point modes
	1 sec	 If two gyros fail, no reconfiguration, and go to zero gyro Sun-point All counters are reset to zero when test disables/not fail or gyro mode change
4-MTS command echo	120 sec	When failure occurs, the failed unit powers off and enters S/W Sun-point
PSEA power supply	10 sec	Close aperture door and enter inertial hold
Valid ephemeris check/magnetic field model	6 days	Ephemeris is computed at 1-Hz rate. Before ephemeris is computed, check if the time since epoch is greater than ephemeris delta time limit, then set delta time limit exceeded flag; otherwise, it should always be zero. Magnetic field ephemeris check is activated when model is on
Minimum energy overflow (deleted)		
Second RWA momentum failure	30 sec	Each RWA momentum change is computed at 40-Hz rate to check against the RWA momentum threshold. When number of active RWAs is 2 or less, power off all wheels and enter spin stabilized mode
System momentum limit	40 sec	Squared magnitude of environmental torque is computed once each second If the value is over the limit for 40 consecutive seconds, then the system momentum test fails and zero gyro control is activated
3-MTS command echo	120 sec	If two or more MTSs have failed, spin stabilization is activated
Bus minimum voltage	60 sec	Three bad bus voltages below 26.5 V results in safe-mode response to stop KAs and enter H/W Sun-point
Battery state of charge first limit	30 sec	When this test is active, EPS calculation must be ON for more than two orbits. Battery SOC first limit 285 A-hr is set to allow 30 A-hr for 486 S/W Sun-point capture and 120 A-hr for the battery SOC second limit. Safe-mode response is to enter S/W Sun-point; and SI load shedding safing
Battery state of charge second limit	30 sec	When this test is active, EPS calculation must be ON for more than two orbits. Battery SOC second limit 205 A-hr is set to allow 90 A-hr for GGM capture, 10 A-hr for two RWA reasonableness failures, and 20 A-hr for the uncertainty in estimating battery capacity. Safe-mode responses to stop KAs
Total load current	2 sec or 360 sec	The failure occurs when the total load current exceeds 109.8 A for 360 sec or 145 A for 2 sec. The diode bus A, B disconnects from bus C, and bus C is activated
Structure current	2 sec or 10 sec	The failure occurs when the structure current exceeds 15 A for 10 sec or 48 A for 2 sec. The diode bus A, B disconnects from bus C, and bus C is activated
SADE torque failure	5 sec	SA processing must be in a maneuver mode. In normal mode, SADE electronics should switch to secondary SADE and enter S/W Sun-point. In Sun-point mode, power off SADE. If the vehicle is in drift mode, 4-3 RWA Sun-point mode, or zero gyro Sun-point mode, then activate to power off the secondary SADE or the primary SADE. If the vehicle is in spin stabilized mode, power off the secondary SADE or power off the primary SADE. This test remains enabled to monitor for second SADE
Solar array position	SFSACL (5 sec) FSLEWT (13 min) for normal, Sun-point or zero gyro FSLEWT (23 min) for spin stab	Vehicle must be in Sun-point mode, 13 min for normal, Sun-point, or zero gyro; or 23 min. for spin stab. is the minimum time to allow ongoing SA slew completion before executing the next SA slew. When failure occurs, SADE electronics should switch to nonfailed SADE

Item	Sub- system	Test	Failure	Safing action	Notes
1	DMS	486 Memory Test (Parity Error Interrupt Count)	Number of parity errors exceeds 2	Stop KA signal and transi- tion to hardware Sun-point	Counters resettable by ground command
2	DMS	486 Memory Test (Write Protect Violation)	Number of write protect errors exceeds 90	Stop KA signal and transi- tion to hardware Sun-point	Counters resettable by ground command
3	DMS	Special Instruction (CPU)	Computation check changes KA bit pattern	Stop KA signal and transi- tion to hardware Sun-point	Specific tests of CPU instructions to ensure health of CPU
4	DMS	Processing Rate Time- out	Fails to complete cycle twice	Stop KA signal and transi- tion to hardware Sun-point	Ensure that processing on timing cycle completes within its cycle time
5	DMS	1 kHz vs. 40 Hz Timing Test	Timing signals out of sync two times	Stop KA signal and transi- tion to hardware Sun-point	Interrupts within ± 300 nanoseconds
6	DMS	No STOCC Commands for N hours	No STOCC command for 100 hours	Transition to software Sun-point and close aperture door	If no commands have been received within a ground- specified period, the safing system transitions to soft- ware Sun-point; macro action (S)
7	DMS	Stored Program Command (SPC) List	(A) SPC list exhausted = 1(B) SPC list exhausted = 2	Transition to inertial hold mode Transition to software Sun-point	Transition caused by last command loaded; macro action (S)
8	DMS	Interrupt Branch Checksum	Checksum not zero	Reset branches and transition to software Sun-point	Macro action (S)
8A	DMS	Unprotected Memory Checksum	Checksum not zero	Transition to software Sun-point	Macro action (S)
9	DMS	Sun Protection	 (A) In normal mode: Sun ephemeris less than 32° from optical boresight for consecutive 5 seconds (B) In Sun-point mode: Either composite Sun presence monitor for CSS1 or CSS2 is set for consecutive 5 seconds (C) Upon transition to Sun-point mode: the buffered Sun ephemeris less than 46° 	Close aperture door and transition to inertial hold	The third test runs continu- ously in Sun-point mode, but buffered value, if out of limits, will cause immediate failure upon entry into Sun-point and close aperture door only Ephemeris check performed only in normal mode and macro action (S) and set PIT flag
10	DMS	Earth, Moon Protection	Less than 5° from Earth or 10° Moon disc and FGSs not in look-away position	FGS default ON, FGE and OCE high voltage OFF and transition to inertial hold	Macro action (S) and set PIT Flag
11	DMS	No Ephemeris for 5 and 11 Days	The computed data time from the epoch is greater than 6 days. (Vehicle ephemeris delta time limit) or 11.5 days (Sun, Moon, TDRSA and TDRSB ephemeris delta time limit)	Transition to software Sun-point	Macro action (S)

Table 2.9-3. 486 safe-mode tests and actions

Note: *** Software Macro (S); applies to all occurrences of (S). **** For large OFF roll angles, requires change in threshold or disabling test. * Modified by STOCC based on EPS conditions.

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ltem	Sub- system	Test	Failure	Safing action	Notes
12	DMS	SIC&DH Toggle	The SIC&DH "OKAY" toggle bit fails to toggle for 21 consecutive seconds	Deactivate SIC&DH proc- essing. Safe the SIs	Macro action (S); SIC&DH toggle samples at 0.5 second rate
13A	EPS	Battery State of Charge (Level 1)	Below 150 A-hr state of charge for 30 sec	Transition to software Sun-point mode	Trigger macro actions (S) and time interval setable by ground command
13B	EPS	Battery State of Charge (Level 2)	120° A-hr state of charge for 30 sec	Stop KA signal and transi- tion to hardware Sun-point mode	
14	EPS	Bus Minimum Voltage	Three bus voltages less than 26.5 volts for 30 consecutive seconds	Stop Kas and transition to hardware Sun-point mode	The main bus C if off-line
15A	EPS	Total Load Current	Greater than 125 A for 10 sec or 145 A for 2 sec	 Disconnect diode bus A from C and diode bus B from C Activate bus C 	Macro action (H)
15B	EPS	Total Structure Current	Greater than 15 A for 10 sec or 48 A for 2 sec	 Disconnect diode bus A from C and diode bus B from C Activate bus C 	Macro action (H)
16	EPS	SADE Torque Failure Flag and SADE Input Power Monitor	Logic 0 indicates SADE torque failure and Logic 1 indicates power ON (SADE input power mon) for consecutive 5 seconds	In normal mode: Switch to redundant SADE and transition to software Sun-point In Sun-point mode: Power off SADE	Macro action (H), (S). The test is only active when SA processing is in a maneuver mode
16A	EPS	Solar Array Command Profile Test	Difference between actual SA position and command profile exceeds threshold 14° for 3 consecutive seconds	In normal mode: Switch to redundant SADE and transition to software Sun-point In Sun-point mode: Power off SADE	Macro action (S), (H)
17	EPS	Solar Array Position Test	Exceeds SA required position by $\pm 9^{\circ}$ for 5 consecutive seconds after 13 minutes start of SA wing commands	Switch to nonfailed SADE	Test performs in Sun Point Safe-mode only; macro action (S)
18	EPS	Solar Array Position Limit	Within -15° and +140°; if out for 3 sec, fail test	Switch to redundant SADE and transition to software Sun-point	Macro action (S)
19	EPS	Rate of Charge Test	Exceeds separate thresholds for ⁺ 300 consecutive seconds or EPS has not been in step to trickle charge for 3 orbits	Transition to software Sun-point****	Thresholds for maneuvering, nonmaneuvering during full charge, trickle charge, and eclipse periods; macro action (S)
20	EPS	Solar Array/Sun Position	Greater than 9° during stationary period, 35° during off nominal for consecutive 10 seconds	Transition to software Sun-point****	Macro action (S)
21	I&C	HGA Torque Limit	Exceed torque command limit of 10 in–oz for 300 1-second samples over any time interval. If within limit for 10 minutes (consecutive), reset counter to zero	Transition to inertial hold; deactivate HGA control law	Macro action (S)

Table 2.9-3. 486 safe-mode tests and actions (continued)

Note: *** Software Macro (S); applies to all occurrences of (S). **** For large OFF roll angles, requires change in threshold or disabling test. * Modified by STOCC based on EPS conditions.

Item	Sub- system	Test	Failure	Safing action	Notes
22	PCS	Body Rate Error Check (BREC)	Threshold (incremental angle check) (A) low mode-2.0 arcsec/sec for2 sec (B) high mode-80.0 arcsec/sec for 2 sec	Switch to RGA high mode Transition to inertial hold mode	Macro action (S)***
23A	PCS	RWA Torque Limit Test	(Torque CMD)-(torque from RWA) greater than limit 1.0 N-m- s for consecutive 3 cts in 10 sec/count	Turn OFF anomalous wheel and transition to software Sun-point mode	Macro action (H, S)
23B	PCS	RWA Speed Test	Exceed 2200 rpm for 60 sec	Transition to software Sun-point mode	Macro action (H, S)
24	PCS	PSEA Monitor Power Supply Test	Logic zero indicates failure of two power supplies in monitor electronics for consecutive 10 seconds	Close aperture door and transition to inertial hold	Macro action (H, S)
25	PCS	Gyro Reasonableness	 Three tests: (A) Gyro disparity: performs in normal mode only. Incremental error exceeds 1500 counts for 1 second in low mode; 120 counts for 1 second in high mode (B) Gyro H/W error failure: performs in normal and Sun-point modes: 29/30 counts in 25 msec period not met for 1 second (C) Gyro counts saturation test: performs in normal and Sun-point modes. Gyro count data exceed 3600 counts for 1 second 	If four gyros active and only one gyro failure detected, go from four to three gyros. If back in limit, fourth gyro is returned to calculations. If one gyro fails and second gyro fails later, configure from four to three gyros, no recovery back to four gyros and go to software Sun-point mode. If two gyros fail simultaneously, no reconfiguration and go to software Sun-point mode	Gyro 4 to Gyro 3 transition is coded in software. Transi- tion to software Sun-point is a macro action (S) Currently, the database value for gyro reinstatement is set to maximum, 2.47 days Therefore, the failed gyro has to remain within limits for 2.47 days before returning to calculations
26	PCS	Magnetic Torquers Test	Torque error exceeds 500 A-m ² for 2 consecutive minutes indicating invalid PIT processing	Turn OFF faulty channel and transition to software Sun-point mode	Macro action (H, S)
27	PCS	Minimum Energy Overflow	The momentum management Minimum Energy (ME) law performs an overflow check	A failure flag is set indicating an overflow and transition to software Sun- point mode	Macro action (S)
28	PCS	Two RWA torque failure	Two of four RWAs (torque cmd) - (torque from RWA) greater than limit 1.0 N-M-S for consecutive 3 cts in 10 sec/count	Turn off anomalous wheel, and transition to software Sun-point mode and stop KA signal	Macro action (H, S)

Table 2.9-3. 486 safe-mode tests and actions (concluded)

Note: **** Software Macro (S); applies to all occurrences of (S). **** For large OFF roll angles, requires change in threshold or disabling test. * Modified by STOCC based on EPS conditions.

3.0 OPTICAL TELESCOPE ASSEMBLY

3.1 INTRODUCTION

The heart of the HST is the OTA, where the reflecting 24 Ritchey-Chretien Cassegrain telescope is located. As illustrated in Figure 3.1-1, this type of telescope features a system where light from a star or other object travels through the aperture, down the assembly past the smaller Secondary Mirror (SM), and strikes the larger 94.5-inch PM. The light is then reflected 16 feet to the 12.2-inch SM, where it is narrowed and intensified into a small diameter beam. The beam travels through the 24-inch hole in the PM to the focal plane just behind it. The focal plane is 4.9 feet behind the front surface of the PM.



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At the focal plane the light originally captured by the big mirror is converted into a focused image. This light image is divided among 11 sensing devices: 4 axial SIs, 3 optical control sensors, 1 radial SI, and 3 FGSs. Parts of the image that enter the apertures of the SIs are transmitted as data to Earth. These pictures and other SD are converted to electronic signals and are transmitted via the HGAs at a rate up to 1 Mbps. After being received on Earth, the data can be reconstructed into images and spectrograms.

The mirrors are kept at a nearly constant temperature so that images at the focal plane will have no distortion in their resolution due to environmentally induced surface changes. Structural stability is required in order to minimize spacecraft orbital thermal distortions affecting the alignment of mirrors and sensors. Three kinds of movement must be compensated as shown in Figure 3.1-1: (1) despace, a change between mirrors that defocuses faint objects; (2) decentering, a lateral movement such that PM and SM optical axes, even though parallel, are not coincident; and (3) tilt, angling of optical axes that produces smearing images.



Figure 3.1-2. Mirror motion compensation

The OTA includes the following major components:

- a. Primary Mirror Assembly (PMA) *Contains* the main ring that is the primary load carrying structure, the reaction plate assembly, the main and central PMA baffles, and finally the PM.
- b. Secondary Mirror Assembly (SMA) *Contains* the SM subassembly, the SM baffle, and the metering truss.
- c. Focal Plane Structure Assembly (FPSA) *Contains* the SIs, fine guidance/optical-control sensors, the rate gyros, and acquisition star trackers.
- d. Fine-guidance/optical control sensors *Supply* fine guidance data for HST vehicle control and for adjustment of optics in orbit.
- e. OTA ES *Contains* most of the OTA electronic components mounted in seven of the nine available equipment bays. Other OTA-related electronic boxes and cables are located in the SSM ES of the HST.

An exploded view of the OTA subsystems is shown in Drawing 3.1 sheet 1. Drawing 3.1 sheet 2 is a layout of the OTA and defines the location of the major structural component parts.



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3.2 OTA ELEMENTS

3.2.1 **Primary Motor Assembly**

The PMA provides the principal OTA optical element and maintains it in an environment such that its optical performance is not compromised. This includes an optimum thermal environment for dimensional stability, the elimination of stray light, a structural cell to support the mirror during launch and landing and accurately locate it within the OTA, as well as provide controlled forces to optimize the mirror figure. The PMA also provides the OTA structural interface to the SSM and the structural interface for the OTA SMA and FPSA subsystems.

The PMA is composed of the MR, the Reaction Plate Assembly (RPA), PMA baffles, and the PM. The PMA mounts to the SSM ES aft bulkhead through stainless steel axial and tangential link fittings (three each) fixed to the MR, as depicted in Drawing 3.2 sheet 1. The principal PMA subelements are described below.

3.2.1.1 Main Ring

The MR is the primary load-carrying structure to which all other OTA subassemblies are attached. It consists of a 118-inch outer diameter by 100-inch inner diameter by 15-inch-deep annular box beam. Two concentric stainless steel cylinders form the inner and outer diameters, separated by forward and aft steel channels riveted in place, with the rivets staggered 1° apart. Internal titanium fittings provide the load-bearing interface to other PMA elements. The cavity between the inner and outer cylinders is closed out by titanium skin panels.

The MR interfaces to the SSM ES aft bulkhead through stainless steel axial and tangential (shear) links as shown in Drawing 3.2 sheet 1. The PM mounts to the MR through steel and titanium tangential shear fittings bonded to the mirror aft surface by epoxy resin, and axial fittings preloaded between the forward and aft PM surfaces. These links allow for minor changes in the PM position caused by the thermal environment and operation of the PM figure control system.

3.2.1.2 Reaction Plate Assembly

The RPA provides the reaction point for the figure control actuators, which control the shape of the PM figure. Should the PM figure change due to the launch environment or gravity release, its optical performance will degrade if not corrected. The figure control actuators of the RPA can be utilized, in conjunction with dedicated electronics, to correct the PM figure to within acceptable tolerances.

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SHEET 1 JSC-29029 The RPA is constructed of a magnesium alloy and consists of 15 radial I-beams interconnected by 48 intercostal ribs. The 15 radial beams form the spokes of a wheel, with the PM feed- through aperture forming the center. The end of each radial member interfaces to the MR through a flexure, which allows the RPA to give slightly (+ 0.0058 in. maximum) as the 24 figure control actuators correct the PM shape. Each actuator mounts between two intercostal ribs. Placement of the actuators is depicted in Drawing 3.2 sheet 1. The PMA central baffle assembly mounts within the hub of the RPA.

3.2.1.3 PMA Baffles

Two baffle structures are connected to the PMA to provide OTA stray light control. The main baffle extends forward from the MR, concentric with the PM optical centerline. It is an aluminum cylinder, 104.75 inch in diameter by 191.45 inch long, which mounts to the forward face of the MR with 17 bolts. The inner surface of the cylinder, which includes 14 concentric, light-gathering ribs, is finished with light-absorbing Chemglaze flat black paint.

The central baffle provides protection from stray light interference with the focused image transmitted from the SM. This baffle is an aluminum, truncated conic tube, 119.45 inch long and 15.5 inch in diameter at the forward end, opening to a 23.5-inch diameter at the aft end. As with the main baffle, the inner surface of the tube, which includes nine concentric, light-gathering ribs, is finished with light-absorbing Chemglaze flat black paint. The central baffle mounts to the hub of the RPA and extends through the center aperture hole in the PM.

3.2.1.4 Primary Mirror

The PM, the primary OTA optical element, is a sandwich construction with back plate, front plate, and edgebands fused together. The mirror is made of Corning ultralow expansion glass, composed of 92.5 percent silicon oxide (SiO_2) and 7.5 percent titanium oxide (TiO_2) . The PM is a 2.4-m (94.5-inch)-diameter hyperboloid with an 11.04-m (434.6-inch) radius of curvature. The mirror surface is vacuum coated with 650 angstroms of aluminum and 250 angstroms of magnesium fluoride (MgF₂). The MgF₂ boosts the UV performance of the mirror and protects the underlying aluminum layer.

The PM is structurally mounted to the MR through axial and tangential (shear) steel and titanium fittings. These fittings allow for minor adjustment of the PM position and shape. The tangential fittings are bonded to the aft surface of the PM by epoxy resin. The bonding interface is sensitive to cold temperatures and nominally protected from temperature extremes by the PM and RPA heaters. Failure of these heater systems could result in losing the bond integrity, making the HST potentially unsafe for Earth return. Each of the 24 figure control actuators of the RPA is bonded to the PM aft surface in a similar manner, but these bonds are not as critical because the actuators do not structurally support the PM.

The stability of the PM thermal environment is maintained by a system of operational and contingency heaters. The PM has redundant operational heaters. The heaters receive power from, and are controlled by, the EP/TCE within the OTA ES. In the event of a loss of the operational heaters, a separate contingency heater system (known as the Contingency Gravity Gradient (CGG) heaters) will thermally condition critical PMA components and surfaces, but cannot maintain an operational thermal environment. Placement of the PM heaters is shown in Figure 3.2-1.





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Figure 3.2-1. Primary mirror heaters

In the event electrical power for thermal conditioning of the PMA is lost, the HST cannot be returned to Earth if OTA temperatures fall below those specified in Table 3.2-1. This is due to the possible loss of structural integrity at the composite metal interfaces within the OTA structure. Differences in the coefficients of thermal expansion between the composite and the steel and titanium fittings can compromise the epoxy bond between the two materials.

	Lower temperature lir	nits (°F)
	On-orbit safety factor >1	Landing
FPS bathtub fittings	+6.8	+17.6
FPS "D" latch fittings	-25.6	+6.8
V3 truss fittings	-81.4	-8.6
EQT shelf "A/B" flexure at FPS	-25.6	-23.8
EQT shelf "C" flexure at FPS	-25.6	-23.8
FGS "B/C" latch fittings	-38.2	-31.0
FGS "D" latch fittings	-36.4	-2.2
FGS bench/shroud retainer	-38.2	-31.0
FGS keel/EXO interface	-95.8	-63.4
Pickoff mirror flexure	-11.2	+8.6

Table 3.2-1. OTA thermal limits

3.2.2 Secondary Mirror Assembly

The SMA, in concert with the PMA, completes the OTA two-element optical system. The assembly is constructed to minimize changes in position of the SM relative to the PM caused by the varying HST thermal environment. Optical performance of the SMA can be optimized by the STOCC by minor adjustment of the SM tilt, decenter, and despace parameters using an integral actuator system.

The SMA consists of the Secondary Mirror Subassembly (SMSA), the secondary mirror baffle, and the metering truss, as shown in Figure 3.2-2. The SMA interfaces through the metering truss to the forward face of the MR, concentric with the HST optical axis and PMA main baffle. The principal SMA subelements are described below.

3.2.2.1 SMA and Baffle

The SMSA houses the 0.31-m (12.2-inch) diameter convex SM, manufactured from Zerodur glass and coated with aluminum and magnesium fluoride in the same proportion as the PM. The SM focuses and reflects the light from the PM, directing it through the PMA central baffle to the focal plane within the FPSA. The SMSA contains an actuator system and SM support structure, diagnostic instrumentation, and thermal control components. The SM baffle encloses the SMA to provide stray light protection and is finished with light-absorbing flat black paint.



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Figure 3.2-2. Secondary mirror subassembly

3.2.2.2 Metering Truss

The 193-inch-long metering truss provides for the accurate placement of the SM relative to the PM in a thermally dynamic environment. The truss is a structure of graphite/epoxy members and rings, connected through titanium fittings. The graphite/epoxy members are selected according to their measured coefficients of thermal expansion. The selection criteria differ with the member position in the truss network, as shown in Figure 3.2-3. The truss structure is thermally passive, using no heaters, and individual members are wrapped in a layer of MLI.

The metering truss is required to maintain dimensional control between the PM and the SM in despace, limited to 118 microinches; decenter, limited to 394 microinches; and tilt, limited to 2 arcseconds. The truss supports the SMSA and SM baffle at the forward end and bolts to the forward face of the MR through eight fittings at the aft end, concentric with the HST optical axis and PMA main baffle.

3.2.3 Focal Plane Structure Assembly

The FPSA provides the integrating OTA structure supporting the five SIs, three FGS radial bay modules (RBMs), and an SSM equipment shelf. The principal function of the FPSA is to maintain the pointing alignment of each of the SIs relative to the FGSs under variable orbital environments. To accomplish this, the FPSA is primarily constructed of graphite/epoxy material, which exhibits low dimensional variation over a range of temperatures. Active thermal control elements (heaters) augment the thermal stability of the system.

The FPSA comprises a focal plane deck assembly (FPDA), an SI support structure (SISS), and an SSM equipment shelf, as depicted in Drawing 3.2 sheets 2 and 3. The FPDA consists of the central hub and beam structure and the FPSA structural interface to the PMA. Four bipod flexures, designed to accommodate radial displacements associated with the thermal expansion and contraction of the PMA, mount to the aft face of the MR at eight points. All interfaces with the FGS/RBMs and the radial SI are contained within the FPDA, including the guide rail systems and crew aids for on-orbit replacement of those instruments. The FPDA also forms the forward face of the axial SI guide rail system.



Figure 3.2-3. Metering truss





D



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The SISS is a graphite/epoxy truss structure that supports the aft end of the axial SIs in the lateral directions and provides the reaction point for proper mounting of the instruments in the FPSA. The four axial SI B-latches (one for each instrument) mounted to the aft truss near the OTA centerline provide an 800-pound preload against the B-fittings of each SI. The aft axial SI guide rail system is mounted to the truss to provide proper alignment of the instrument for installation and removal during orbital servicing. The SI cabling is distributed on the SISS truss to prevent cable interference with the SI changeout activity. Cable routing for the axial SIs is depicted in Drawing 3.2 sheets 2 and 3.

The SSM equipment shelf is a 425-pound, 4-inch-thick graphite/epoxy composite structure consisting of top and bottom panels over a honeycomb core. The SSM equipment shelf attaches to the FPDA and SISS and provides mounting and alignment provisions for the three FHSTs and three RGA RSUs, attaches to FPDA and SISS. The SSM equipment shelf is depicted in Figure 3.2-4.

3.2.3.1 Axial SI Mounting Provisions

Accommodations for mounting and maintaining structural alignment of the axial SIs are provided by both elements of the FPSA (FPDA and SISS). Each axial instrument is latched into the FPSA structure through three registration fittings shown in Figure 3.2-5. Each fitting constrains the movement of the instrument in at least one axis upon installation into the OTA system. Two latch mechanisms (the A and C registration fittings) are provided on the FPDA (forward) end of each SI cavity, and one mechanism (the B registration fitting) is provided on the SISS (aft) end, as shown in Figure 3.2-5. The following paragraphs provide an overview of the axial SI mounting interfaces. The changeout task for an axial instrument is described in Section 3.3.

1. Point A registration fitting

The A-fitting (SI) and A-latch (OTA FPDA) are a ball and jaw arrangement, as shown in Figure 3.2-6, Figure 3.2-7, and Drawing 3.2 sheet 4. The A-fitting on the SI, shown in Drawing 3.2 sheet 5 and Figure 3.2-8, is a metallic truncated ball fitting. The A-latch mechanism, operated through a manual drive interface located at the outboard end of the SI, consists of two jaws that clamp around the ball fitting, providing constraint in all three HST axes. The A-latch is equipped with A-latched and A-seated microswitches that can be monitored through corresponding indicator lights near the manual drive interface of the A- and B-latches. Operation of the B-latch places the A-fitting in the A-latch jaw capture envelope, opening the A-seated microswitch and extinguishing the primary and redundant indicators (Figure 3.2-9). When the A-latch has fully closed, a microswitch opens to extinguish the redundant A LATCH indicators.


(a) Overview

Figure 3.2-4. SSM equipment shelf

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(b) AS interior (SSM equipment shelf installed)

Figure 3.2-4. Concluded

3.2-17



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Figure 3.2-5. FPSA axial SI cavity (HSP shown)

3.2-18



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(a) A-fitting of axial instrument



(b) A-latch jaw mechanism mounted on OTA FPDA

Figure 3.2-6. Axial instrument A registration fitting pair

3.2-19



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(b) Detail



3.2-20



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		WHICH MAY BE MONITORED THR THE MANUAL DRIVE INTERFACE	OUGH CORRESPONDING IND OF A- AND B-LATCHES.	ICATOR LIGHTS NEAR RESPECTIVELY.	
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		AFFROMMATELY AS DEVICES AND AS A PIN, CONSTRAINING ROT	ACTS THROUGH THE C-F	TTTING OF THE SI T ABOUT THE A OR B	
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		FOR THE AXIAL SI A- AND B- A-LATCH AND BLIND-MATE CON A-LATCH	ATCHES, AS WELL AS TH	E RADIAL SI THE FGS/RBM	
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(b) B-fitting



3.2-25



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(a) Axial SI A-latch



(b) Axial SI B-latch manual drive interface

Figure 3.2-9. Axial SI fitting manual drive interfaces

3.2-26

2. Point B registration fitting

The B-latch (OTA SISS) and B-fitting (SI) are a plunger and seat arrangement, as shown in Figure 3.2-10and Figure 3.2-11. The B-fitting on each axial instrument is a spherical bearing seat capable of swiveling within its mount to allow for minor initial misalignments during the installation of the instrument. The B-fitting is shown clearly in Drawing 3.2 sheet 5 and in Figure 3.2-8. The B-latch plunger, mounted near the V1 axis in the SI cavity, as shown in the drawing and in Figure 3.2-5, is a spring-loaded device capable of a nominal preload of 800 lb on the B-fitting of an SI. When engaged, the B-latch pushes the SI into the A- and C-latches and constrains SI movement in the V2 and V3 axes. The preload may be removed through a manual drive interface, similar to that of the A-latch, withdrawing the plunger from the B-fitting. A B-ready microswitch is provided on the FPSA SI forward guide rail to indicate adequate positioning of the SI during installation to allow operation of the B-latch. The microswitch may be monitored through corresponding indicator lights on the SISS near the B-latch manual drive interface, as shown in the drawing provided and in Figure 3.2-9.

3. Point C registration fitting

The C-latch (OTA FPDA) and C-fitting (SI) are a ball-and-flexure type interface, as shown in Figure 3.2-12 and Figure 3.2-13. The term latch is somewhat misleading with this type of arrangement because there are no active drive mechanisms involved. The C-latch is a simple cylinder with rounded edges, resembling a doorknob. It is mounted to the FPDA at 45° between the V2 and V3 axes. It is a totally passive device and acts through the C-fitting of the SI (Figure 3.2-8) as a pin, constraining rotation of the instrument about the A- or B-latches, as shown in Drawing 3.2 sheet 4. The C-fitting on the instrument provides a positive capture flexure insert for the C-latch doorknob when the B-latch preload is applied to the SI and forces it forward (in the +V1 direction) to place the A-fitting in the A-latch capture envelope. A positive indicator lights, which also indicates that the A-fitting is provided by the A-seated indicator lights, which also indicates that the A-fitting is properly seated for capture by the A-latch jaws.



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(a) B-fitting "seat" of axial instrument

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(b) B-latch plunger mechanism mounted on the OTA SISS Figure 3.2-10. Axial instrument B registration fitting pair

3.2-28

UNLATCHED POSITION



(a) Concept

Figure 3.2-11. Axial SI latch B concept and detail

3.2-29



(b) Detail

Figure 3.2-11. Concluded



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(a) C-latch "ball" fitting mounted on OTA FPDA



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(b) C-latch fixture insert on axial instrument

Figure 3.2-12. Axial instrument C registration fitting pair

3.2-31





Figure 3.2-13. Axial SI latch C concept and detail

3.2-32

4. Point A and B drive interfaces

The A- and B-latches are actuated through manual drive interfaces on the FPDA and SISS structures, respectively. The crew interface is a 7/16-inch twelve-point ratchet fitting recessed into a housing that provides a spring-loaded driveshaft locking mechanism. This shaft-locking device comprises two jaws that engage gear teeth on the shaft as shown in Drawing 3.2 sheet 4. These jaws are spring-loaded to engage the shaft whenever the EVA ratchet tool is not present. When the tool is inserted into the drive interface housing, the jaws are forced open, allowing the shaft to rotate for manual operation. When the tool is subsequently removed, the jaws re-engage the shaft as before, locking the mechanism in place. This same arrangement is used for the axial SI A- and B-latches, as well as the radial SI A-latch and blind-mate connector interface and the FGS/RBM A-latch. The manual drive points for the axial SI are shown in Figure 3.2-9.

5. Axial SI indicator lights

The axial SI indicator lights operate in reversed logic, which means that positive indication of an event or configuration is the light going off. With no instrument present, all indicator lights are ON. This arrangement allows the EVA crewmember to verify proper operation of the indicator light circuit before using it for instrument installation. The SI indicator lights are powered from the orbiter by the SSM work light circuit. Indicator light panel locations are shown in Figure 3.2-9. A simplified schematic of the indicator light circuits for both the axial and radial instruments is described in Section 2.4.2.

3.2.3.2 Radial SI Mounting Provisions

The radial SI is mounted in the -V3 radial bay of the FPDA as shown in Figure 3.2-14. It is supported within the structure through three registration fittings (A, B, and C) and installed using a guide rail system to ensure proper alignment. The -V3 radial bay is shown in Figure 3.2-15 before instrument installation. An overview of these interfaces is provided in the following paragraphs. The installation and removal of a radial instrument is discussed in Section 3.3.

a. Point A registration fitting

The A-latch (SI) and A-fitting (FPDA) support the radial SI in all three HST axes. The A-latch, shown in Figure 3.2-16 and Figure 3.2-17, is a lead screw interface operated through a torque (jacking) rod that passes through the instrument to a manual drive interface on its external (outboard) surface (Figure 3.2-14).



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Figure 3.2-14. WF/PC exterior overview



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Figure 3.2-15. Radial SI cavity



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Figure 3.2-16. Radial instrument A registration fitting pair

3.2-36



Figure 3.2-17. Radial SI latch A concept

After the radial SI has been inserted into the radial bay cavity, along the guide rails, the A fitting in the FPDA is prepared to receive the A-latch. The A-latch is then manually driven to its latched position, drawing the instrument inward and seating the SI B- and C-fittings on their corresponding FPDA latches. Proper seating of the SI at the preload springs and subsequent latching can be verified by the torque required to drive the latch (32 to 35 ft-lb) and by counting the number of bolt turns following bolt engagement (approximately 22 revolutions). In addition, a plunger-actuated microswitch at the FPDA side (Drawing 3.2 sheet 6) is automatically opened approximately 20 revolutions after A-latch engagement, removing power to the redundant LATCHED lamp indicators. The indicator lights, located in the +V2 radial bay, behind the FGS doors, provide the microswitch status to the EVA crewmember (Figure 3.2-18). The manual drive interface is the same type as described for the axial SI A- and B-latches.

b. Point B and C registration fittings

The B- and C-fittings of the radial SI are simple ball and flexure arrangements similar to the axial SI C-latch. Latches B and C, which require no EVA operation, engage sockets with flexure fittings at each sidewall of the SI (Figure 3.2-19). As shown in Drawing 3.2 sheet 6, the C-latch has two flexures to provide two-axis constraint of the SI, while the B latch has one flexure to provide only single-axis (forward and aft) constraint. The latches are located on the FPDA and insert into the flexures upon partial engagement of the A-latch screw. As with the axial SI C-latch, deactivation of the A LATCH indicator lights is positive indication that the B- and C-fittings are properly engaged. Once all three latches are engaged, the SI no longer contacts the FPDA guide rails. All loads are then transferred through the three latches.

c. Radial SI blind-mate connector

The radial SI also incorporates a blind-mate connector system. The four connectors of the SI electrical interface are mounted to a fixed interface plate on the instrument structure as shown in Figure 3.2-19. They are mated through the manual operation of a lead screw through the center of the corresponding FPDA interface plate within the radial bay cavity, as shown in Figure 3.2-15and Figure 3.2-20. The FPDA plate is a floating interface to allow operation of the lead screw to draw the FPDA plate up to the interface for mating the connectors, while bullet-shaped guides correct any minor misalignment. A torque rod connects the manual drive interface on the exterior of the instrument (Figure 3.2-14) to the connector plate lead screw. This is the same type of drive interface employed for the A-latch and the axial SI A- and B-latches; however, the external interface requires 5.3 revolutions for mating or demating the connector.





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Figure 3.2-18. WF/PC indicator lights behind +V2 doors



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Figure 3.2-19. Radial SI B- and C-fitting and blind-mate connector

3.2-42



Figure 3.2-20. Blind-mate connector interface

d. Radial SI indicator lights

Proper mating of the four blind-mate connectors may be verified by the EVA crewmember using a circuit test switch and indicator light at the radial SI indicator light panel within the +V2 radial bay, as shown in Drawing 3.2 sheet 6 and Figure 3.2-18. The simple test circuit uses one pin of each of the four connectors in series with an indicator light and the SSM work light power source. The indicator light will illuminate when the test switch is placed in the CONN TEST position only if all four connectors have properly mated.

3.2.3.3 FGS RBM Mounting Provisions

The FGS RBMs are installed in the +V3 and \pm V2 radial bays, through a latch and guide rail arrangement similar to the radial SI. However, four fittings (A, B, C, and D) instead of three are used by the FGS RBM, which is completely enclosed in the aft shroud. Access to each radial bay is through access doors; electrical connections to the instrument must be manually demated individually. Figure 3.2-21 shows the FGS RBM before and after installation into the FPSA. The following paragraphs provide an overview of the FGS RBM mounting provisions, and a description of the FGS RBM changeout task is discussed in Section 3.4.

1. Point A registration fitting

The A-latch and A-fitting of the FGS RBM is identical to that of the radial SI shown in Figure 3.2-19. Figure 3.2-22 shows the A-latch lead screw relative to the FGS RBM pickoff mirror. The manual drive interface for operating the lead screw is on the top (outboard) surface of the instrument, and is the same type as used on the radial SI A-latch and the axial SI A- and B-latches.

2. Point B, C, and D registration fittings

The B-, C-, and D-fittings are all ball and flexure arrangements similar to the B- and C-fittings of the radial SI. The B- and C-fittings are shown in Figure 3.2-23. However, the doorknob portion of the B- and C-fittings on the FGS RBM is mounted to the instrument structure, and the flexure inserts are mounted within the radial bay cavity in the FPDA. For the D-fitting, the doorknob is mounted along the radial bay cavity centerline; the D-fitting flexure insert is mounted to the FGS structure, along the instrument centerline. Placement of all the registration fittings for the FGS RBM is shown in Drawing 3.2 sheet 7.

3. FGS RBM indicator lights

Each radial bay includes indicator lights that provide the microswitch status of the Alatch for FGS RBM changeout. As with the axial SI lights, these operate in reversed logic where proper instrument installation is verified by the indicator light turning OFF. Indicator light locations in each FGS radial bay are shown in Drawing 3.2 sheet 7.





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Figure 3.2-21. FGS installation

3.2-45



Figure 3.2-22. FGS detail



Figure 3.2-23. Fine guidance sensor B- and C-fittings

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3.2.4 Fine Guidance Sensor/Optical Control Subsystem

The FGS plays a dual role in the HST optical system, posing as an SI for astrometric data collection as well as a critical PCS sensor. The FGS system, working though the primary OTA optics, allows the HST to achieve its high-precision pointing requirements for science observations. Its role in the SSM PCS is discussed in Section 2.7. Included in each FGS RBM is an OCS that is controlled by the optical control electronics (OCE) mounted in the OTA ES. The FGS/OCS data is used by the STOCC to measure changes in OTA optical performance. Such changes may result from thermal distortion of the PM shape or small changes in the SM-to-PM relative position and orientation. The STOCC uses this information to correct the mirror figure through the PMA and SMA actuator systems.

An overview description of the FGS optical system is provided in Drawing 3.2 sheet 8. The physical characteristics of the FGS RBM are provided in Section 2.7. Instrument mounting provisions within the OTA FPDA are described in paragraph 3.2.3.

FGS Optical Description

The following discussion traces the optical light path through the FGS as described in Drawing 3.2 sheet 8. The starlight passes through the OTA primary optics and reaches the focal plane. The FGS total FOV is shown in detail A of sheet 8. The light is then reflected from the FOV of the main optics into the FGS by the pickoff mirror to an aspheric collimating mirror that is effectively placed behind the focal point of the primary optics. The collimator, in making the diverging light rays parallel, achieves an angular gain of 57.25 (the ratio of the primary focal length to the collimator focal length). In other words, if a star moves 1 arcsecond across the FOV of the primary optics, the light exiting the collimator will show an angular change of 57.25 arcseconds.

The light then enters the first star selector, a cylindrically shaped unit containing two mirrors and a refractive group of lenses. The whole cylindrical unit is then rotated for star selection, positioning the A-vector as shown in detail B.

The light then enters a second star selector that uses a modified double rhomb prism instead of mirrors. This star selector can be rotated independently of the first star selector and corresponds to the position of the B-vector in detail B of sheet 8. By adjusting both star selectors, the line of sight of the instantaneous FOV can be maneuvered anywhere in the 60-arcminute² FGS FOV. These two star selectors are the only moving parts in the FGS.

The light is then passed through a bandpass optical filter and a beamsplitter. The beamsplitter separates the light for the two independent x-y measurements and polarizes the light for the Koesters prism. The interferometer with Koesters prisms and final optical stages is shown in detail C.

The optical interferometer is a Koesters prism made up of two right-angle prisms adjoined at a coated semireflective surface, which reflects half of the light energy that strikes it and transmits half of the light through the interface with a quarter wave (90°) phase lag. Detail D shows the Koesters prism with a planar light wave impinging parallel to the prism face. Points A, B, and C represent the same point on the sinusoidal light wave; e.g., the peak. As ray A enters the prism, it is reflected and strikes the semireflective surface. The portion of light that has lagged 90° and passed through is A', and A" is the sinusoidal peak of the portion that is reflected. The rays A' and A" then exit the prism. A ray entering on the exact opposite side of the prism (point C) does the reverse, as shown in Drawing 3.2 sheet 8.

Notice that the rays C' and A" are 90° out of phase and when combined show up as a gray image on the Photomultiplier Tube (PMT). The rays A' and C" are also out of phase by the same amount and when combined will show up as the same shade of gray image on their PMT.

If the incoming light wave is rotated by a small angle (e.g., 0.01 arcseconds), as indicated in detail E, light ray A now enters the prism before the light ray C, and its respective components exit the prism earlier than before, while the light ray C' and C" exits later than in the previous case. Thus, the rays A" and C' move farther apart in phase, and when combined tend to interfere, causing the PMT image to be a darker shade of gray. On the other side of the prism, reinforcement occurs, causing a lighter shade of gray. The two PMT signals are then compared to determine the angle of the incoming light plane. A doublet lens and a mask are placed between the Koesters prism and the PMTs to focus the light and define precisely the 3- by 3-arcsecond² instantaneous FOV.

The FGE contain a microprocessor to handle feedback to the Star Selector Servo Motors (SSS). When the HST is star pointing, the FGS signal is fed back through a discrete proportional integral derivative control loop to the SSS, and the star selectors then track apparent motion of the star. The position of the star selectors is used by the PCS for fine pointing, together with the fine error signal that is performed by the PCS from the PMT counts.



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FGS OPTICAL AND MECHANICAL ARRANGEMENT



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SHEET 8 OF 11

3.2.5 OTA Equipment Section

The OTA ES is a 150° toroidal structure that mounts to the SSM ES forward bulkhead on the -V3 side of the spacecraft. As depicted in Drawing 3.2 sheet 9, the OTA ES consists of nine equipment bays, seven of which are used to house much of the OTA avionics. These components include the following:

- 1. EP/TCE and fuse modules bay H (shown in Figure 3.2-24)
- 2. DIU bay B (shown in Figure 3.2-25)
- 3. Actuator Control Electronics (ACE) bay E
- 4. OCE bay C (shown in Figure 3.2-25)
- 5. FGE (nos. 1, 2, and 3) bays D, F, and G (shown in Figure 3.2-26)

Of these units, only the EP/TCE and the DIU are mounted within the bay cavities. The rest are mounted on the hinged doors of their corresponding OTA ES bay. The doors are honeycomb structures insulated with MLI and/or silver flexible optical solar reflector material and latched closed by three EVA-compatible J-hooks. Mounting provisions for each avionics unit are detailed in Drawing 3.2 sheets 10 and 11.

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Figure 3.2-24. OTA ES bay H - EP/TCE and fuse modules

3.2-59



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Figure 3.2-25. OTA ES bay B (DIU) and bay C (OCE)

3.2-60



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Figure 3.2-26. OTA ES bay D (FGE)

3.2-61

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VIEW B-B DIU-1 BOTTOM PANEL CONNECTOR MAP







OCE



OCE INSTALLATION DA BAY C

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3.3 OTA POWER DISTRIBUTION

This section provides a detailed electrical schematic of the power distribution to all of the OTA subsystems (see Drawing 3.3). Power is received from the HST main buses and distributed to the OTA end items. Individual subsystem electrical schematics can be found in other portions of Section 3.

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3.4 BACKGROUND

The following two sections describe generic procedures for on-orbit changeout of the axial and radial SI components. It details specific procedures necessary to perform these operations, with supporting figures and operational precautions. These procedures lay the groundwork for flight-specific procedural development and are subject to change based on flight-specific requirements and constraints.

3.4.1 Radial SI (WF/PC) Changeout

Before beginning the EVA changeout, the WF/PC is powered down. Commands are sent from the ground or the crew, if the crew is given command and monitoring capability on the crew spec.

The +V2 aft trunnion scuff plate is removed by releasing the four captive fasteners 4 to 6 revolutions CCW each, as shown in Figure 3.4-1. The scuff plate is pulled forward to clear the trunnion pin support mounts, rotated 90° CCW, and pushed back onto the trunnion support mounts as shown in Figure 3.4-2. The two +V1 captive fasteners are then re-engaged to the trunnion support mounts. The two -V1 fasteners do not attach to the trunnion support mounts in this configuration. The scuff plate is tethered to the HST so there is no concern for losing the scuff plate during this procedure.

The two-bolt latches and the handle latch bolt on the +V2 FGS doors are disengaged, the handle latch PIP pin removed, the handle latch disengaged, and the +V2 FGS doors opened (see Figure 3.4-3). The WF/PC blind-mate connector indicator lights are tested by holding the momentary test switch to verify light integrity (see Figure 3.4-3).

Four WF/PC guide studs are then installed into the original WF/PC inserts (Figure 3.4-4). The WF/PC I portable handhold (Figure 3.4-5) is installed onto the original WF/PC by engaging the four zip nuts onto the WF/PC studs and hand tightening the nuts. It is important to avoid touching the WF/PC radiator because this could damage the unit.

After receiving verification that the WF/PC blind-mate connector has been deadfaced, the blind-mate connector drive mechanism is demated by turning the standard hexhead interface 5.3 revolutions CCW. The WF/PC groundstrap is disconnected and temporar-ily installed on the temporary stowage clip on the SSM ES (Figure 3.4-6 and Figure 3.4-7).

The WF/PC A-latch is then disengaged (22.5 revolutions CCW). The A-latch indicator lights activate after approximately two revolutions.

The original WF/PC is then pulled out of the WF/PC cavity using the portable handhold and the guide rail system. Caution should be exercised to avoid contacting the pickoff mirror during and after removal.

Prior to installation of the replacement WF/PC, the tangs of the WF/PC blind-mate drive connector mechanism are verified in the closed position.

The replacement WF/PC is inserted into the cavity and repositioned appropriately to engage the guiderails. The replacement WF/PC is fully inserted into its seated position, and the A-latch engaged (22.5 revolutions CW) and torqued. A light pressure of 3 to 5 pounds against the portable handhold may be needed to start engagement of the A-latch. The A-latch indicator lights deactivate after approximately 20 revolutions of the A-latch drive.

The groundstrap is then reconnected and torqued to approximately 10 ft-lb. The blindmate connector is mated and torqued by rotating the drive mechanism 5.3 revolutions CW. Once the blind-mate connector is mated, the connector test switch can be actuated to verify a good electrical connection. The WF/PC portable handhold is removed by disengaging the four zip nuts, restowing them on the adjacent stowage studs, and removing the handhold from the replacement WF/PC.

The +V2 FGS doors are closed by latching the handle latch and installing the PIP pin. The remaining two latches are latched, and all three doorlatch bolts torqued to approximately 23 ft-lb.

The scuff plate is rotated back into position by releasing the two +V1 captive fasteners, pulling the scuff plate to clear the trunnion support mounts, rotating the scuff plate back to its launch and landing configuration, and re-engaging all four captive fasteners (four to six revolutions each).

An aliveness test is performed after the SI installation to check the electrical path to the instrument.



Figure 3.4-1. +V2 aft trunnion scuff plate in launch/landing position



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Figure 3.4-2. +V2 aft trunnion scuff plate in servicing mission configuration



Figure 3.4-3. +V2 FGS doors and latches



Figure 3.4-4. Guide studs installed on WF/PC I



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Figure 3.4-5. WF/PC I portable handhold

3.4-7



Figure 3.4-6. WF/PC-I groundstrap (installed)



Figure 3.4-6. WF/PC-I groundstrap (installed)



Figure 3.4-7. WF/PC-I groundstrap (stowed)

3.4.2 Axial SI Changeout

Axial SI changeout procedures are identical for all axial SIs. Generic changeout procedures are described here with specific photo references to high-speed photometer (HSP)/COSTAR changeout operations as examples.

Before beginning the EVA changeout, the axial SI is powered down. Commands are sent from the ground or the crew (if the crew is given command and monitoring capability on the crew spec).

The aft shroud doors (Figure 3.4-8) are opened by disengaging the 3-bolt latches, disengaging the handle latch bolt, releasing the handle latch, and opening the doors. The doors may pop open approximately 2 inches when the handle latch is released. The doors are opened to the fully open position at the bump stop and tethered open. A view of the HSP and FOC is shown in Figure 3.4-9. The portable worklight assembly is installed into a worklight receptacle (Figure 3.4-10) and mounted on an interior HST handrail to illuminate the worksite area. The Intravehicular (IV) crewmember is instructed to turn on the worklight by throwing switch S24 on the SSP, activating the worklight and the SI indicator lights.

Upon verification of axial SI powerdown, the SI power and data connectors are disconnected and the cable harness tethered out of the way (Figure 3.4-11). The SI groundstrap is then disconnected and tied to the aft bulkhead under the MLI cover (Figure 3.4-12). The A-latch is then released to the hardstop position (12 to 16 revolutions CCW). The A LATCH indicator lights are verified (lights on indicates unlatched). The B-latch cover (Figure 3.4-13) is removed by pulling on the outermost ring to release the Velcro attachment and swinging the cover to the fully open Velcro attachment position (Figure 3.4-14). The B-latch is then released to the hardstop position (10 to 14 revolutions CCW). The SI is lowered 1-7/16 inches in the -V1 direction past the detent springs in the upper guiderail (Figure 3.4-15) until the bottom guiderail is contacted, in order to position the SI to be removed along the OTA guiderails. A constant force toward the middle wall separating the two SIs should be applied to maintain SI contact along the aft guiderail during lowering to prevent potential jamming. At this point, the A SEATED indicator lights (located adjacent to the B-latch interface) can be verified. The SI is then removed straight out of the cavity along the OTA guiderails, using the SI hand-rails for removal. The B READY indicator lights turn on during SI removal. Figure 3.4-16 shows the empty HSP/FOC cavity.



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3.4-11



Figure 3.4-9. AS (-V2) interior overview (HSP, FOC)



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3.4-13



Figure 3.4-11. AS (-V2) floor overview (+V3/-V2) for HSP/COSTAR changeout



Figure 3.4-12. AS groundstrap tie-back detailed view for HSP/COSTAR changeout



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Figure 3.4-13. SI B-latch cover (stow position) (-V2/-V3 axes)



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Figure 3.4-14. SI B-latch cover (open position)



Figure 3.4-15. HST upper guiderail details





Figure 3.4-16. AS (-V2) interior view (empty bay)

The new SI is inserted into place along the OTA guiderails until the B READY indicator lights are deactivated (the lights deactivate when the SI is in position for B-latch closure). Once again, a constant force toward the middle wall separating the two SIs should be applied to the SI to maintain SI contact along the aft guiderail during insertion to prevent potential jamming. The B-latch is then closed (10 to 14 revolutions CW) and torqued to 9 ft-lb. The SI raises 1-7/16 inches as the B-latch is engaged, and the A-and C-latches are seated. The A SEATED indicator lights turn OFF when the B-latch is torqued. The A-latch is then engaged (12 to 16 revolutions CW), torqued to 35 ft-lb, and verified latched from the A LATCHED indicator lights. The B-latch cover is released from the Velcro open position and reattached to the Velcro closed position. The SI groundstrap is attached and the power and data connectors remated using the identification and clocking markings on the connectors.

Worklight receptacle power is then removed by throwing switch S24 on the SSP, followed by removal of the worklight. The left aft shroud door is closed and held in position while the right door is partially closed and the handle latch engaged. The 4-bolt latches are then latched.

An aliveness test is performed after the SI installation to check the electrical path to the instrument.

4.0 SCIENTIFIC INSTRUMENTS

4.1 INTRODUCTION

A group of five versatile SIs are flown on the HST mission to conduct scientific observations. The SIs perform their primary function by converting the OTA focal plane light energy into digital data. The current complement of SIs comprises

- a. Wide Field/Planetary Camera II (WF/PC II)
- b. Space Telescope Imaging Spectrograph (STIS)
- c. Near Infrared Camera and Multi-Object Spectrometer (NICMOS)
- d. Advanced Camera for Surveys (ACS)
- e. COSTAR

These SIs are located in the OTA and are shown in their corresponding positions in Figure 4.1-1.

The WF/PC is the only radially mounted SI; the ACS, STIS, NICMOS, and COSTAR are all axially mounted. The typical weight of an axial bay instrument is about 700 pounds with approximate dimensions of 35 by 35 by 84 inches; the WF/PC is somewhat smaller and lighter. All instruments draw approximately 110 to 170 watts during observations.

The individual instruments are designed for modular replacement both on the ground and in orbit by a suited astronaut operating from the orbiter. Within each instrument are detectors, mechanisms, and submodules that can be interchanged without major disassembly or optical realignment.

In addition to the five dedicated SIs, the FGS (Sections 2.7 and 3.2) in the OTA is sometimes considered the sixth SI when used as an astrometry SI. Since only two of the three FGSs are required for the Pointing Control System (PCS), the third unit can be used for astrometric observations.

Power, command, and data interfaces to the SIs are shown in Drawing 4.1. A description of each SI follows.



Figure 4.1-1. Location of SIs in the OTA[k1]

4.1-2


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4.2 SI DESCRIPTIONS

4.2.1 Wide Field/Planetary Camera

The WF/PC is a high resolution imaging system capable of operation with a wide field focal ratio of f/12.9 as well as a planetary ratio of f/30. The WF/PC (shown in Figure 4.2-1 is the only radially mounted SI and is located on the -V3 side of the SSM. The WF/PC provides a sensitive and highly linear detector with a broad wavelength region (1150 A to 1.1 μ m) and a wide dynamic range (8 m to 28 m in the visual band). A filter carousel carrying up to 50 spectral filters, polarizers, or transmission gratings is also included. The resolution of this instrument surpasses that of ground-based instrument resolution by a factor of at least 10.

The WF/PC can be operated in two modes that are characterized by the two names Wide Field Camera (WFC) mode and Planetary Camera (PC) mode. WFC mode is used primarily for deep sky surveys (field: 2.7- by 2.7-arcminutes). PC mode provides higher spatial resolution capabilities over a smaller field view (1.2 by 1.2 arcminutes) than the WFC. Both camera modes use their wide dynamic ranges to photograph astronomical scenes of both faint and bright sources.

The scientific objectives of the WF/PC include

- a. Determination of the cosmic distance scale over longer distances larger than possible from the ground (by a factor of 10)
- b. Tests of world models and cosmic evolution
- c. Comparative evolutionary studies of distant and local galaxies
- d. Stellar population studies to very faint levels
- e. High-resolution luminosity profiles of galactic nuclei
- f. Energy distribution of stars and compact sources
- g. Dynamic motions in supernova remnants and protostars
- h. Synoptic studies of planetary atmospheres
- i. Search for extrasolar planets
- j. High resolution and Ultraviolet (UV) studies of comets

4.2.1.1 Background

The WF/PC was one of five scientific instruments launched aboard the HST on April 24, 1990. WF/PC occupies the center of the HST focal plane and uses its on-axis position to acquire wide- and narrow-angle images (WF and PC, respectively).



(a) V3 radical instrument installation (WF/PC I)



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(b) WF/PC I detail - UV flooding port

Figure 4.2-1. WF/PC installation

4.2-2

Before the launch of WF/PC I, JPL had already embarked on building a second camera (WF/PC II) for changeout at the end of the first camera's life. Originally WF/PC II (Figure 4.2-2) was planned to be identical to WF/PC I; however, because of problems identified during prelaunch testing (six thermal vacuum tests were performed on WF/PC I), modifications were proposed for WF/PC II (dealing mainly with improving ultraviolet throughput by avoiding self-induced optical contamination). Shortly after launch it was discovered that HST did not meet its specified optical performance criteria. Extensive analysis indicated that the telescope primary mirror has approximately 0.3 waves of spherical aberration, a considerable amount by optical standards; this aberration significantly compromised several planned scientific investigations.

Further analysis indicated that the originally intended image quality could be attained by refiguring key WF/PC optical surfaces to cancel out the telescope wavefront aberration. With the instrument already in the initial build phase, a rapid optical retrofit was undertaken on WF/PC II. Subsequent analysis indicated that the alignment of the HST exit pupil to WF/PC entrance pupil, an alignment that had not been critical on WF/PC I, was critical to the WF/PC II image quality (pupil shear). To ensure that pupil shear was kept acceptably small in all four optical channels, mechanisms were added to articulate the pickoff mirror and three of the four fold mirrors (pupil alignment of the fourth optical channel can be performed by the pickoff mirror alone).

WF/PC I	WF/PC II
Dual mode (four WFs or 4 PCs) selected by pyramid rotation	Single mode (three WFs + PC) fixed pyramid
Four WF optics/camera heads	Three WF optics/camera heads
Four PC optics/camera heads	One PC optic/camera head
Fixed pickoff mirror	Articulated pickoff mirror
Eight fixed fold mirrors nonarticulated	Fold mirrors + electronics; three articulated, one fixed
Texas Instrument charge-coupled devices (CCDs) require more cooling; higher read noise	Loral CCDs require less cooling; lower read noise
UV system throughput degraded by contamination	Contamination control: bakeouts, Zeolite contamination trap, venting
UV light pipe	Internal calibration subsystem
Quantum efficiency hysteresis (QEH) problem	No QEH

In a compromise between science requirements and project budget/schedule constraints, the instrument was downscoped as follows:

4.2.1.2 WF/PC Design Overview

The WF/PC II design switches between the wide-field and planetary-camera modes by a four faceted pyramid mirror illuminating three wide-field cameras and one planetary camera. The pyramid mirror splits the incoming telescope optical beam into four pieces and then relays these beams to four separate sets of optics and camera heads. These beams are then mosaicked back together to reconstruct the image as seen on the sky. This approach was dictated because the largest CCD detectors in the 1970 era (when

WF/PC I was being designed) were only 800 by 800 pixels; thus, they were not large enough to accommodate the full field-of-view.



Figure 4.2-2. WF/PC II

In WF/PC II, Loral CCDs have replaced the WF/PC I Texas Instrument (TI) CCDs. These devices require less cooling, have a lower read noise, and are cosmetically superior. Additionally, UV flooding is no longer needed to solve the QEH problem and so the associated flip mirror is now being used for a nonmission-critical internal calibration capability that subsystem assemblies have been exhaustively baking out prior to system assembly. In addition, Zeolite molecular absorber material is mounted at several locations within WF/PC II to capture any residual contaminates that remain after various bakeouts. A new filter set, containing narrowband UV and far UV filters, is also included in WF/PC II.

The remaining major elements of WF/PC II, the optical bench, the housing, and the radiator, remain essentially unchanged from the original design. Modifications were required, however, to permit mounting of the larger PC camera head in a WF optical bench slot. This relocation also necessitated the rebending of the associated heat pipe and the elimination of four unused heat pipes and the unused cabling for the deleted camera heads (the associated signal chain electronics are not deleted; they are wired as redundant units). Additionally, the Articulated Fold Mirror (AFM) electronics were mounted to the outside of the housing and a Remote Interface Unit (RIU) was provided for the command and telemetry interface of the AFMs. A protective cover for the pickoff mirror and two permanently fixed handles on either side of the WF/PC II for astronaut use during on-orbit replacement (Figure 4.2-3) have also been added to WF/PC II. The guiderails on WF/PC II have been lengthened by approximately 2 inches to further aid in the alignment and capture of the latch fittings.

A redesign of the internal and external vent paths for the WF/PC II was undertaken with the purpose of protecting the CCDs as much as possible from contamination from other parts of the instrument and to allow these contaminants to vent externally. WF/PC II redesign provided for removal of the aperture window, thus improving the venting of contaminants. In addition, WF/PC I was provided with a UV light pipe to aid in the on-orbit flooding of the CCDs using UV light from the Sun. In WF/PC II, the light pipe has been eliminated by using newly developed bias-gated CCDs, which do not require UV flooding.

The front bulkhead of the optical bench for WF/PC II contains four flat-fold mirrors and three mirror mechanisms. A set of four heat pipes is utilized in the WF/PC II in order to transfer the waste heat from the thermoelectric coolers to the radiator. WF/PC II contains several electrically powered mechanisms: the pyramid mechanism (rotation and focus), the shutter, the Selectable Optical Filter Assembly (SOFA), the CA flip mirror mechanism, the pickoff mirror mechanism, and the AFMs.

The pickoff mirror mechanisms are added to the WF/PC II design to ensure alignment of the optical axes of the HST and WF/PC, given the uncertainty of the latch interfaces and the OTA collimation. The mechanism provides a two-axis control of the pickoff mirror, with a step size of 0.2 arcmin and a total range of $\pm 0.25^{\circ}$.



sfoc_fl2705_165.tif

Figure 4.2-3. WF/PC handles

The purpose of the pickoff mirror cover (Figure 4.2-4) is to protect the pickoff mirror and mechanism from being bumped during the on-orbit EVA changeout of WF/PC II. This cover weighs approximately 7.5 pounds and is rectangular in shape, with five sides closed. The sixth side adjacent to the WF/PC is open to allow removal of the cover. The cover has four attachment points for securing to the WF/PC. The design uses the four existing lugs on the WF/PC housing instrument cover for attachment. The cover is rotated in a downward motion for release of the lugs.

Physical Description

The WF/PC is installed in the OTA and is radially oriented in the AS of the HST. It measures 52.5 by 43.0 by 21.5 inches and weighs 543.5 pounds. The WF/PC is fitted with a tether ring and portable handle to facilitate EVA servicing. Provisions for mounting of the WF/PC in the OTA are described in Section 3.2.

Electrical Power

The WF/PC is redundantly powered by PDU-4 (PRI) and PDU-3 (RED). The WF/PC RIUs are powered by PDU-4 (PRI) and PDU-3 (RED). The normal power requirement of the WF/PC is 62 to 150 watts. Drawing 4.2 sheet 1 provides the WF/PC electrical schematic.

Thermal Control

The WF/PC is operational for +50° to +68° F and will survive temperatures from -13° to +68° F. Thermal control of the WF/PC is maintained by heat pipes that conduct heat from the image detectors to an external radiator. This system forms part of the external surface of the AS. An annular gap around the radiator (with a light seal) provides a pressure vent for the AS during ascent and entry.

Command and Data Interfaces

The WF/PC scientific data is transmitted to the SSM via the CU/SDF within the SIC&DH module. Commanding and engineering TLM for SI status and operation is provided by the SIC&DH supervisory bus via BCU 5 (Drawing 4.1)

EVA Changeout Operations

See Section 3.4.1.



Figure 4.2-4. WF/PC pickoff mirror cover



sfoc_fl2705_167.tif

Right side

Figure 4.2-4. Continued



sfoc_fl2705_168.tif

Figure 4.2-4. Concluded

4.2-10



4.2.2 Space Telescope Imaging Spectrograph

The STIS is an axial *SI* that replaces and improves current HST spectrographic capabilities. STIS replaced the GHRS in AS bay 1. On Due to on-orbit failures in both Side 1 and 2 of the Power supply the STIS is currently in Safe mode. STIS Side 1 failed on May 16, 2001, and STIS Side 2 failed on August 3, 2004. See Appendix D for details on the individual power failures. Side 2 will be repaired on STS-125/SM 4, see section 11.3.3 for details on the repair.

The overall configuration of the STIS is shown in Figure 4.2-5. The entire STIS is mounted on the A, B, and C fittings. This interface is common to all of the axial SIs. These fittings support the STIS enclosure, which is the outer skin of the instrument. The enclosure provides the interface for the STIS, including the connector panel, astronaut handles, other necessary guides, rails, and handling points.

STIS is an extremely flexible advanced spectrograph with a large wavelength range (115 to 1000 nanometers (nm)). The STIS consists of three large area, two-dimensional detectors (1K by 1K). Two detectors are Multianode, Microchannel Arrays (MAMAs); the third is a CCD. The STIS is capable of a wide range of resolution and field of view with integrated correction for the HST aberration on the mirror. The STIS contains redundant electronics with an on-board processor and on-board system calibration.



Figure 4.2-5. STIS overall configuration

Physical Description

STIS is axially oriented in the AS in position 1 of the HST. The total weight of the instrument in the orbiter bay is approximately 798 pounds. The STIS static envelope measures 86.1 by 34.8 by 34.8 inches. Provisions for mounting of axial instruments in the OTA are described in Section 3.2.

The structural subsystem consists of two major components: optical bench and enclosure.

The main optical bench is the primary load bearing structure in the instrument. The main optical bench is a graphite-epoxy truss structure consisting of trapezoidal tubing longerons, square tubing cross members, and plate bulkheads. The X, Y, and Z fittings are the structural interface between the enclosure and the bench. Bulkheads of the bench also support the optical elements and detectors as well as providing light isolation barriers.

The enclosure is a rectangular box structure with one longitudinal edge chamfered. The enclosure consists of forward and aft bulkheads with joining panels. There are six removable panels and three fixed panels. The removable panels provide access to the

4.2-14

optical bench and internal electronics and mechanisms. The two rear outboard panels provide clearance for thermal radiators that are part of the TEC system for the CCD detector. An EVA handle is attached to one side of the enclosure. The electrical connector interface panel is on the top panel. The enclosure is vented through openings in the aft bulkhead. The enclosure is fabricated from aluminum honeycomb with graphite-epoxy face sheets. The MEB is supported by an integral honeycomb structural saddle that bolts into the enclosure. Two remote interface units attach to the MEB aft side plate. A CCD Electronics Box (CEB) attaches to the enclosure top plate. The instrument half of the A, B, and C latch fittings bolt respectively to the X, Y, and Z fittings on the enclosure bulkheads.

Electrical Power

The STIS is powered by PDU-2 (PRI) and PDU-1 (RED). The STIS RIUs are powered by PDU-2 (PRI) and PDU-1 (RED). The normal power requirement for the STIS is 113.0 watts. Drawing 4.2 sheet 2 provides the STIS electrical schematic.

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Mechanisms

There are seven mechanisms in the mechanical subsystem: mode select mechanism, slit wheel mechanism, corrector mechanism, mode isolation mechanism, Echelle blocker mechanism, CCD shutter mechanism, and calibration input mechanism. The STIS uses the mechanisms to focus, position, and condition incident light energy to the optical detectors. The corrector mechanism moves a mirror along one axis of linear motion and about two axes of rotational motion. The mode select mechanism positions one of 21 optics in the light path and scans the light path over a $\pm 5^{\circ}$ range. The slit wheel mechanism moves various science slits and target acquisition apertures into the optical beam. The CCD Shutter positions a shutter to control the exposure time of the CCD detector. The mode isolation mechanism prevents unwanted light input to the MAMA detectors. The Echelle blocker mechanism prevents light from reaching the unused set of Echelle gratings. The calibration lamp input mechanism positions a mirror to relay a calibration source beam into the optical path.

Optics

The optical subsystem consists of three detectors and the optical elements necessary to process the input image received from the HST and focus the processed image onto one of the detector apertures. The system uses two MAMA detectors and a CCD detector to convert light energy to electronic signals that are processed and transmitted to Earth.

The light entering the instrument is focused by the corrector mechanism and a correction mirror onto a slit or aperture in the slit wheel. Light passing through the slit wheel is collimated and the collimated beam is focused on an optic in the mode select mechanism. Light from the mode select mechanism is directed to one of the detectors directly or by relay mirrors and Echelle gratings. Light from calibration lamps (not shown) is directed to the entrance slit in place of the HST input image.

Thermal Control

The STIS uses passive and active thermal control to maintain proper on-orbit temperatures. In the payload bay the active controls are nonoperational. The thermal environment is established by the SIPE. The STIS passive thermal control system consists of MLI and selected surface finishes. The external surfaces of the instrument are graphiteepoxy. All MLI are in the interior of the enclosure.

The survival time of STIS during EVA is driven by the temperature of the graphite epoxy enclosure panels. The lower limit is between -50° and -75° C.

Command Data Interfaces

The STIS science data is transmitted to the SSM via the CU/SDF within the SIC&DH. Commanding and engineering TLM for SI status and operation is provided via the SIC&DH supervisory bus. The STIS interfaces to the supervisory bus via the BCU 1 (Drawing 4.1).

4.2.3 Near Infrared Camera and Multi-Object Spectrometer

The NICMOS is the first spaceborne astronomical near-infrared instrument. NICMOS is an axial scientific instrument that was installed into aft shroud bay position 2 during SM-2, replacing the *Faint Object Spectrograph (FOS)*.

The overall configuration of the NICMOS is shown in Figure 4.2-6. The entire NICMOS is mounted on the A, B, and C fittings. This interface is common to all the axial SIs. These fittings support the NICMOS enclosure, which is the outer skin of the instrument. The enclosure provides the interface for the NICMOS, including the connector panel, astronaut handles, other necessary guides, rails, and handling points.



Figure 4.2-6. NICMOS scientific instrument

NICMOS is an optical sensing instrument designed to operate in the near infrared spectrum. The instrument uses a solid nitrogen cryogenic system to cool sensors in order to improve their sensitivity. The instrument consists of three cameras that can operate simultaneously and asynchronously at three different magnifications and fields of view. The three 256 by 256 HgCdTe arrays are maintained at 58 K inside a 5-year dewar with solid N₂ and thermoelectric coolers. The spectral coverage ranges from 0.8 to 2.5 μ m with wavefront correction provided internally by a corrector optic at the common intermediate pupil plane. Prisms are included in the filter suite for the widest field camera (camera 3) to provide multiobject spectroscopy with a resolution of about 200. The occulting spot in the camera 2 intermediate image plane provides coronagraphic capability.

Physical Description

The total weight of the instrument in the orbiter bay is approximately 847 pounds. The NICMOS static envelope measures 86.1 by 34.8 by 34.8 inches. Provisions for mounting of axial instruments in the OTA are described in Section 3.2.

The NICMOS structural subsystem consists of two major components, the optical bench and enclosure.

The main optical bench is the primary load bearing structure in the instrument. The bench is a graphite-epoxy truss structure consisting of trapezoidal tubing longerons, square tubing cross members, and plate bulkheads. The X, Y, and Z fittings are the structural interface between the enclosure and the bench, while the K, L, and M fittings are the structural attachment points for the dewar. A truss-type bracket assembly is attached to the main bench to support and position optical elements external to the dewar.

The enclosure is a rectangular box structure with one longitudinal edge chamfered. The enclosure consists of forward and aft bulkheads with joining panels. There are three removable panels and two fixed panels. The two removable outboard panels have removable covers to provide access to internal electronics. The outboard panels also have openings for thermal radiators on the dewar. One of the outboard panels will have EVA handles attached. The chamfered edge has openings for thermal radiators and a cryogen service panel. The electrical connector interface panel is also on the chamfered end. The enclosure is fabricated from aluminum honeycomb with graphite-epoxy face sheets. The MEBs are supported by an integral aluminum honeycomb structural saddle that bolts into the enclosure. The instrument half of the A, B, and C fittings bolt respectively to the X, Y, and Z fittings on the enclosure bulkheads.

Mechanisms

There are three types of mechanisms in the mechanical subsystem, the *Pupil Adjust Mechanism* (PAM), the field offset mechanism (FOM), and the filter wheel mechanism. These mechanisms focus, position, and condition incident light energy to the optical detectors. The PAM moves a mirror along one axis of linear motion and about two axes of rotational motion and is mounted on the main optical bench. The FOM moves a mirror about two axes of rotational motion and is mounted on the fore-optics bracket. There are three filter wheel assemblies that move various filters into the light path of one of three detectors and are internally mounted in the dewar vacuum shell.

Optics

The optical subsystem consists of three cameras and the optical elements necessary to focus the input image received from the HST onto one of the camera detector arrays. Figure 4.2-7 depicts the overall optical subsystem described as follows:



Figure 4.2-7. Optical system stages

A relay system forms an initial image of the telescope focal plane and is common to all instruments (stage 1). Stage 1 consists of the Turning Mirror (TM) mounted on the PAM, the Reimaging Mirror (RM), and the FOM. The output of stage 1 is directed to the Field Dividing Mirror (FDM) assembly by the FOM mechanism. A uniform illuminator is also located on the FOM mechanism for calibration purposes.

Stage 2 consists of three separate light relay paths, one for each camera. Each path consists of a FDM, two focusing mirrors that form a telescope, and a folding mirror that positions the light beam on the appropriate aperture window in the dewar vacuum shell front cover.

The third and final stage of the optical subsystem consists of the three cameras and the other dewar-mounted optical elements, which include the aperture windows, cold shield, window filter wheels, and cold masks for each camera. The aperture windows form part of the vacuum shell. The other third stage elements all operate in the vacuum space of the dewar.

Cryogenics

The cryogenic subsystem is a solid subliming cryocooler that uses solidified nitrogen in conjunction with Thermoelectric Coolers (TECs) to maintain a 58 K environment for the instrument detectors. Figure 4.2-8 shows the dewar configuration, while Table 4.2-1 summarizes the characteristics of the dewar. The cryogen is stored in a cylindrical tank surrounded by MLI, three cold shields, and the outer vacuum shell. The cryogen tank internal volume is filled with a low-density, open-pore aluminum foam that improves thermal stability and coupling. A cold well in the cryogen tank provides the cold environment for the instrument detectors. The vacuum shell is closed by a front cover which supports drive motors for three optical filter wheels inside the dewar. Light to the three cameras enters the dewar through fused silica windows in the front cover.



Figure 4.2-8. Dewar external configuration

Cryogen	Solidified nitrogen
Cryogen fill	238 pounds (108.2 kg)
Operating temperature	40-58 K maintained by periodic cooling
Weight	419 pounds fully loaded
Vacuum shell size	51 by 27 inches
Cryogen tank size	38 by 17.8 inches
Cryo tank usable volume	7240 cubic inches (118.5 liters)
Cryogen fill efficiency	Approximately 95 percent
Cryogen operating pressure	0.6 psia
Overpressure protection	14 + 1 psid burst disks on cryo tank and vacuum shell

Table 4.2-1. Dewar characteristics

Figure 4.2-9 depicts the dewar plumbing schematically. The plumbing components serve three functions, cryogen fill and vent, dewar cooling, and vacuum acquisition and maintenance. Three filtered lines in the forward dome of the cryogen tank are manifolded together to a single fill/vent line that is routed along and thermally bonded to the innermost vapor-cooled shield. The line then exits the dewar through the aft girth ring. Outside the dewar, the line divides. One branch passes through a manual valve to the capped capillary vent line. The other branch leads to a burst disk and an emergency vent line. For filling, the capillary vent line is disconnected at the valve and a liquid nitrogen supply is connected in its place. Following completion of cryogen fill and

solidification, the manual valve is closed and the capillary vent line is reconnected causing the system to be "locked up" at an internal pressure of 0.6 psia.



Figure 4.2-9. Dewar plumbing configuration

Cryogen solidification and recooling to maintain the cryogen at 40-58 K is achieved by flowing cold helium gas through a cooling loop consisting of cooling coils attached to the inner and outer surfaces of the cryogen tank aft dome. The lines pass into and out of

the dewar through vacuum-jacketed lines and valves. The vacuum jackets form part of the total evacuated volume of the dewar.

The dewar is evacuated through a line that is welded into the aft girth ring. Outside the dewar, the line branches, one branch going to the vacuum acquisition fitting and the other to a burst disk. Initially, the dewar is evacuated through the vacuum acquisition fitting by a mechanical pump. Vacuum is maintained and can be measured through the electrical vac ion pump during ground operations. The vac ion pump and pressure transducer will be nonoperational for launch and instrument installation in the HST.

The vacuum line burst disk provides pressure relief capability in the event of cryogen tank failure and release of cryogen into the vacuum shell. Identical burst disks and outlet line operate at warm temperatures to prevent blockage due to ingestion and freezing of moisture. The vent outlet port will be covered by a diffusing debris cap to prevent blockage.

The burst disks are a "reversing membrane" design. Inlet pressure against the membrane works against the force of a belleville washer that establishes the burst pressure. When the inlet pressure exceeds the preset burst pressure, the membrane snaps over against a cutting edge that penetrates the membrane allowing flow from the inlet to the outlet vent. Relief pressure is preset by adjusting the preload on the belleville washer.

Electrical Power

The NICMOS is powered by PDU-3 (PRI) and PDU-4 (RED). The NICMOS RIUs are powered by PDU-3 (PRI) and PDU-4 (RED). The NICMOS power ranges from 62.5 to 158 watts. Drawing 4.2 sheet 3 provides the NICMOS electrical schematic.

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ASSEMBLY CLEARS HANDHOLD BRIDGE LATCH 1 BETWEEN GUIDE RAIL STANDOFFS



ASSEMBLY OVER HANDHOLD BRIDGE LATCH 1 BEGINS MOVING UNDER GUIDE RAIL



LATCHES ENGAGED



ASSEMBLY IN POSITION

CRYOCOOLER INSTALLATION SEQUENCE (ESM INSTALLATION IS SIMILAR)





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2 NICMOS CLNG SYS AFT Shroud Inst det 4

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DSGN	J. WEED	3/6/01					
DR			NICMOS COOLING SYSTEM				
ENGR	T. ARNOLD						
ENGR			DETAILS				
APP			DETAILS				
APP			HST CSM	- DWG NO 4. 2			
CADS			ALL				
LTR	PCN	-	82.5 X 34		SHEET 4 OF 6		
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Thermal Control

NICMOS uses passive and active thermal control to maintain proper on-orbit temperatures. In the payload bay the active controls will be nonoperational. The thermal environment is established by the SIPE. The NICMOS passive thermal control system consists of MLI and selected surface finishes. The external surfaces of the instrument are graphite epoxy. All MLI are in the interior of the enclosure. *Unfortunately due to a thermal short between the cold baffles and the surrounding Vapor Cooled Shield (VCS) reduced the lifetime of the cryogen and thus the anticipated period of science operations from 4.5 to 1.5 years.*

The NICMOS Cooling System (NCS) was installed to restore NICMOS to operation and extend the life of the instrument beyond the original plans. NCS is designed to maintain a temperature between 58 and 72 K (the goal is 70 K) with stability of 0.1 K/orbit and long-term stability of 0.5 K/year.

NCS Hardware Overview

NCS comprises three major elements: NICMOS Cryocooler (NCC), NCS Capillary Pump Loop (CPL)/radiator, and the Electronics Support Module (ESM). The NCC cools the NICMOS detectors by circulating neon through existing plumbing originally used, prelaunch, to circulate helium to freeze liquid nitrogen in the dewar. The connectors have EVA friendly bayonet fittings. The NCS CPL/radiator removes heat from the NCC to the external environment. The heat transport capability is in excess of that required for the NCC under any operating condition and can be utilized for future needs via an interface plate on the radiator. The ESM provides the command, control, and telemetry functions.

There are three NCS coolant loops. The circulator loop flows neon to carry heat from NICMOS to the NCC. The cryocooler loop, housed within the NCC, uses neon to carry heat from the circulator loop to the CPL. The CPL uses ammonia (NH_3) to carry heat from the NCC to the radiator, where the heat is given up into the space environment.

The NCC circulator loop uses a centrifugal circulator to pump a constant flow of neon between the NCC and NICMOS. The circulator has a permanent magnet synchronous motor driven by the Power Conversion Electronics (PCE), operating at 1200 rev/s and nominally yielding a flow rate of 0.4 g/s of neon at 4 atm. The circulator loop removes heat from the NICMOS instrument and conductive parasites.

Heat flows from the circulator loop to the cryocooler loop via the Cold Load Interface (CLI). The CLI is a stainless steel counter flow fin-type heat exchanger with a thermal effectiveness of 0.963. The CLI provides structural support for the circulator, turboalternator, and recuperator.

The cryocooler loop uses two high-speed turbomachines, the turboalternator and the centrifugal compressor, to provide expansion and compression of the neon as it transfers heat to the Heat Rejection Interface (HRI). The HRI provides a conductive

4.2-31

coupling between the NCC, PCE, and CPL. Neon gas expands on the face of a turbine rotor that turns an alternator. The electricity generated by the alternator flows through a selectable resistive load in the PCE. Heat created in the load is rejected into the HRI and later rejected by the radiator. In this way heat is extracted from the cryocooler loop. A centrifugal compressor, at the warm end of the loop, provides compression to the neon. The compressor is driven by a three-phase induction motor with a maximum operating speed of 7500 rev/s and is bolted to the HRI. A recuperator provides heat transfer between the high- and low-pressure streams of gas in the cryocooler loop. The cold end of the recuperator is supported by the CLI, and the warm end is supported by the NCC structure.

The NCS CPL/radiator interfaces with the NCC at the HRI via the CPL evaporator (CPL pump) that carries heat from the NCC to the external radiator. Cold liquid coming from the radiator flows into the evaporator and is vaporized transferring heat from the NCC. This warm vapor carries heat out of the evaporator to the radiator via stainless steel tubing. The vapor enters a heat pipe heat exchanger in the radiator and condenses in the header heat pipe. Heat from the header heat pipe is conducted to a series of spreader heat pipes throughout the radiator, where heat is then radiated to the space environment. The condensed vapor moves through a subcooler region to cool the liquid prior to exiting the radiator and returning to the HRI.

Detailed descriptions of each component of the NCS can be found in Section 10.3.5.2 through 10.3.5.6.

Command Data Interfaces

The NICMOS science data is transmitted to the SSM by the CU/SDF within the SIC&DH. Commanding and engineering TLM for SI status and operation is provided by the SIC&DH supervisory bus. The NICMOS interfaces to the supervisory bus through BCU-2.

4.2.4 Advanced Camera for Surveys (ACS)

The Advanced Camera for Surveys (ACS) is a third generation imager for HST that replaces the Faint Object Camera (FOC) in the -V2/-V3 quadrant of the HST aft shroud. ACS has a field of view greater than twice as large as WFPC2 and is greater than five times more sensitive. The ACS is launched in the Second Axial Carrier (SAC) Axial Science Instrument Protective Enclosure (ASIPE). The ACS will be unpowered before installation in the HST because the only electrical connection to the ASIPE is the grounding strap.

The instrument has three optical channels: the Wide Field Channel (WFC), the High Resolution Channel (HRC), and the Solar Blind Channel (SBC). The WFC has a large field of view for rapid surveying. The HRC has a smaller field of view with a corresponding increase in resolution. The SBC blocks visible light to operate in the far-UV spectrum. ACS can also perform spectroscopy, coronography, and imaging polarimetry. The ACS is designed to increase the capability of the HST for surveys and

discovery by at least a factor of 10. Its mission is to study the evolution of galaxies, galaxy clusters, and the early universe.

ACS is a versatile instrument that can be applied to broad range of scientific programs. The high sensitivity and wide field of the WFC in the visible and near-infrared will make it the instrument of choice for deep imaging programs in this wavelength region. The HRC, with its excellent spatial resolution, can be used for high-precision photometry in stellar population programs. The HRC coronograph can be used for the detection of circumstellar disks and Quasi-Stellar Object (QSO) host galaxies.

Physical Description

The ACS is very similar to previous HST axial science instruments such as STIS, NICMOS, and COSTAR. It measures 86.5 by 35.1 by 35.1 inches, roughly comparable to a telephone booth, and masses 875 pounds. The exterior surfaces are composed of graphite epoxy and metal. A door approximately 3 inches inside the ACS aperture blocks light but does not seal the enclosure.

ACS has numerous interfaces. For EVA manipulation, an EVA handhold and L-Handle are installed on the -V2 face of the instrument. Mechanically, ACS interfaces HST via three kinematic latches, labeled A, B, and C. Jumpers J1 and J2 (power), J3 and J4 (data), and a groundstrap electrically connect ACS to the telescope. Finally, ACS is designed to be cooled via the HST Aft-Shroud Cooling System (ASCS) for greater performance. The ACS to ASCS thermal interface is via two capillary pump loop connectors. ASCS has been demanifested from flight SM3B, but installation is expected on SM4.

See Figure 4.2-10 to Figure 4.2-13.



Figure 4.2-10. ACS in a ground fixture, showing outboard panels (note the EVA handhold)



Figure 4.2-11. ACS in a ground fixture, showing inboard panels



Figure 4.2-12. ACS Electrical Connector panel with nonflight shorting plugs

4.2-35



Figure 4.2-13. ACS -V2 Outboard Panel (this face contains CCD radiator; small diagonal face contains connector panel on left and two CEB radiators to right)

The ACS structural subsystem consists of two major components: the fixed truss optical bench and the enclosure.

The enclosure provides the interfaces for the ACS, including the connector panel, astronaut handles, and other necessary guides, rails, and handling points. Inside the enclosure the electronics and thermal shelf are supported from enclosure structural members. The truss assembly is kinematically mounted to the A, B, and C fittings using X, Y, and Z fittings. Note that the truss assembly is often referred to as the optical bench, and these two terms are used interchangeably.

The optical bench is the primary load bearing structure in the instrument. The bench is a graphite-epoxy truss structure consisting of trapezoidal tubing longerons, square tubing cross members, and plate bulkheads. Truss members are joined with bonded fingerplates. Bonded-in metal fittings provide load points for interfacing with other components of the instrument. The optical bench bulkheads support the optical elements and detectors. The X, Y, and Z fittings are the structural interface between the enclosure and the bench. The X fitting provides three-axis constraint. The Y axis fitting constrains lateral motion but allows axial motion through a sliding spherical bearing. The Z fitting constrains rotation about the X fitting.

The electrical connector interface panel is on the top chamfered panel. The enclosure is vented through openings in the aft bulkhead as well as through the aperture. These vents are sized to the pressure profiles of the orbiter ascent and descent rates.

The enclosure is fabricated from aluminum honeycomb with graphite-epoxy face sheets and is designed to withstand launch and loading loads and to have axial stiffness of 75,000 lb/in. The panels are not vented but are designed to withstand the pressure differentials of ascent/descent.

An integral honeycomb structural saddle that bolts into the enclosure supports the Main Electronics Box (MEB). The honeycomb is 0.5 inches thick with 0.015-inch aluminum face sheets. Two Remote Interface Units (RIUs) attach to the MEB aft side plate. Two CCD Electronics Boxes (CEBs) attach to the enclosure top plate.

Optical Subsystem

The optical subsystem provides three channels: a WFC, an HRC, and an SBC. Note that in this context, the words "channel" and "camera" are used interchangeably. The WFC has a large 202 by 202 arcsecond field of view with a resolution of 0.05 arcsecond/pixel for rapid surveying. The HRC has a 26- by 29-arcsecond field of view with a resolution of 0.027 arcsecond/pixel. The SBC blocks visible light to operate in the far-UV spectrum. It has a 31 by 35 arcsecond field of view at a resolution of 0.032 arcsecond/pixel.

Each channel consists of a detector and the optical elements necessary to process the input image received from the HST and focus the processed image onto one of the detector apertures. The system uses one Multi-Anode Microchannel Array (MAMA) detector and two Charge Coupled Device (CCD) detectors to convert light energy to electronic signals that are processed and transmitted to Earth. There are six mirrors, 56 filters, and four windows used in the optical system. These optical elements are made of various glasslike materials, including fused silica, MgF₂, CaF₂, BaF₂, and LiF.

The ACS uses nine electromechanical assemblies in the optical train to focus, position, and condition or block incident light energy to the optical detectors when operating in the HST. The mechanisms will be unpowered and nonoperational during prelaunch and launch operations of the shuttle. Each mechanism will be operated briefly during the functional test after ACS installation.

There are two types of calibration lamps used in the optical system: one deuterium lamp and four tungsten lamps. The deuterium lamp has a Ni63 oxide source that is slightly radioactive.

Electrical Subsystem

The electrical system consists of two MEBs, RIUs, two CCD electronics boxes located inside the top panel, and an Auxiliary Power Box (APB) on the -V3 enclosure panel.

Thermal Subsystem

Before installation, the ACS is thermally maintained by the ASIPE. After installation, ACS has both an active and passive thermal control subsystem to maintain the various components with acceptable operating limits. Active heaters, thermal straps, thermal blankets, surface finishes, heat pipes, and special radiators are utilized in this subsystem. The optical bench is maintained at $20 \pm 2^{\circ}$ C to minimize thermally induced structural deformations that would degrade optical performance. The electronic compartment operating temperature range is from -25 to +40°C, depending upon spacecraft and instrument operating conditions.

The heat energy dissipated by the MAMA detector is transported to the aft bulkhead by four heat pipes. The CCD detectors are mounted on thermoelectric coolers to allow the detector chip operating temperature to be controlled at -80°C. Two heat pipes from each CCD assembly transport heat to a heat transfer plate. Two additional heat pipes transport the heat from the transfer plate to an external radiator on an outer panel of the enclosure. The working fluid in the heat pipes is anhydrous ammonia.

For passive thermal control, the instrument exterior finish is the natural black of the graphite-epoxy except for radiator surfaces. The MEBs and CCD electronic boxes have aluminum radiator plates, anodized black mainly with small areas of chromate conversion coating.



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4.2.5 COSTAR

The mission of COSTAR, a second-generation HST instrument, is to place and align mirror corrective optics for the FOC, the FOS, and the GHRS. COSTAR replaced the original *High-Speed Photometer* (HSP); *however, as the new SIs have been developed and installed, they have incorporated the corrective optics into their design, and thus have overcome the need for the COSTAR SI. This SI is due to be replaced on SM-4.*

The overall configuration of the COSTAR is shown in Figure 4.2-14. The entire COSTAR is mounted on the A, B, and C fittings. This interface is common to all of the axial SIs. These fittings support the COSTAR enclosure, which is the outer skin of the instrument. The enclosure provides the interface for the COSTAR, including the connector panel, astronaut handles, other necessary guides, rails, and handling points.

Inside the enclosure, the electronics and thermal shelf are supported on rafters. The enclosure also supports the fixed optical bench on a set of kinematic mounts. The Deployable Optics Bench (DOB) rail deployment mechanism and its actuators are mounted on the fixed optical bench. The DOB deploys to place mirrors in front of the other SIs. The DOB is shown in Figure 4.2-15.

The optical subsystem provides for imaging correction for the FOC, FOS, and GHRS. The electronic subsystems provide for communication to and from the HST and for control of COSTAR functions. The thermal subsystem provides for both passive and active thermal control of COSTAR.

Physical Description

The COSTAR is attached to the OTA. The COSTAR is axially oriented in the AS in position 4 of the HST. It measures 86.5 by 34.6 inches and weighs less than 700 pounds. The COSTAR is equipped with a handle for EVA servicing. Provisions for mounting axial instruments in the OTA are described in Section 3.2.

Electrical Power

The COSTAR is powered by PDU-4 (PRI) and PDU-3 (RED). The COSTAR RIUs are powered by PDU-4 (PRI) and PDU-3 (RED). The normal power requirement for the COSTAR is 43 to 64 watts. Drawing 4.2 sheet 5 provides the COSTAR electrical schematic.

Thermal Control

The COSTAR is operational in temperatures from -4° to +104° F. The COSTAR will survive temperatures from -4° to +104° F. Thermal control is passive; i.e., operating power is dissipated by radiation through the HST AS.

Command/Data Interfaces

The COSTAR SD is transmitted to the SSM by the CU/SDF housed within the SIC&DH. Commanding and engineering TLM for SI status and operation is provided by the SIC&DH supervisory bus. The COSTAR interfaces to the supervisory bus through BCU-4 (Drawing 4.1).



(a) Cutaway view

Figure 4.2-14. COSTAR configuration



(b) COSTAR installed in OTA SISS

Figure 4.2-14. Concluded



Figure 4.2-15. DOB



4.2.5.1 COSTAR Structural Elements

The COSTAR structural subsystem consists of three major components, the enclosure, the Fixed Optical Bench (FOB), and the DOB.

The COSTAR enclosure is composed of 6061-T651 aluminum and is configured as two bulkheads. The bulkhead rib thickness gauge is 0.050 to 0.750 inches. In depth, the bulkhead wall thickness is approximately 0.062 to 0.380 inches. The FOB is mounted to the bulkheads by a set of kinematic mounts.

The removable outboard panel of the COSTAR enclosure is 5052-H39 aluminum honeycomb with a thickness of 0.550 inches and a cell size of 3/16 inches. The face sheets are 6061-T6 aluminum. The fixed inboard panel of the COSTAR enclosure is 5052-H39 aluminum honeycomb with a thickness of 0.550 inches. and a cell size of 3/16 inches. The face sheets are 6061-T6 aluminum. The panels are designed to withstand launch loads and to provide a stiffness to 75,000 in-lb.

Longitudinal stringers run from the forward to the aft panel, at the corners to which rafters are attached. The rafters serve to stiffen the outboard panel and to provide a place to mount the electronics. Handles are attached to the rafters to assist in EVA activities.

A kinematic mounting system is used to mount the fixed bench within the enclosure. The front ball mount restrains the fixed bench in all three lateral axes. The rear support links the bench in two lateral axes, but allows it to float in the longitudinal direction. The final fitting is an antirotational link that prevents rotation about the other two fittings.

4.2.5.2 COSTAR Mechanical Systems

The three mechanism types are as follows:

- a. DOB deployment
- b. Arm deployment
- c. Tip/tilt/focus

The DOB deployment drive mechanism support, deploys, stows, adjusts, and provides an electrical connection for the DOB. The deployment mechanism is mounted between two bulkheads on the aft end of the FOB. The track used to guide the DOB is cinematically mounted to the FOB. Seven rollers constrain five degrees of freedom; a sixth degree of freedom is constrained by drive linkage and a compliant stop.

The deployment drive mechanism extends the DOB 19.49 inches through the opening in the front bulkhead. Adjustment provisions can extend the DOB out an additional 0.3937 inches. During normal operations, the crank arm is rotated on a set of bearings by the motor/gearbox assembly. In the event of an electrical or mechanical failure within the assembly, the DOB can be manually retracted. During manual retraction,

4.2-47

the manual retract mechanism rotates the entire housing which, in turn, retracts the crank arm and the DOB.

The housing bearings provide redundancy for manual show. The flex links, which provide support to the electrical connections for the DOB, have a wet-lubed, slip-fit interface between the shaft and inner race of the bearings that facilitates redundancy. This design is also utilized to achieve redundancy in the spherical rod end bearings in the connecting link between the DOB and the crank arm.

The arm deployment mechanisms deploy the FOC M1, FOC M2, FOS M2, and the GHRS M2 arms to place the mirrors in the desired light paths for the FOC, FOS, and the GHRS. The arms are designed using beryllium for stability. An *overcenter* spring cartridge ensures positioning of the arm against the stowed or deployed stops. A motor/gear head drive through a bevel gear pair to raise and lower the arms. Arm deployment positions are sensed by a potentiometer and sent back to the remote interface units to provide position verification. Heaters are attached to the arms to actively provide thermal stability. An emergency stow linkage system and shear pin ensures storing of the arms in the event of electrical failure. Shearing the pins requires 33 in-lb. The snubber springs in the primary sprint assist in preventing the arms from vibrating off the stowed stops during launch.

The tip/tilt focus mechanism provides for on-orbit alignment and focusing of the FOS and GHRS M1 mirrors. Titanium and CRES with matching CTEs are used for structural stability and stiffness. The mirrors are mounted on a titanium U-joint or two-axis flexure pivots for tip/tilt adjustment. Tip/tilt is actuated by motor driven screws (motor micrometers). Nonjamming stops in the actuator limit motion to +30 arcminutes. Differential impedance transducers are used for sensing tip/tilt position of the mirrors. All actuators (tip/tilt/focus) are preloaded to eliminate free play. The GHRS and the FOS M1 mechanisms are mounted to a titanium structure, thus ensuring a thermal match. Invar flexures adapt this structure to the low CTE DOB. Adjustable shims located at the mirror/mechanism interface allow for precise adjustment of the mechanism.

4.2.5.3 COSTAR Avionics

The electrical subsystem employs redundant, open loop systems to ensure reliability. The redundant subsystems provide operational versatility and minimal impact in the event of a failure. The electrical subsystem will remain inactive until after the COSTAR has been installed in the HST. There are no exposed voltages associated with the COSTAR.

The electrical subsystem comprises the following:

- a. RIU
- b. Main Electronics Box (MEB)
- c. Position sensors

d. Stepper motors

The RIUs are provided as Government Furnished Equipment (GFE). The two RIUs receive commands from the HST and transmit those commands to the MEBs. The RIUs receive telemetry from the MEBs and position sensors, then transmit the data to the HST.

The MEB is divided into two redundant subsystems, side 1 and side 2, allowing each side to provide open-loop control for the stepper motors for each mechanism. Two separate converters are used for operate mode and hold mode. Power to the LVPS is provided by the HST. Electrical isolation is provided between the spacecraft power bus and the electronic subsystems by the use of isolation transformers at the input of both the operate and hold mode converters.

The position sensors monitor the mechanisms and provide position data to the RIUs.

Except for the DOB deployment mechanism, the stepper motors are not duplicated for redundancy. However, provisions have been made to ensure redundancy is achieved through dual windings in these motors.

The electronic subsystems will remain inactive until the COSTAR has been installed in the HST. There will be no exposed voltages associated with the COSTAR.

EMI configurations include common mode and differential mode filtering at the input of each converter. EMI testing will ensure compliance and compatibility with the STS environment.

4.2.5.4 COSTAR Thermal Control

The COSTAR thermal control subsystem design has two controlled areas, the electronics and the bench compartment. The internal surfaces of the electronics region are black to enhance the heat transfer between the electronics and the panels.

A low emissivity MLI efficiency blocks the energy exchange with the stable bench compartment. Conductive isolation is achieved by inserting a strip of isolating material between the outboard panels and the rest of the enclosure. MLI on the bulkheads effectively isolates the bulkheads from radiation exchange with the outboard panels.

The bench compartment thermal design provides a stable environment during the external fluctuations of the HST temperatures. An effective MLI system is used to surround the compartment. The MLI has a low emissivity finish on both sides to maximize the isolation system.

5.0 SCIENTIFIC INSTRUMENT CONTROL AND DATA HANDLING

5.1 INTRODUCTION

The SIC&DH subsystem provides the communications link between the SSM/OTA and the SIs. The components of the SIC&DH are all located in bay 10 of the SSM ES, except for 10 of the RMs and their BCUs. The RMs, which consist of an RIU and up to two Expander Units (EUs), provide the command and Engineering Data (ED) interface with the SIC&DH and the SIs. The SI RMs are physically attached to the SIs. The SI BCUs are in the OTA. The other components of the SIC&DH are mounted in the SSM on a removable shelf designed for on-orbit replacement by an EVA crewmember as a single unit (Figure 5.1-1).

All information to or from the SIs passes through the CU/SDF of the SIC&DH. Therefore, it is not possible to send commands to the SIs or to collect data from them unless the CU/SDF is on and operating. It is possible to command the SIs and to collect SD and a very limited amount of ED from them without involving the NSSC-I in the SIC&DH. The NSSC-I is involved in the monitoring and control of the SIs. Operation of the SIs in anything but the hold or safe mode without having the NSSC-I fully operational should be considered only under the most extreme circumstances.

The SIC&DH is a variation of the communication and data handling module of the Multimission Modular Spacecraft (MMS). The SIC&DH PCU, the STINT for computer, RIU, EU, BCU, and NSSC-I computer are all based on Standard Telemetry and Command Component (STACC) designs developed for MMS. In some cases, modifications to STACC designs have been made to accommodate the unique requirements of the HST. Although the CU/SDF is a substantially new design, parts of it are based on the STACC central unit design. The NSSC-I software executive for the SIC&DH is based on the NSSC-I executive developed for MMS and contains many of the same features.

The SIC&DH subsystem is required to perform the following functions:

- a. Provide system hardware redundancy under ground control
- b. Receive +28 V unregulated power from the SSM
- c. Control power conditioning and distribution within the SIC&DH
- d. Receive and distribute clock and synchronization signals
- e. Receive, distribute, and execute real-time commands
- f. Communicate with the SSM DMS through PITs
- g. Report SIC&DH status to the SSM by ED



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Figure 5.1-1. SIC&DH

5.1-2

- h. Collect SI ED for onboard use of transmission to the SSM
- i. Receive, format, and encode SD from the SIs
- j. Transmit SD to the SSM or process it onboard
- k. Provide a computer facility for SI control and data manipulation
- I. Issue a prestored payload safing command sequence

Physical Description

The SIC&DH is located on the door of bay 10 of the SSM ES. It measures 21.5 by 32.5 by 9.5 inches and weighs 135.85 pounds. The SIC&DH is equipped with a handle and a tether ring mounted by four ORU bolts and six captive fasteners to facilitate EVA servicing.

Electrical Power

The SIC&DH is powered by PDUs 1 and 2. The normal power requirement for the SIC&DH is 39 to 104 watts.

Thermal Control

The SIC&DH is operational in temperatures from +14° to +105.8° F and will survive temperatures from -58° to +140° F. The thermal control of the SIC&DH is passive. After lift-off, the air/nitrogen within the SIC&DH boxes and bay 10 of the SSM ES is vented and ceases to be a factor in the thermal environment. Heat transfer between the SIC&DH and SSM occurs primarily through radiation exchange with the interior surfaces of bay 10 and through conduction by the SIC&DH baseplate.

Command/Data Interfaces

The SSM DMS provides the command, timing, ED, and SD interfaces with the SIC&DH (see Section 2.5). The SSM DMS to SIC&DH interfaces are shown in Figure 5.1-2.

The SIC&DH module interface consists of Master Timing Pulse (MTP) enables (via Telemetry Format Control (TFC)), transfer of SSM PIT words and command(s), acquisition of SIC&DH PIT words, and TLM data (via a DIU).

The MTP is used for engineering TLM synchronization and time code update validation. Three serial digital channels of a DIU are used for SSM PIT word transfer/16-bit command, SIC&DH PIT word acquisitions, and TLM data acquisition. The serial digital interface between the DIU and the SIC&DH module is not cross-strapped. Hence, the correct hardware configuration of the DIUI module command and data register for the DIU element and for the SIC&DH module is required.

Drawing 5.1 illustrates the command and data flows in the SIC&DH.

5.1-3



Figure 5.1-2. SSM DMS-to-SIC&DH interfaces



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5.2 SIC&DH ELEMENTS

NASA Standard Spacecraft Computer - Model I

The NSSC-I consists of a Central Processor Module (CPM) and eight memory modules. Each memory module contains 8192 18-bit words. The NSSC-I software consists of an executive and SI-unique application processors. The executive processor provides the functions of input/output control, delayed command storage and execution, ED collection and limit checking, PIT processing, diagnostics and processor scheduling, and control. The SI-unique application processor is defined by the SI teams and performs required functions of SI control, monitoring, and data manipulation/analysis.

Standard Interface for Computer

The STINT is a STACC unit used without modification on the HST. The STINT provides the interface between the CU/SDF and the NSSC-I. The STINT is functionally transparent to the SIC&DH subsystem user.

Control Unit/Science Data Formatter

The CU/SDF is the central hub of the SIC&DH. It performs the following functions:

- a. Receives a 1.024-MHz square-wave clock signal from the SSM DMU and transmits it to the SIs via the MDB. This signal is used as the basis for all SIC&DH clock and synchronization signals.
- b. Receives an MTP from the SSM DMU and uses the MTP for TLM and processing synchronization.
- c. Receives ground commands in a 27-bit format from a three-signal (enable, clock, and data) interface with the SSM DMU, reformats the commands, and transmits them either to the SIs through the MDB or to the NSSC-I through the STINT.
- d. Receives SSM commands in a 16-bit format from a three-signal (enable, clock, and data) interface with the SSM DIU, reformats the commands, and transmits them to either the SIs through the MDB or the NSSC-I through the STINT.
- e. Reads sets of ED requests from the NSSC-I through the STINT, reformats them, and transmits them to the SIs by the MDB.
- f. Reads ED replies (one per request) from the SIs through the MDB and transmits them to the NSSC-I by the STINT.
- g. Reads completed minor frames of SI and SIC&DH ED from the NSSC-I through the STINT and transmits them to the SSM by a three-signal (enable, clock, and data) interface with the SSM DIU.

- h. Reads sets of 64 fixed ED requests from an internal ROM, transmits them to the SIs through the MDB, collects the replies from the SIs from the MDB, and provides the replies to the SSM through the ED interface in lieu of the normal minor frames of ED (under certain abnormal conditions).
- i. Receives SD from the SIs from dedicated six-signal interfaces and transfers data to the NSSC-I through the STINT.
- j. Receives SD from the SIs from the six-signal interfaces or SD logs and NSSC-I memory dumps from the NSSC-I through the STINT; formats the data into packets; optionally adds a PN sequence and/or Reed-Solomon error correction encoding; and transmits the resulting data as a continuous bit stream to the SSM by a two-signal (clock and data) interface with the SSM DMU. SD from the SIs can be raw or processed detector outputs, SI health and status information, or SI microprocessor memory dumps. Data logs from the NSSC-I can contain any information (including forward linked data and SI microprocessor memory dumps) collected in the NSSC-I under control of NSSC-I executive processor or an SI-unique application processor. NSSC-I memory dumps each contain the contents of one of sixteen 4096-word banks of NSSC-I memory.

Power Control Unit

The PCU is a modification of a STACC unit. The PCU performs power switching and distribution for all the components on the SIC&DH ORU tray. The PCU also performs power conditioning for some of those components. The PCU is internally redundant. It receives +28 V dc unregulated power from the SSM on two independent redundant power buses. It also receives four HLD commands from each of the redundant SSM CDIs. The PCU provides +28 V unregulated power to the CU/SDF and RMs; +5 V regulated power to the CPMs, STINTs, and oscillators; and +5 V, -5 V, and +12 V regulated power to the NSSC-I memory modules. The PCU contains redundant 1.8-MHz oscillators that generate clock signals provided to the CPMs via the STINTs.

Remote Module

The RM is a STACC unit that has been modified somewhat for use on HST. Each RM consists of an RIU and zero to two EUs. Each RIU provides 8 lines for input of serial digital commands, 64 lines for output of discrete (relay or logic) commands, and 64 lines for sampling serial digital, active analog, conditioned analog, and bilevel TLM points.

On HST, the RMs are used in redundant pairs. Only one unit of each pair is fully active at a time. There are six such pairs used on HST, one for each of the five SIs and one for the SIC&DH. All active RMs monitor all messages on the supervisory bus. Only the RM that is addressed responds to a given message, except for the SD.

The RMs provide the interface between the SIs and the SIC&DH except for the SD. They accept serial digital command messages from the MDB and provide the commands to the SIs as either 16-bit digital messages or 35-ms pulse/ground connections on discrete lines. In response to ED requests, they clock in 8-bit serial digital samples from the SIs or provide bilevel sampling and analog-to-digital conversion as required and transmit 9-bit ED replies on the MDB.

Bus Coupler Unit

The BCU couples two RMs to the MDB. The BCU does this so that a failure in an RM will not affect the MDB or other RMs attached to the MDB. The BCU is a STACC unit to which connectors have been added for use with cabling provided by the OTA.

Multiplexed Data Bus

The MDB consists of a supervisory bus and a reply bus, both of which operate at 1.024 MHz. The supervisory bus is operated in a repeating sequence of four 32-bit slots that contain synchronization messages, commands, and ED requests addressed to the various RMs.

6.0 HST SOLAR ARRAY

6.1 INTRODUCTION

The HST SA3 system generates a total of 4730 watts to support HST mission operations. The SA output is passed to the SSM EPS at a nominal operating voltage of 34 volts via two diode boxes for battery charging and distribution to the HST load. The system consists of two wings and control electronics to allow positioning of the SA solar cell panels for optimum solar illumination.

The SA3 physical configuration is depicted in Drawing 6.1 sheet 1. Each wing is equipped with four panels attached to a Panel Support Structure (PSS) and a Solar Array Drive Mechanism (SADM) that slews the array to face the Sun when it is deployed. The weight of one SA is 640 pounds.

Deployment of each wing is only accomplished via EVA, which will be described later in this section. The orientation of each wing is controlled by the STOCC using the SA ECA. A functional overview of the SA ECA is provided in Drawing 6.1 sheet 3. The ECA system consists of the DCE and redundant SADE units, housed within SSM ES bay 7. The ECA receives operating power from the SSM EPS and exchanges commands and telemetry with the SSM DMS. The ECA provides for SADM motor control and for conditioning of all wing instrumentation. Previously, the ECA also provided control for the PDM and SDM, which is no longer required with the SA3 design.

During the SM1 mission, the original SA1s were replaced with SA2s. SA2 was similar in design to SA1 with a few redesign exceptions. The problem with the SA1 and SA2 design was that they were susceptible to temperature-induced jitter. The new rigid SA3 design is intended to provide for the long-term power needs of the HST while reducing the jitter experienced with flexible arrays. A structural overview of the SA3 system is shown in Drawing 6.1 sheet 1.

The crew will also replace the existing DBAs with second generation DBA2s. In addition to providing diode isolation between the solar arrays and batteries, each DBA2 includes relays that are commanded by a respective Diode Box Controller (DBC). These additional relays could be used to recover from PCU relay failures. Control of these relays also requires the DBA2s to be connected to the HST P601 EPS test connector (at the EPS test panel). The DBA2s also include arc suppression networks to extend the life of the PCU relays and to provide wingtab connectors for easier installation.

6.1.1 SA3 System Hardware

6.1.1.1 Panel Support Structure

The PSS, shown in Drawing 6.1 sheet 1, provides the mounting interface for four solar panels and the associated harnessing. The PSS consists of two support frames joined by two panel hinge assemblies. The two panel hinge assemblies serve as the primary

structural interface between the two frames. It is at this interface that the PSS is folded for stowage on the Rigid Array Carrier (RAC) for launch.

Each PSS frame provides the main structural support for its two attached solar panels. The support frames are made of six 3- by 5-inch rectangular and one 3-inch diameter round Aluminum-Lithium Alloy X2096-T8A3 tubes. The tubes are connected via Aluminum 7075-T73 fittings. The fittings are fastened to the tubes via A286/Monel rivets. The fixed panel support frame incorporates the interface to the mast assembly and is the frame that is directly mounted to the RAC. The rotating frame is hinged to the fixed frame via the panel hinge assemblies. The two frames are near mirror images of one another with the exception of the mast interface and the tip latch fittings.

Each solar panel is mounted to its respective PSS support frame by three titanium flexures. These flexures, shown in Figure 6.1-1, are designed to isolate motion of the solar panels from the support structure to minimize solar array induced jitter of HST. They also provide thermal isolation and reduce the potential for "thermal snap" caused by builtup strains between the PSS and solar panel.



Figure 6.1-1. Solar panel flexure detail

6.1.1.2 SA Deployment Operations

Deployment of the SA system is performed by the EVA crew on SM-3B as part of the installation. Due to the rigid design of the SA-3 the SA deploy and retraction functions are no longer available for ground commanding and will require an EVA crewmember to perform manually deploy and retract the wing halves.

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(a) +V2

Figure 6.1-2. SA EVA worksite location



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(b) -V2 SA



6.2 SA SYSTEM HARDWARE

6.2.1 Solar Array Solar Cell Blanket

The individual solar panels are procured directly from the Iridium commercial production line and are built using a 5056 Al honeycomb core assemblies with front and backside graphite epoxy face sheets. The front face sheet is covered with Kapton sheeting and is then populated with solar cells. The rear face sheet of the panel is cobonded with a Tedlar coating. The honeycomb core edges are covered with Tedlar tape. Each panel weighs 33.5 pounds.

Each of the four solar panels on an SA wing contains 1000 6.5- by 5.5-cm Gallium Arsenide on Germanium solar cells, shown in Figure 6.2-1. These solar cells are wired within the solar panels into 20 circuits of 50 series-connected cells each. These 20 circuits are grouped electrically into 10 strings of 2 parallel-connected circuits, as shown in Figure 6.2-2. Each string is capable of producing a maximum short circuit current (lsc) of 2.4 amps. The power from each solar panel string is carried individually through redundant wiring all the way to the SA wing DBA2 where the strings are electrically combined into individual SPAs.



Figure 6.2-1. SA3 solar cell detail



Figure 6.2-2. SA3 circuit detail

The electrical interface between each solar panel and its SA wing is a single 37-pin sub-D connector, which is supported by the interface connector-mounting bracket located on the backside of the panel (total of four panels per wing). Solar panel power passing through this sub-D connector is carried from the panel to the coupler through a wiring harness bundle consisting of 20 twisted pairs (i.e., 40 wires per bundle). This allows the power from each panel's 10 strings to be carried redundantly through two separate twisted wire pairs within the bundle; either pair is rated to carry the full string current by itself.

The four SA wing wiring harnesses are combined within the coupler and are passed on to the SADM flex harness, as shown in Figure 6.2-3. The flex harness design for SA3 is intentionally unchanged from SA1/SA2 because this is a proven design, and also, it ensures SADE torque compatibility. The overall design is unchanged. However, the flex harness wiring of circuits passing through it had to be changed relative to SA1/SA2 because the individual SPAs in SA1/SA2 are electrically wired within the solar array blanket itself and carried through the interconnecting wiring as intact SPAs. In SA3, 40 individual strings are carried from the four solar panels (10 strings per solar panel) down the wing through the SADM flex harness, all the way to a DBA2, where the SPAs are then electrically formed.



Figure 6.2-3. SA3 wiring/harness overview

The SADM flex harness terminates in connectors that mate to the DBA2. The current SA1/SA2 DBA design utilizes ganged rack-and-panel Deutsch connectors on a common drive stud bracket, with the Y- and Z-ganged connectors dedicated to passing SA power to the DBA X-connector. The X-connector passes power from the DBA to the spacecraft. The W-connector is dedicated to SA and DBC command and telemetry and heater power (see Figure 6.2-4).



Figure 6.2-4. SADM to DBA2 Detail

Panel Hinge Assemblies

Two panel hinge assemblies on each SA wing join the two support frames, shown in Drawing 6.2 sheet 1. The inboard hinge (i.e., the one closest to the mast) contains the components for latching the wing in the deployed position. The two EVA latch screws incorporate a soft dock device to assist the crewmembers while deploying the PSS. No EVA access is required for the outboard hinge. The hinges contain belleville washers that allow for longitudinal (in-axis) expansion and contraction. They also provide friction in the hinge joint to allow a smooth, controlled rate of panel rotation. A combination of a snubber on the inboard hinge and a stop on the outboard hinge are adjusted at assembly to provide a preload to ensure stability and rigidity across the hinge interface.

6.2.2 Solar Array Drive Mechanism

The SADM orients the deployed SA wing toward the Sun and is capable of rotation in either direction. Two mechanisms are required for the SA system, one for each wing, and each is capable of independent or simultaneous operation. The SADM is shown in *Figure 6.2-5* and detailed in *Drawing 6.2 sheets 1, 2, and 3*.

Physical Description

The SADM consists of a central drive module containing the ball bearings, motorresolver assemblies and solenoid brake, and an OLD for protection of the bearings during the launch environment. A Flexible Wire Harness (FWH) unit provides power and signal interface across the rotating joint between the SA and SSM. A manual brake is incorporated to provide an EVA crewman the means to position the SA wing and hold it at that orientation in the event of an SADM failure.



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6.2-5

The SA PDM bolts directly to the SADM output shaft flange. The SADM drive module is contained within the SAD housing by means of a bolted flange. An adapter provides the structural interface between the SA wing and the SSM FS. The SAD Adapter (SADA) consists of the conical base attached to the SADM housing and a corresponding conical receptacle bolted to the SSM FS. The two sections are held together with a manacle (marmon) clamp ring, which can be opened for on-orbit changeout or SA jettison. Figure 6.2-6 shows the SADA during SA assembly.

The drive module consists of two angular contact ball bearings, liquid-lubricated and maintained under constant preload during on-orbit conditions by means of a diaphragm. The drive torque is generated by one direct drive torque motor, mounted together with a single-speed and multispeed resolver on common structural rings. A second motor/resolver assembly provides redundancy, and both motors can be used simultaneously in a contingency. The angular position of the shaft is measured by the single speed windings of the resolver. The multispeed resolver windings are used to produce the signal for commutation of the torque motor by the SADE.

The solenoid brake consists of 12 brake pads equally spaced around the circumference of the drive unit. Each pad is attached to the housing and acts on a common brake disk attached to the SADM shaft. The solenoid brake is provided to hold the SA wing orientation fixed in orbit, protecting against the FWH torque and on-orbit disturbances during normal and maintenance operations. The brake system is nominally engaged in the unpowered condition. When motor power is applied by the SADE to reorient the SA wing, power is also applied to the solenoid brake using electronic ramping circuitry to avoid mechanical disturbances. When wing slew is complete, brake power is ramped down to re-engage braking torque to the SADM brake disk.

In launch configuration, the SADM bearings are protected by an OLD positioned at the outboard end of the SADM (Drawing 6.2 sheet 2).

Four latches engage radially into a V-groove on the circumference of the brake disk. The latches are kept in position by U-shaped levers that are hinged to a flange fixed to the SADM housing. In the locked position, the levers are overcentered and tied by a steel rope. Release is initiated on command from the STOCC by the SADE. It is performed by heating a memory alloy element that deflects and, thereby, moves away one of the levers. The resulting slack in the steel rope, together with the supporting retention springs, releases the other levers and latches. The steel rope is retained by the springs against a sheet metal collar. A redundant memory alloy element is located at a second lever. The locked and released status of the OLD latches can be sensed by redundant microswitches wired in parallel.



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- Notes: 1. Marmon (EVA) clamp is in OPEN position for installation.
 - 2. Photo is not current configuration (refer to Figure 6.2-12).

Figure 6.2-6. SADA interface (view is just before insertion of SADM base cone into SADA receptacle)

6.2-7

The electrical lines from the SA pass through the hollow shaft of the SADM (power and signal lines separated by a screen) and into the FWH unit. The cables within this unit are not laced together; this exploits the higher flexibility of the single cables, which are allowed to twist and bend as the SADM rotates. The range of angular rotation of the SADM is +260° to -80°, as shown in Drawing 6.2 sheet 2. The cables outboard of the SADM shaft are bundled and routed along the PDM. Those inboard of the mechanism are bundled and connected to the diode box via EVA connector brackets, which are capable of being released in the event of SA jettison or changeout.

EVA Interfaces

If the memory alloy elements fail to release the OLD on orbit, a manual override capability enables release by an EVA crewmember. Rotation of a hexagon fitting through 3-1/2 full turns CCW releases the OLD, allowing the SADM to slew. A subsequent a 5-2/3 rotation CW closes the OLD release microswitches. Nominal torque values for normal manual OLD release should be approximately 2 ft-lb. The SA torque limiter set to 5 ft-lb should be used for this operation.

Two manual brakes are incorporated into the SADM to fix rotation of the SA during on-orbit maintenance, if required. This device consists of a lever system that acts on the shaft flange collar in a manner similar to that of the OLD. Either brake is actuated by rotating its associated 7/16-inch hex head EVA interface 1/2 turn to the ON or OFF position (Figure 6.2-7) under a nominal torque of approximately 1.5 ft-lb.

If an electrical failure of the SADM motors occurs, the mechanism is designed so that the failed motors offer no resistance to manual orientation by an EVA crewman. The only torque resistance to overcome is that of the SADM bearing friction, the solenoid brake, and the FWH. The maximum torque applied by an EVA crewmember to reposition the array should be less than 22 ft-lb (30 N+m). The nominal torque to rotate the SADM is less than 1 ft-lb. The SA torque limiter set to 5 ft-lb should be used for this operation.

If on-orbit wing changeout or jettison becomes necessary, the SADA clamp can be opened and closed by cutting the lockwire on the spindle lock and rotating the standard 7/16-inch hex head EVA fitting on the spindle lock through 22 turns CW or CCW, depending on which of the two spindle lock interfaces is driven. The SADA marmon clamp is shown in Drawing 6.2 sheet 1 and Figure 6.2-8 through Figure 6.2-10. Manual operation of the SADA clamp EVA interface drives the clamp open and closed enough to allow removal of the SADM from the SADA adapter on the SSM FS. The nominal torque for opening is less than 2.0 ft-lb (2.75 N+m). For closing after SA installation, the SA torque limiter set at 2.0 ft-lb (2.75 N+m) must be used to ensure that correct pretension in the clamp is achieved.



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6.2-9



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Figure 6.2-8. SA SADA clamp overview

6.2-10



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Figure 6.2-9. SADA clamp spindle connection hex I/F detail view

6.2-11



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Figure 6.2-10. SA SADA clamp contingency I/Fs


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SHEET 3 OF 6

6.2.3 Diode Box Assembly

6.2.3.1 Diode Box Assembly 2

Each wing is supported by a Diode Box Assembly 2 (DBA2) that provides isolation diodes for power from each of the SPAs and the interconnect interface between the electronics and the deployment and positioning mechanisms. The design is based on the original HST DBAs with design enhancements. The enhancements include relay arc suppression networks, and the ability to recover solar array power lost because of failed relays in the PCU and additional relays for charge control capability. The DBA2 is an ORU and also provides the capability to disconnect the cables from SA3 if jettison of the array is required. Connector brackets are the means of disconnect. The brackets allow the removal of two or four connectors at a time using an EVA power tool on the connector bracket drive stud provided. Each connector bracket has a tether loop to restrain the wire harness after disconnect.

Physical Description

Each DBA2 measures approximately 5 by 6 by 34 inches and weighs 24 *pounds*. As shown in *Drawing* 6.2 *sheet* 1 and *Figure* 6.2-11, they are mounted externally to the HST on the forward bulkhead of the SSM ES near the SADA/SSM interface, using four ORU bolts and associated J-hooks. Each DBA contains 80 stud-mounted power diodes, two per string. The two power isolation diodes of each string are connected in parallel and mounted together on an aluminum heatsink.

The fixing brackets for the heatsinks are attached to the side flanges of the DBA2 ORU tray. The tray also supports 12 electrical connectors that provide the SSM-to-SA3 interface and the interface between the ECA and SA mechanisms. These connectors are grouped in two sets of four and two sets of two. The W-connectors provide telemetry from the SA3, and the X-connectors output SPA power to the SSM. Both X and W sets are the same as for the original diode box and are mated using a drive-stud bracket. The CCW rotation of the connector bracket drive stud through five full turns automatically disconnects the bracket connectors, allowing the bracket and harness to be retained to the SA3 wing using a provided keyhole slot and bolt. The Y- and Z-connectors provide raw input power from the SA3 panels to the DBA2 and are the more traditional, EVA-friendly, wingtab connectors. EVA tether rings are affixed to each end of the DBA2 for temporary restraint of the unit while detached from the SSM during installation or SA3 on-orbit maintenance. A fiberglass cover is attached to the DBA2 on the front side to minimize inadvertent contact with the diode plates.

Electrical Description

The DBA2 isolation diodes are JANTXV1N5816 high-efficiency diodes. As mentioned above, the two diodes of a single SPA circuit are mounted on the same heatsink and in electrical contact with it. Isolation from the other heatsinks is provided by an insulating plate onto which the heatsinks are bolted.

Within the DBA2, the SPAs are electrically formed and are then fed to the PCU. The 40 solar panel strings fed to each DBA2 from their respective solar array wings will be wired into SPAs as follows:

- a. Three five-string SPAs: D, E, and AA
- b. Six *four*-string SPAs: A, B, C, BB, CC, and DD
- c. One one-string SPA: EE

The SPA assignments are configured to protect the HST wire deratings and the derating of the PCU and DBA2 relays.

Each DBA2 contains 28 relays, as shown in Drawing 6.2 sheet 4 : 10 latching disconnect relays, 9 nonlatching charge control relays, and 9 latching bypass relays. A set of three relays (one disconnect, one charge control, and one bypass) is designated for each of the nine SPAs that provide battery charging. The \pm EE SPAs are not connected to a battery and, therefore, have only disconnect relays in the DBA2. The disconnect relays will be used to electrically disconnect SA3 from HST and may be used to deadface the SA connectors. The charge control relays are provided for each SA3 SPA, except the \pm EE SPAs, and are used to manage battery charging. The bypass relays will be used to recover charge control relays that have failed in the disconnect position. The operation of these relays will be described in the DBC section.

An Arc Suppression Network (ASN) is included in the DBA2 for each SPA section to eliminate arcing across the contacts of the PCU relays. The way the ASN operates is as follows: In sunlight, with charge control relays closed, a capacitor is charged to the SPA voltage level. The charge control relay starts to open, and as the contact resistance increases (arcing across the contacts), the SPA voltage increases, thus causing the capacitor to absorb the SPA current, which limits the rate of voltage rise, causing the arc to extinguish. Then the capacitor charges up to the open circuit SPA voltage. Once orbit night arrives, the capacitor discharges through the 100 k Ω resistor down to the bus voltage.

Thermal Control

Heat generated by the DBA2 power diodes is radiated out of the unit through the silverized Teflon-coated heatsink plates. The inside face of the shield surrounding the plates is painted black. All other external surfaces are taped with aluminized Kapton except where they are covered by the mounting bracket.

Active thermal control of the DBA2 is provided during HST mission phases when the SA3 is not generating power. Survival heaters are installed on each of the 10 power diode heatsink plates in a DBA2 and controlled by two parallel sets of two series-connected thermostats. Heater and thermostat placement is shown in *Drawing 6.2 sheet 1*. Each heater element is approximately 256 ohms. A DBA2 survival heater circuit consists of the thermostats controlling 10 heater strips connected in

parallel. The instantaneous maximum power required by the DBA survival heaters is 80 watts (40 watts per DBA2). The survival heaters are powered off during nominal HST operations because heat generated by the power diodes warms the DBA2 while the SA3 is generating electricity.

6.2.3.2 Diode Box Controller

The DBC is an addition to the DBA2 that provides a means to recover from certain PCU relay failures. The DBC provides battery charge control external to the PCU, using relays in the DBA2.

The DBC enables HST to regain the previously lost SPAs by reconfiguring the PCU and performing battery charge control in the DBA2/DBC (Software Charge Mode (SCM)). It provides an Autonomous Charge Mode (ACM) to provide battery charge control while in HWSP. The DBC provides an SCM to reduce usage of PCU relays during normal operation and provides disconnection of SA power for PCU changeout.

Physical Description

The DBC is integrated into the DBA2 ORU assembly, as shown in *Drawing 6.2 sheet 1* and *Figure 6.2-11*. The DBC power is supplied via the cross-strap harness from the J601 interface by tapping and fusing command and telemetry buses CT 1/2 and CT 3/5. The EVA crew installs a cross-strap harness between the two DBA2s and extends it to the J601 connector on the HST EPS test panel. The DBC cross-strap harness is routed along a handrail attached to the top of the SSM ES. The harness interconnects the two DBCs to provide redundancy on critical functions. The harness is a T-connection to J601 to provide for power, ACM battery voltage, and battery temperature signals when the VIK jumper plugs are installed.



Figure 6.2-11. DBC/DBA2 installed on HST

Functional Description

The DBC has three modes of operation: SCM, ACM, and Ground Command Mode (GCM). In SCM, the DBA2 relays are commanded by the 486 and EPS FSW based on battery voltage and temperature. ACM commands the DBA2 relays based on battery voltage and temperature also, but the commands originate at the DBC. GCM allows the DBA2 relays to be commanded in real time by the STOCC.

SCM is enabled by a Keep-Alive Signal (S K/A) sent from the DCE and by enabling the EPS charge control routine in FSW. The DCE S K/As are cross-strapped to both DBCs for redundancy using the cross-strap harness. The 486 monitors battery voltages and temperatures and compares them against a virtual V/T curve for chargeoff and charge-on determination. When a given battery has reached the chargeoff or charge-on V/T threshold, the 486 sends a relay specific pulse-width modulated signal to both DBCs to offline the respective SPA. The pulse-width command is decoded by the intended DBC and the specific DBC relay driver is commanded to open or close the respective nonlatching charge control relay in the DBA2. Loss of the S K/A from the DCE results in the disabling of SCM, and the CC relays in the DBA2 are autonomously closed.

ACM is enabled by the loss of the Autonomous Mode Disable (AMD) signals from the DCE, which are cross-strapped to both DBCs. Nine individual DBC hardware charge controllers, one for each SPA (except the EE), compare the respective temperature compensated battery voltage against a fixed chargeoff and charge-on threshold. When a given battery reaches the chargeoff or charge-on threshold, the hardware charge controller commands the respective relay driver to open or close the respective nonlatching charge control relay in the DBA2. Resumption of the AMD signals from the DCE results in the disabling of ACM.

ACM mode is disabled to launch and would be enabled in the event the PCU was not changed out. Enabling the ACM requires the EVA crew to install the V/T signal jumper plug on the cross-strap harness. This provides the battery voltage signals to DBC. In addition, the crew must install the six VIK jumper plugs to provide battery temperature signals to the DBC.

GCM is a subset of SCM and permits the following: the opening of each DBA2 charge control relay; the collective opening of the D, E, and AA DBA2 charge control relays in each or both DBA2s; the collective closing of all DBA2 charge control relays in each or both DBA2s; the capability to collectively open or close all 10 SPA disconnect relays in each or both DBA2s; the capability to collectively open or close all 9 bypass relays in each or both DBA2s. GCM requires the S K/A signals to be enabled, the 486 to be in control, and the critical command enable for disconnect and bypass relay operations.

Because of a limitation of available telemetry channels, the collective status of the 10 disconnect relays and the collective status of the 9 bypass relays in the DBA2 are provided in telemetry by using one spare bilevel telemetry channel in each SADE. SADE-1 conditions the collective status of the 9 bypass relays, and SADE-2 conditions the collective status of the 10 disconnect relays. For each relay type, the telemetry indicates if all are open or one or more are closed. The status of the nine DBA2 charge control relays is provided in telemetry by using nine spare analog telemetry channels in the DCE.

EVA Operations

The diode box connectors are designed for EVA SA changeout operations.

The Velcro tiebacks (Figure 6.2-13), connecting the Ty-raps on the W-harness and the Y/Z harness with the Ty-rap on the SADA mounting flange, are disconnected. The six keyhole bolts on the diode box assembly are then disengaged, two for each of the three connector drive mechanisms (Figure 6.2-12). Care must be taken not to damage the protective DBA grid cover (Figure 6.2-13). The SA power connectors, labeled "Z," "Y," and "SA POWER OFF 5t," are then disconnected by rotating the 7/16-hex head connector bracket drive stud five revolutions (Figure 6.2-14). The P5A and P7A connectors from "Z" are then capped and stowed on the SA I temporary stowage bracket (Figure 6.2-15). The P6A and P8A connectors from "Y" are tethered to "I." Care must be taken not to touch the interior of these sockets because open circuit voltage might be present. The SA data connectors labeled "W" and "SA DATA OFF 5t" (Figure 6.2-16) are disconnected in the same manner and stowed on the SA I temporary stowage bracket.

The connector drive mechanisms on the new SAs are connected in the same manner, reverse order, and torqued to approximately 10 ft-lb.





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DBA-2 PHYSICAL DETAIL



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Figure 6.2-12. SA I (+V2) diode box and connector bracket



Figure 6.2-13. SA I (+V2) diode box



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Figure 6.2-14. SA I (+V2) "Z" and "Y" connector bracket detail view (+V2/+V3 axes)



Figure 6.2-15. SA I (+V2) "Z" and "W" temporary mounting brackets



Figure 6.2-16. SA I (+V2) "W" connector bracket detail view (+V2/-V3 axes)

6.2.4 Electronics Control Assembly

The SA ECA is responsible for control and monitoring of all functions of the SA system except for switching of the operational and survival heaters. The ECA consists of the DCE and the SADE. The DCE is configured as one box with redundant units provided internally. The SADE is made up of two redundant boxes. The ECA is detailed in Drawing 6.2 sheets 4 and 5.

6.2.4.1 Deployment Control Electronics

Physical Description

The DCE, comprising the main unit housing and filter box, is mounted on the door of SSM ES bay 7, as shown in *Figure 6.2-17*. The filter box resides underneath the main unit, utilizing the same hole pattern. The DCE measures 9.45 by 8.08 by 11.9 inches and weighs 22.4 pounds. The upper compartment of the DCE unit contains 18 printed circuit boards wired to 14 Cannon Royal D subminiature 50-way connectors (two captive screws per connector). Of these 14 connectors, 10 are positioned at one end section to provide the interface with the SSM harness. The remaining four connectors, stationed on the opposing end of the unit, are not used in flight. The lower compartment of the unit houses the PCB connector receptacles and two PCBs holding the main bus filters, 10 power transistors, and 12 crystal relays. The DCE is classified as a block II ORU.

Electrical Power

Redundant DCE electronics are provided within the unit to enhance system reliability. Each DCE half (each hereafter referred to as side A and side B) receives fused, switched HST main bus power from separate SSM sources. Side A is powered via relay K19 of PDU-1 and side B from relay K19 of power distribution unit PDU-2. These relays are controlled by the STOCC using HLD commands from DIU-5 (K19 of PDU-1) and DIU-2 (K19 of PDU-2). Each PDU feed has short circuit protection from a 10-A fuse housed inside the corresponding PDU.

Unregulated input power is applied to the DCE dc-dc converters and the SA mechanism motor drivers in parallel, as shown in *Drawing 6.2 sheet 5*.



Figure 6.2-17. DCE mounting

6.2-33

The dc-dc converters regulate main bus power before application to the DCE control circuitry and are commanded on by CDI HLD command from the STOCC. The SDM and PDM motor drivers apply main bus power directly to the motors under control of the DCE logic circuitry. DCE power utilization for operation of the PDM and SDM motors is summarized in Table 6.2-1 for nominal and worst-case conditions.

Operati DCE st DCE or DCE tw DCE or DCE tw	ion andby ne stepper motor /o stepper motors ne dc-motor /o dc-motors	Consumption at 32 V 3.42 W 16.32 W 29.18 W 18.22 W 31.92 W	Dissipation at 32 V 3.42 W 9.42 W 16.22 W 7.57 W 10.12 W									
Note: A	Note: Assuming dc motor current = 400 mA											
2. Wor	st Case											
Operati	ion	Consumption at 32 V	Dissipation at 32 V									
DCE st	andby	3.65 W	3.65 W									
DCE or	ne stepper motor	17.2 W	11.8 W									
DCE tw	vo stepper motors	30.6 W	20.0 W									
DCE or	ne dc-motor	27.5 W	8.2 W									
DCE tw	vo dc-motors	50.3 W	11.2 W									
Note:	Jote: Assuming dc motor current = 600 mA Assuming MB = 32 V Assuming stepper motor resistance = 31Ω Assuming stepper motor current = 360 mA + 10 percent											

Table 6.2-1. DCE power consumption/dissipation

During mission phases when inadvertent operation of the SDM or PDM may cause a safety concern (e.g., HST in the orbiter payload bay on the deployment mission), the K19 relay of PDU-1 and -2 must remain open to provide the sufficient number of inhibits (three) to powering the SDM motors. Although two of the three inhibits are required to be monitorable, in this configuration only the K19 status is available to verify a safe condition.

Command Interfaces

1. Nominal Case

The DCE accepts serial digital command messages and HLD (power pulse) commands directly from the DMU CDIs and outputs bilevel and analog TLM signals from the SA instrumentation to DIU-4, as shown in *Drawing 6.2 sheet 5*. A CDI special serial

digital interface from both CDI-A and CDI-B is provided to the digital command interface of each signal conditioner for reception of motor control command words from the STOCC. The motor control information provided in these command messages allows the DCE to select up to two mechanisms simultaneously for operation (primary winding of one motor and redundant winding of the other). Upon reception, the DCE control logic validates the contents of the message and automatically executes the command, once verified.

The DCE dc-dc converter configuration is controlled through DCE relays K1 and K2 using three HLD commands from either CDI. These commands allow only one converter enable relay to be closed at a time, preventing sides A and B from being powered on simultaneously. CDI HLD commands also allow the STOCC to switch the DCE instrumentation interface between the side A and side B signal conditioners.

Data Interfaces

The signal conditioners output conditioned discrete and analog TLM signals from the SDM and PDM instrumentation to DIU-4. SA system instrumentation includes the following:

- a. 32 microswitches (PDM and SDM deploy/stow status)
- *b.* 10 thermistors (mechanism temps)
- c. 6 platinum (Pt) resistors (blanket temps)

The functional location of these items is shown in Drawing 6.2 sheet 5. One potentiometer is provided in each SA blanket spreader bar to monitor blanket tension. Of the 16 microswitches provided per wing, 8 reside in the SDM actuators to sense full deployment or stowage of their associated bistem elements, 5 in the drum brake, 2 for PDM deployment, and 1 for OLD release. No microswitches are provided to sense full retraction of the PDM. The thermal status of the PDM and SDM actuators is provided by five thermistors per wing. These five transducers provide monitoring of PDM drive and hinge temperatures, the inboard SDM actuator, and the outboard SDM motor. Finally, three Pt resistors are integrated into the blanket pair for each wing to provide TLM of blanket temperature.

In addition to wing instrumentation, each conditioner provides a DCE FAIL FLAG and a DCE MOTOR OPERATE discrete derived from the DCE internal status. The DCE FAIL FLAG asserts under any of the following conditions:

- a. Overcurrent of any DCE motor
- b. Undercurrent of any PDM stepper motor
- c. HST main bus voltage outside the range of 24 to 32 V dc
- d. Converter current consumption too high

6.2-35

- e. Internal voltage supply (+10 V dc) erroneous
- f. Invalid content of command register

The digital command interface processing logic outputs the DCE MOTOR OPERATE discrete to the DIU while motor operation is in progress and monitors microswitch status of the selected mechanism. Motor operation is terminated automatically by the DCE upon indication from those microswitches that the mechanism has fully deployed (SDM or PDM) or stowed (SDM only).

Note: The DCE control logic terminates motor drive upon sensing any microswitch transition, regardless of the source.

Motor Drivers

The motor drivers and their interfaces to the other DCE components are shown in *Drawing 6.2 sheet 5*. Each PDM comprises redundant stepper motors, and each SDM contains inline redundant dc motors. Eight motor driver circuits (four stepper motor drivers and four dc motor drivers) apply and remove drive motor power directly under control of the signal conditioner control logic. One driver is provided for each of the eight motors in the SA system. Either signal conditioner can command the primary or redundant motor drivers of any mechanism. For both the PDM and SDM, a single motor can operate the mechanism. The motor drivers are grouped and selectively connected to motors, control signals, and HST main buses to allow full functional performance even if a single failure affects the control signals, power supply, or an open circuit of any drive motor.

6.2.4.2 Solar Array Drive Electronics

The SADE provides the SSM with a command and TLM interface to the SADM for slewing the array to the best possible Sun-pointing orientation. The SADE provides closed-loop motor and solenoid brake control using motor command shaping based on the output of SADM motor resolver data. The SADE also provides the circuitry required to command release of the SADM OLD and condition SADM thermistor signals for output to the SSM DMS. Two SADEs are provided for redundancy. Drawing 6.2 sheet 6 provides a functional overview of the SADE.

SADE-1 was replaced during the SM-1 mission. The new unit is identical to the original, which was replaced to correct a transistor failure caused by junction overtemperature.

Physical Description

Each SADE measures 13.9 by 5.9 by 8.7 inches and weighs approximately 15.9 pounds. Both SADE units are mounted to the door of SSM ES bay 7, as shown in *Figure 6.2-18* and both are classified as ORUs. The mounting hardware consists of six noncaptive 3/16-inch torque-set hex bolts with washers and 16 captive screws (two on each of eight D-type connectors).

Electrical Power

The SADEs receive operating power from the SSM EPS via STOCC command to their corresponding PDU relays, as shown in Drawing 6.2 sheet 6. SADEs 1 and 2 receive HST main bus power via the K19 relay of PDU-3 and PDU-4, respectively. The K19 relay of PDU-3 (SADE-1 power on/off) is controlled via command from DIU-4 or by the PSEA during safe-mode operations. The corresponding relay of PDU-4 (SADE-2 power on/off) is commanded by the STOCC from DIU-2 and may also be controlled by the PSEA. Each unit consumes 13.2 watts at average power and 85 watts at peak power.

Separate PDU circuits are used for release of the SADM OLD, as shown in *Drawing 6.2 sheet 5*. OLD release is achieved by the electrical heating of a memory alloy element, deforming the component to allow the OLD latches to disengage. The OLD is mechanically described in Section 6.2.4. Both SADEs provide the interface circuitry to apply electrical power to redundant OLD elements. Power is supplied to these circuits from the K2 relay of PDU-1 (SADE-1) and PDU-2 (SADE-2) commanded on by DMU CDI special discrete command. These relays are commanded open by STOCC via DIU HLD (DIU-5 for SADE-1 OLD POWER OFF and DIU-2 for SADE-2 OLD POWER OFF) when verification of OLD release has been obtained. Operation of the OLD consumes approximately 40 W per wing.

Command Interface

The SADE accepts SA slew commands from the 486 flight computer, the PSEA SMCs, or the STOCC and outputs motor position, OLD status, and SADM thermistor data to the DIUs and PSEA, as shown in Drawing 6.2 sheet 5. The SADE controls slewing of the arrays based on the incremental control inputs it receives from these sources. Specifically, each SADE accepts inputs from DIU-5 (elements A and B), CDI-A and -B, and SMC-A and -B. The output from all the A elements (e.g., DIU-5A, CDI-A, and SMC-A) are wire-OR'd together within the vehicle harnessing, as are the B elements. The discrete angle commands received by the SADE shall slew the wings in increments of $\pm 5.625^{\circ}$ and $\pm 0.088^{\circ}$. The 486 software issues the incremental array pointing commands to the SADE using DIU HLDs corresponding to these angle values. In the event of entry into safe mode, the PSEA SMCs shall issue slew commands to automatically reposition the arrays for optimum Sun illumination in hardware Sun-point mode. Finally, the ground may bypass the 486 and issue slew commands directly using DMU CDI special discretes. Each SADE also receives CDI special discretes from both CDI-A and -B to enable or disable release of the SADM OLD. This function can, therefore, be performed only by direct STOCC command.



Figure 6.2-18. SADE units installed on SSM bay 7 door

6.2-38

Data Interface

Instrumentation and motor resolver data gathered by the SADEs is passed on to the DMS via DIU-4 analog and serial digital input channels, as shown in Drawing 6.2 sheet 5. Redundant temperature transducers are installed within the SADM housing of each wing on the motor casing and output shaft. These thermistors are conditioned by dedicated circuitry in each SADE unit, and their output signals are provided to DIU-4 analog channels.

The SADE also provides two 16-bit serial digital words to a DIU-4 serial digital input channel (one word for each wing). As depicted in the drawing provided, the 11 least significant bits of each word are used for wing position angle, and the 3 most significant bits are used as status indicators (SADE FAILURE FLAG, MOTOR OPERATING FLAG, OLD STATUS FLAG). The remaining bit is a spare. When polled, the DIU clocks the digital data words out of the SADE serial buffer and transfers them to the DMU for access by the 486 for processing and/or the DMU TFC module for downlink.

The SADE provides bilevel wing position data to the PSEA for contingency control of the SA wing positions by the SMCs. These data consist of the seven most significant bits of the wing position data and status flags discussed above. The SMC firmware utilizes the wing position data in generating slew commands to reorient the array for optimum solar illumination during safe-mode operations.

Functional Description

The purpose of the SADE system is to rotate the SADM smoothly over a commanded angle within a prescribed time. The control system loop is designed so that the SA dynamics are not overexcited during a slew. To achieve this, motor ramping circuitry is provided to shape the motor current profile at the start and end of each slew, avoiding interactive torque disturbances caused by sudden application or removal of motor power. This, combined with modulated resolver outputs, minimizes the time at the end of the slew for the HST/SA to settle before the HST returns to science observation.

Either motor resolver can slew the SA wing to a predetermined orientation over the range of +260° to -80°, as shown in *Drawing 6.2 sheet 5*. The STOCC shall utilize the following considerations when planning an SA reorientation:

a. The HST SA pointing accuracy of the SADM is better than 3.5°

b. Time for rotation is defined by and will not exceed

 $T = 2 \times \sqrt{\{900 + [(0.125E04) \times \Theta]\}}$ seconds

where Θ = slew angle > 5.625° Actual slew time is T = 65.3042 $\sqrt{\Theta}$

- *c.* After completion of a slew, the SA nominally requires approximately 5 minutes to settle to below ICD levels for SA-induced torques into the HST
- d. All slews shall be greater than 5.625°
- e. Command initiation of a slew should begin approximately 16 seconds before actual slewing
- *f.* Accumulative slews can be programmed only using target angles (The use of sequential angles could result in a summation of 3.5° of error.)
- *g.* For complete stability of the SA drive system, all slews shall be performed with the SA blankets fully unfurled

The SADE is functionally depicted in *Drawing 6.2 sheet 5*. The interface electronics receive rotation angle commands, convert them to an SA wing position, and decode the rotational direction. From this input, a shaping function module then generates a rotation angle versus time sinusoidal shaping function in a Time (T) that depends on the commanded angle. Control system compensation provides the SADE a measure of the FWH resistive torque value between slews of the SA wings. The output of the shaping function module feeds the motor drive stage, generating the necessary motor currents for SA drive operation. The decoded motor-resolver output provides feedback to the motor driver stage to allow closed-loop operation of the SADM. A motor power ramping module gradually increases or reduces power to the motors to avoid sudden disturbances, while a brake control circuit generates a current versus time profile to disengage the brake during a slewing maneuver. At the end of the slew maneuver, the motor and brake power ramping circuits will operate to gradually decrease motor current and re-engage the solenoid brake.

EVA Operations

The SADE is designed for EVA changeout, if necessary.

The bay 7 door is opened by releasing the six J-bolts (five turns CCW) and the handle knob. The door is then tethered open no more than 100°. Power is removed from the SADE and the eight SADE power and data connectors (P1 through P8) are demated. The SADE is then removed by releasing the six noncaptive bolts and washers (six turns).

The replacement SADE is installed in reverse order of removal as defined above.





6.2.5 Safety Considerations

DCE Safety

Hazardous periods for inadvertent operation of the DCE are during RMS grapple, HST berthing and deployment, and SA EVA changeout operations.

During HST RMS grapple, and HST berthing/deployment, the PDM inadvertent operations are inhibited by at least two inhibits. Single fault tolerance is provided by configuring the DCE primary and redundant power to OFF (i.e., inhibit no. 1), which is monitored in real-time telemetry. The second inhibit resides within the DCE PDM motor relay (relay S0).

During nominal operations the DCE will be configured ON, removing one of two inhibits leaving one relay (inhibit no. 2) to prevent PDM inadvertent operation.

Crew detection (visual and telemetry) and procedural control will stop SDM/PDM inadvertent operation. Recovery of the SA to a safe state is achieved by crew commanding the DCE to OFF in the case of inadvertent PDM retraction. In the case of inadvertent PDM retraction, the following steps are required to restore an SA to a safe state:

- 1. IVA crew commanding DCE to OFF (to terminate PDM motion).
- 2. IVA crew goes to free drift (to preclude motion of partially retracted PDM caused by RCS loads).
- 3. Option A: STOCC ground commanding redeploys SA PDM using redundant electronics (to restore a stable PDM configuration).

Option B: EVA crew manually redeploys SA PDM (to restore a stable PDM configuration).

4. IVA crew restores RCS control.

The inherently slow speed of the SDM (4.0 ft/min) and PDM (10.0 deg/min) provides sufficient crew reaction time to preclude any hazard resulting from an undesired SA wind or boom operation. The PDM provides 1.25 ft-lb (1.7 N+m) torque and SDM drive torque is 500 in-oz. These low contact forces and rates do not present a catastrophic hazard to the orbiter or crew.

During non-EVA periods when SAs are operated, adequate HST-to-orbiter clearance exists.

SADM Safety

Hazardous periods for inadvertent slewing of the SAs are during RMS grapple, HST berthing and deployment, and anytime the HST is pivoted down to the reboost position when the crew is not available to monitor SA position. During these periods, inadvertent SADM operations are prevented by at least two inhibits. Single fault tolerance is provided by configuring the SADE primary and redundant power to OFF (i.e., inhibit no.1), which is monitored in real-time telemetry. Inhibit no. 2 resides in the SADE gating circuit, which inhibits slew commands if no enable pulse is issued or if slew commands are not in the correct sequence. The input register will reset in 393 seconds if a partial or incorrect command is received.

During EVA or periods when HST is pivoted down to the reboost position and the crew is available to monitor SA position, only inhibit no. 2 will prevent inadvertent SADM operation.

Crew detection (visual and telemetry) and procedural control will stop SADM inadvertent operation. Recovery of the SA to a safe state is achieved by crew commanding SADE to OFF.

The inherently slow speed of the SADM (25 deg/min after 8 minutes) provides sufficient crew reaction time to preclude any hazard resulting from an undesired SA wing-commanded movement. Additionally, the STOCC has similar monitoring and commanding capability.

During nominal non-EVA SM periods, with SADE set to ON and the HST in the berthing position, adequate SA wing clearance exists to preclude a collision hazard in the event of inadvertent SA rotation.

7.0 SERVICING MISSION 1

7.1 INTRODUCTION

7.1.1 Background

The HST was launched on the STS-31 mission on April 24, 1990, and deployed the next day at an altitude of 337 n. mi. with a 28.5° inclination. The HST currently sits at a 317 n. mi. altitude with a 28.5° inclination. HST anomaly history since deployment is documented in Appendix D of this CSM.

The HST was designed for on-orbit servicing and refurbishment to meet and possibly exceed the anticipated 15-year lifetime of the spacecraft.

7.1.2 SA/ORI/ORU Replacement Identification

The following is a list of the replacement units for the first servicing mission with additional HST servicing components, tools, and aids:

1. Replacement Components

SA II WFPC II COSTAR RSU 2 and RSU 3 ECU 1 and ECU 3 DF-224 coprocessor Goddard High Resolution Spectrometer (GHRS) repair kit MSS 1 and MSS 2 P15 and P16 fuse plugs for PDUs 1-4 SADE 1

2. Additional Components

RSU 1 EP/TCE fuse plugs PDU P17 fuse plugs

For a list of EVA tools and crew aids, see volume II of this CSM.

7.1.3 STS-61 Payload Configuration

The payload complement for the first servicing mission included the MSFC Orbital Replacement Unit Carrier (ORUC) with a mission-specific complement of ORUs and ORIs, the Solar Array Carrier (SAC), and the GSFC FSS.

For full details of the aforementioned HST Space Support Equipment (SSE) and payload bay configuration, see the HST SSE CSM, volume II.

Figure 7.1-1 shows the HST mounted atop the FSS BAPS. The HST centerline coordinates with respect to the orbiter are at Xo = 1149.20 and Yo = 0.0 when berthed to the FSS. The face of the aft shroud is located at Zo = 496.594.

The HST interfaced to the orbiter payload primary power bus and SSP via the Interface Power Control Unit (IPCU) on the FSS and the remotely mateable/demateable umbilical on the BAPS. The IPCU received regulated orbiter dc power from two Power Distribution and Switching Units (PDSUs) that contain six Direct Power Converters (DPCs) each. The outputs of the IPCU were routed directly to the HST through the FSS maintenance umbilical (J101) nominally or through the deploy/return umbilicals (J107 and J108) in a contingency when the maintenance umbilical could be mated.

7.1.4 Mission Overview

For the first servicing mission, the orbiter launched on a direct insertion trajectory to the HST orbit, estimated at 315 n. mi. with a 28.45° inclination. Once the spacecraft had been properly configured, the orbiter rendezvoused with the HST, grappled and berthed the HST to the FSS, connected external orbiter power to the HST via umbilical, and serviced the spacecraft accordingly.

STS-61 carried a crew of 7 for an 11-day mission, with 5 planned EVA days to accomplish mission objectives. The SSE was activated on FD1 during postinsertion. On FD2, the orbiter RMS was activated and checked out. The orbiter rendezvoused, grappled, and berthed the HST on FD3. FDs 4 through 8 were used for performing five EVAs with duration of 6.0 hours each devoted to HST servicing (Figure 7.1-2). HST deployment activities took place on FD9, with FD10 reserved as an unscheduled EVA/deploy day. The mission also allowed 2 additional days on orbit for protecting the shuttle from weather conditions or for shuttle contingency delays before landing.

Two EVA teams, consisting of two members each, provided alternating support for each EVA day. The primary team performed EVAs on FD4, FD6, and FD8, with the secondary team performing EVAs on FD5 and FD7. FD10 was reserved for an unscheduled EVA if required to properly configure the HST before deployment. Nominal deployment took place on FD9. The EVA teams were crosstrained for all servicing activities to accommodate off-nominal situations that required changing EVA teams and team members.



Figure 7.1-1. HST berthed atop the FSS

7.1-3

	6-hour scheduled EVA period																
	*Inc	ludes insta	Ilat	ion of LGA cove	r, umbilical co	onnect	ctor cover ingress			ids							
EVA Day 1	Setup* RSUs 2 & 3 0:45 2:15		SAC prep/ ECU 0:50/ 1:35/ 0:45			CUs :45 f	CUs 3 & 1/Fuses Clos 45 fi 2:35 :25										
	0:40			2:00		AS d 0:45	AS doors):45		2:30		A 1		AS 1:0	6 doors 00	0:3	0	
EVA Day 2	Set SA (-) :20 2:50				SA (+) 2:10 Clos :25					6							
	:15 2 SA incl jettison 5:20					0:30				0:30							
EVA Day 3	Set :20	Set WFPC :20 4:15					MSSs 1 & 2 1:00 :25					Clos :25					
	:10	10 3:45				1:40 0:			0:45								
EVA Day 4	Set :20	Set COSTAR 20 3:10				Coproc Clos 1:40 :25											
	:10	3:20						2:20		N 1		MLI r 1:00	emoval	:25			
EVA Day 5																	
	Set :20	t SADE GH 1:30 0:5		GHRS 0:55		Close° 1:00											
	:20	2:55	5				PI 0:	DM 30		1:15		MLI i 0:50	nstall		1:00		

° Includes removal of LGA cover, ingress aids

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Figure 7.1-2. EVA timeline as flown vs. plan

The STS-61 mission provided corrective optics for the HST by changing out some of the original instrumentation. These replacement instruments compensated for the spherical aberration in the primary mirror. In addition, deteriorating system components were replaced with new units. The mission also allowed changeout of the original solar arrays. All replaced components were stowed on their carriers for landing, with the exception of one solar array, which could not be retracted and had to be jettisoned.

Propulsion quantity and time permitting, the HST was granted a reboost opportunity as agreed to by the Flight Control teams.

7.2 MISSION OPERATIONS

7.2.1 Rendezvous and Capture

Rendezvous and proximity operations leading to capture of the HST on FD3 occurred at a time based on orbiter phasing.

SSE activation procedures took place within 2 hours of payload bay door opening. This included activation of the FSS and ORUC heaters and the FMDM for monitoring of thermal telemetry. The primary prerendezvous activities included checkout of the RMS, the orbiter Ku-band antenna, the HST SSE, and the ground command system.

STOCC configured the HST for rendezvous operations with the orbiter before the terminal phase of the rendezvous, that is, before the orbiter TI burn. At this time, the STOCC had maneuvered the HST to a solar inertial attitude so that the +V3 axis is pointed toward the Sun with the -V1 axis coincident with the velocity vector at orbit noon on FD2. This was the HST rendezvous attitude. Following the maneuver, the onboard recorders were dumped, the DF-224 software configured, the aperture door closed and latched, the telemetry system configured to LGA-to-direct-TDRSS, the high-gain antennas deactivated and retracted, and the FHSTs and FGSs deactivated. The HST solar arrays were not retracted until after the HST had been successfully berthed to the FSS. Proximity operations and berthing could be performed with all the appendages deployed, if necessary.

After the orbiter TI burn, the HST initiated a nonessential load reduction to prevent loss of direct sunlight on the arrays. The orbiter and HST were then configured for S-band payload operations via PSP bypass while the Ku-band was used for radar during rendezvous activities.

Just before the HST roll maneuver to the capture attitude, the HST EPS and SA safemode tests were disabled. Shortly after the roll had been completed, the PCS safemode tests were disabled. The PSEA keep-alive monitoring was also disabled and the SADEs turned off.

As shown in Figure 7.2-1, the orbiter approached the HST just past the R-bar after orbital sunset on an inertial approach along the HST -V1 axis using approach techniques that minimized potential contamination. To facilitate grapple operations, the HST performed the aforementioned roll maneuver before grapple to align the grapple fixture on the -V3 axis with the orbiter RMS end effector. The size of the roll depended on the angle to the Sun and ranged from about 70° to 180°.





Figure 7.2-1. HST capture attitude definition

7.2-2

The flight crew nominally grappled the HST via the -V2/-V3 grapple fixture and deactivated the PCS, sending the HST into free drift mode. The LGAs were then swapped from the aft end to the forward end of the satellite to allow communications during berthed operations. The HST was berthed to the FSS using the orbiter mounted CCTV system and the FSS top mounted CCTV with a berthing target on the aft end of the HST. The three HST berthing pins were engaged by three corresponding FSS berthing latches as the HST was secured to the FSS BAPS. The orbiter then maneuvered to the battery charge attitude to top off the batteries before transfer to external power. The remotely operated umbilical on the FSS was then connected using the orbiter MCDS and the A6U panel and external power is applied to the HST via the orbiter SSP.

The onboard GPCs and the ground received telemetry from the forward HST LGA through the orbiter PI via the PSP bypass mode. Ku-band bent-pipe telemetry was used as a hot backup in case PDI telemetry is lost. Ku-band bent-pipe telemetry could be seen only at the STOCC. The SSM and OTA heaters were activated and the SAs remained deployed until the batteries were fully charged. The day's activities concluded with the FGEs turned back on and the SIC&DH, WFPC, HSP, and GHRS placed in warmup mode.

During priority operations, the orbiter supported RF communications with the HST as available. Specifically, Ku-band bent-pipe service for relaying HST telemetry (0.5, 4, 32, or 1024 kbps) to the ground was supported, on a noninterference basis, by the Ku-band in radar mode when the HST was outside of 5 n. mi. Within this range, the orbiter supported HST telemetry at 32 kbps via the PDI and orbiter operational downlink. The HST was commanded through the PSP/PI.

7.2.2 EVA Operations

This section describes the EVA operations necessary to accomplish the mission objectives.

All commanding activity originates from the STOCC unless otherwise specified. The IV crew could send certain commands from the onboard GPC SPEC and can monitor the status of certain HST systems during EVA operations and the rest of the mission.

All EVA activities were presented assuming the EV crew had already performed the payload bay setup procedures appropriate to the specific task.

7.2.2.1 SA Changeout

The SA changeout operation was identical for each array with the few exceptions that are noted. For detailed SA changeout operations, see Section 6.

a. Prepare for SA Changeout

For each array, the IV crewmember rotated the BAPS to the SA changeout position. Proper position is verified by lining up the SA markings on the HST with that on the FSS BAPS (90° for the -V2 SA, and 270° for the +V2 SA).
Before the EVA began, the SADE was turned ON and the stowage position of the PDM verified. The PDMs were then retracted, the aft latches closed, and the SADE and DCE turned off. Once the SAs are properly configured for changeout, the EV crew begins the changeout procedures.

During SM1, the -SDM was retracted after several attempts while the +SDM failed to retract. As a result, the EVA crew jettisoned the SA +wing. During the -SDM retraction, the solar array -wing drive brake slipped outside the allowable PDM stowage range resulting in a manual retraction of the -PDM.

The EV crew started by translating to the SAC and unlatching the SA forward latch. On the opposite end of the SAC, the SA II PDM handle was installed and the SADA clamp released. The SAC SADA clamp was identical to the HST SADA clamp.

b. Remove SA I from HST

At this point, one of the EV crewmembers retrieved the jettison handle and ingressed the MFR on the RMS. As this crewmember was being maneuvered to the SA I worksite, the other crewmember was installing a PFR at the HST worksite location. While the crewmember on the RMS was installing the jettison handle on the SDM cushion roller bracket, the crewmember in the PFR engaged the PDM locking device and the SAD brake. The same crewmember then disconnected the SA power and data connectors and stowed them on the temporary stowage bracket for transport. The SADA clamp spindle lock lockwire was removed and the SADA clamp released.

The SA I aft latch was then opened and the RMS EV crewmember removed and translated the SA I to the SAC using the jettison handle as a translation aid. As was mentioned earlier, the +wing could not be retracted and was therefore jettisoned.

c. Install SA I into the SAC TSB

The RMS crewmember installed the SA I onto the temporary stowage bracket by inserting the SA I forward handhold into the stowage bracket blade, while the other crewmember strapped the SA I down using the strapping cleat. It was necessary to position the SADA facing outboard for proper installation into the TSB.

d. Remove SA II from the SAC

A second jettison handle was placed onto SA II and the aft latch was opened. SA II was removed by the RMS EV crewmember using the jettison handle with assistance from the other crewmember using the PDM handle.

e. Install SA II onto the HST

The RMS EV crewmember carefully positioned the SA II into its seated position on the HST while the other crewmember closed the SADA clamp. The aft latch was then closed. The non-RMS crewmember then removed the power and data connectors from their temporary stowage brackets and connected them to their corresponding HST connectors. The PDM locking device was then released to complete SA II installation.

f. Install SA I onto the SAC

The TSB strap on SA I was then released. The SA I was placed onto the SAC and the forward latch partially closed. The SADA clamp and aft latch were fully closed, followed by complete closure of the forward latch. The jettison handle was then stowed and the three connector brackets connected to the bracket receptacles on the SAC. SA closeout procedures followed, including payload bay cleanup tasks and an aliveness test performed on the new SA.

g. SA Changeout without RMS

If the RMS had failed, the SA I arrays could be changed out using the jettison handle and the EVA handle on the SA. The SA II arrays used the PDM handle, the EVA handle, and the SA II spines as translation handholds.

7.2.2.2 WF/PC Changeout

The generic radial SI changeout operation is described in Section 3.4.

The BAPS was rotated to the WF/PC changeout position using the switch on the A6U panel. Proper position is verified by lining up the SA markings on the HST with that on the FSS BAPS (180° for WF/PC).

Before the EVA changeout, the WF/PC was powered down. WF/PC commands were nominally sent from the STOCC, although the crew could issue these commands from the spec if necessary. The IV crewmember powered on the SI indicator lights from the SSP.

The EVA began by rotating the aft fixture outboard to the 180° position. While one crew– member was rotating the scuff plate and opening the +V2 FGS doors to test the WF/PC A LATCH indicator lights, the other crewmember was installing the WF/PC I guide studs into the WF/PC I receptacles. The WF/PC I portable handhold were then retrieved from the aft fixture and installed onto the WF/PC I guide studs.

After receiving verification that the WF/PC blind-mate connector had been deadfaced, the blind-mate connector drive mechanism was demated and the WF/PC groundstrap disconnected.

The WF/PC A latch was then disengaged.

The WF/PC I was pulled out of the WF/PC cavity using the portable handhold and the guiderail system. This required a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with guidance from the non-RMS crewmember. This allowed for any repositioning necessary to ensure a level extraction of the WF/PC I. The WF/PC I removal continued until it was fully removed. WF/PC I was then translated to and stowed on the temporary parking fixture.

The radial SIPE door was then opened and the WF/PC II portable handhold removed from the forward fixture and installed onto WF/PC II.

The WF/PC blind-mate connector in the SIPE was demated and the ground strap released and stowed. The WF/PC A latch in the SIPE was then disengaged and the A LATCH indicators verified activated.

WF/PC II was removed from the SIPE by the RMS-mounted EV crewmember and translated to the HST worksite. The radial SIPE door was then closed and one of the latches latched. The tangs of the WF/PC II blind-mate drive connector mechanism were verified in the closed position, and the WF/PC pickoff mirror cover was removed from WF/PC II. To remove the cover, the PIP pin was removed first. By placing one hand on the grab bar and pulling on the lever arm with one hand, the cover was removed by pivoting the cover approximately 90° about the lower pins until the lower pins release. Delicate maneuvering was necessary to avoid contact with the pickoff mirror to prevent damage to the instrument.

WF/PC II was nominally installed into the HST WF/PC cavity using a coordinated EV/IV effort with an EV crewmember on the RMS and another EV crewmember providing guidance. This allowed for any repositioning necessary to ensure level insertion of WF/PC II. WF/PC II was inserted into the cavity and repositioned appropriately to engage the guiderails. It was then fully inserted into its seated position, the A latch engaged, and the A LATCH indicator lights verified deactivated.

The WF/PC ground strap was then reconnected and the blind-mate connector mated. Once the blind-mate connector was mated, the connector test switch was actuated to verify a good electrical connection. The WF/PC II portable handhold was removed and restowed on the handhold stowage fixture.

The +V2 FGS doors were closed and the scuff plate rotated back into position.

The pickoff mirror cover was installed on the WF/PC I by pushing the engagement pins inward toward each other on the lower hinge. The lever arm should have been in its lowest detent on the cams with the upper pins fully retracted. The cover was engaged on the WF/PC I lower hinge pins with the cover rotated away from the pickoff mirror. The cover was pivoted about its lower hinge pins using the lever arm. As the cover reached its stowage position, the upper pins engaged and the lever arm seated properly in its detent position. The PIP pin was then installed to lock the lever arm in place. The radial SIPE door was then reopened. WF/PC I was removed from the temporary parking fixture and installed in the radial SIPE in the same manner that the WF/PC II was installed in the HST with the same precautions. The WF/PC I blind-mate connector was not remated in the radial SIPE. The WF/PC I portable handhold was removed and restowed on the aft fixture. The radial SIPE doors were then closed and latched and cleanup procedures performed.

An aliveness test was performed after WF/PC II installation to check the electrical path to the instrument.

7.2.2.3 HSP/COSTAR Changeout

The generic axial SI changeout operation is detailed in Section 3.4.

The BAPS was rotated to the HSP/COSTAR changeout position using the switch on the A6U panel. Proper position was verified by lining up the SA markings on the HST with that on the FSS BAPS (90° for HSP/COSTAR).

Before the EVA changeout, the HSP was powered down. Commands were nominally sent from the STOCC, although the crew could issue these commands from the spec if necessary.

The EVA began by rotating the aft fixture outboard to the 180° position, and removing and storing the handhold on the forward fixture.

The -V2 aft shroud doors were opened and the portable worklight assembly installed. The IV crewmember activated the worklights and SI indicator lights from the SSP.

Upon verification of HSP powerdown, the four power and data connectors and ground strap were disconnected. The A latch was then released and the A LATCH indicator lights verified. The B-latch cover was removed and the B latch released. The HSP was lowered until the bottom guide rail was contacted to position the HSP for removal along the OTA guide rails. At this point, the A SEATED indicator lights were verified. The HSP was then pulled out of the axial cavity via the guide rail system and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with guidance from the non-RMS crewmember. This allowed for any repositioning necessary to ensure a level extraction of the HSP. The HSP removal continued along the OTA guiderails until fully removed. The B READY indicator lights turned on during HSP removal. Once removed, the HSP was stowed on the temporary parking fixture.

The axial SIPE door was opened and the axial SIPE groundstrap disconnected and stowed. The A-latch and B-latch were then opened, verifying that the A SEATED indicator light activated during B-latch opening. The COSTAR then pulled out of the axial SIPE via the guide rail system and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with guidance from the non-RMS crewmember. This allowed for any repositioning necessary to ensure a level extraction of the COSTAR. The B LATCH indicator lights were activated during COSTAR removal.

COSTAR was then translated back to the HST worksite for installation. Prior to installation, the COSTAR contamination cover was removed from the aperture opening. COSTAR was inserted into place along the OTA guide rails until the B READY indicator lights were deactivated. The B-latch was then engaged and torqued. COSTAR raised as the B-latch is engaged and the A-and C-latches were then seated. The A SEATED indicator lights turned OFF when the B-latch was torqued. The A-latch was then closed and torqued and verified latched by the A LATCH indicator lights. The B-latch cover was then closed. The COSTAR ground strap was attached and the four power and data connectors remated using the identification and clocking markings on the connectors.

The HSP was removed from the aft fixture and installed into the SIPE for landing in the same manner that the COSTAR was installed into the HST. The axial SIPE door was then closed. The forward fixture and the portable handhold were also stowed and the payload bay cleaned up.

An aliveness test was performed after COSTAR installation to check the electrical path to the instrument.

7.2.2.4 RSU Changeout

The RSU changeout operation is identical for each RSU. For specific hardware details and generic EVA operations, see Section 2.7.1.5.

The IV crewmember rotated the BAPS to the RSU changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (180° for all of the RSUs).

Before the EVA changeout, the RSU survival heaters were powered off. RSU commands were nominally sent from the DIU and DMU from the STOCC, although the crew could issue these commands from the spec if necessary, or from the SSP for power to the RSU survival heaters.

At the -V3 aft shroud worksite, the FHST 2 and 3 shade seals were retracted, the latches on the -V3 aft shroud doors and the handle latch were disengaged, and the doors were opened and tethered at the padded stop position. Once the -V3 doors were opened, the RSUs were powered off.

The portable worklight assembly was then installed and powered. After a GO was received from the IV crewmember, the two wingtabbed RSU power connectors were disconnected and the RSU removed. The replacement RSU was then retrieved from the SOPE and installed and the original RSU stowed for landing.

After the replacement RSU was installed the power connectors were remated. The worklight was then powered down and removed, and the doors closed. The changeout procedure concluded with re-engaging the FHST 2 and 3 shade seals.

An aliveness test was performed after RSU installation to check the electrical path to the instrument.

During SM1, the crew experienced difficulty in closing the -V3 aft shroud doors. The top of the doors would not come close enough together to engage the latches. The crew eventually used the Payload Retention Device (PRD) to winch the doors together, successfully latching the doors closed.

7.2.2.5 ECU Changeout

The ECU changeout operation is identical for each ECU. For specific hardware details and generic EVA operations, see Section 2.7.1.5.

The IV crewmember rotated the BAPS to the ECU changeout position using the switch on the A6U panel. Proper position is verified by lining up the markings on the HST with those on the FSS BAPS (80° for all of the ECUs).

Before the EVA changeout, the RSU survival heaters were powered off. ECU commands were nominally sent from the DIU and DMU from the STOCC, although the crew could issue these commands from the spec if necessary.

At the -V2 SSM ES bay 10 worksite, the bay 10 door was opened to the full open position. After a GO was received from the IV crewmember, the original ECU power connectors were demated by driving the ECU away from its connector bracket. The ECU was then removed. The replacement ECU was then retrieved from the SOPE and installed, and the power to the ECU and RSU heaters reapplied. The bay 10 door was then closed, the original RSU stowed for landing, and the worksite cleaned up.

An aliveness test was performed after ECU installation to check the electrical path to the instrument.

7.2.2.6 GHRS Repair Kit Installation

The IV crewmember rotated the BAPS to the EVA changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (270° for the GHRS repair kit).

The GHRS repair kit was retrieved from the SOPE and translated to the +V2 aft shroud worksite. Before opening the aft shroud doors, the orbiter attitude was verified. The +V2 aft shroud doors were then opened by disengaging the three latch bolts, disengaging the handle bolt, releasing the handle on the door, and opening the door. The GHRS was then deactivated.

GHRS commands were nominally sent from the STOCC, although the crew could issue these commands from the spec if necessary.

GHRS repair kit installation was accomplished by demating the four OTA connectors from the four connectors on the GHRS connector panel. The four repair kit connectors designated as P1, P2, P3, P4 were connected to the GHRS connector panel. The four repair kit connectors designated J1, J2, J3, J4 were connected to the OTA connectors that were previously connected to the GHRS connector panel. The installation was followed by closeout photos of the worksite. The +V2 aft shroud doors were then closed, the handle latched, and the three latch bolts engaged.

An aliveness test was performed after GHRS changeout to test the integrity of the hardware.

7.2.2.7 Coprocessor Installation

The IV crewmember rotated the BAPS to the EVA changeout position using the switch on the A6U panel. Proper position is verified by lining up the markings on the HST with those on the FSS BAPS (150° for the coprocessor changeout).

The coprocessor was retrieved from the LOPE and translated to the bay 1 worksite. The bay 1 door was then opened by releasing the six J-bolts, opening the bay door, and setting the door stay. The DF-224 handles were then removed by extracting the noncaptive handle bolts. On MCC GO, the P1/P2 heater connectors were demated from the DF-224 and the T-harness cable was installed. The coprocessor was attached to the DF-224 with four bolts. The P9 and P8 connectors were then mated. Closeout photos were taken of the bay 1 worksite, and the bay 1 door was then closed by fastening the six J-bolts. The DF-224 handles were stowed in the LOPE for re-entry.

When power was removed from the DF-224, the crew and MCC lost telemetry; however, the STOCC was configured to receive Ku-band bent-pipe telemetry.

An aliveness test was performed as soon as coprocessor installation was complete. The bay 1 door was not closed until the aliveness test had been satisfactorily completed.

7.2.2.8 Installation

The IV crewmember rotated the BAPS to the MSS changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (330° for both MSS 1 and MSS 2). In addition, the FSS BAPS pivoter was pivoted to the 75° position for access to the MSS worksite.

Before MSS changeout operations, the MSS 1 was powered off. MSS 1 and MSS 2 commands were nominally sent from the STOCC, although the crew could issue the MSS 1 power off commands from the spec if necessary.

The new MSSs were retrieved from the ORUC and installed atop the already existing MSS.

The new MSS 1 was structurally mounted atop the original MSS 1 by removing the MLI cover, demating the power connector, hand tightening the four knobs on the new MSS mounting structure, reconnecting the power connector to the new MSS, and replacing the MLI cover. While the MSS 1 aliveness test was in progress, the MSS 2 was structurally mounted atop the original MSS 2, but the power connector was not removed from the original MSS 2 until the MSS 1 aliveness test checked out successfully. Upon completion of the MSS 1 aliveness test, the MSS 2 power connector was demated from the original MSS 2 and remated to the new MSS 2. The MLI was then replaced over MSS 2 and the aliveness test is performed.

During SM1, the EVA crew discovered that all old MSS 2 sensor housing aluminum face sheets had delaminated from the foam potting encasing the sensor. Prolonged UV degradation of the exposed foam was considered a contamination threat to the HST aperture. As a result, the EVA crew constructed MSS covers taken from the WF/PC handhold MLI blankets and the RSIPE contingency latch release covers.

7.2.2.9 Fuse Plug Changeout

For specific hardware details and generic EVA operations, see Section 2.4.1.6.

The IV crewmember rotated the BAPS to the PDU fuse plug changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (210° for the fuse plugs).

Before starting the changeout, the circuits downstream of the fuse plug were verified deactivated. However, the fuse plug remained HOT with 28 V at the input side of the fuse plug. The RSU survival heater was placed on orbiter-supplied power from SSP S16, bypassing the PDU relay.

Once configured, the fuse plugs were ready for changeout at the bay 4 worksite. When the fuse plugs had been changed out, the PDU relays were reconfigured and an RSU aliveness test was performed to test the integrity of the new fuse plugs.

7.2.2.10 SADE 1 Changeout

For specific hardware details and generic EVA operations, see Section 6.2.6.2.

The IV crewmember rotated the BAPS to the SADE 1 changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (180° for the SADE).

Before SADE 1 changeout operations, the SADE 1 was powered off. SADE 1 commands were nominally sent from the STOCC, although the crew could issue the SADE 1 power off command from the spec, if necessary.

At the bay 7 door worksite, the bay 7 door was opened and tethered. After a GO was received from the IV crewmember, the original SADE 1 connectors were demated and the SADE 1 removed. The replacement SADE 1 was retrieved from the COPE and installed in bay 7. The new SADE 1 had four mounting bolts compared to six mounting bolts on the original SADE 1. Once the connectors were remated, the bay 7 door was closed and the worksite closed out.

An aliveness test was performed after SADE 1 installation to check the electrical path to the instrument.

7.2.3 Redeployment

After HST servicing was completed, the telescope was redeployed. Some activities for configuring HST for deployment occurred on the FSS and some occurred on the RMS.

The solar arrays and high gain antennas were deployed while the HST was berthed to the FSS. The orbiter assumed an attitude to allow the HST batteries to charge. After the batteries had reached the required state of charge, the RMS grappled the telescope and the crew transferred from orbiter power to HST batteries. The umbilical was demated when the HST was verified on internal power. The HST was unberthed from the FSS and maneuvered to the release position.

In the release position, the STOCC commanded the HST aperture door open and enabled the pointing and safe-mode electronics assembly. The telescope was released; after the first separation burn, the crew commanded the HST to S/W Sunpoint. The STOCC recovered the HST and configured it for orbit operations.

8.0 SERVICING MISSION 2

8.1 INTRODUCTION

8.1.1 Background

The HST was launched on April 24, 1990, and deployed the next day at an altitude of 337 n. mi. with a 28.5° inclination. The telescope was first serviced during the STS-61 mission in December 1993. HST anomaly history since the deployment mission is documented in Appendix D.

The HST has been designed for on-orbit servicing and refurbishment to meet, and quite possibly exceed, the anticipated 15-year lifetime of the spacecraft.

8.1.2 ORU Replacement Identification

The following is a list of the replacement units for the second servicing mission with additional HST servicing components, tools, and aids:

- a. Planned replacement components (by priority)
 - 1. NICMOS (FOS)
 - 2. FGS (FGS-1)
 - 3. SSR (ESTR-1)
 - 4. STIS (GHRS)
 - 5. RWA (RWA-1)
 - 6. DIU-2
 - 7. ESTR-2
 - 8. OCE Kit
 - 9. SADE-2
 - 10. MSS Covers
- b. Contingency components
 - 1. Rate Sensor Unit (RSU)
 - 2. Electronics Control Unit (ECU)
 - 3. PCU essential bus harness

- 4. SSE "J" harness
- 5. PDU fuse plugs
- 6. OTA fuse modules
- 7. SA bistem braces
- 8. SA thermal shield repair devices
- 9. MLI patches

For a list of EVA tools and crew aids, see the HST SSE CSM.

8.1.3 STS-82 Payload Configuration

The payload complement for the second servicing mission included the MSFC ORUC with a mission-specific complement of ORUs and ORIs, the SAC, and the GSFC FSS.

For full details of the aforementioned HST SSE and payload bay configuration, refer to the HST SSE CSM.

Figure 8.1-1 shows the payload bay configuration. The HST centerline coordinates with respect to the orbiter are at Xo = 1149.20 and Yo = 0.0 while berthed to the FSS. The face of the aft shroud is located at Zo = 496.594.

The HST is interfaced to the orbiter payload primary power bus and SSP via the IPCU on the FSS and the remotely mateable/demateable umbilical on the BAPS. The IPCU received regulated orbiter dc power from two Power Distribution and Switching Units (PDSUs) that feed six DPCs each. The outputs of the IPCU were routed directly to the HST through the FSS maintenance umbilical (J101) nominally or through the deploy/return umbilicals (J107 and J108) in a contingency where the maintenance umbilical could not be mated.

8.1.4 Mission Overview

For the second servicing mission, the orbiter launched on a direct insertion trajectory to the HST orbit, estimated to be approximately 319 n. mi., with a 28.45° inclination. Once the spacecraft had been properly configured, the orbiter rendezvoused with the HST, grappled and berthed the HST to the FSS, connected external orbiter power to the HST via umbilical, and serviced the spacecraft accordingly.

STS-82 carried a crew of 7 for an 11-day mission, with 4 scheduled and 2 unscheduled EVA days to accomplish mission objectives. The SSE was activated on FD1 during post insertion. On FD2 the orbiter RMS was activated and checked out. The orbiter rendezvoused, grappled, and berthed the HST on FD3. FDs 4 through FD8 were used for performing five EVAs of 6.0 hours each for HST servicing. An overview of the EVA can be seen in figure 8.1-2. HST deployment activities took place on FD9. The mission

also allowed 2 additional on-orbit days for the shuttle to protect against weather conditions or shuttle contingency delays before landing.

Two EVA teams, consisting of two crewmembers each, provided alternating support for each EVA day. One team performed EVAs on FD4, FD6, and FD 8, and the other team performed EVAs on FD5 and FD7. FD8 and FD9 were reserved in case an unscheduled EVA was required for contingency mission success and to properly configure the HST before deployment. The contingency FD8 EVA was used to repair some damaged MLI on HST. The EVA teams were crosstrained for all servicing activities to accommodate off-nominal situations requiring shuffling of the EVA teams and crewmembers.

The STS-82 mission replaced two axial science instruments, with new capability instruments that contain their own compensators for the spherical aberration in the primary mirror. The remaining axial science instrument, FOC, still requires the COSTAR for its correction to the aberration in the mirror. In addition, deteriorating system components were be replaced by their respective replacement units. All replaced components were be stowed on their respective carriers for landing.

Due to propulsion margin and available time, the HST was granted a VRCS reboost opportunity. The reboost could only utilize orbiter vernier thrusters due to SA loads, and the HST was stabilized by installing and rigidizing the BAPS post on the FSS.



Figure 8.1-1. Cargo bay arrangement

8.1-4



Hubble Space Telescope Flight Systems and Servicing Project

HST SM-2 EVA Scenario As Flown vs. Planned





Figure 8.1-2. HST SM-2 EVA timeline as flown vs. plan

8.1-5

8.2 MISSION OPERATIONS

8.2.1 Rendezvous and Capture

Rendezvous and Proximity Operations leading to capture of the HST on FD3 would have occurred at a time based on HST-orbiter phasing.

Primary pre-rendezvous activities included checkout of the RMS, EMUs, the orbiter Ku-band antenna, the PGSC (which was be used to enhance HST PSP bypass communications), the HST SSE, and the ground command system. SSE activation procedures took place within 2 hours of PLBD opening. This included the activation of the FSS, SAC, and ORUC heaters, as well as the FMDM and use of the orbiter MDMs for monitoring of the thermal telemetry.

HST activities prior to rendezvous included dumping of the onboard tape recorders, configuration of the DF-224 flight software, closure of the AD, configuration of the communications system to LGA-direct-TDRS, HGA deactivation and retraction, and FHST and FGS deactivation. Should the AD not have closed as expected, HST would have maneuvered to a -V1 VV at orbit-noon attitude and the orbiter would have performed a V-bar approach.

The STOCC configured the HST for rendezvous operations with the orbiter prior to and during the terminal phase of the rendezvous with all operations except load shedding, roll-to-grapple attitude, and PSP-bypass communications configuration performed prior to the orbiter TI burn. The STOCC maneuvered the HST to the rendezvous attitude. This attitude was designed to maintain HST power position and minimize plume loading on the SAs at the 150-foot normal Z braking gate. The rendezvous attitude was determined by these conditions: Sun in the V1 -V3 plane and the +V1 axis pointing down the R-bar at an orbit angle of 59° before orbit noon.

Following the orbiter TI burn, the HST initiated a nonessential load reduction in order to minimize the power-negative effects of the roll-to-capture attitude. S-band communications between the orbiter and HST was established via the orbiter payload communications system in the PSP-bypass mode augmented with the HST PGSC. Contingency HST direct-TDRSS communications links remained available throughout the rendez-vous and capture. The Ku-band was utilized for radar during rendezvous activities.

Prior to grapple, HST performed a roll maneuver to the capture attitude to align the -V3/+V2 grapple fixture on HST with the orbiter RMS. The roll maneuver placed the -V3 axis 52° out of plane north as measured on the Sun side of HST. The magnitude of the roll maneuver about V1 = β -52°, resulting in a roll that ranged from 0° to -104°. HST EPS and SA safe-mode tests were disabled. Shortly after the roll had been completed, the PCS safe-mode tests were disabled. At that time a 180-minute backup Sun-point timer was activated to protect the HST in the event of rendezvous break-off. The PSEA keep-alive monitoring was also disabled and the SADEs turned off.

As shown in Figure 8.2-1, the orbiter approached the HST using the R-bar approach technique that minimized propellant usage and potential contamination. The capability existed to perform proximity operations, grapple, and berthing with all of the appendages deployed. However, the -V3 HGA would have to be positioned to maximize clearance for RMS grapple. The flight crew nominally grappled the HST via the -V2/-V3 grapple fixture, and deactivated the PCS by commanding the HST into drift mode.

HST-orbiter communications were reconfigured from the aft LGA to the forward LGA to allow for communications during berthed operations. The HST was berthed to the FSS using the orbiter-mounted CCTV system and the FSS top mounted CCTV with a berthing target on the aft end of the HST. The three HST berthing pins were engaged by three corresponding FSS berthing latches as the HST was secured to the FSS BAPS. The remotely operated umbilical on the FSS was then connected, utilizing the orbiter MCDS and the A6U panel, and external power was applied to the HST via the orbiter SSP. The orbiter then maneuvered to the battery charge attitude to top off the batteries.

The onboard GPCs and the ground received telemetry from the forward HST LGA through the orbiter PI and PDI via the PSP-bypass mode. Ku-band bent-pipe telemetry was utilized as a backup in the event that the PDI telemetry was lost. Ku-band bent-pipe telemetry can be displayed only at the STOCC.

The SSM and OTA heaters were activated and the SAs remained deployed throughout the mission. At the conclusion of rendezvous day activities, the FGEs were turned back on, and the SIC&DH, WF/PC, STIS, FOC, and NICMOS were be placed in warmup mode.

The orbiter supported HST telemetry at 32 kbps via the PDI (500 kbps, 4 kbps, and 32 kbps when augmented by the HST PGSC) and orbiter operational downlink. The HST was commanded through the PSP/PI. Ku-band bent pipe service for relaying HST telemetry (0.5, 4, 32, or 1024 kbps) to the ground was supported as needed whenever Ku-band TDRS coverage was available.





Figure 8.2-1. HST R-bar approach

8.2-3



Figure 8.2-1. Concluded

8.2.2 EVA Operations

This section describes the EVA operations performed to accomplish the mission objectives.

All commanding activity was done from the STOCC unless otherwise identified. The IV crew had the capability to send certain commands from the onboard GPC SPEC and to monitor the status of the certain HST systems during EVA operations and the rest of the mission.

8.2.2.1 STIS/GHRS Changeout

Note: The nominal plan was to install STIS and NICMOS, respectively, during the same EVA.

For more detailed information concerning latching go to the generic axial SI changeout operation in Section 3.4.2 of this document. Prior to the beginning of the EVA changeout, the BAPS was rotated to the GHRS/STIS changeout position using the switch on the A6U panel. Proper position was verified by lining up the SA markings on the HST with that on the FSS BAPS. The GHRS was thermally preconditioned and the COSTAR deployable optical bench was retracted. Commands were nominally sent by the STOCC.

The EVA began by rotating the aft fixture outboard, and deploying the aft fixture on the ORUC. The IV crewmember activated the worklights and SI indicator and SI indicator lights from the SSP.

Next, the +V2 aft shroud doors were opened to the fully open position at the bump stop, and tethered open. The IV called for an MCC GO for GHRS powerdown. Upon verification of GHRS powerdown, the eight GHRS repair kit connectors were disconnected, then mated to one another and tethered to a GHRS handle. Next, the groundstrap was disconnected and the GHRS purge line was disconnected and stowed on the aft bulk-head purge line fitting. Then, the A-latch was released and the A LATCHED indicator lights verified. After the A-latch was released, the B latch cover was removed and the B latch released. The crew verified that the A SEATED indicator lights illuminated when the GHRS A-latch was unseated. Next, the GHRS was lowered until the bottom guide-rail was contacted to position the GHRS for removal along the OTA/SSM guiderails. The B READY indicator lights were monitored during the extraction from the B-ready-to-latch position. The GHRS was then pulled out of the axial cavity via the guiderail system and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with the non-RMS crewmember providing guidance support. This allowed for any repositioning necessary to ensure a level extraction of the GHRS. Once removed, the GHRS was stowed on the ORUC aft fixture and secured with one PIP pin.

The ORUC ASIPE door was opened and the ASIPE groundstrap was disconnected and stowed. The A-latch was disengaged and the A LATCHED indicator lights illuminated. Then, the B-latch MLI cover was opened and the B latch was released. STIS was then

translated back to the HST worksite for installation. STIS was inserted into place along the OTA/SSM guiderails until the B READY indicator lights were extinguished. The B-latch was then engaged and torqued. STIS rose as the B-latch was engaged and the A- and C-latches were seated. The A SEATED indicator lights were extinguished when the B-latch was torqued. The B-latch cover was then closed. Then, the A-latch was closed, torqued, and verified latched by the A LATCHED indicator lights extinguishing. Next, the STIS groundstrap was attached, and the four power and data connectors remated using the identification and clocking markings on the connectors.

Upon completion of the STIS installation, IV gave MCC a GO for STIS aliveness test. The STOCC performed the aliveness test to check the electrical path to the instrument.

The GHRS was removed from the aft fixture and installed into the ASIPE until B READY indicator lights extinguished. A SEATED lights were extinguished when B-latch torquing was complete. After the B-latch was fully engaged, the A-latch was engaged and torqued. The A-latch was verified latched with the extinguishing of the A LATCHED lights. Finally, the ASIPE lid was closed and latched.

8.2.2.2 NICMOS/FOS Changeout

For more detailed information concerning latching, go to the generic axial SI changeout operation in Section 3.4.2 of this document. Before beginning the EVA changeout, the BAPS was rotated to the FOS/NICMOS changeout position using the switch on the A6U panel. Proper position was verified by lining up the AS markings on the HST with those on the FSS BAPS.

The COSTAR deployable optical bench was retracted, the aft fixture on the ORUC was deployed, the SI indicator lights were activated from the SSP via the SM worklights switch, and the +V2 aft shroud doors were opened to the full open position. The changeout began by the IV calling MCC to GO for FOS powerdown. The powerdown was nominally performed by the STOCC. Upon verification of FOS powerdown, the four power and data connectors were disconnected. Next, the groundstrap was disconnected from the FOS and stowed. Then, the FOS purge line was disconnected and stowed on the aft bulkhead purge line fitting.

The FOS A-latch was then released and the A LATCHED indicator lights were verified to be illuminated. The B-latch MLI cover was removed and the B-latch released. As the B-latch was released, the status of the A SEATED indicator lights were be verified to be illuminated and the B READY light verified as deactivated. The FOS was then pulled out of the AS, via the guiderail system and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with the non-RMS crewmember providing guidance support. The FOS was kept parallel with the V1 axis to position the FOS for removal along the OTA guiderails. Once removed, the FOS was temporarily stowed on the ORUC aft fixture.

The SAC ASIPE door was opened and the ASIPE groundstrap disconnected and stowed. The A-latch was disengaged causing the A LATCHED indicator lights to illuminate. Then the B-latch was opened, the A SEATED indicator lights verified illuminated, and the B READY light verified extinguished. NICMOS was then pulled out of the A SIPE, via the guiderail system, and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with the non-RMS crewmember providing guidance support. This allowed for any repositioning necessary to ensure a level extraction of the NICMOS. The B READY indicator lights were activated during NICMOS removal. After removal, the ASIPE lid was temporarily closed and secured with one latch.

NICMOS was then translated back to the HST worksite for installation. NICMOS was inserted into place along the OTA/SSM guiderails until the B READY indicator lights were extinguished. NICMOS was kept parallel to the HST V1 axis. The B-latch was then engaged and torqued. NICMOS rose as the B latch was engaged and the A- and C-latches were seated. The A SEATED indicator lights were extinguished when the B-latch was torqued. The B-latch MLI cover was then closed. Then, the A-latch was closed, torqued, and verified latched from the A LATCHED indicator lights extinguishing. Next, the NICMOS groundstrap was attached, and the four power and data connectors remated using the identification and clocking markings on the connectors.

After the connectors were mated, IV notified MCC GO for NICMOS aliveness Test. The STOCC performed the aliveness test after NICMOS installation to check the electrical path to the instrument.

The cryo vent cover was removed from NICMOS and the cryo vent line was retrieved from the ORUC Multimission ORU Protective Enclosure (MOPE) stowage location where the cryo vent cover was stowed. The MOPE lid was closed and secured with all five latches.

The cryo vent line was routed under the lower SSM guiderail. The three NICMOS cryo vent line interface bolts were disengaged. The NICMOS capillary line plug was opened and the cryo vent line was routed through the bulkhead handrail. The cryo vent line was installed onto the NICMOS and the three-keyway slot bolts engaged and torqued. The NICMOS cryo vent line valve was opened and the cryo vent line strap was secured to the handrail.

The +V2 AS doors were closed with IV disabling the SI indicator lights via the SSP SM worklights switch.

The FOS was removed from the aft fixture and installed into the ASIPE until B READY indicator lights extinguish. A SEATED lights were extinguished when B-latch torquing completed. After the B-latch was fully engaged, the A-latch was engaged and torqued. The A-latch was verified latched with the extinguishing of the A LATCHED lights. Finally, the ASIPE lid was closed and latched.

8.2.2.3 FGS-1 Changeout and OCE Kit

Note: The nominal plan was to install the FGS, ESTR-2, and the OCE Kit, respectively, during the same EVA.

For more detailed information concerning latching, go to the generic radial SI changeout operation in Section 3.4.1 of this document. Before the EVA, the IV crewmember rotated the BAPS to the FGS-1/OCEK changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS.

IV activated S24 SSM WK LIGHTS to activate the FGS latch indicator lights and the HST electrical receptacles.

The changeout began by deploying the aft fixture. Next, the scuff plate was repositioned by disengaging four fasteners, rotating the plate 90°, and tightening one upper bolt. The -V2 FGS doors were opened by disengaging the two latch bolts and releasing the handle. Four FGS guide studs were installed on the FGS.

Then IV called for MCC to GO for FGS powerdown. The STOCC nominally performed the powerdown. After receiving verification that the FGS blind-mate connector had been deadfaced, four left wingtab connectors and groundstrap were demated, and tethered back to the left tether loop. Then, the four right wingtab connectors and groundstrap wingtab connectors were demated, and tethered back to the right tether loop.

The FGS portable handhold was then retrieved from the forward fixture and installed onto the FGS guide studs via the four zipnuts with the spacers and MLI A latch cutout on the FGS handhold at the bottom (-V1).

The FGS A latch was then disengaged with the A LATCHED indicator lights illuminating.

The FGS was pulled out of the FGS cavity using the portable handhold and the guiderail system and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with the non-RMS crewmember providing guidance support. This allowed for any repositioning necessary to ensure a level extraction of the FGS. The FGS may have needed to be pitched up (+V1) to avoid A-latch contact with the coplanar D-latch fitting. The FGS could only be touched at the portable handhold, ground support equipment brackets, and FGS guiderails.

FGS was translated to and temporarily stowed on the aft fixture.

The FSIPE door was opened with the groundstrap disconnected and stowed. The FGS-1R portable handhold was removed from the forward fixture and installed onto the FGS-1R via four zipnuts with the MLI cutout on the FGS-1R handhold at the bottom. The FGS A-latch in the FSIPE was then disengaged. The A-latch indicators were verified with the A LATCHED lights illuminated. The FSIPE groundstrap was disconnected and stowed in the FSIPE groundstrap stowage receptacle. The FGS A-latch in the FSIPE was then disengaged 1/2 revolution to break the seating torque followed by the remaining 22+ revolutions (until the A-latch was fully disengaged). The A-latch indicators were verified activated after the first 2.5 revolutions CCW of the A-latch, indicating a not fully latched condition.

The FGS-1R was removed from the FSIPE guiderails. The FGS-1R may have needed to be pitched up (orbiter forward) to avoid A-latch contact with the coplanar D-latch fitting. During removal, the FGS-1R pickoff mirror could not be touched due to sharp edge and contamination concerns. The FSIPE door was then closed and one of the latches latched. Next, the FSIPE lid on-orbit latches were verified to be armed. The FGS-1R pickoff mirror cover was then removed from FGS-1R by releasing the locking device and moving the handle downward and pivoting the mirror cover away from the FGS-1R mirror. Care was taken to avoid contacting the pickoff mirror and the absorbers near the pickoff mirror to prevent possible damage to the instrument.

The FGS-1R was nominally installed into the HST FGS-1R cavity using a coordinated EV/IV effort with an EV crewmember on the RMS and another EV crewmember providing guidance. This allowed for any repositioning necessary to ensure a level insertion of FGS-1R. However, the FGS-1R may have needed to be pitched up (+V1) to avoid A-latch contact with the coplanar D-latch fitting. The FGS-1R was inserted into the cavity and repositioned approximately to engage the guiderails. It was then fully inserted into its seated position. The A-latch was engaged and torqued and the A LATCHED indicator lights were extinguished. The four FGS-1R handhold zipnuts and the handhold on the new FGS were removed leaving only the four FGS-1R guide studs to remain.

The EV crewmember then mated the FGS-1R groundstrap wingtab connector to the replacement FGS-1R. Then the eight wingtab connectors were mated to the FGS-1R.

After completing the FGS-1R installation, IV will gave MCC a GO for FGS-1R aliveness test. The STOCC performed the aliveness test to verify electrical integrity of the instrument connectors.

With both doors fully closed, the handle latch and two door latches were reinstalled and torqued. Then, the three FGS-1R door latches were retorqued. Next, the FGS-1R doors were visually inspected for closure including the resilient door light seal. Then, the scuff plate was repositioned by rotating 90° and engaging the four fasteners.

The FGS was removed from the aft fixture and inserted into the FSIPE guiderails. The FGS may have needed to be pitched up (orbiter forward) to avoid A-latch contact with the coplanar D-latch fitting. The FGS was inserted into the FSIPE and repositioned appropriately to engage the guiderails. It was then fully inserted into its seated position. The A-latch was engaged and torqued with the FGS A LATCHED indicator lights extinguishing.

The FSIPE groundstrap wingtab connector was mated to the FGS and the four FGS handhold zipnuts and the handhold on the FGS were removed leaving only the four guide studs on the FGS. The handhold was installed on the forward fixture.

After the worksite was clear, the FSIPE lid was closed and all three lid latches latched and verified. The aft fixture was also stowed.

To perform the OCE EK Installation, the five SAC SOPE lid latches were released and the SOPE lid fully opened. The SOPE tool board was raised and the OCE EK was retrieved from the SOPE. The SOPE tool board was then lowered and the SOPE lid closed and five lid latches engaged.

The HST bay C door is opened by releasing the three J-hooks and opening the door. The door was secured via either the OTA door stay or tethering. The bay C door opening angle could not exceed 118°.

Then IV called for MCC to GO for FGS powerdown. The STOCC nominally performs the powerdown. After receiving a GO from IV, the EV crewmember demated OCE connectors P6 and P11.

After receiving a GO from IV, the EV crewmember demated OCE connectors P6 and P11. Any additional OCE connectors could have been demated at crew discretion if required to improve crew access to the connectors.

OCE EK connector P11A was mated to the corresponding OCE connector J11. OCE EK connector P6A mated to OCE connector J6 with OCE EK connector J6A mated to HST harness connector P6. OCE EK connector J11A was mated to the corresponding HST connector P11. Any additional connectors demated to improve access were remated at this point. The untethered connector protective cap on OCE J13 was removed and stowed. OCE EK connector P13 was mated to OCE connector J13. The remaining OCE EK connectors remained mated to each other. The OCE EK was positioned to ensure no interference with bay C door closure. EV notified IV after the kit was installed.

IV notified MCC GO for OCE aliveness test. The STOCC performed the aliveness test to verify electrical integrity of OCE EK connectors.

The OTA door stay or tethers were removed from the bay C door and the bay C door closed and the three J-hooks engaged and torqued.

8.2.2.4 ESTR-2 Changeout

Prior to ESTR-2 changeout operations, the IV crewmember rotated the BAPS to the ESTR-2 changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS.

The HST bay 8 door was opened by releasing the six J-hooks. The door could have been kept in the fully open position by installing the SSM adjustable door stay or by tethering the door. The bay 8 door could not exceed 118°.

8.2-10

The IV called MCC GO for ESTR-2 powerdown. ESTR-2 commands were nominally sent by the STOCC. After deadfacing the ESTR-2, IV gave EV a GO to demate the ESTR-2 P1, P2, and P3 connectors. The P1 connector sockets were electrically energized and could not be contacted. No structure or tool could contact the pins of the P1 connectors. To minimize the probability of inadvertent shorting, the crew had to maintain positive control of the P1 connector during temporary stowage.

The four ESTR-2 fasteners were released, allowing removal from the bay. The original ESTR-2 could not be subject to angular accelerations that exceeded 40 rad/sec/sec at any time. ESTR-2 was translated to the SOPE and stowed.

The Small ORU Protective Enclosure (SOPE) housing the replacement ESTR-2 was disengaged with the lid fully opened. The empty bay interior box was opened and the original ESTR-2 was placed inside the box and stowed in the SOPE. The replacement ESTR-2R interior SOPE box was opened and the ESTR-2R was removed from the SOPE. The ESTR-2R could not be subjected to angular accelerations that exceeded 40 rad/sec/sec at any time. The interior box was closed and stowed inside the SOPE. The SOPE lid was closed and secured with all three latches.

The ESTR-2R was installed on the HST bay mounting structure and the four fasteners were engaged and torqued. The three ESTR-2 connectors were then mated.

Upon completion of the ESTR-2R installation, IV notified MCC GO for ESTR-2R aliveness test. The STOCC performed the aliveness test to verify the integrity of the ESTR-2R connectors.

The bay 8 door was then closed and the six J-hooks engaged and torqued.

8.2.2.5 DIU-2 Changeout

Note: The nominal plan was to install the DIU-2, SSR, and RWA, respectively, during the same EVA.

Prior to DIU-2 changeout operations, the IV crewmember rotated the BAPS to the DIU-2 changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with the B target on FSS.

The changeout began with the EV crewmembers opening the MOPE and retrieving the ORU handle and gender changer cables. The EV crew then translated to HST bay 3.

The EV crewmember opened the HST bay 3 door by releasing the six J-hooks and opening the door to the integral door stay fully open position. HST bays 2 and 3 battery thermal louvers and the WF/PC II radiator were NO TOUCH ZONES and cannot be contacted. The DIU-2 portable handhold is installed on the original DIU-2.

IV calls MCC GO for DIU-2 powerdown. DIU-2 commands were nominally sent by the STOCC. The STOCC also performed a battery isolation test to ensure that battery 4 is isolated from the DIU-2 P3 connectors.

8.2-11

The P3 connector was non-scoop-proof and was electrically energized by the SA. The P1 and P2 connectors were also electrically energized (fused at 3 amps). The pins on these connectors could not be shorted to ground. No structure or tool could contact the pins of the HST P1, P2, or P3 connectors. To minimize the probability of inadvertent shorting, the crew maintained positive control between the connector plug and receptacle during the demating and mating of P1, P2, or P3. The inline gender changer had to be installed immediately after these connectors were demated. The DIU-2 connectors could be demated in any order. After demating the connectors, the DIU was removed from bay 3 and translated to the MOPE.

The MOPE housing the replacement DIU-2 was disengaged and the lid fully opened with the tool board raised. The replacement DIU-2 was then removed and the original DIU-2 placed inside the MOPE with the portable ORU Handhold removed and restowed. The tool board was lowered in the MOPE with the lid closed and secured with all five latches.

The replacement DIU-2 was installed on the HST bay 3 mounting structure and the two center soft dock alignment pins engaged. Then, the 18 DIU connectors were inspected and mated. Following that, the four corner captive fasteners were engaged and torqued. Next, the four connector caps on the gender changers were removed just prior to remating each respective connector and stowed. The cable restraint device was then removed and stowed in the MOPE for return. EV notified IV after the DIU-2 connectors had been remated.

IV notified MCC GO for DIU aliveness test.

The bay 3 door was closed and the six J-hooks engaged and torqued.

8.2.2.6 ESTR-1/SSR Changeout

Prior to SSR changeout operations, the IV crewmember rotated the BAPS to the ESTR-1/SSR changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS.

The changeout began with the EV crew translating to HST bay 5. The HST bay 5 door was opened by releasing the six J-hooks. The door could have been kept in the fully open position by installing the SSM adjustable door stay or by tethering the door. The bay 5 door could not exceed 118°.

IV called MCC GO for ESTR-1 powerdown. ESTR-1 commands were nominally sent by the STOCC.

After deadfacing the ESTR-1, IV gave EV a GO to demate the ESTR-1 P1, P2, and P3 connectors. The P1 connector sockets were electrically energized and could not be contacted. No structure or tool could contact the pins of the P1 connectors. To minimize the probability of inadvertent shorting, the crew maintained positive control of the P1 connector during temporary stowage.

8.2-12

The four ESTR-1 fasteners were released allowing removal from the bay. The ESTR-1 could not be subjected to angular accelerations that exceeded 40 rad/sec/sec at any time. The ESTR-1 was removed and translated to the SOPE.

The SOPE housing the SSR was disengaged with the lid fully opened. The empty bay interior box was opened and the ESTR-1 was placed inside the box and stowed in the SOPE. The SSR interior SOPE box was opened and the SSR was removed from the SOPE. The T-harness was also removed from the SOPE. The SSR could not be subject to angular accelerations that exceed 40 rad/sec/sec at any time. The interior box was closed and stowed inside the SOPE. The SOPE lid was closed and secured with all three latches.

The SSR was installed on the HST bay mounting structure and the four fasteners were engaged and torqued. The four tethered connector caps were removed and the SSR T harness retrieved from the SOPE. The SSR J4 connector cover is removed and stowed. Then, the SSR J4 connector was mated to the T harness P4 connector. The ESTR-3 P2 connector was demated, the T harness J2 connector cap removed, and the ESTR-3 J2 connector mated to T-harness P2. The T-harness J2 connector was then mated to the HST P2. EV notified IV when complete.

Upon completion of the SSR installation, IV notified MCC GO for SSR aliveness test. The STOCC performed the aliveness test to verify the integrity of the SSR/T-harness connectors.

The bay 5 door was then closed and the six J-hooks engaged and torqued.

8.2.2.7 RWA Changeout

The RWA changeout operation is identical for each RWA. For specific hardware details, see Section 2.7.2.1.

The IV crewmember rotated the BAPS to the RWA changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS.

Prior to the beginning of the EVA changeout, the STOCC powered down the RWA allowing the RWA to spin down.

At the +V3 aft shroud worksite, the EVA crewmember translated to bay 6, released the six J-bolts, and opened the bay 6 door.

After a GO was received from the IV crewmember, the P1, P2, P3, and P4 connectors were disconnected, the three keyway bolts disengaged, and the original RWA removed. The RWA was translated to the LOPE where it was exchanged with the replacement RWA (RWA-1R). The RWA-1R was removed from the LOPE by disengaging four J-bolts, disengaging three keyway bolts, and removing the RWA-1R from the LOPE.

The RWA-1R was translated to the +V3 bay 6 worksite where the three keyway bolts were engaged, the connectors remated, following by EV giving IV a GO for the RWA aliveness test.

An aliveness test was performed by the STOCC to check the electrical path to the instrument.

8.2.2.8 SADE-2 Changeout and SADE-1 Screws

Note: The nominal plan was to install SADE-2 and MSS covers, respectively, during the same EVA.

For specific hardware details, see Section 6.2.6.2.

Prior to SADE-2 changeout operations, the IV crewmember rotated the BAPS to the SADE-2 changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with the B target on the FSS.

The SOPE housing the replacement SADE-2 was opened. Then, the appropriate SOPE interior box was opened and the replacement SADE-2R was removed. The SOPE lid was then closed and secured with one latch. The EV crew then translated to the HST bay 7 worksite.

At the bay 7 door worksite, the bay 7 door was opened by releasing the six J-hooks and tethered. The bay 7 door opening angle could not exceed 101°.

EV crewmembers secured the SADE-2R adjacent to the original SADE-2. IV called MCC GO for SADE-2 powerdown. SADE-2 commands were nominally sent from the STOCC. After a GO was received from the IV crewmember, the eight SADE-2 D connectors were individually demated from the SADE-2 by disengaging its two 4-40 screws and fully disengaging the connector. This connector was immediately mated to the SADE-2R extension.

Next, the six SADE-2 noncaptive fasteners (with one semicaptive washer each) were disengaged and stowed on the bolt block. The SADE-2 was then removed from the bay 7 door. Following the removal of the SADE-2, the SADE-2R was installed on the HST bay 7 door and the four captive fasteners were engaged and torqued. Then, the SADE cable restraint device was removed from the SADE-2R and the fastener retention block was transferred from the SADE-2R to the SADE-2. EV notifies IV of SADE-2R installation complete.

IV notified MCC GO for the SADE-2R aliveness test. The STOCC performed the aliveness test after SADE-2R installation to check the electrical path to the instrument.

Next, the EV crew installed T-fasteners on P1 lower, P4 lower, P7 upper, and P8 lower connectors on SADE-1. This replaced the missing clips on the P1 lower and P4 lower connectors.

The bay 7 door was closed by engaging the six J-bolts and the EV crew translated back to the SOPE.

The SOPE lid was reopened and the SADE-2, SADE cable restraint device, and the bolt block were stowed. Finally, the SOPE lid was closed and fully secured with all three latches.

8.2.2.9 MSS Covers

Prior to the operation, IV rotated the BAPS to the MSS cover position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS.

The operation began with the EV crew removing the BAPS post center pins and translating to the COPE and retrieving the MSS MLI covers. IV pivoted the BAPS to the 75° position and calls MCC GO for MSS MLI cover installation. The COPE cover was closed with all five latches engaged.

The EV crew then translated to the MSS 1 worksite and installed the MSS MLI covers. The crew then translated to the MSS 2 worksite and installed the MSS MLI covers. EV notified IV of MLI cover installation complete. IV pivoted the BAPS to the 90° position.

8.2.3 Redeployment

After the HST servicing was completed, the telescope was redeployed. Part of the activities for configuring the HST for deployment occurred on the FSS and part occurred on the RMS.

The high gain antennas were deployed while the HST was berthed to the FSS. The orbiter assumed an attitude to allow the batteries on the HST to charge. After the batteries had reached the required state-of-charge, the RMS grappled the telescope and the crew transferred from orbiter power to HST batteries. The umbilical was demated when the HST was verified to be on internal power. The HST was unberthed from the FSS and maneuvered to the release position.

In the release position the STOCC commanded the HST aperture door open and enabled the pointing and safe-mode electronics assembly. Immediately prior to release, the crew started a macro procedure in the DF-224 that would command the HST to software Sun-point when a 5-minute timer expires. The timer would expire postrelease, and the STOCC would recover the HST and configure it for orbit operations.

9.0 SERVICING MISSION 3A

9.1 INTRODUCTION

9.1.1 Background

The HST was launched on April 24, 1990, during the STS-31 mission onboard the Space Shuttle Discovery and serviced during the STS-61 and STS-82 missions in December 1993 and February 1997, respectively. It was deployed on April 25, 1990, at an altitude of 337 n. mi., with a 28.5° inclination. The HST currently sits at a 324 n. mi. altitude with a 28.5° inclination. HST anomaly history since the deployment mission is documented in Appendix D.

The HST has been designed for on-orbit servicing and refurbishment to meet, and quite possibly exceed, the anticipated 15-year lifetime of the spacecraft.

Since SM-2 HST has had several failures in the PCS, which have put the telescope in a configuration that could jeopardize the primary science mission, STS-103 was designed as a quick response mission to replace the failed components, and also to install some hardware upgrades that were already flight ready.

9.1.2 SI/ORU Replacement Identification

The following is a list of the replacement units for the third servicing mission with additional HST servicing components, tools, and aids:

- a. Planned replacement components and tasks (by priority)
 - 1. RSU-1, 2, 3
 - 2. VIK
 - 3. 486 computer (DF-224)
 - 4. FGS (FGS-2)
 - 5. SSAT (SSAT-2)
 - 6. SSR (ESTR-3)
 - 7. NOBL
 - 8. Open NICMOS valves
 - 9. SSRF
- b. Contingency components
 - 1. Aft shroud latch repair kits
 - 2. Electronics Control Unit (ECU)
 - 3. PCU essential bus harness

- 4. DMU to 486 harnesses
- 5. OCE-EK spare cable
- 6. SSR "J" and "T" harnesses
- 7. CT bus plug
- 8. CDU cable harness
- 9. PDU fuse plugs
- 10. OTA fuse modules
- 11. SA bistem braces
- 12. SA thermal shield repair devices
- 13. MLI patches

For a list of EVA tools and crew aids, see the HST SSE CSM.

9.1.3 STS-103 Payload Configuration

The payload complement for the third servicing mission included the MSFC ORUC with a mission-specific complement of ORUs and ORIs, and the GSFC FSS.

For full details of the aforementioned HST SSE and payload bay configuration, refer to the HST SSE CSM.

Figure 9.1-1 shows the payload bay configuration. The HST centerline coordinates with respect to the orbiter are at Xo = 1149.20 and Yo = 0.0 while berthed to the FSS. The face of the aft shroud is located at Zo = 496.594.

The HST interfaces to the orbiter payload primary power bus and SSP via the IPCU on the FSS and the remotely mateable/demateable umbilical on the BAPS. The IPCU receives regulated orbiter dc power from two Power Distribution and Switching Units (PDSUs) that feed six DPCs each. The outputs of the IPCU are routed directly to the HST through the FSS maintenance umbilical (J101) nominally or through the deploy/return umbilicals (J107 and J108) in a contingency where the maintenance umbilical cannot be mated.



Figure 9.1-1. Payload bay configuration

9.1.4 Mission Overview

For the third servicing mission, the orbiter launched on a direct insertion trajectory to the HST orbit, estimated to be approximately 324 n. mi., with a 28.45° inclination. Once the spacecraft had been properly configured, the orbiter rendezvoused with the HST, grappled and berthed the HST to the FSS, connected external orbiter power to the HST via umbilical, and serviced the spacecraft accordingly.

STS-103 carried a crew of seven for a 10-day mission, with 4 scheduled and 1 unscheduled EVA days to accomplish mission objectives. The SSE was activated on FD1. On FD2 the orbiter RMS was activated and checked out. The orbiter rendezvoused, grappled, and berthed the HST on FD3. FD4 through FD6 was used for performing three EVAs of 6.0 hours each for HST servicing. An overview of the EVA can be seen in Figure 9.1-2. HST deployment activities took place on FD7, with FD8 reserved as an unscheduled EVA/deploy day. The mission also allows for 2 additional on-orbit days for the shuttle to protect against weather conditions or shuttle contingency delays before landing.

Two EVA teams, consisting of two crewmembers each, provided alternating support for each EVA day. One team performed EVAs on FD4 and FD6, and the other team performed an EVA on FD5. FD7 was reserved in case an unscheduled EVA was required to properly configure the HST before deployment. The EVA teams were cross-trained for all servicing activities to accommodate off-nominal situations requiring shuffling of the EVA teams and crewmembers.

The STS-103 mission replaced all three RSUs, which should ensure the reliability of the PCS to support science observations. An SSAT that failed after SM-2 was replaced with a refurbished unit to restore science downlink redundancy. Hardware upgrades included replacing FGS-2 with a refurbished unit, the DF-224 with a 486 computer, and ESTR-3 with an SSR. These upgrades should dramatically increase the science/engineering data storage and processing capability.



° Includes BAPS Post installation

asd = aft shroud door latch contingency

mfr = crew change positions in manipulator foot restraint

hga = high gain antenna deployment

RM August 12, 1999

Figure 9.1-2. Proposed HST SM-3A EVA scenario

9.1-5 Verify that this is the correct version before use

9.2 MISSION OPERATIONS

9.2.1 Rendezvous and Capture

Rendezvous and Proximity Operations leading to capture of the HST on FD3 would have occurred at a time based on HST-orbiter phasing.

Primary prerendezvous activities included checkout of the RMS, EMUs, the orbiter Ku-band antenna, the PGSC (which was used to enhance HST PSP bypass communications), the HST SSE, and the ground command system. SSE activation procedures took place within 2 hours of PLBD opening. This included the activation of the FSS and ORUC heaters, as well as the FMDM and use of the orbiter MDMs for monitoring of the thermal telemetry.

Because of another RGA failure prior to launch, HST had been placed in the Zero Gyro Sun-Point (ZGSP) mode. The telescope was placed into Hardware Sun-Point (HWSP) approximately 8 hours prior to the Ti burn and stayed in that mode until RMS grapple. Prior to initiating HWSP, both sides of the PSEA were configured and the RMGA was powered on. Refer to Section 2.9.2.2 for a description of the HST configuration under HWSP. The DF-224 was brought up to process 32K telemetry, which provided data to the crew. Even though the DF-224 was powered on, the PSEA was in control of the vehicle.

The crew followed a nominal rendezvous timeline, except that at approximately 150 feet from HST, a flyaround was performed to line up the grapple fixture with the RMS end effector because HST was in HWSP. The SPA trim relays were all commanded open to minimize the effects of solar array shadowing from the orbiter. Once grappled, HST was commanded to drift mode by resetting both sides of the PSEA. The nominal drift mode command would not have worked because the HST is under PSEA control. This required a patch to the flight software to provide the command to the crew. The PSEA was then reconfigured and the RMGA was powered down, and the crew returned to a nominal mission timeline.

HST-orbiter communications were reconfigured from the aft LGA to the forward LGA to allow for communications during berthed operations. The HST was berthed to the FSS using the orbiter-mounted CCTV system and the FSS top mounted CCTV with a berthing target on the aft end of the HST. The three HST berthing pins were engaged by three corresponding FSS berthing latches as the HST was secured to the FSS BAPS. The remotely operated umbilical on the FSS was then connected, utilizing the orbiter MCDS and the A6U panel, and external power was applied to the HST via the orbiter SSP. The orbiter was then maneuvered to the battery charge attitude to top off the batteries.

The onboard GPCs and the ground received telemetry from the forward HST LGA through the orbiter PI and PDI via the PSP-bypass mode. Ku-band bent-pipe telemetry was utilized as a backup in the event that the PDI telemetry was lost. Ku-band bent-pipe telemetry can be displayed only at the STOCC.

The SSM and OTA heaters were activated and the SAs remained deployed throughout the mission. At the conclusion of rendezvous day activities, the FGEs were turned back on, and the SIC&DH, WF/PC, STIS, NICMOS, and FOC were placed in warmup mode.

The orbiter supported HST telemetry at 32 kbps via the PDI (500 kbps, 4 kbps, and 32 kbps when augmented by the HST PGSC) and orbiter operational downlink. The HST was commanded through the PSP/PI. Ku-band bent pipe service for relaying HST telemetry (0.5, 4, 32, or 1024 kbps) to the ground was supported as needed whenever Ku-band TDRS coverage is available.

9.2.2 EVA Operations

This section describes the EVA operations performed to accomplish the mission objectives.

All commanding activity was done from the STOCC unless otherwise identified. The IV crew had the capability to send certain commands from the onboard GPC SPEC and to monitor the status of the certain HST systems during EVA operations and the rest of the mission.

9.2.2.1 RSU-1, 2, 3 Changeout

Note: The nominal plan was to install RSUs 1, 2, 3, open the NICMOS cooling valves, and install the VIK during the same EVA.

The RSU changeout operation was identical for each RSU. For specific hardware details and latching information, see Section 2.7.1.5.

The IV crewmember rotated the BAPS to the RSU changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (180° for all of the RSUs).

Prior to the beginning of the EVA changeout, the FHST shutters were verified closed from SSP tb DS15, and the RSU was deactivated by commanding OFF the RSU control and survival heaters as well as the rate sensor and support electronics. RSU commands were nominally sent from the DIU and DMU from the STOCC, although the crew had the capability to issue these commands from the spec, if necessary, or the SSP for power to the RSU survival heaters.

RSU-2R was retrieved from the COPE, stowed in an ORU transfer bag, and maneuvered to the -V3 aft shroud. At the -V3 aft shroud worksite, the FHST 2 and 3 shade seals were retracted, the latches on the -V3 aft shroud doors and the handle latch were disengaged, and the doors were opened and tethered at the padded stop position. At least 5 minutes prior to the -V3 doors opening, all three RSUs were powered off.
After a GO was received from the IV crewmember, the two wingtabbed RSU power connectors were disconnected, the three captive blots disengaged, and the original RSU removed. The replacement RSU was tilted slightly for easier installation. Then, the three captive bolts were reengaged, and the two power connectors reconnected. The original RSU was maneuvered to the COPE and stowed for landing.

After RSU-2R installation was complete, EV notified MCC GO for RSU-2R aliveness test. The STOCC then performed an aliveness test to verify the integrity of the RSU-2R connections.

This sequence was repeated for RSU-3R and then for RSU-1R. Once all three RSUs were installed, the NICMOS COOLANT IN and OUT VALVES were opened 2.5 turns. The -V3 doors were closed with the changeout procedure, concluding with reengaging the FHST 2 and 3 shade seals.

9.2.2.2 Voltage/Temperature Improvement Kit Installation

Note: The nominal plan was to install RSUs 1, 2, 3, open the NICMOS cooling valves, and install the VIK during the same EVA.

The installation began by retrieving the VIK transport caddy from the airlock and maneuvering to SSM bay 2.

EV translated to the SSM bay 3, disengaged all the J-hooks on the bay 3 door but one, and then maneuvered to SSM bay 2. The J-hooks on the bay 2 door were disengaged, and the door was opened to the fully open integral door stay position. The manual door stay was then engaged.

IV verified with MCC GO for battery connector demate.

Once the battery connectors were deadfaced, IV gave EV a GO for connector demate, and the VIK gender changer connector was demated from the VIK J2 connector. The battery harness P2X (X being A, B, or C) was demated and connected to the VIK J2X connector. Next, the VIK pre-integrated gender changer GCP2 connector was mated to the battery connector J2X. The VIK was secured to the battery P1X harness using Velcro.

This procedure was repeated for the installation of the second and third VIKs. Once all three VIKs were installed in bay 2, the bay 2 door was closed and secured by engaging and torquing the J-hook bolts.

Next the SSM bay 3 door was opened and the above procedure was performed to install all three bay 3 VIKs.

After all the VIK installations were complete, EV notified MCC GO for VIK aliveness test. The STOCC performed an aliveness test to verify the integrity of the VIK connections.

9.2.2.3 DF-224/486 Computer Changeout

Note: The nominal plan was to install the 486 computer and FGS-2R, respectively, during the same EVA.

Prior to DF-224 changeout operations, the IV crewmember rotated the BAPS to the DF-224/486 changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (90° or -V2 fwd).

The changeout began by retrieving the 486 computer Y-harness from the LOPE lid and the Y-harness scoop-proof P10 connector was mated to the 486 J10 receptacle. Once mated, the Y-harness was restrained to the top of the 486 using Velcro.

The EV crew translated to the SSM bay 1 and opened the door by releasing the six J-hooks. The door must be kept to the fully open position of 94° to avoid potential EMU backpack contact.

IV verified MCC GO for DF-224 connector demate. DF-224 powerdown commands could only be sent by the STOCC. After the STOCC deadfaced the DF-224 connectors, IV gave EV a GO to demate the nine HST connectors. Next, the six DF-224 J-hooks were disengaged and the DF-224/Coprocessor was removed from the bay 1 mounting bracket.

Once the DF-224/Coprocessor had been removed, the seven connector converters were mated to their respective HST harness connectors. The connector converters were required because scoop proof connectors had been incorporated into the 486 computer to reduce risk to mission success.

Next, the LOPE lid was opened by releasing the four J-hooks, and the 486 computer was removed from the LOPE.

The DF-224/Coprocessor was then stowed in the LOPE and the six J-hooks were engaged. The LOPE lid was closed and secured for landing by engaging the four lid J-hooks and torquing the J-bolts.

The 486 was fully seated in bay 1 and secured by engaging the six J-hooks, then seating and torquing the J-hook bolts. Once secured, each of the seven left side and two top Y-harness connectors were mated to the HST harnesses.

Immediately upon completion of the 486 computer connector matings, EV notified IV GO for 486 computer aliveness test. The STOCC performed the aliveness test to verify the integrity of the 486 connections.

After a successful aliveness test, the bay 1 door was closed and secured by engaging the six J-hooks.

Once the lower powered 486 computer had been installed, an SSM bay 1 NOBL was installed to reduce the heat rejection of bay 1. The bay 1 NOBL and the NOBL plugs were retrieved from the NOBL Protective Enclosure (NPE) and the NOBL was placed over the existing FOSR on bay 1. The four NOBL plugs were installed in the four bay 1 vent holes and engaged by pivoting the handle down 90°, and then the NOBL grounding cap was placed over the J-hook bolt.

9.2.2.4 FGS-2 Changeout

Note: The nominal plan was to install the 486 computer and FGS-2R, respectively, during the same EVA.

For more detailed information concerning latching, go to the generic radial SI changeout operation in Section 3.4.1 of this document.

Before the changeout, the IV crewmember rotated the BAPS to the FGS-2 changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (360° or the +V3 fwd).

IV activated S24 SSM WK LIGHTS to activate the FGS latch indicator lights and the HST electrical receptacles.

The changeout began by deploying the aft fixture. The +V3 FGS doors were opened by disengaging the two latch bolts and releasing the handle.

Then IV verified MCC was GO for FGS-2 connector demate. The STOCC performed the powerdown. The crew had the capability to issue the poweroff commands, if necessary. After receiving verification that the FGS blind-mate connector had been deadfaced, four left wingtab connectors and groundstrap were demated and tethered back to the left tether loop. Next, the four right wingtab connectors and groundstrap wingtab connectors were demated and tethered back to the right tether loop.

Four FGS guide studs were installed on the FGS. The FGS portable handhold was then retrieved from the forward fixture and installed onto the FGS guide studs via the four zipnuts with the spacers and MLI A-latch cutout on the FGS handhold at the bottom (-V1).

The FGS A-latch was then disengaged with the A LATCHED indicator lights illuminating.

The FGS was pulled out of the FGS cavity using the portable handhold and the guiderail system and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with the non-RMS crewmember providing guidance support. This allowed for any repositioning necessary to ensure a level extraction of the FGS. The FGS may have needed to be pitched up (+V1) to avoid A latch contact with the coplaner D-latch fitting. The FGS can only be touched at the portable handhold, ground support equipment brackets, and FGS guiderails.

FGS was translated to and temporarily stowed on the aft fixture.

The FSIPE door was opened with the groundstrap disconnected and stowed. The FGS-2R portable handhold was removed from the forward fixture and installed onto the FGS-2R via four zipnuts with the MLI cutout on the FGS-2R handhold at the bottom. The FGS A-latch in the FSIPE was then disengaged.

The FSIPE groundstrap was disconnected and stowed in the FSIPE groundstrap stowage receptacle. The FGS A-latch in the FSIPE was then disengaged 1/2 revolution to break the seating torque followed by the remaining 22+ revolutions (until the A-latch was fully disengaged). The A-latch indicators were verified activated after the first 2.5 revolutions CCW of the A-latch, indicating a not fully latched condition. The A-latch indicators were verified with the A LATCHED lights illuminated.

The FGS-2R was removed from the FSIPE guiderails. The FGS-2R may have needed to be pitched up (orbiter forward) to avoid A-latch contact with the coplanar D-latch fitting. During removal, the FGS-2R exterior adsorber module mirror, just above the pickoff mirror, could not be touched because of sharp edge and contamination concerns. The FSIPE door was then temporarily closed and one of the latches latched. Next, the FSIPE lid on-orbit latches were verified armed. The FGS-2R pickoff mirror cover was then removed from FGS-2R by releasing the locking device, moving the handle downward, and pivoting the mirror cover away from the FGS-2R mirror. Care was taken to avoid contacting the pickoff mirror and the absorbers near the pickoff mirror to prevent possible damage to the instrument. Once removed, the mirror cover was stowed on the ORUC stowage bracket.

The FGS-2R was nominally installed into the HST FGS-2R cavity using a coordinated EV/IV effort with an EV crewmember on the RMS and another EV crewmember providing guidance. This allowed for any repositioning necessary to ensure a level insertion of FGS-2R. However, the FGS-2R may have needed to be pitched up (+V1) to avoid A-latch contact with the coplanar D-latch fitting. The FGS-2R was inserted into the cavity and repositioned approximately to engage the guiderails. It was then fully inserted into its seated position. The A-latch was engaged and torqued and the A LATCHED indicator lights were extinguished. The four FGS-2R handhold zipnuts and the handhold on the new FGS were removed, leaving only the four remaining FGS-2R guide studs.

Next, the EV crewmember mated the FGS-2R groundstrap wingtab connector to the replacement FGS-2R. The eight wingtab connectors were mated to the FGS-2R.

After completing the FGS-2R installation, IV gave MCC a GO for FGS-2R aliveness test. The STOCC performed the aliveness test to verify electrical integrity of the instrument connectors.

With both doors fully closed, the handle latch and two door latches were reinstalled and torqued. Then, the three FGS-2R door latches were retorqued. Next, the FGS-2R doors were visually inspected for closure including the resilient door light seal.

The FSIPE door was opened, and the FGS was removed from the aft fixture and inserted into the FSIPE guiderails. The FGS may have needed to be pitched up (orbiter

forward) to avoid A-latch contact with the coplanar D-latch fitting. The FGS was inserted into the FSIPE and repositioned appropriately to engage the guiderails. It was then fully inserted into its seated position. The A-latch was engaged and torqued with the FGS A LATCHED indicator lights extinguishing.

The FSIPE groundstrap wingtab connector was mated to the FGS and the four FGS handhold zipnuts and the handhold on the FGS were removed, leaving only the four guide studs on the FGS. The handhold was installed on the forward fixture.

After the worksite was clear, the FSIPE lid was closed and all three lid latches latched and verified. The aft fixture was also stowed.

Because of EVA timeline constraints, the OCE-EK task was performed on the following day during EVA3. To perform the OCE EK connection, the HST bay C door was opened by releasing the three J-hooks and opening the door. The door was secured via either the OTA door stay or tethering. The bay C door opening angle cannot exceed 118°.

Next, IV verified MCC GO for OCE connector demate. The STOCC performed the powerdown. The crew had the capability to issue the poweroff commands, if necessary. After receiving a GO from IV, the EV crewmember demated OCE connector P11.

Once OCE-EK P11A/J11A was connected, EV informed IV GO for OCE aliveness test. When the aliveness test was complete, the bay C door was closed and latched.

9.2.2.5 SSA Transmitter Changeout

Note: The nominal plan was to install OCE-EK, SSAT-2R, SSR-3, and bay 5-10 NOBL, respectively, during the same EVA.

Prior to SSAT changeout operations, the IV crewmember rotated the BAPS to the SSAT changeout position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (270° for the SSAT).

The changeout began with the EV crew translating to HST bay 5. The HST bay 5 door was opened by releasing the six J-hooks. The door was kept in the fully open position by installing the SSM adjustable door stay or by tethering the door.

IV verified MCC GO for SSAT-2 connector demate. SSAT-2 should have been powered off and all three safety inhibits put in place during the rendezvous timeframe, and this was verified prior to demating the SSAT-2 connectors.

After deadfacing the SSAT, IV gave EV a GO to demate the two circular connectors and the three coax connectors from the SSAT-2.

Next, the Fastener Retention Board (FRB) was attached to the SSAT to provide a place to stow the noncaptive bolts and washers. The SSAT-2 was removed from the HST by disengaging the eight noncaptive bolts and washers.

Once the SSAT was disengaged from the HST, the COPE lid was opened and the SSAT-2R was retrieved from the COPE. The SSAT-2 was then stowed in the COPE and the COPE lid was closed.

SSAT-2R was installed on the bay 5 door mounting position and secured by engaging and torquing all seven captive bolts. The SSAT-2 coax harness P3, P4, and P5 connector extensions were fully seated and mated to the SSAT-2R receptacles J3, J4, and J5. Next, SSAT-2 harness P1 and P2 connectors were mated to SSAT-2R receptacles J1 and J2.

Upon completion of the SSAT-2R connector mate, EV notified MCC GO for SSAT-2R aliveness test. The STOCC performed the aliveness test to verify the integrity of the SSAT-2R connections.

9.2.2.6 ESTR-3/SSR-3 Changeout

Note: The nominal plan was to install OCE-EK, SSAT-2R, SSR-3, and bay 5-10 NOBL, respectively, during the same EVA.

Proper BAPS position was verified by lining up the markings on the HST with those on the FSS BAPS (270°).

The changeout began with the EV crew translating to HST bay 5. The HST bay 5 door was opened by releasing the six J-hooks. The door was kept in the fully open position by installing the SSM adjustable door stay or by tethering the door.

IV verified MCC GO for SSR-1 and ESTR-3 connector demate. SSR-1/ESTR-3 commands were sent by the STOCC. The crew had the capability to issue the SSR-1/ESTR-3 poweroff commands, if necessary.

After deadfacing SSR-1/ESTR-3, IV gave EV a GO to demate the ESTR-3 P1, P2, and P3 connectors. The P1 connector sockets were electrically energized and could not be contacted. No structure or tool should contact the pins of the P1 connectors. To minimize the probability of inadvertent shorting, the crew maintained positive control of the P1 connector during temporary stowage.

The four ESTR-3 fasteners were released, allowing removal from the bay. The ESTR-3 could not be subject to angular accelerations that exceed 40 rad/sec/sec at any time. The ESTR-3 was removed and translated to the COPE.

The COPE was opened and SSR-3 was removed. Next, ESTR-3 and the T-harness was stowed in the COPE and on the COPE lid, respectively. The J-harness was removed from the COPE lid, and the COPE was closed and secured with all three latches. EV then translated back to SSM bay 5 with SSR-3 and the J-harness.

The SSR was installed on the HST bay mounting structure and the four fasteners were engaged and torqued. The four tethered connector caps were removed and stowed. SSR J1, J2, and J3 connectors were then mated to the HST harness. The SSR-1 J4

connector was mated to the J-harness P4 connector, and the SSR-3 J4 connector was mated to the other J-harness P4 connector.

Upon completion of the SSR installation, IV notified MCC GO for SSR aliveness test. The STOCC performed the aliveness test to verify the integrity of the SSR/T-harness connectors.

The bay 5 door was closed and the six J-hooks engaged and torqued.

9.2.2.7 NOBL Installation

Note: The nominal plan was to install OCE-EK, SSAT-2R, SSR-3, and bay 5-10 NOBL, respectively, during the same EVA.

Because the EVA timeline for EVA3 was long, the EV crew was able to install NOBL only for SSM bays 9 and 10. In addition, a NOBL was installed on SSM bay 1 as part of the HST-486 task, to provide temperature control to that bay rather than to effect MLI repair.

The bay 5 and 10 NOBLs and a set of four NOBL plugs were retrieved from the NPE and translated to the SSM bay worksite. The bay 9 NOBL was installed over the MLI by attaching the doorstop and doorknob clips to their respective interfaces. The bay 10 NOBL task began by first removing the two MLI (original) blankets. The NOBL was then placed in position and secured with a vent plug in each door vent hole.

This procedure was repeated for SSM bay 10.

9.2.3 Redeployment

After the HST servicing was completed, the telescope was redeployed. Part of the activities for configuring the HST for deployment occurred on the FSS and part occurred on the RMS.

The high gain antennas were deployed while the HST was berthed to the FSS. The orbiter maneuvered to the battery-charging attitude. After the batteries had reached the required state-of-charge, the RMS grappled the telescope and the crew transferred from orbiter power to HST batteries. The umbilical was demated when the HST was verified on internal power. The HST was unberthed from the FSS and maneuvered to the release position.

In the release position, the STOCC commanded the HST aperture door open and enabled the pointing and safemode electronics assembly. Immediately prior to release, the crew will start a macro procedure in the 486 that commanded the HST to software Sunpoint when a 5-minute timer expired. The timer expired postrelease, and the STOCC recovered the HST and configured it for orbit operations.

10.0 SERVICING MISSION 3B

10.1 INTRODUCTION

10.1.1 Background

The HST was launched on April 24, 1990, during the STS-31 mission onboard the Space Shuttle Discovery and serviced during the STS-61, STS-82, and STS-103 missions in December 1993, February 1997, and December 1999, respectively. It was deployed on April 25, 1990, at an altitude of 337 n. mi. with a 28.5° inclination. *At SM3B launch, t*he HST *was located* was located at approximately 324 n. mi. altitude with a 28.5° inclination.

The HST was designed for on-orbit servicing and refurbishment to meet the anticipated 15-year lifetime of the spacecraft.

HST anomaly history since deployment is documented in Appendix D of this CSM.

10.1.2 SI/ORU Replacement Identification

The following list provides the replacement units for the third servicing mission with additional HST servicing components, tools, and aids:

- a. Planned replacement components and tasks (by priority)
 - 1. Advanced Camera for Surveys (ACS)
 - 2. Power Control Unit (PCU)
 - 3. +V2 SA3/Diode Box Assembly 2
 - 4. -V2 SA3/Diode Box Assembly 2
 - 5. NCS (NICMOS Cryocooler (NCC), Electronic Support Module (ESM), and Radiator)
 - 6. NOBL (bays 5 and 6) (only the NOBL for bay 6 was installed on 3B)
 - 7. NOBL (bays 7 and 8)
- b. Contingency components
 - 1. Power Distribution Unit (PDU) fuse plugs R&R (FP 15(2), and 16(2) and 17(2))
 - 2. OTA fuse module R&R (OTA P25, P26, P27A, and P28A)
 - 3. PCU essential bus harness installation (old PCU)
 - 4. Rate sensing unit R&R

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- 5. Electronics control unit R&R
- 6. Aft shroud latch repair mechanism installation (latch standard (6) and handle (2))
- 7. Solar Array II Bistem brace installation and thermal cover repair (4)
- 8. Spare SADA clamps (2)
- 9. V/T signal jumper plug (1)
- 10. MLI repair kit patches
- 11. VIK jumper plugs (6)

For a list of EVA tools and crew aids, see the EVA Support Equipment List (ESEL).

10.1.3 STS-109 Payload Configuration

The payload complement for the fourth servicing mission (SM3B) included the RAC, SAC, FSS, and MULE, with a mission-specific complement of ORUs and ORIs.

For full details of the payload bay configuration, refer to the *Cargo Systems Manual* (*CSM*) Hubble Space Telescope Space Support Equipment Servicing Mission 3B (USA001948).

Figure 10.1-1 shows the payload bay configuration. The HST centerline coordinates with respect to the orbiter were at Xo = 1149.20 and Yo = 0.0 while berthed to the FSS. The face of the aft shroud was located at Zo = 496.594.

The HST interfaced to the orbiter payload primary power bus and SSP via the IPCU on the FSS and the remotely mateable/demateable umbilical on the BAPS. The IPCU received regulated orbiter dc power from two FSS Enhanced Power Distribution and Switching Units (EPDSUs) that feed six DPCs each. The outputs of the IPCU were routed directly to the HST through the FSS main umbilical (J101) nominally or through the deploy/return umbilicals (J107 and J108) in a contingency where the maintenance umbilical cannot be mated.



Figure 10.1-1. Payload bay launch configuration

10.1.4 Mission Overview

For the 3B servicing mission, the orbiter *was* launched on a direct insertion trajectory to the HST orbit, approximately 324 n. mi., with a 28.45° inclination. Once the spacecraft had been properly configured, the orbiter *rendezvoused* with the HST, grappled and berthed the HST to the FSS, connected external orbiter power to the HST via umbilical, and serviced the spacecraft accordingly.

STS-109 carried a crew of seven for an 11-day mission, with 5 scheduled and 1 unscheduled EVA days to accomplish mission objectives (only 5 were performed). On FD1, the SSE was activated and during FD2 the SEE and orbiter RMS was checked out. The orbiter rendezvoused, grappled, and berthed the HST on FD3. FD4 through FD8 were used to perform five EVAs for HST servicing. An overview of the EVA sequence can be seen in Figure 10.1-2. HST deployment activities took place on FD9, with FD10 reserved as an unscheduled EVA/deploy day. The mission also allowed for 2 additional on-orbit days for the shuttle to protect against weather conditions or shuttle contingency delays before landing.

Two EVA teams, consisting of two crewmembers each, provide*d* alternating support for each EVA day. One team performs EVAs on FD4, FD6, and FD8, and the other team performs EVAs on FD5 and FD7. Nominal deployment *took* place on FD9. The EVA teams *were* crosstrained for all servicing activities to accommodate off-nominal situations requiring shuffling of the EVA teams and crewmembers.

The objectives of the STS-109/SM-3B mission *were* to replace the FOC with the ACS, remove and replace the PCU, replace the second generations SAs with third generation rigid SAs, install the NICMOS cooling system, and install NOBL on bays 5 through 8.

Note: Only the NOBL for bay 6 was actually installed on SM-3B due to timeline constraints.



Figure 10.1-2. Proposed HST SM-3B EVA scenario

10.2 MISSION OPERATIONS

10.2.1 Rendezvous and Capture

Rendezvous and Proximity Operations *lead* to capture of the HST on FD3. Primary prerendezvous activities included a checkout of the RMS, EMUs, the orbiter Ku-band antenna, the PGSC (which *was* used to enhance HST PSP bypass communications), the HST SSE, and the ground command system. SSE activation procedures *took* place within 2 hours of PLBD opening. The procedures included the activation of the FSS and Multi-Use Lightweight Equipment (MULE) FMDMs, as well as the heaters for the MULE, FSS, and SAC. The orbiter MDMs *were* used for monitoring SAC thermal telemetry.

HST activities prior to rendezvous included dumping of the onboard recorders, configuration of the 486 flight software, closure of the AD, configuration of the communications system to LGA-direct-TDRS, HGA deactivation and retraction, and FHST and FGS deactivation.

The STOCC configured the HST for rendezvous operations with the orbiter prior to and during the terminal phase of the rendezvous with all operations except load shedding, roll-to-grapple attitude, and PSP-bypass communications configuration performed prior to the orbiter TI burn. The STOCC maneuvered the HST to the rendezvous attitude. This attitude *was* designed to maintain HST power position and minimize plume loading on the SAs at the 1500-foot normal Z braking gate. The rendezvous attitude is determined by these conditions. The Sun in the V1-V3 plane and the +V1 axis pointing down the R-bar at an orbit angle of 59° before orbital noon.

Following the orbiter TI burn, the HST initiate*d* a nonessential load reduction in order to minimize the power-negative effects of the roll-to-capture attitude. S-band communications between the orbiter and HST *were* established via the orbiter payload communications system in the PSP-bypass mode augmented with the HST PGSC. Contingency HST direct-TDRSS communications links remain*ed* available throughout the rendezvous and capture. The Ku-band was utilized for radar during rendezvous activities.

Prior to grapple, HST performed a roll maneuver to the capture attitude to align the -V3/+V2 grapple fixture on HST with the orbiter RMS. The roll maneuver placed the -V3 52° out of plane north as measured on the anti-Sun side of HST. The magnitude of the roll maneuver about V1= β -52°, resulting in a roll that ranges from 0° to -104°. HST EPS and SA safe-mode tests *were* disabled. Shortly after the roll *was* completed, the PCS safe-mode tests *were* disabled. At that time a 165-minute backup sunpoint timer *was* activated to protect the HST in the event of a rendezvous breakoff. The PSEA keep-alive monitoring *was* also disabled and the SADEs turned off.

As shown in Figure 10.2-1 the orbiter approached the HST using the R-bar approach technique that minimized propellant usage and potential contamination. The capability existed to perform proximity operations, grapple, and berthing with all of the appendages deployed. However, the HGAs *would* have had to be retracted to maximize clearance for RMS grapple. The flight crew nominally grappled the HST via the -V2/-V3 grapple fixture, and deactivated the PCS by commanding the HST into drift mode.

HST-orbiter communications *were* reconfigured from the aft LGA to the forward LGA to allow for communications during berthed operations. The HST *was* berthed to the FSS using the orbiter-mounted CCTV system and the FSS top mounted CCTV with a berthing target on the aft end of the HST. The three HST berthing pins *were* engaged by three corresponding FSS berthing latches as the HST *was* secured to the FSS BAPS. The remotely operated umbilical on the FSS *was* then connected, utilizing the orbiter MEDS and the A6U panel, and external power *was* applied to the HST via the orbiter SSP. The orbiter *was* then maneuvered to the battery charge attitude to top off the HST batteries.

The orbiter GPCs and the ground received telemetry from the forward HST LGA through the orbiter PI and PDI via the PSP-bypass mode. Ku-band bent-pipe telemetry was utilized as a backup in the event that the PDI telemetry was lost. Ku-band bent-pipe telemetry can be displayed only at the STOCC.

The HST SSM and OTA heaters *were* activated. At the conclusion of rendezvous day activities, the FGEs *was* turned back on and the SIC&DH, WF/PC 2, STIS, NICMOS, and FOC *were* placed in warmup mode.

The orbiter supported HST telemetry at 32 kbps via the PDI (500 kbps, 4 kbps, and 32 kbps when augmented by the HST PGSC) and orbiter operational downlink. The HST *was* commanded through the PSP/PI. Ku-band bent pipe service for relaying HST telemetry (0.5, 4, 32, or 1024 kbps) to the ground *was* supported as needed whenever Ku-band TDRS coverage is available.



(a) R-bar approach and inertial grapple

Figure 10.2-1. HST R-bar approach

10.2-3



Figure 10.2-1. HST R-bar approach (concluded)

10.2.2 EVA Operations

This section describes the EVA operations necessary to accomplish the mission objectives. Detailed steps were provided in the STS-109 EVA checklist.

All commanding activity was performed from the STOCC unless otherwise identified. The IV crew had the capability to send certain commands from the onboard GPC SPEC displays and to monitor the status of the certain HST systems during EVA operations and the rest of the mission.

All EVA activities are presented with the assumption that the EV crew has already performed the payload bay setup procedures appropriate to the specific task.

10.2.2.1 SA3 Installation

Refer to Section 6.0 for details on SA2/DBA removal and SA3/DBA-2 installation.

10.2.2.2 PCU Replacement

Before HST was powered down, several thermal covers were installed to prevent HST ORUs from reaching their nonoperating thermal limits during the course of the PCU changeout. Also, a Fixed Head Star Tracker (FHST) cover was installed over the FHST apertures to prevent stray light from entering the FHSTs while HST was powered down. The FHST shutters automatically open when power is removed from HST. The covers and the FHST cover modification were installed on an EVA prior to PCU changeout as a get-ahead, but were not deployed until just prior to HST powerdown. This ensured that HST can survive a rapid deploy contingency. Other activities that are required prior to PCU changeout are battery power connector demates via EVA and SA3 power isolation, which is performed from the ground. These activities are required to deadface several of the PCU connectors as a safety measure.

Once the thermal covers and the FHST cover modification are deployed, the next task is to demate the HST Battery power connectors. The J-hooks on the SSM bay 2 door are disengaged, the door is opened fully, and the doorstay is engaged. IV confirms with the MCC that the vehicle is configured for battery connector demate, and each one of the three battery P1A connectors are demated and battery caps are installed. The bay 2 doorstay is then disengaged, the door is closed, and one J-hook is engaged. This process is repeated for SSM bay 3 to isolate the remaining three batteries. Following battery demate, IV will complete powerdown of HST using the Standard Switch Panel (SSP).

Next, the J-hooks on the SSM bay 4 door are disengaged, the door is opened, and the doorstay is engaged. The door must not be opened any further than 110° to prevent the large wiring bundles running across the bay 4 hinge from becoming overstressed. The PDU fuse plugs for the two inboard PDUs (PDU-2 and PDU-4) are removed and stowed in the PDU caddy. Caps are substituted in their place, which will provide better EVA access to the PCU connectors. The Harness Retention Device (HRD) is installed in bay

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4, which will be used to hold the PCU cables out of the envelope for installing the new PCU.

The Single Point Ground (SPG) lug will be disconnected and restrained out of the PCU translation path. The PCU transport handhold is soft-docked onto the PCU, the 10 keyhole bolts are released, and the PCU is removed from HST. At the same time, the other EV crewmember retrieves the new PCU-R from the SAC by releasing the six keyhole bolts and removing it from the ORU Protective Adapter (OPA). The old PCU is installed on the SAC by engaging the six keyhole bolts, and the thermal cover is reinstalled. Next the new PCU-R is mounted on HST by engaging the 10 keyhole bolts.

Once the SPG lug is engaged, the EV crewmember begins inspecting and mating the 36 PCU connectors. When all 36 connectors have been successfully mated, the PDU fuse plugs may be reinstalled onto the inboard PDUs. The HRD is removed from bay 4, the doorstay disengaged, and the bay 4 door is closed and secured.

Upon completion of the PCU-R installation, IV crewmembers and MCC begin the PCU-R aliveness test. This test is an integration of switch throws and ground commanding to check the electrical path to the new PCU-R.

Upon completion of aliveness, the bay 2 and bay 3 battery power cables must be remated, and the thermal covers and the FHST cover modification retracted. The thermal covers and the FHST cover modification will be removed from HST the following EVA day because of timeline constraints.

10.2.2.3 ACS Installation

Note: The nominal plan is to install the ACS, ESM, and perform PCU cleanup, respectively, during the same EVA.

For more detailed information concerning latching go to the generic axial SI changeout operation in Section 3.4.2 of this document. The ACS EVA worksite is shown in Figure 3.4-8.

Prior to the beginning of the EVA changeout, the BAPS is rotated to the FOC/ACS changeout position (-V2 Forward) using the switch on the A6U panel. Proper position is verified by lining up the AS markings on the HST with that on the FSS BAPS. The STOCC will thermally condition the FOC the night before and the COSTAR deployable optical bench will have been already retracted (~ one month prior to launch).

The EVA begins by deploying the aft fixture on the SAC. The IV crewmember activates the worklights and SI indicator from the SSP. Next, the -V2 aft shroud doors are opened, if not yet completed. Figure 3.4-9 reveals the ACS detailed worksite.

As the -V2 aft shroud doors are being opened, the IV calls the MCC to power down the FOC. The STOCC normally performs the powerdown; however, the crew has the capability to issue these commands, if necessary. In addition to the powerdown

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of FOC, the STOCC will also reconfigure the NASA Standard Spacecraft Model-1 (NSSC1) Flight Software (FSW) to support the new ORI.

Once the doors are opened to the fully opened position at the bump stop and tethered open, and upon verification of FOC powerdown, the four FOC power (J1, J2) and signal data (J3, J4) connectors are disconnected.

Next the FOC purge line will be demated from the FOC be gently squeezing on the wire-locking clip to release it, and pulling the fitting away from the FOC. The purge line will then be mated to the purge line stowage fitting on the HST aft bulkhead. Then the FOC ground strap will be disconnected and temporarily stowed.

Next, the A latch is released and the A LATCHED indicator lights verified. After the A latch is released, the B latch cover is removed and the B latch released. The crew should verify that the A SEATED indicator lights illuminate when the FOC A latch is unseated. Next, the FOC is lowered until the bottom guiderail is contacted to position the FOC for removal along the OTA/SSM guiderails. The B READY indicator lights should be monitored because they will illuminate during the extraction from the B ready-to-latch position. The FOC is then pulled out of the axial cavity via the guiderail system and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with the non-RMS crewmember providing guidance support. This allows for any repositioning necessary to ensure a level extraction of the FOC. Once removed, the FOC is stowed on the SAC aft fixture and secured with one pip pin. Figure 10.2-2 shows the HST -V2 aft shroud (empty bay) overview.

The SAC ASIPE door is opened and the ASIPE and NCC ESM groundstraps are disconnected and stowed. The A latch is disengaged, and the A LATCHED indicator lights should illuminate. Then, the B latch MLI cover is opened, and the B latch is released. The A SEATED light should be monitored for ACS and is then translated back to the HST worksite for installation. ACS is inserted into place along the OTA/SSM guiderails until the B READY indicator lights are extinguished. The B latch is then engaged and torqued. ACS rises as the B latch is engaged, and the A and C latches are then seated. The A SEATED indicator lights are extinguished when the B latch is torqued. The B latch cover is then closed. Next, the A latch is closed, torqued, and verified to be latched by noting that the A LATCHED indicator lights have extinguished. Next, the ACS groundstrap is attached, and the four power and data connectors remated using the identification and clocking markings on the connectors.

Upon completion of the ACS installation, IV gives MCC a GO for ACS aliveness test. The STOCC performs the aliveness test to check the electrical path to the instrument.

The FOC is removed from the aft fixture and installed into the ASIPE until B READY indicator lights extinguish. The A SEATED lights will be extinguished when B latch torquing is complete. After the B latch is fully engaged, the A latch is engaged and torqued. The A latch is verified latched with the extinguishing of the A LATCHED lights. Finally, the ASIPE lid is closed and latched.

10.2.2.4 ESM Installation

Note: The nominal plan is to install the ACS and ESM and to perform PCU cleanup, respectively, during the same EVA.

Prior to the beginning of the ESM installation, the BAPS should be rotated to the -V2 forward using the switch on the A6U panel. Proper position is verified by lining up the AS markings on the HST with that on the FSS BAPS. The ACS should be completely installed prior to installing the ESM.

The ESM will be retrieved from the MULE by opening the contamination cover, securing it open by pulling the three restraint straps, and releasing the four keyhole bolts. Direct sunlight should not be allowed to fall on the black surfaces of the ESM during the installation; there is concern that contamination could bake off the black surfaces and deposit on critical OTA surfaces.

The ESM is brought into the -V2 Aft Shroud position and located such that the bridge latch 1 is in the middle holes of the inboard SSM guiderail standoffs. Bridge latch 1 is then engaged by moving the pawl release level down approximately 10° to the latch extend position and rotating the operating handle to extend the plunger until the plunger nose cone is firmly seated in the middle hole of the outboard SSM guiderail standoff and the white marking on the shaft is visible. The crew verifies that bridge latch 2 is in the open position and lowers the ESM assembly until bridge latch 2 contacts the SI connector EVA handhold. The hex plunger handle interface is torqued down. The locking bolt is engaged and torqued down. Next, the two bridge latch 1 lockdown hex bolts are engaged and torqued down.

The IV calls for MCC to powerdown COSTAR. The STOCC normally performs the powerdown; however, the crew has the capability to issue these commands if necessary. Upon verification of COSTAR powerdown, the ACS to ESM groundstrap is attached. The four (4) COSTAR harnesses are demated and the COSTAR -Y harness is installed and mated to the ESM. The groundstrap should be attached first; the connectors can be mated in any order. The four Cross Aft Shroud Harness (CASH) power and signal connectors are then mated at the ESM.

Upon completion of the ESM installation, the IV gives MCC a GO for the ESM aliveness test. The STOCC performs the aliveness test to check the electrical path to the ORU.

10.2.2.5 NCS Installation

Note: The nominal plan is to install the NCS and NOBL, respectively, during the same EVA.

Prior to the beginning of the NCS installation, the BAPS should be rotated to the +V2 forward using the switch on the A6U panel. Proper position is verified by lining up the AS markings on the HST with the markings on the FSS BAPS.

Next, the +V2 aft shroud doors are opened, if not yet completed. Figure 10.2-3 reveals the NCS detailed worksite.

The doors are fully opened to the limit of the doorstop extension and tethered open.

Figure 10.2-4 shows the floor overview and detail of the AS for NCS installation.

The NICMOS cryovent line will be removed by loosening three keyhole bolts at the HST aft bulkhead interface and three keyhole bolts at the NICMOS interface. The cryovent line will be removed and stowed in a transport bag, shown in Figure 10.2-4. The cryovent insert will be retrieved and installed into the aft bulkhead by positioning and rotating it into its three keyhole slots with the conduit lock ring handle parallel to the front face of STIS and pointing towards the +V3 axis. The three keyhole bolts are engaged and torqued down, and the three cryovent insert-over-center latches are retracted.

The IV calls for MCC to powerdown NICMOS and COSTAR. The STOCC normally performs the powerdown; however, the crew has the capability to issue these commands, if necessary.

Upon verification of NICMOS powerdown, the NICMOS groundstrap is removed and stowed, and the NCC groundstrap adapter is installed in the NICMOS groundstrap location, as shown in Figure 10.2-5. Then the NICMOS groundstrap is reinstalled on the NCC groundstrap adapter. The groundstrap adapter should be installed before the NCC to allow better access to the NICMOS groundstrap interfaces.

The NCC will be retrieved from the SAC by opening the contamination cover, rigid cover, and releasing the four keyhole bolts. Direct sunlight should not be allowed to fall on the black surfaces of the NCC during the installation; there is concern that contamination could bake off the black surfaces and deposit on critical OTA surfaces. During translation, the two neon bypass valves will be opened.

The NCC is brought into aft shroud +V2 position 2 and placed so that the bridge latch 1 is in the middle hole of the inboard SSM guiderail standoff. Bridge latch 1 is then engaged by moving the pawl release level down approximately 10 degrees to the latch extend position, rotating the operating handle to extend the plunger until the plunger nose cone is firmly seated in the middle hole of the outboard SSM guiderail standoff and the white marking on the shaft is visible. The crew verifies that bridge latch 2 is in the open position, and lowers the ESM assembly until bridge latch 2 contacts the SI connector EVA handhold. The center guide latch plunger bolt is preloaded, and the inboard lockdown bolt is engaged followed by the outboard bolt. Then the NCC groundstrap adapter is untethered from the NICMOS handrail and installed on the NCC. Upon verification of COSTAR power down, the four CASH terminator plugs may be removed and the four CASH connectors mated to the NCC.

Once the radiator is brought over to the aft shroud, it is aligned and positioned over the HST guiderails. The two upper and one lower radiator quarter-turn latches are engaged

onto HST handrails and then the two shear ties are engaged to the lock position. Once the radiator is installed, the radiator handrails cannot be used for transportation.

Before harness and Capillary Pumped Loop (CPL) mate, the two cryovalve heaters are attached to the Coolant In and Coolant Out valves. The two neon lines are installed into their respective NICMOS Coolant In and Coolant Out interfaces by hand-tightening the wingnuts.

The NCC power harness is retrieved and mated to the radiator diode box J1 connector. The EV then peels back the MLI covering the radiator diode box and verifies that the NCC A and NCC B indicators are not illuminated. The MLI around the EPS test panel is peeled back; the connector cap is removed from P600.

Next the conduit is released from the NCS radiator by opening the two quarter-turn conduit latches. The conduit is pivoted under the HST aft bulkhead and locked into place.

The three primary and one secondary sock restraints are released and the CPL sock assembly is routed through the +V2 AS cryovent port. Once the sock is clear of the cryovent port, the three spring-loaded pawls are reengaged. The conduit light seal is engaged to the cryovent port insert, and the crew will verify that the light seal alignment mark is visible inside the AS. The conduit lock ring is rotated until the pins on the stovepipe are engaged, and two hex bolts on the conduit lock ring are engaged. The CPL sock assembly is removed and stowed.

On MCC GO, the EV crew takes the "A" and "B" switches to Enable and NCC A and NCC B to the ON position, verifies that the associated indicators turn OFF, then switches "A" and "B" to the OFF position and replaces the MLI.

Upon completion of the connections, the IV gives MCC a GO for the NCS aliveness test. The STOCC performs the aliveness test to check the electrical paths to the system.



Figure 10.2-2. +V2 aft shroud interior



Figure 10.2-3. NICMOS cryo and connector panel

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Figure 10.2-4. Removing NICMOS cryovent line



Figure 10.2-5. Installing NCC groundstrap adapter from NICMOS groundstrap lug



Figure 10.2-6. Engaging bridge latch 1 - lockdown bolts



Figure 10.2-7. NCC with evaporator, CASH, and radiator harness installed



Figure 10.2-8. NCS P600 cable routing to EPS test panel



Figure 10.2-9. NCS radiator diode box detail

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10.3 NEW HARDWARE FOR SM3B

10.3.1 Advanced Camera for Surveys

10.3.1.1 Overview

The Advanced Camera for Surveys (ACS) is a third generation imager for HST that replaces the Faint Object Camera (FOC) in the -V2/-V3 quadrant of the HST aft shroud. ACS has a field of view greater than twice as large as WFPC2 and is greater than five times more sensitive. The ACS is launched in the Second Axial Carrier (SAC) Axial Science Instrument Protective Enclosure (ASIPE). The ACS will be unpowered before installation in the HST because the only electrical connection to the ASIPE is the grounding strap.

The instrument has three optical channels: the Wide Field Channel (WFC), the High Resolution Channel (HRC), and the Solar Blind Channel (SBC). The WFC has a large field of view for rapid surveying. The HRC has a smaller field of view with a corresponding increase in resolution. The SBC blocks visible light to operate in the far-UV spectrum. ACS can also perform spectroscopy, coronography, and imaging polarimetry. The ACS is designed to increase the capability of the HST for surveys and discovery by at least a factor of 10. Its mission is to study the evolution of galaxies, galaxy clusters, and the early universe.

ACS is a versatile instrument that can be applied to broad range of scientific programs. The high sensitivity and wide field of the WFC in the visible and near-infrared will make it the instrument of choice for deep imaging programs in this wavelength region. The HRC, with its excellent spatial resolution, can be used for high-precision photometry in stellar population programs. The HRC coronograph can be used for the detection of circumstellar disks and Quasi-Stellar Object (QSO) host galaxies.

10.3.1.2 Structural Description

The ACS is very similar to previous HST axial science instruments such as STIS, NICMOS, and COSTAR. It measures 86.5 by 35.1 by 35.1 inches, roughly comparable to a telephone booth, and masses 875 pounds. The exterior surfaces are composed of graphite epoxy and metal. A door approximately 3 inches inside the ACS aperture blocks light but does not seal the enclosure.

ACS has numerous interfaces. For EVA manipulation, an EVA handhold and L-Handle are installed on the -V2 face of the instrument. Mechanically, ACS interfaces HST via three kinematic latches, labeled A, B, and C. Jumpers J1 and J2 (power), J3 and J4 (data), and a groundstrap electrically connect ACS to the telescope. Finally, ACS is designed to be cooled via the HST Aft-Shroud Cooling System (ASCS) for greater performance. The ACS to ASCS thermal interface is via two capillary pump loop connectors. ASCS has been demanifested from flight SM3B, but installation is expected on SM4.

See Figure 10.3-1 to Figure 10.3-11.



Figure 10.3-1. ACS in a ground fixture, showing outboard panels (note the EVA handhold)



Figure 10.3-2. ACS in a ground fixture, showing inboard panels



Figure 10.3-3. ACS in the ASIPE, showing the EVA handhold, L-handle, and ASCS saddle



Figure 10.3-4. ACS electrical connector panel with nonflight shorting plugs



Figure 10.3-5. ACS forward end (C-latch in upper left corner of instrument, A-latch and ACS aperture with nonflight cover in lower right corner)



Figure 10.3-6. Detail of ACS A-latch and aperture with nonflight aperture cover



Figure 10.3-7. Detail of ACS C-latch



Figure 10.3-8. ACS aft end B-latch in lower left corner and nonflight vent covers installed (this face also contains SBC radiator)



Figure 10.3-9. Detail of ACS B-latch



Figure 10.3-10. ACS forward end view showing aperture on left and two guide blocks along upper right edge



Figure 10.3-11. ACS -V2 Outboard Panel (this face contains CCD radiator; small diagonal face contains connector panel on left and two CEB radiators to right)
The ACS structural subsystem consists of two major components: the fixed truss optical bench and the enclosure. ACS is installed into the HST using the A, B, and C fittings common to all of the axial scientific instruments. These fittings support the outer skin of the instrument, the ACS enclosure.

The enclosure provides the interfaces for the ACS, including the connector panel, astronaut handles, and other necessary guides, rails, and handling points. Inside the enclosure the electronics and thermal shelf are supported from enclosure structural members. The truss assembly is kinematically mounted to the A, B, and C fittings using X, Y, and Z fittings. Note that the truss assembly is often referred to as the optical bench, and these two terms are used interchangeably.

The optical bench is the primary load bearing structure in the instrument. The bench is a graphite-epoxy truss structure consisting of trapezoidal tubing longerons, square tubing cross members, and plate bulkheads. Truss members are joined with bonded fingerplates. Bonded-in metal fittings provide load points for interfacing with other components of the instrument. The optical bench bulkheads support the optical elements and detectors.

The X, Y, and Z fittings are the structural interface between the enclosure and the bench. The X fitting provides three-axis constraint. The Y axis fitting constrains lateral motion but allows axial motion through a sliding spherical bearing. The Z fitting constrains rotation about the X fitting.

The fittings are not to be considered mechanisms but are rather structural mounts or joints that kinematically constrain the bench to the scientific instrument latch hard points of the enclosure. When installed in the HST, these fittings allow thermal distortions of the enclosure relative to the truss to occur without impacting the precision alignment of the optics.

The ACS enclosure is a rectangular box honeycomb structure with one longitudinal edge chamfered and consists of forward and aft bulkheads with joining panels. The enclosure provides the mechanical interface to the ASIPE/HST through the A, B, and C fittings. The enclosure supports the electronics boxes (mounted inside the enclosure) and astronaut handles (mounted externally). It is also a contamination and light shield for the instrument.

The electrical connector interface panel is on the top chamfered panel. The enclosure is vented through openings in the aft bulkhead as well as through the aperture. These vents are sized to the pressure profiles of the orbiter ascent and descent rates.

The enclosure is fabricated from aluminum honeycomb with graphite-epoxy face sheets and is designed to withstand launch and loading loads and to have axial stiffness of 75,000 lb/in. The panels are not vented but are designed to withstand the pressure differentials of ascent/descent. An integral honeycomb structural saddle that bolts into the enclosure supports the Main Electronics Box (MEB). The honeycomb is 0.5 inches thick with 0.015-inch aluminum face sheets. Two Remote Interface Units (RIUs) attach to the MEB aft side plate. Two CCD Electronics Boxes (CEBs) attach to the enclosure top plate.

The instrument half of the A, B, and C latch fittings bolt, respectively, to the X, Y, and Z fittings on the enclosure bulkheads. Optical bench flight loads are carried directly into the ASIPE latches. The enclosure carries only the flight loads from its own weight and the weight of the MEBs. The ASIPE latches between the A and B fittings apply a static compressive preload.

10.3.1.3 Optical Subsystem

The optical subsystem provides three channels: a WFC, an HRC, and an SBC. Note that in this context, the words "channel" and "camera" are used interchangeably. The WFC has a large 202 by 202 arcsecond field of view with a resolution of 0.05 arcsecond/pixel for rapid surveying. The HRC has a 26 by 29 arcsecond field of view with a resolution of 0.027 arcsecond/pixel. The SBC blocks visible light to operate in the far-UV spectrum. It has a 31 by 35 arcsecond field of view at a resolution of 0.032 arcsecond/pixel.

Each channel consists of a detector and the optical elements necessary to process the input image received from the HST and focus the processed image onto one of the detector apertures. The system uses one Multi-Anode Microchannel Array (MAMA) detector and two Charge Coupled Device (CCD) detectors to convert light energy to electronic signals that are processed and transmitted to Earth. There are 6 mirrors, 56 filters, and 4 windows used in the optical system. These optical elements are made of various glasslike materials, including fused silica, MgF₂, CaF₂, BaF₂, and LiF.

The ACS uses nine electromechanical assemblies in the optical train to focus, position, and condition or block incident light energy to the optical detectors when operating in the HST. The mechanisms will be unpowered and nonoperational during prelaunch and launch operations of the shuttle. Each mechanism will be operated briefly during the functional test after ACS installation.

There are two types of calibration lamps used in the optical system: one deuterium lamp and four tungsten lamps. The deuterium lamp has a Ni63 oxide source that is slightly radioactive.

10.3.1.4 Electrical Subsystem

The ACS electronics will not be powered until installation into HST. The electrical system consists of two MEBs, RIUs, two CCD electronics boxes located inside the top panel, and an Auxiliary Power Box (APB) on the -V3 enclosure panel.

10.3.1.5 Thermal Subsystem

Before installation, the ACS is thermally maintained by the ASIPE. After installation, ACS has both an active and passive thermal control subsystem to maintain the various components with acceptable operating limits. Active heaters, thermal straps, thermal blankets, surface finishes, heat pipes, and special radiators are utilized in this subsystem. The optical bench is maintained at $20^{\circ} \pm 2^{\circ}$ C to minimize thermally induced structural deformations that would degrade optical performance. The electronic compartment operating temperature range is from -25° to +40° C, depending upon spacecraft and instrument operating conditions.

The heat energy dissipated by the MAMA detector is transported to the aft bulkhead by four heat pipes. The CCD detectors are mounted on thermoelectric coolers to allow the detector chip operating temperature to be controlled at -80° C. Two heat pipes from each CCD assembly transport heat to a heat transfer plate. Two additional heat pipes transport the heat from the transfer plate to an external radiator on an outer panel of the enclosure. The working fluid in the heat pipes is anhydrous ammonia.

For passive thermal control, the instrument exterior finish is the natural black of the graphite-epoxy except for radiator surfaces. The MEBs and CCD electronic boxes have aluminum radiator plates, anodized black mainly with small areas of chromate conversion coating.

10.3.1.6 ACS Installation

ACS installation is the fourth replacement of an axial scientific instrument on the HST. The installation task, therefore, is similar to that of COSTAR, STIS, and NICMOS. The duration of each of those installations closely matched preflight plans.

ACS installation consists of five major tasks. First, the worksite must be prepared and the FOC removed and temporarily stowed. Second, although technically not a part of ACS, the Cross Aft Shroud Harness (CASH) must be installed in support of the NCS. Third, the ACS instrument must be retrieved from the ASIPE on the SAC. Fourth, ACS is mechanically installed in the -V2/-V3 quadrant of the aft shroud and electrical connectors, including the groundstrap, are mated. Fifth, the FOC is stowed in the ASIPE.

Section 10.2.2 has a more detailed description of ACS installation in context of the entire servicing mission.

10.3.2 Power Control Unit (PCU)

The HST PCU has experienced several relay failures and is to be replaced on SM3B. The new unit is electrically similar to the old unit described in Section 2.4.1.8 except that the connector panel is angled to allow easier EVA connector mates. There are two integral handholds for crew handling. Connector savers on P13 and P14 and the ground plug have been modified.

10.3.3 HST Solar Array 3

10.3.3.1 Introduction

The HST SA3 system generates a total of 4730 watts to support HST mission operations. The SA output is passed to the SSM EPS at a nominal operating voltage of 34 volts via two diode boxes for battery charging and distribution to the HST load. The system consists of two wings and control electronics to allow positioning of the SA solar cell panels for optimum solar illumination.

Each wing of the SA3 is equipped with four panels attached to a Panel Support Structure (PSS) and a Solar Array Drive Mechanism (SADM) that slews the array to face the Sun when it is deployed. The weight of one SA is 640 pounds.

Deployment of each wing is only accomplished via EVA, which will be described later in this section. The orientation of each wing is controlled by the STOCC using the SA ECA. A functional overview of the SA ECA is provided in Drawing 6.1 sheet 3. The ECA system consists of the DCE and redundant SADE units, housed within SSM ES bay 7. The ECA receives operating power from the SSM EPS and exchanges commands and telemetry with the SSM DMS. The ECA provides for SADM motor control and for conditioning of all wing instrumentation. Previously, the ECA also provided control for the PDM and SDM, which is no longer required with the SA3 design.

During the SM1 mission, the original SA1s were replaced with SA2s. SA2 was similar in design to SA1 with a few redesign exceptions. The problem with the SA1 and SA2 design was that they were susceptible to temperature-induced jitter. The new rigid SA3 design is intended to provide for the long-term power needs of the HST while reducing the jitter experienced with flexible arrays.

The crew will also replace the existing DBAs with second generation DBA2s. In addition to providing diode isolation between the solar arrays and batteries, each DBA2 includes relays that are commanded by a respective Diode Box Controller (DBC). These additional relays could be used to recover from PCU relay failures. Control of these relays also requires the DBA2s to be connected to the HST P601 EPS test connector (at the EPS test panel). The DBA2s also include arc suppression networks to extend the life of the PCU relays and to provide wingtab connectors for easier installation.

10.3.3.2 SA3 System Hardware

Panel Support Structure

The PSS provides the mounting interface for four solar panels and the associated harnessing. The PSS consists of two support frames joined by two panel hinge assemblies. The two panel hinge assemblies serve as the primary structural interface between the two frames. It is at this interface that the PSS is folded for stowage on the Rigid Array Carrier (RAC) for launch.

Each PSS frame provides the main structural support for its two attached solar panels. The support frames are made of six 3- by 5-inch rectangular and one 3-inch diameter round Aluminum-Lithium Alloy X2096-T8A3 tubes. The tubes are connected via Aluminum 7075-T73 fittings. The fittings are fastened to the tubes via A286/Monel rivets. The fixed panel support frame incorporates the interface to the mast assembly and is the frame that is directly mounted to the RAC. The rotating frame is hinged to the fixed frame via the panel hinge assemblies. The two frames are near mirror images of one another with the exception of the mast interface and the tip latch fittings.

Each solar panel is mounted to its respective PSS support frame by three titanium flexures. These flexures, shown in Figure 10.3-12, are designed to isolate motion of the solar panels from the support structure to minimize solar array induced jitter of HST. They also provide thermal isolation and reduce the potential for "thermal snap" caused by built-up strains between the PSS and solar panel.



Figure 10.3-12. Solar panel flexure detail

10.3-12

Solar Panels

The individual solar panels are procured directly from the Iridium commercial production line, and are built using a 5056 AI honeycomb core assemblies with front and backside graphite epoxy face sheets. The front face sheet is covered with Kapton sheeting and is then populated with solar cells. The rear face sheet of the panel is co-bonded with a Tedlar coating. The honeycomb core edges are covered with Tedlar tape. Each panel weighs 33.5 pounds.

Each of the four solar panels on an SA wing contains 1000 6.5- by 5.5-cm Gallium Arsenide on Germanium solar cells, shown in Figure 10.3-13. These solar cells are wired within the solar panels into 20 circuits of 50 series-connected cells each. These 20 circuits are grouped electrically into 10 strings of two parallel-connected circuits, as shown in Figure 10.3-14. Each string is capable of producing a maximum short circuit current (Isc) of 2.4 amps. The power from each solar panel string is carried individually through redundant wiring all the way to the SA wing DBA2, where the strings are electrically combined into individual SPAs.



Figure 10.3-13. SA3 solar cell detail

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Figure 10.3-13. SA3 solar cell detail



Figure 10.3-14. SA3 circuit detail

The electrical interface between each solar panel and its SA wing is a single 37-pin sub-D connector, which is supported by the interface connector-mounting bracket located on the backside of the panel (total of four panels per wing). Solar panel power passing through this sub-D connector is carried from the panel to the coupler through a wiring harness bundle consisting of 20 twisted pairs (i.e., 40 wires per bundle). This allows the power from each panel's 10 strings to be carried redundantly through two separate twisted wire pairs within the bundle; either pair is rated to carry the full string current by itself.

The four SA wing wiring harnesses are combined within the coupler and are passed on to the SADM flex harness, as shown in Figure 10.3-15. The flex harness design for SA3 is intentionally unchanged from SA1/SA2 because this is a proven design and also it ensures SADE torque compatibility. The overall design is unchanged. However, the flex harness wiring of circuits passing through it had to be changed relative to SA1/SA2 because the individual SPAs in SA1/SA2 are electrically wired within the solar array blanket itself and carried through the interconnecting wiring as intact SPAs. In SA3, 40 individual strings are carried from the four solar panels (10 strings per solar panel) down the wing through the SADM flex harness, all the way to a DBA2, where the SPAs are then electrically formed.



Figure 10.3-15. SA3 wiring/harness overview

The SADM flex harness terminates in connectors that mate to the DBA2. The current SA1/SA2 DBA design utilizes ganged rack-and-panel Deutsch connectors on a common drive stud bracket, with the Y- and Z-ganged connectors dedicated to passing SA power to the DBA X-connector. The X-connector passes power from the DBA to the spacecraft. The W-connector is dedicated to SA and DBC command and telemetry and heater power (see Figure 10.3-16).



Figure 10.3-16. SADM to DBA2 Detail

Panel Hinge Assemblies

Two panel hinge assemblies on each SA wing join the two support frames. The inboard hinge (i.e., the one closest to the mast) contains the components for latching the wing in the deployed position. The two EVA latch screws incorporate a soft dock device to assist the crewmembers while deploying the PSS. No EVA access is required for the outboard hinge. The hinges contain belleville washers that allow for longitudinal (in-axis) expansion and contraction. They also provide friction in the hinge joint to allow a smooth, controlled rate of panel rotation. A combination of a snubber on the inboard hinge and a stop on the outboard hinge are adjusted at assembly to provide a preload to ensure stability and rigidity across the hinge interface.

Solar Array Drive Mechanism

The SADM/coupler/damper assembly, consists of the ESA supplied SADM, the mast/SADM coupler, the SADA soft dock assembly, and the damper assembly. The SADM assemblies (one per wing) are provided by ESA and

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are essentially identical in design to SA1/SA2. They include a SADM and Solar Array Drive Adapter (SADA) upper half or cone. The cone is keyed to correctly clock SA3 to HST. The SADA outer cone mates to an ESA supplied SADA clamp pickup assembly on the HST. The SADA serves as the primary structural interface between the SA wing and the HST. The SADM provides the motive force for slewing the SA3 wing about its mast axis.

The handedness of the SADM/coupler assemblies is established when this assembly is mated to the mast tube assembly. During this operation, the SADM rotating shaft, in the neutral harness position (shaft stop is 180° away from the housing toggle stop), is clocked 90° CW (+V2 wing) or CCW (-V2 wing) with respect to the centerline between the manual brake actuation bolts. The +V1 side of the mast tube assembly is centered on the line between the manual brake actuation bolts and bolted to the SADM flange.

Provisions have been made to allow the installation of either SA3 on either side of HST in a contingency. A second keyway has been provided in the SADA outer cone. By releasing the EVA brake on the SADM and rotating the SADM 180°, the secondary keyway may then be used to clock SA3 properly.

Coupler

The coupler is located outboard the SADM and connects the SADM motor output shaft to the damper and solar array mast assembly. The coupler contains four EVA compatible power feedthrough connectors, and electrical redundancy bus blocks. The coupler also provides mounting points for the Y-, Z-, and W-connector temporary stowage brackets from launch until the new wing is installed on HST. The connectors are mated to the stowage brackets, which in addition to providing mechanical restraint, act to electrically short the individual solar panel strings and selected W-connector functions.

Damper

The damper provides additional structural damping for the primary bending modes of SA3. It is located between the mast tubes assembly and the coupler. The damper consists of four quarter-cylindrical shear lap assemblies mounted circumferentially to a central post. The shear lap assemblies consist of a layer of Visco-Elastic Material (VEM) sandwiched between 6AI-4V titanium cylindrical shear laps. The inner and outer shear laps are mounted to opposite flanges of the central post. Motion transmitted through the damper passes through the shear laps into the VEM where the energy is absorbed. Launch loads are carried through a 6AI-4V titanium central post internal to the shear laps. Active heater control is used to maintain the temperature environment for the VEM in each shear lap. The damper to solar array mast assembly bolt circle joint provides an EVA interface for future replacement of a damaged or failed solar array wing.

SADA Soft Dock

The SADA soft dock feature consists of three spring fingers mounted on a ring between the SADA outer cone and the SADM. The purpose of this feature is to temporarily hold SA3 in place once the crew has inserted it into the SADA clamp pickup assembly on HST. The spring fingers latch around a flange on the pickup assembly and hold the SADA halves within the clamp capture range. Each spring may be independently overridden if SA3 must be removed. Each spring also may be configured in one of two locations, allowing for either SA3 wing to be installed on either side of HST in a contingency condition.

Solar Array Mast Assembly

The SA mast assembly is designed to place the SA panels away from the body of HST. The mast tubes assembly interfaces with the damper. The mast assembly is articulated for stowage on the RAC. The solar array mast is deployed while the SA wing is still stowed in the RAC. Two EVA-actuated captive bolt assemblies latch the mast in place.

The mast hinge consists of a shaft (integral to the mast tube end fitting) supported by two angular contact bearing pairs. The bearing pairs are each mounted in a 6AI-4V-titanium ear. One ear is fixed to the hinge base; the other ear is somewhat flexible. This design allows longitudinal (in-axis) thermal growth, while still maintaining joint stiffness. Two captive mast-locking bolts secure the mast hinge. A soft dock holds the mast temporarily in the deployed position. This feature enables the crewmembers to deploy the mast and easily secure it with the locking bolts.

The four SA mast wiring harness bundles (one from each solar panel) run down the backside of the SA wing and mast, through EVA contingency connectors on the coupler, through the SADM flex harness, and finally to the DBA2.

The EVA contingency connectors provide the ability at some later servicing mission to change out either an SA wing without the associated SADM, or vice versa. In this scenario the SA mast would be mechanically separated from the damper at the bolted interface just above the damper, while the four wing wiring harness bundles would be separated at the four EVA contingency connectors located on the coupler.

Diode Box Assembly 2

Each wing is supported by a DBA2 that provides isolation diodes for power from each of the SPAs and the interconnect interface between the electronics and the deployment and positioning mechanisms. The design is based on the original HST DBAs with design enhancements. The enhancements include relay arc suppression networks, and the ability to recover solar array power lost because of failed relays in the PCU and additional relays for charge control capability. The DBA2 is an ORU and also provides the capability to disconnect the cables from SA3 if jettison of the array is required. Connector brackets are the means of disconnect. The brackets allow the removal of two or four connectors at a time using an EVA power tool on the connector bracket drive

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stud provided. Each connector bracket has a tether loop to restrain the wire harness after disconnect.

Physical Description

Each DBA2 measures approximately 5 by 6 by 34 inches and weighs 24 pounds. As shown in Figure 10.3-17, they are mounted externally to the HST on the forward bulkhead of the SSM ES near the SADA/SSM interface, using four ORU bolts and associated J-hooks. Each DBA contains 80 stud-mounted power diodes, two per string. The two power isolation diodes of each string are connected in parallel and mounted together on an aluminum heatsink.

The fixing brackets for the heatsinks are attached to the side flanges of the DBA2 ORU tray. The tray also supports 12 electrical connectors that provide the SSM-to-SA3 interface and the interface between the ECA and SA mechanisms. These connectors are grouped in two sets of four and two sets of two. The W-connectors provide telemetry from the SA3, and the X-connectors output SPA power to the SSM. Both X and W sets are the same as for the original diode box and are mated using a drive-stud bracket. The CCW rotation of the connector bracket drive stud through five full turns automatically disconnects the bracket connectors, allowing the bracket and harness to be retained to the SA3 wing using a provided keyhole slot and bolt. The Y- and Z-connectors provide raw input power from the SA3 panels to the DBA2 and are the more traditional, EVA-friendly, wingtab connectors. EVA tether rings are affixed to each end of the DBA2 for temporary restraint of the unit while detached from the SSM during installation or SA3 on-orbit maintenance. A fiberglass cover is attached to the DBA2 on the front side to minimize inadvertent contact with the diode plates.

Electrical Description

The DBA2 isolation diodes are JANTXV1N5816 high efficiency diodes. As mentioned above, the two diodes of a single SPA circuit are mounted on the same heatsink and in electrical contact with it. Isolation from the other heatsinks is provided by an insulating plate onto which the heatsinks are bolted.

Within the DBA2, the SPAs are electrically formed and are then fed to the PCU. The 40 solar panel strings fed to each DBA2 from their respective solar array wings will be wired into SPAs as follows:

- a. Three 5-string SPAs: D, E, and AA
- b. Six 4-string SPAs: A, B, C, BB, CC, and DD
- c. One 1-string SPA: EE

The SPA assignments are configured to protect the HST wire deratings and the derating of the PCU and DBA2 relays.

Each DBA2 contains 28 relays: 10 latching disconnect relays, 9 nonlatching charge control relays, and 9 latching bypass relays. A set of three relays (one disconnect, one charge control, and one bypass) is designated for each of the nine SPAs that provide battery charging. The \pm EE SPAs are not connected to a battery and, therefore, have only disconnect relays in the DBA2. The disconnect relays will be used to electrically disconnect SA3 from HST and may be used to deadface the SA connectors. The charge control relays are provided for each SA3 SPA, except the \pm EE SPAs, and are used to manage battery charging. The bypass relays will be used to recover charge control relays that have failed in the disconnect position. The operation of these relays will be described in the DBC section.

An Arc Suppression Network (ASN) is included in the DBA2 for each SPA section to eliminate arcing across the contacts of the PCU relays. The way the ASN operates is as follows: In sunlight, with charge control relays closed, a capacitor is charged to the SPA voltage level. The charge control relay starts to open, and as the contact resistance increases (arcing across the contacts), the SPA voltage increases, thus causing the capacitor to absorb the SPA current, which limits the rate of voltage rise, causing the arc to extinguish. Then the capacitor charges up to the open circuit SPA voltage. Once orbit night arrives, the capacitor discharges through the 100 k Ω resistor down to the bus voltage.

Thermal Control

Heat generated by the DBA2 power diodes is radiated out of the unit through the silverized Teflon-coated heatsink plates. The inside face of the shield surrounding the plates is painted black. All other external surfaces are taped with aluminized Kapton except where they are covered by the mounting bracket.

Active thermal control of the DBA2 is provided during HST mission phases when the SA3 is not generating power. Survival heaters are installed on each of the 10 power diode heatsink plates in a DBA2 and controlled by two parallel sets of two seriesconnected thermostats. Each heater element is approximately 256 ohms. A DBA2 survival heater circuit consists of the thermostats controlling 10 heater strips connected in parallel. The instantaneous maximum power required by the DBA survival heaters is 80 watts (40 watts per DBA2). The survival heaters are powered off during nominal HST operations because heat generated by the power diodes warms the DBA2 while the SA3 is generating electricity.

Diode Box Controller

The DBC is an addition to the DBA2 that provides a means to recover from certain PCU relay failures. The DBC provides battery charge control external to the PCU using relays in the DBA2.

The DBC enables HST to regain the previously lost SPAs by reconfiguring the PCU and performing battery charge control in the DBA2/DBC (Software Charge Mode (SCM)). It provides an Autonomous Charge Mode (ACM) to provide battery charge control while in

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HWSP. The DBC provides an SCM to reduce usage of PCU relays during normal operation and provides disconnection of SA power for PCU changeout.

Physical Description

The DBC is integrated into the DBA2 ORU assembly, as shown in Figure 10.3-17. The DBC power is supplied via the cross-strap harness from the J601 interface by tapping and fusing command and telemetry buses CT 1/2 and CT3/5. The EVA crew installs a cross-strap harness between the two DBA2s, and extends it to the J601 connector on the HST EPS test panel. The DBC cross-strap harness is routed along a handrail attached to the top of the SSM ES. The harness interconnects the two DBCs to provide redundancy on critical functions. The harness is a T-connection to J601 to provide for power, ACM battery voltage, and battery temperature signals when the VIK jumper plugs are installed.



Figure 10.3-17. DBC/DBA2 installed on HST

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Functional Description

The DBC has three modes of operation: SCM, ACM, and Ground Command Mode (GCM). In SCM, the DBA2 relays are commanded by the 486 and EPS FSW based on battery voltage and temperature. ACM commands the DBA2 relays based on battery voltage and temperature also, but the commands originate at the DBC. GCM allows the DBA2 relays to be commanded in real time by the STOCC.

SCM is enabled by a Keep-Alive Signal (S K/A) sent from the DCE and by enabling the EPS charge control routine in FSW. The DCE S K/As are cross-strapped to both DBCs for redundancy using the cross-strap harness. The 486 monitors battery voltages and temperatures and compares them against a virtual V/T curve for chargeoff and charge-on determination. When a given battery has reached the chargeoff or charge-on V/T threshold, the 486 sends a relay specific pulse-width modulated signal to both DBCs to offline the respective SPA. The pulse-width command is decoded by the intended DBC and the specific DBC relay driver is commanded to open or close the respective nonlatching charge control relay in the DBA2. Loss of the S K/A from the DCE results in the disabling of SCM, and the CC relays in the DBA2 are autonomously closed.

ACM is enabled by the loss of the Autonomous Mode Disable (AMD) signals from the DCE, which are cross-strapped to both DBCs. Nine individual DBC hardware charge controllers, one for each SPA (except the EE), compare the respective temperature compensated battery voltage against a fixed chargeoff and charge-on threshold. When a given battery reaches the chargeoff or charge-on threshold, the hardware charge controller commands the respective relay driver to open or close the respective nonlatching charge control relay in the DBA2. Resumption of the AMD signals from the DCE results in the disabling of ACM.

ACM mode is disabled to launch and would be enabled in the event the PCU was not changed out. Enabling the ACM requires the EVA crew to install the V/T signal jumper plug on the cross-strap harness. This provides the battery voltage signals to DBC. In addition, the crew must install the six VIK jumper plugs to provide battery temperature signals to the DBC.

GCM is a subset of SCM and permits the following: the opening of each DBA2 charge control relay; the collective opening of the D, E, and AA DBA2 charge control relays in each or both DBA2s; the collective closing of all DBA2 charge control relays in each or both DBA2s; the capability to collectively open or close all 10 SPA disconnect relays in each or both DBA2s; the capability to collectively open or close all 9 bypass relays in each or both DBA2s. GCM requires the S K/A signals to be enabled, the 486 to be in control, and the critical command enable for disconnect and bypass relay operations.

Because of a limitation of available telemetry channels, the collective status of the 10 disconnect relays and the collective status of the 9 bypass relays in the DBA2 are provided in telemetry by using one spare bilevel telemetry channel in each SADE. SADE-1 conditions the collective status of the 9 bypass relays, and SADE-2 conditions

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the collective status of the 10 disconnect relays. For each relay type, the telemetry indicates if all are open or one or more are closed. The status of the nine DBA2 charge control relays is provided in telemetry by using nine spare analog telemetry channels in the DCE.

10.3.3.3 EVA Operations

At the start of the solar array changeout, the SA2 SDMs will already be retracted. The PDMs will still be deployed because the SA2 SADM brake is expected to slip away from the PDM stow position during SDM retraction, as it did during SM1. Once this slippage takes place, the array cannot be slewed back to the PDM stow position because the reduced inertia of the rolled up SA2 will prevent stable operation of the SADM control system. For this reason the PDM retraction will be performed via EVA only.

10.3.3.4 SA2 Removal

The RAC SA2 forward and aft latches will be prepared for SA2 stowage by removing the launch lock pip pins and inserting them in the adjacent stowage holes. Next, the SA2 PDM is manually retracted by rotating the PDM override interface. The orbiter must be in free drift from the beginning of manual PDM retraction until the crew has control of the array using the integral handrails. As the SA2 approaches the stowed position, the second EV crewmember aligns and seats the SA2 in the HST forward and aft latches. The crew may manually rotate the SA2 SADM as necessary to align the SA2 with the latches during PDM retraction and then engage the SA2 PDM lock by rotating the 7/16-inch hex interface until the indicator pin is aligned with the ON position hardstop. During the retraction, the crew should avoid any contact with displaced SA2 hinge pins, the SA2 blanket, bistem thermal shields, blanket tension assembly springs, and limit straps.

Once the PDM is locked, the crew will install a Portable Connector Tray (PCT); will demate and stow the DBA W-, Y-, and Z-connector brackets; and will stow them on the PCT. An MCC GO is required prior to connector demate to ensure that all the connectors have been deadfaced. If the SA2 blankets are not fully retracted, the demating and stowage of the Z and Y brackets must occur while the orbiter flies an attitude that shadows the SA. These connectors are energized when the solar cells are illuminated.

Next the SADA clamp is released by driving the hex EVA interface 19 to 20 turns. At 16 turns from the fully engaged position, the SADA clamp should be open far enough for the SA2 to be removed and the SA3 to be installed. The SA2 will now be removed from HST and translated over to the RAC. An ESA-imposed operational requirement states that the HST SADA clamp spindle should not be driven faster than 20 rpm.

The SA2 is seated on the RAC SADA clamp, rotated into the aft and forward latches, and held in that position. The SADA clamp spindle lock will be released by folding the lever up 90° to the OFF position; the clamp will then be engaged by rotating the EVA hex interface until the indicator pin is aligned with the CLOSED position.

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The spindle lock is then re-engaged by folding the lever down 90° to the ON position. An ESA-imposed operational requirement states that the RAC SADA clamp spindle should not be driven faster than 20 rpm. The RAC aft and forward latches will now be driven to their hardstop, and the crew will verify that the pointers on the hex interfaces are aligned with the PRELOAD marks on the housing MLI. The aft latch is the main load-bearing interface of the SA2; so this latch must be engaged prior to the forward latch. Finally, the crew will engage the forward constraint by installing the pip pin through theSA2 GSE lift point and seating the two bolts that secure the sliding bracket.

10.3.3.5 DBA Changeout

Once the SA2 is stowed on the RAC, the EV crewmembers will remove the DBA2 from the RAC by peeling back the thermal cover and releasing the four J-hook bolts. The DBA2 has a thermal exposure limit of 5 hours 50 minutes that begins after the thermal cover is removed.

The free float crewmember will demate the HST DBA X-connector bracket keyhole bolts and then disengage the connector bracket drive stud by rotating the EVA hex interface. The free floater then removes the DBA from HST by releasing the four J-hook bolts. The crew swaps DBAs, and the RMS crewmember stows the DBA on the RAC using the four J-hook bolts. The thermal cover is then reinstalled and secured.

At the same time, the free float crewmember mounts DBA2 on HST and secures the four J-hook bolts. The connectors are inspected for debris or damage, and the HST DBA X-connector drive stud is rotated until the back of the connector bracket is flush with the top of the DBA tray rail. Once the DBA X-connector bracket is mated, the EV crew will give the ground a GO to enable the DBA2 heaters.

10.3.3.6 SA3 Installation

This task begins by releasing RAC latch 5 and deploying the SA3 mast 90° until the soft-dock engages. Two 7/16-inch hex mast-locking bolts are engaged to lock the mast into place.

RAC latch 4 is released by loosening the 7/16-inch hex interface 10 to 15 turns CCW. Either latch 4 or latches 2 and 3 must be released after the SA3 mast is removed from latch 5, because they assist in reacting moments about the orbiter Z axes that may otherwise cause binding at latch 5. Next latch 2 and latch 3 are released, and the crew will stabilize the SA3 to prevent contact with adjacent structure. The RAC/SA3 conductive Velcro ground strap is released and secured to the RAC stowage location. This strap must be in place for deorbit with SA3; so it cannot be released on a previous EVA day. One crewmember will stabilize the SA3 while the other releases latch 1 by rotating the 7/16-inch hex interface CCW until the threads disengage. The SA3 is guided out of the RAC, and the IV crewmember will monitor clearance to the second SA3 stowed on the opposite side of the RAC. While SA3 is being controlled by a single crewmember, the orbiter will be in free drift.

The RMS crewmember translates with the SA3 toward HST, and the other crewmember helps guide the mast into the HST SADA clamp. Once the SA3 is seated with the primary keyway on the lip of the SADA aligned with the key in the SADA clamp, all three softdocks will be engaged. This ensures that the array is aligned properly for SADA clamp engagement. With the SADA fully seated, the SADA clamp will be fully closed by driving the 7/16-inch hex interface until the indicator pin is aligned with the CLOSED position hardstop. The orbiter will be in free drift from SA3 final approach until the SADA clamp is engaged. An ESA-imposed operational requirement states that the HST SADA clamp spindle should not be driven faster than 20 rpm.

10.3.3.7 SA3 Connector Mating and Panel Deployment

Once the solar array has been successfully mated to HST, the power and data connectors can be mated. The ground will give the EV crew a GO prior to engaging the W-connector bracket, in order to prevent mating an energized connector. The free float crewmember unstows the W-connector bracket by rotating the drive stud 5 to 7 turns CCW until the connector bracket is free. The connector bracket is then mated by positioning the connector bracket above the DBA2 W-connector interface and rotating the drive stud CW until the top of the bracket is flush with the top of the DBA tray rail. Then the P5A-P8A connectors will be demated from the SA3 connector stowage bracket and mated to the corresponding receptacles on the DBA2. The P5A-P8A connectors cannot be mated while the solar cells are illuminated. Nominally, this will not be a problem because the panels will still be folded during this timeframe. Next the DBC telemetry harness will be mated to SA3 W bracket J3, and the DBC cross-strap harness will be routed around the +V3 SSM bay handrails and mated at the DBA2 J1 and EPS test panel J601 connectors. The P601 connector contacts are energized; however, no GO is required from the ground because this connector is tied directly to the HST main buses and, operationally, cannot be deadfaced.

After all connections have been successfully mated, the EV crew will remove the SA3 panel restraint strap by squeezing the buckle levers, extending the strap, and removing the strap from the handrail hooks. At this time, the orbiter will be placed in free drift until the panel locking soft-dock is engaged. The SA3 panels will be deployed by hand. The deploy speed will be limited to a maximum of 5 deg/sec, which will allow time for the crew to react to an unexpected motion. At the end of travel, the deployment speed will be less than 1.0 deg/sec to preclude the SADA bracket from slipping. Then the two SA3 panel locking bolts will be rotated CW until the gap between the two halves of the inboard hinge is closed and both bolts reach their hardstop. Finally, the SADM bracke will be released by rotating either of the two EVA hex interfaces 11 to 14 turns CCW.

See Figure 10.3-18 through Figure 10.3-30.

10.3.4 Rate Sensor Unit (RSU)

Rate Sensor Unit (RSU) 3 has had failures on Gyros 5 and 6. See Appendix D. The replacement of RSU3 on SM3B is under review.

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Figure 10.3-18. Stowing SA2 on the RAC SADA clamp



Figure 10.3-19. DBA2 removal from the port RAC shelf



Figure 10.3-20. +V2 DBA2 installation on HST



Figure 10.3-21. Engaging -V2 SA3 mast locking bolts



Figure 10.3-22. Releasing RAC -V2 SA3 latch 4



Figure 10.3-23. Releasing -V2 SA3RAC latch 2



Figure 10.3-24. -V2 SA3 removal from the RAC



Figure 10.3-25. Engaging SA3 SADA clamp



Figure 10.3-26. +V2 SA3/DBA2 W-connector mate



Figure 10.3-27. +V2 SA3/DBA2 connectors 5A-8A mated



Figure 10.3-28. SA3 deploy

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Figure 10.3-29. Engaging +V2 SA3 panel locking



Figure 10.3-30. V2 SA3 brake release

10.3.5 NICMOS Cooling System (NCS)

Figure 10.3-31 is the NCS hardware layout in HST.



Figure 10.3-31. NCS hardware layout in HST

NICMOS is the only instrument on HST capable of detecting Infrared (IR) radiation. It is a critical trailblazer to future IR missions and is a direct predecessor of NGST. The NICMOS detectors are mounted to the cold optics bench housed in a hybrid dewar (solid nitrogen with aluminum foam) designed to maintain their temperature at 58 ± 2 K with a stability of ± 2 K. NICMOS was designed for a 4.5-year mission lifetime. NICMOS was successfully installed on HST during SM2 in February 1997, but the instrument failed because of a thermal short between the cold baffles and the surrounding Vapor Cooled Shield (VCS). This thermal short significantly increased the parasitic heat loading, which in turn reduced the lifetime of the cryogen and thus the anticipated period of science operations from 4.5 to 1.5 years.

The NICMOS Cooling System (NCS) is designed to restore NICMOS to operation and extend the life of the instrument beyond the original plans. NCS is designed to maintain a temperature between 58 and 72 K (the goal is 70 K) with stability of 0.1 K/orbit and long-term stability of 0.5 K/year. NCS is an addition to HST with systems that are completely new; they do not replace any existing hardware. HST interfaces have not been used previously for the purposes required by these systems.

10.3.5.1 NCS Hardware Overview

NCS comprises three major elements: NICMOS Cryocooler (NCC), NCS Capillary Pump Loop (CPL)/radiator, and the Electronics Support Module (ESM). The NCC cools the NICMOS detectors by circulating neon through existing plumbing originally used, prelaunch, to circulate helium to freeze liquid nitrogen in the dewar. The connectors have EVA friendly bayonet fittings. The NCS CPL/radiator removes heat from the NCC to the external environment. The heat transport capability is in excess of that required for the NCC under any operating condition and can be utilized for future needs via an interface plate on the radiator. The ESM provides the command, control, and telemetry functions.

There are three NCS coolant loops. The circulator loop flows neon to carry heat from NICMOS to the NCC. The cryocooler loop, housed within the NCC, uses neon to carry heat from the circulator loop to the CPL. The CPL uses ammonia (NH_3) to carry heat from the NCC to the radiator, where the heat is given up into the space environment.

The NCC circulator loop uses a centrifugal circulator to pump a constant flow of neon between the NCC and NICMOS. The circulator has a permanent magnet synchronous motor driven by the Power Conversion Electronics (PCE), operating at 1200 rev/s and nominally yielding a flow rate of 0.4 g/s of neon at 4 atm. The circulator loop removes heat from the NICMOS instrument and conductive parasites.

Heat flows from the circulator loop to the cryocooler loop via the Cold Load Interface (CLI). The CLI is a stainless steel counter flow fin-type heat exchanger with a thermal effectiveness of 0.963. The CLI provides structural support for the circulator, turboalternator, and recuperator.

The cryocooler loop uses two high-speed turbomachines, the turboalternator and the centrifugal compressor, to provide expansion and compression of the neon as it transfers heat to the Heat Rejection Interface (HRI). The HRI provides a conductive coupling between the NCC, PCE, and CPL. Neon gas expands on the face of a turbine rotor that turns an alternator. The electricity generated by the alternator flows through a selectable resistive load in the PCE. Heat created in the load is rejected into the HRI and later rejected by the radiator. In this way heat is extracted from the cryocooler loop. A centrifugal compressor, at the warm end of the loop, provides compression to the neon. The compressor is driven by a three-phase induction motor with a maximum operating speed of 7500 rev/s and is bolted to the HRI. A recuperator provides heat transfer between the high- and low-pressure streams of gas in the cryocooler loop. The cold end of the recuperator is supported by the CLI, and the warm end is supported by the NCC structure.

The NCS CPL/radiator interfaces with the NCC at the HRI via the CPL evaporator (CPL pump) that carries heat from the NCC to the external radiator. Cold liquid coming from the radiator flows into the evaporator and is vaporized transferring heat from the NCC. This warm vapor carries heat out of the evaporator to the radiator via stainless steel tubing. The vapor enters a heat pipe heat exchanger in the radiator and

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condenses in the header heat pipe. Heat from the header heat pipe is conducted to a series of spreader heat pipes throughout the radiator, where heat is then radiated to the space environment. The condensed vapor moves through a subcooler region to cool the liquid prior to exiting the radiator and returning to the HRI.

10.3.5.2 NICMOS Cryocooler

Figure 10.3-32 is the +V2 interior view of the NICMOS cryocooler installation location.



Figure 10.3-32. +V2 interior view of the NCC installation location

The NCC consists of two closed flow loops; both loops contain high purity neon gas. The cryocooler loop provides refrigeration by cooling the neon gas. Neon gas is circulated in a separate circulator loop between the cold load interface heat exchanger and the existing cooling coil plumbing located inside the NICMOS cryostat.

An induction motor driven compressor supplied work to the neon gas in the cryocooler loop to increase its pressure. This high-pressure gas is then expanded across a turboalternator to a lower pressure that cools the gas, and the cold gas is used to remove heat from the circulator loop at the cold load interface. The gas is returned to the compressor inlet through a recuperator, completing the cryocooler loop.

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In the circulator loop, a circulator pumps the second stream of neon gas from the cold load interface heat exchanger to the cooling coil inside the NICMOS cryostat. There it removes heat from the cryostat to maintain the instrument detector temperature. The gas is returned from the cryostat to the circulator inlet. Two male titanium bayonet fittings attach, via EVA operations, to the Coolant In and Coolant Out female bayonet interfaces on the NICMOS cryopanel (see Figure 10.3-33 and Figure 10.3-34).



Figure 10.3-34. NCC connection to NICMOS cryopanel

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The cryocooler is designed to remove approximately 7 watts of heat, including approximately 0.5 watts from the NICMOS instrument, from the circulator loop at the cold load interface. The intended operating temperature for the cryostat is 70 K. The majority of the heat load removed by the cryocooler is the result of parasitic heat sources at the bayonet connectors, at the cryostat inlet and outlet valves, and from heat transfer into the flexible lines between the cold load interface and the connections at the NICMOS cryostat.

Figure 10.3-35 shows the NCC CPL thermal joint.



Figure 10.3-35. NCC CPL thermal joint

As a thermal interface, a copper saddle provides a conductive joint between the NCC HRI and CPL evaporator. The CPL evaporator is installed via EVA operations. Pressure required to meet conduction requirements is achieved by EVA installation of a cover plate that bolts to the copper saddle. See Figure 10.3-36. Activation/verification of CPL survival heaters may be required before seating the evaporator pump.



Figure 10.3-36. NCC saddle cover

The NCC Housing Assembly is a 16- by 16- by 30-inch (41 cm by 41 cm by 76 cm) aluminum housing that encloses the cryocooler (Figure 10.3-37) and part of the circulator loop and provides the thermal interface between the CPL and cryocooler loop. The CPL interfaces and the circulator loop flexlines exit on the top of the housing to maximize EVA accessibility. Eight EVA electrical connectors are also on top of the housing.



Figure 10.3-37. NICMOS cryocooler

The housing attaches to the HST aft shroud via bridge latches attached to an aluminum-faced, honeycomb core baseplate. The baseplate ties into the supports of the lower SSM guiderails with Bridge Latch 1 (BL1) (Figure 10.3-38) and the SI connector handrail with Bridge Latch 2 (BL2) (Figure 10.3-39).

Bridge latch 1 (Figure 10.3-38) is an EVA-actuated rack-and-pinion plunger latch with a dual ratchet/pawl assembly to control allowable travel direction. The latch deploys a compliant nose code that engages the center 2.0-inch (5.1-cm) diameter hole in the support standoffs for the aft SSM guiderails, providing 5 degrees of constraint. The preloaded ratchet/pawl locks the plunger into position.



Figure 10.3-38. Center guide latch/bridge latch 1

10.3-45

Bridge latch 2 (Figure 10.3-39) is an EVA-actuated latch mechanism with fixed and movable jaws that clamp onto an EVA handrail, providing one degree of constraint.



Figure 10.3-39. Handrail latch/bridge latch 2

The NCS N1 handrail latch and NST1 shear tie are shown in Figure 10.3-40.

10.3-46



Figure 10.3-40. NCS N1 handrail latch and NST1 shear tie

The NCC housing encloses all of the NCC except the circulator loop flex lines, valve heaters, and bayonet fittings. A Kevlar 29 braid suspension is incorporated to support the cold end components, including the recuperator, to minimize parasitic heat loss.

The ESM, radiators, and NCC must be removed from the telescope for the return of the HST to Earth because they cannot withstand the landing loads. The radiators also exceed the allowable orbiter payload bay envelope.

The NCC has three high-speed rotating parts: compressor, circulator, and turboalternator. The compressor and circulator are driven at their operating rotational speeds by ac inverters that provide the required frequency and voltage to the motors. The turboalternator shaft rotates because of the expansion of gas through the turbine rotor that produces shaft work. All shafts are supported at each end during operation by self-acting gas journal bearings and by thrust bearings located at each end for axial positioning of the shafts. The neon process gas is the working fluid in the bearings, with no other means of lubrication provided.

The nominal value of the compressor rotational speed is about 7000 rev/sec at the design operating condition. The turboalternator rotates at approximately 4000 rev/sec and the circulator is driven at 1500 rev/sec. Each of the turbomachines is capable of a wider range of operating speeds as required during cooldown, for standby operation, and for other operating states. The turbomachines use high strength materials for the shafts (titanium), impellers (titanium), and housings (stainless steel).

The NCC will not operate until HST deployment.

Electrical power for the compressor and the circulator is provided by inverter power supplied to convert HST unregulated dc power to three-phase ac power of the correct voltage and frequency required. At the design operating conditions, the total electrical power input to the PCE is approximately 350 watts from the HST dc power system.

The dc power is provided to the PCE by the P600 harness via the J600 diode box to J1 and J2 connectors on NCC (see Figure 10.3-41).



Figure 10.3-41. NCS diode box

The expansion of neon gas through the turbine produces shaft power that is converted to approximately 11 watts of electrical power by the alternator located on the turbine shaft. The alternator output is connected to load resistors located in the PCE module where the power is dissipated as heat. The value of load resistance is increased during cooldown to maintain the highest possible turbine speed to maximize work output and the cooldown rate.

Heat produced by the compressor, the PCE, and the turboalternator load resistance is removed by conduction to the HRI located on the NCC support frame. The waste heat is carried away from the heat rejection interface by the NCS CPL, using ammonia as the heat transfer fluid. There is no transfer of neon or ammonia between the NCC and the NCS CPL.

This thermal management system maintains the heat rejection temperature at a nominal value of 300 K, although the temperature could be higher or lower than 300 K, depending upon the total amount of heat rejected and depending upon the thermal environment in HST.

The gas in the NCC loops is high purity neon that is tested and verified to meet the purity requirements before filling the loops. The cryocooler loop requires 15 grams of neon to achieve the design operating pressure level in the cryocooler for a 7-watt load at 70 K, with a heat rejection temperature of approximately 300 K.

The gas volume in the circulating loop is approximately 4.9L consisting of the 3-L gas main tank, a 1.5-L refill tank and 377 cm³ for the NICMOS cooling coil, flex lines, and other components. The circulating loop, including both tanks, contains approximately 21.5 g of neon. The nominal pressure is 4 atm. The remainder of the loop is at vacuum when installed. The system is designed so that if the valves between the refill tank, main tank, and the bayonets failed open, the pressure would remain below 135 psi in the worst-case aft shroud environment (30° C).

The 1.5-L refill tank at high pressure (96 psia at 294 K) is connected to the main tank to provide a gas reserve if leakage through the bayonet or NICMOS valve O-rings is excessive. The refill tank would be opened only to the main tank briefly through a remotely actuated solenoid isolation valve (SV2) to recharge the circulator loop and maintain the required operating pressure. The refill tank allows one full recharge of the system.

A purge port allows evacuation of the loop and main tank if contaminants are in the system that affects system performance. Two Solenoid Valves (SV1 and SV3) in series provide one-fault tolerance to loss of all neon resulting in loss of NCC operations. A charge/EVA refill port, with a manual valve (MV2) interface, enables recharge of the main and refill tanks on the ground and allows for recharge on orbit during subsequent servicing missions, if required.

Table 10.3-1 lists the functions of the NCC valves.

NCC valve	Function
MV1	Valve is flown in the open state
MV2	Opened to charge or refill the circulator loop with neon, if required
SV1 and SV3 (in series)	Allows evacuation of circulator loop and main tank, if required
SV2	Allows for refill of circulator loop, if needed

Table 10.3-1. NCC valve names and functions

The operating pressure level in each loop has been selected to achieve high operating efficiency for the critical components. For the cryocooler loop, the pressure at the inlet to the compressor (the lowest pressure in the loop) is approximately 32 psia at the design operating point. The compressor achieves a pressure ratio of approximately 1.63 at the design point so the high pressure in the flow loop is 53 psia.

In the circulating loop, the design operating pressure is 4 atm (44.1 psia) with the NICMOS coil at 70 K and the gas in the fill bottle at 300 K. The fill bottle is connected to the circulating loop through a capillary tube that has very low heat conductance to minimize thermal parasitics from the fill bottle into the cold loop. The pressure drop

caused by gas flow in the circulating loop is small, approximately 0.9 psi; thus, the pressure is nearly uniform through the circulating loop except for the refill tank, which is nominally isolated.

The pressure of the gas will be a function of the temperatures in each loop because each loop is closed and contains a fixed volume of gas. The pressure will be constant in the flow loop and will be a direct function of the volume averaged temperature when the compressor is off and no flow is present. The maximum expected nonoperating pressure in the cryocooler loop would be 53 psia if the gas everywhere were at the maximum temperature of 80° C (emergency/abort landing condition). The pressure at the compressor inlet varies from about 2.2 atm during the cooldown from a nominal 300 K ambient condition to the design inlet pressure of 1.45 atm when the cold load reaches the design temperature of 70 K.

At the nominal design operating condition for the NCC, gas temperatures in the cryocooler loop will vary from a minimum of about 62 K at the outlet of the turbine to a maximum of about 400 K at the outlet of the compressor. The lower temperature is required to provide the desired NICMOS temperature of 70 K, while the high temperature is determined by the heat rejection temperature (305 K nominal) and the compressor work input to the gas.

In the circulating loop, the temperature range is from about 64 K at the outlet of the cold load interface to about 80 K at the inlet to the cold load interface. The heat sink for the cooling loop is the temperature at the turbine outlet.

10.3.5.3 NCS Capillary Pump Loop and Radiator

The NCS system utilizes a single CPL. The NCS CPL runs from the evaporator mounted to the NCC thermal interface joint, with flex lines that exit the HST through the bay 1 cryovent feedthrough, and runs along the aft bulkhead of the HST to the external radiator. The total transport distance is approximately 21 feet. The NCS is controlled and commanded by a shared electronics unit, the ESM, which is installed in bay 3 of the HST aft shroud.

The NCS CPL is a two-phase system that utilizes NH_3 as the working fluid for heat transfer. Heat is transferred from the thermal joint interface into the evaporator, where liquid returning from the radiator is evaporated. This excess power is transported as the vapor phase travels towards the condenser located on the external radiator. In the radiator, heat is rejected to the environment by condensing the vapor.

A fluid reservoir in the CPL stores and manages the working fluid inventory and provides precise thermal control of the system. The CPL has the capability to transport a large amount of heat in excess of that required to transport heat from the NCC out to the external radiator.

The CPL has no moving parts and is one continuous volume. The line and fittings are optimized to transport heat distances that are an order of magnitude greater than heat pipes. Separation of the liquid and vapor phases of the working fluid in the coaxial transport lines eliminates the frictional forces between the phases, providing for increased capillary fluid pumping head performance while minimizing the number of flex lines.

The CPL/radiator subsystem (Figure 10.3-42) is composed of an evaporator pump, a vapor line and a coaxial liquid line, Heat Pipe Heat Exchangers (HPHX), a reservoir, and a radiator. The evaporator pump and the entire CPL system contain no moving parts; it is a passive system. The working fluid is anhydrous NH₃. The evaporator pump and flexible CPL lines up to the cryovent feedthroughs are contained within the HST aft shroud. The radiator panels and the CPL lines from the cryovent feedthroughs to the radiator are external to the HST. The reservoir is mounted to the outside face of the radiator panel.



Figure 10.3-42. +V2 view showing the NCS radiator and conduit

Figure 10.3-43 shows the NCS radiator components (without the shear tie handles).



Figure 10.3-43. NCS radiator components (shear tie handles not shown)

The NCS CPL evaporator pump connects to the thermal joint interface mounted to the NCC. A thermal joint cover (Figure 10.3-36), once it is installed in the thermal joint interface, holds the evaporator in place. The cover provides the required preload for thermal conduction performance. The EVA crew will remove the stowed cover from the NCC, install the evaporators in the thermal joint interface, and then reinstall the cover by tightening the two center bolts. Counterbores on the cover and spring plungers provided an EVA soft dock feature. Frangible bolts are used for contingency removal, and additional bolts are included for installation contingency.

The total CPL transport distance is approximately 252 inches (640 cm). The routing will consist of flexible lines from the evaporator to the cryovent feedthrough, then rigid lines running through the conduit along the bottom of the telescope, and then flexible lines between the end of the conduit and the radiator transition bracket. The rigid lines are stainless steel and are housed in and protected by a rigid aluminum conduit designed to enable EVA installation of the CPL lines as it protects the lines from EVA-induced loads. The flex lines are stainless steel bellows with a braided stainless steel sheath bundled together and covered with a protective sheathing. Once installed, the majority of the flex lines are contained inside the aft shroud. All lines are covered with MLI or with low emissive coatings to enhance their thermal transport performance. The CPL flex lines from the rigid conduit to the evaporator will be fed through the existing 4.5-inch diameter cryovents on the HST aft bulkhead. A cryovent feedthrough secures the CPL to the aft bulkhead and provides a light tight seal and RF protection.

The bay 1 cryovent insert is shown in Figure 10.3-44.



Figure 10.3-44. Bay 1 cryovent insert

The NCS radiator is mounted externally on the aft shroud on the +V2, -V3 quadrant and is approximately 30 ft² (2.8 m²) in size. The radiator is 2 inches thick and constructed of planer aluminum honeycomb panels with embedded heat pipes. The honeycomb core sections were selected for a high stiffness-to-weight ratio versus a solid aluminum plate.

The radiator system consists of heat pipe heat exchangers, 2 header heat pipes, and 16 spreader heat pipes. The heat pipe heat exchangers are part of both the CPL and header heat pipe systems. Heat is transferred from the CPL to the two header heat pipes by the heat exchanger, with no exchange of ammonia. Again with no exchange of ammonia, heat is then transferred to the 16 spreader heat pipes. Heat transferred into the two header heat pipes is then transferred to a series of spreader heat pipes that distribute the heat over the entire radiator surface and hence into the space environment. Both the header and spreader heat pipes are constant conductance heat pipes extruded with axial grooves and are made of 6063 aluminum.

The CPL/radiator systems are completely charged and sealed in final flight configuration at GSFC before integration into the orbiter at KSC. There are no provisions for charging the ammonia heat transport systems at KSC or on orbit.

The CPL is integral to the radiator and is in a stowed position on the radiator for transport and handling. Restraints secure the rigid and flexible portions of the CPL and wiring harness onto the radiator for launch. The radiator will be mounted to the carrier hardware (MULE) for launch by seven latches. The crew will disengage the latches to remove the radiators from the carrier at the beginning of installation. EVA handrails are used for removal from the carrier, translation to the HST, and attachment.

The crew will install the panel by activating three latches that secure the radiator to the HST aft shroud handrails. Each latch employs a shaft-driven elliptical cam that drives a silicon-lined jaw down onto the HST handrail. The handrail is captured between the jaw and the lower body. The latches provide a kinematic interface with the HST, utilizing titanium flexures. Positive indication of the latch status and locking feature is via a quarter turn EVA handle. The radiator latches have the ability to be adjusted with respect to the radiator and handrail positions. Shear ties are used to restrict motion of the latch assembly sliding interface after radiator installation. The EVA-actuated mechanism is designed to accommodate on-orbit geometric uncertainties in the handrail positions. The latches are a fault-tolerant design with EVA accessible bolts holding the latch to the radiator for contingency removal.

Additionally, the radiator has cutouts for upper and lower Portable Foot Restraint (PFR) sockets and is coated with Z93 paint on the radiative side and MLI on the nonradiative side.

The CPL reservoir is mounted to the radiator, as is the radiator diode box. Diode box survival heaters are powered directly from the J600 connector, and the radiator operational heaters are powered from the ESM. The radiator survival heaters are powered by the diode box, but routed through the NCC and ESM.

10.3.5.4 Electronic Support Module

The Electronic Support Module (ESM) contains electronics to monitor and control the NCC and NCS CPL functions. See Figure 10.3-45 and Figure 10.3-46.



Figure 10.3-45. -V2 interior view showing ESM and COSTAR Y-harness



Figure 10.3-46. ESM assembly

10.3-55

Power to the ESM is selectable from the COSTAR power interface or the EPS test connector (J600).

Note: The J600 and J601 connectors are mislabelled on HST as P600 and P601, respectively.

The P600 connector (Figure 10.3-47) is an integral part of the NCS radiator and, with the P600 harness, is carried in the ATM1 for launch. The J600 connector is located on the +V2 side of the HST in the SSM section immediately adjacent to the Astronaut Control Panel (ACP) in the EPS test plug panel. Enclosed are two connectors (J600 and J601) that were originally intended only for ground testing of the HST EPS system. To gain access to the J600 connector, MLI must be removed from its Velcro mountings and pushed out of the worksite. Removal of the MLI allows access to a 5.18- by 11.99-inch sheet metal plate that has two captive 5/32-inch or no. 10 Allen head fastener interfaces. After the plate is removed, access to the J600 connector cover must be removed before the P600 radiator connector can be mated. The P600 is a hot connector mate/demate. The J600 provides power line protection at the J600 EPS test connector. A diode box diode powers the six power feeds from the HST J600 connector into the two power buses that supply the NCS.



Figure 10.3-47. P600 installed in test plug panel next to the ACP

10.3-56

The ESM housing is 16 by 16 by 27.5 inches and is mounted in bay 3 of the HST aft shroud by two EVA-activated latches mounted to the ESM bridgeplate. Bridge latch 1 (Figure 10.3-38) is an EVA-actuated rack-and-pinion plunger design that engages the support standoffs of the aft SSM guiderail. Bridge latch 2 (Figure 10.3-39) is an EVA-actuated latch mechanism that clamps onto the SI connector handhold. The housing is composed of 6061 T-6AI; the base is 1.5-inch aluminum honeycomb plate.

The ESM consists of two microcontrollers, two Remote Interface Units (RIUs) and the Power Switching and Distribution Unit (PSDU). An ion pressure sensor is mounted on the exterior of the ESM housing to detect changes in aft shroud pressure caused by gas leaks. There is an EVA connector panel for the CASH and radiator harnesses.

The microcontrollers provide control and monitoring of NCS thermal transport systems and the NCC (including NCC valves), telemetry processing, command interface and safing functions, and RS-232 interface to ground-support equipment. Heaters are installed on the ammonia systems for operational control of the CPL; they are controlled through the microcontrollers. The microcontroller has upper limits to which the heaters will not overheat the ammonia systems. In addition to the microcontroller, each heater circuit has two hard-wired overtemperature thermostats. These thermostats are normally closed, but open at +51° C. One thermostat is on the positive leg, and one is on the ground leg. The microcontroller and the two thermostats provide two-fault tolerance against heaters failing on and causing the ammonia systems to overpressurize.

The RIUs provide a data and command path between the ESM and the Science Instrument Command and Data Handling Subsystem (SIC&DH). The PSDU houses relays that provide power to the ESM, NCS (operational heaters), neon valve operations, and CPL systems.

10.3.5.5 Cross Aft Shroud Harness

The Cross Aft Shroud Harness (CASH) is shown in Figure 10.3-48; Figure 10.3-49 is the ESM side of the CASH extension hook; and Figure 10.3-50 is the NCC side of the fixed hook.







Figure 10.3-49. CASH extension hook (ESM side)



Figure 10.3-50. CASH fixed hook (NCC side)

10.3-58

10.3.5.6 NCC Diode Box

The main power interface to the NICMOS cooling system is through the HST EPS J600 test connector. The J600 test connectors provide direct access to all six of the HST batteries for charging and discharge during ground processing of the HST and a means of supplying power the transfer switches S1 through S4 and S7 to S10. The six HST battery buses are fused with 15 ampere fuses in P600 of W101 harness and supplied to the NCS Diode Box where they are diode OR'ed together to form redundant (A and B) buses. Each of the buses is split and fused to provide redundant power to the NCS Power Conversion Electronics (PCE), the Electronics Support Module (ESM), and an auxiliary power source for future use.

The S1 and S2 transfer switch power buses are also fused with 15 ampere fuses in P600 and routed to the Diode Box to supply redundant survival heater power for the Diode Box and the NCS. The Diode Box survival heater power is unswitched and supplied through 5 ampere fuses to power thermostatically controlled heaters in the Diode Box. The NCS survival heater, ESM, and PCE power buses are switched through a pair of latching relays to supply NCS power to connectors J2 and J3. The auxiliary power feeds are switched through a second pair of relays in the Diode Box and is terminated at connectors J7 and J8, which have protective caps installed. The Diode Box fuses are installed in two fuses plugs (P5 and P6) to facilitate ground testing and fuse replacement. There are no plans for EVA replacement of P5 and P6.

The power relays in the Diode Box are controlled through six EVA operated switched on the Diode Box – ENABLE A, ENABLE B, NCS A, NCS B, AUX A, and AUX B. The ENABLE A switch enables fused relay power to the NCS A and AUX A control switches. The ENABLE B switch, likewise supplies fused relay power to NCS B and AUX B control switches. Actuation of the control switches (NCS A, NCS B, AUX A, and AUX B) will apply power to the latch coil of the relays if the switch is operated in the direction of the arrow and to the unlatch coil of the relays if the switch is operated in the opposite direction of the arrow.

A Light Emitting Diode (LED) indicator shows the status of each of the four power control relays. When power is applied to the J600 interface of the Diode Box, an illuminated LED will indicate that the associated relay is open and its power connector is de-energized. Once a relay is closed by momentarily operating the control switch in the direction of the arrow, the LED will be extinguished.



Figure 10.3-51. NCS diode box power distribution

10.3-60

10.4 SAFETY

10.4.1 Introduction

This section provides a scaled down version of the safety requirements to which the HST satellite adheres for the fourth servicing mission. This safety information comes directly from HST hazard reports and the accompanying payload inputs to the Payload Safety Review Panel. It focuses primarily on the operational safety workarounds to hazardous situations and the payload inhibits to prevent those hazards from occurring. For generic payload system safety requirements, see the appropriate system section in this document.

Hazardous subsystems that are inadvertently activated or not properly safed during the rendezvous, retrieval, berthing, EVA servicing, and deployment could result in severe damage to the orbiter or injury to the crew. All requirements for safing and monitoring of HST hazardous systems are called out in individual hazard reports. For a complete singular list of the hazards for each mission phase, refer to HR-32.

10.4.2 Rapid Safing

For a rapid safing condition, both sides of the FSS CDUs are utilized for mechanism operation to expedite HST deploy and payload safing (dual motor ops). The HST is released via an orbiter backaway maneuver, and the FSS is pivoted down. The FSS must pivot to 45 degrees to clear the payload bay envelope, and 43.8 degrees (with BSP latched) for landing. If the FSS is in the reboost configuration, the FSS has to be pivoted up before HST backaway. Additionally, any stray or potentially PLBD-offending ORUs/ORIs crew aids or tools will be EVA jettisoned.

For emergency deorbit the payload is safed in less than 20 minutes, and for the Next Primary Landing Site (NPLS) contingency, the payload is safed within 50 minutes.

Two scenarios were chosen as representative worst-case payload bay configurations, and each was analyzed in detail. They include one for an EVA and one for a non-EVA crew sleep. These examples adequately envelop all nominal phases of the servicing mission for rapid safing considerations.

The EVA scenario initiates near the midway point during the solar array 3 installation. Rapid safing functions for this scenario depict the maximum number of crew tasks that would be required to safe the payload for door closure in the event of rapid safing at the worst possible time. See HR-29 for the rapid safing flow for these EVA activities.

The crew sleep scenario initiates from the nominal crew sleep payload bay configuration. See HR-29 for the rapid safing flow during this non-EVA activity.

Based on current estimates, the 20-minute requirement for initiation of PLBD closure cannot be met for the solar array 3 installation. The earliest time to initiate PLBD closure for this scenario is approximately 30 minutes from the call for rapid safing. A Non-Compliance Report (NCR) will be filed for this scenario.

In all cases, dual motor pivoter operation (considered zero fault tolerant) is required to meet or come close to the 20-minute requirement. The 50-minute requirement for the NPLS can be met with single motor pivoter operation (single fault tolerant). Twenty-four minutes is the approximate maximum NPLS rapid safing time for single motor pivoter and rotator operation for activities other than reboost (obtained by doubling the pivot and rotate time).

10.4.3 EVA Operations

During most EVA operations, electrical connectors are deadfaced prior to removal/ installation. For SM-3B, the exceptions to this are the battery connectors during PCU changeout, J600 mating for the NCS radiator installation, and J601 mating for the DBC cross-strap harness.

The connector design and the method of demating/mating the connectors will not pose a hazardous condition to the astronauts. Procedures with the EVA astronauts identify energized connectors and demonstrate the ability to manipulate energized connectors without grounding, mismating, or shorting. Any special EVA protocol necessary to minimize conductive hardware/material in connector worksite vicinity is identified. Procedures will minimize time the demated connector is exposed while not under positive control of an EVA/IVA astronaut.

All remaining connectors have at least one inhibit in place during EVA connector demating/mating activities. Connectors that are mated/demated by an astronaut are deenergized (<500 mA) by opening a minimum of one controlling power relay, or in the PCU changeout case, complete HST powerdown. During mating/demating of all connectors, commands to the HST are inhibited by disabling the remote command panel at JSC. EVA connectors that are energized are designed as scoop-proof, remote/mechanically driven, or use connector covers.

Depending on a connector, the types of inhibits vary in kind. In general, a scoop-proof or rack-and-panel connector design is treated as an inhibit. Any switches or relays upstream with verifiable monitor and downstream are considered inhibits. A high impedance line or any line that can provide no more than 0.5 A is considered nonhazardous. (All the data lines belong in this category and they are considered nonhazardous.) The inhibit-compliant criteria for the connectors are qualifying connector design (scoop-proof or rack-and-panel), upstream inhibit (switches or relays) with verifiable monitor, and downstream inhibit (load removed).

The connectors of the following hardware and/or interfaces have been reviewed for potential hazardous conditions either to the astronaut or to the orbiter during the third servicing mission EVA activities.

At this time, the planned ORU hardware is assumed to be the following:

- a. Replace SA2s with SA3s
- b. Replace DBAs with DBA-2/DBCs
- c. Install DBC cross-strap harness
- d. Replace PCU
- e. Replace FOC with ACS
- f. Install CASH
- g. Install ESM
- h. Install NCC
- i. Install NCS radiator
- j. Install NOBLs on bays 5-8

In addition to the above hardware connector interfaces, there are receptacles in the ORU carrier and HST that are to be used by the astronauts for powering the portable work lights.

The following hardware components have less than the proper number of inhibits required to ensure deadfacing of HOT connectors. Waivers have been provided for demating the connectors on these components.

10.4.3.1 J601/DBC Cross-Strap Harness

As part of the first SA3 installation, the DBC cross-strap is mated to J601 on the HST EPS test panel. Some of the pins on J601 are wired directly to the HST diode buses, which are constantly powered unless the PCU is changed out. During the second SA3 installation, the other end of the cross-strap will be mated to the -V3 DBC. The connector will be energized for the same reason as the J601 mating.

10.4.3.2 PCU Battery Demates/Mates

Prior to PCU removal, the EV crew must demate the J1 (power) connector from each of the six HST batteries. This is required to deadface several PCU connectors that will be demated during the changeout. The six battery power connectors can be remated once the replacement PCU has been fully installed.

10.4.3.3 J600/NCS Radiator

Power to the NCS is obtained from the J600 connector on the HST EPS Panel. Similar to J601, several pins on J600 are wired directly to the HST diode buses, which will remain powered during the NCS installation.

10.4.3.4 PCU Internal Essential Bus Harness Installation

If the HST is unable to transition from external to internal essential power prior to deployment, the following EVA task would be required if the failure was internal to the PCU. PCU connectors J31 and J19 and HST connector P19 are considered high power, which are mated/demated hot, and each connector has the powered side terminated in pins rather than in sockets.

10.4.3.5 ECU Changeout

The EVA task would be required only if a gyro ECU failure were experienced prior to or during SM3A. ECU connectors V13GXP3 and V13GXP4 are considered high power and are mated/demated hot.

10.4.3.6 PDU P15, P16, and P17 Fuse Plug and OTA P27A/P28A Fuse Plug Changeout

The above fuse plugs are carried in the event of a contingency that might require a fuse changeout. The fuse plugs have the following design attributes that mitigate the hazard involved during changeout. All plugs have EVA wingtabs with PDU P15, P16, and P17 fuse plugs that have had their wingtabs and tether loops covered or made from a nonconductive material. The following EVA protocol is implemented to increase safety. An HST PFR is positioned in front of the bay 4 door on which the PDUs are mounted, providing a stable work platform. Any unnecessary conductive hardware within the worksite vicinity is stowed before commencing. The fuse plugs are changed out one at a time.

10.4.4 Touch Temperatures

There are no touch temperature concerns for the HST servicing mission. All HST systems remain within the -180° to +235° F range, even with heaters failed ON or OFF. The SA heaters are the only heaters that can exceed the 235° F limit in a failed ON condition. They are controlled by two thermostats in parallel and can be deactivated after monitoring a high temperature. This temperature range is not a concern in the case of a stalled motor, because inaccessibility of that motor. All surface temperatures are maintained for the nominal EVA orbiter attitudes.

10.5 DISPLAYS, CONTROLS AND VISUALS

This section describes the crew interfaces to the HST. Three SSPs are flown for the mission in the L12U, L12L, and L11U panel locations. Additionally, the A6U panel is used to control the SSE mechanisms and the HST umbilicals. The HST SYSTEMS crew display is also provided with its corresponding FDA tables.

10.5.1 Standard Switch Panel

The crew has control of certain HST systems from the SSPs. The right side of panel L12U controls HST power sources, RSU survival heaters, FHST shutters, and the SSM worklights. The left side of panel L12L controls the capability to switch to PSP bypass mode. Panel L11 controls the MULE and cameras. A functional description of the SSP switches and talkbacks is provided in Table 10.5-1.

10.5.2 Crew Displays and FDA Tables

The HST SYSTEMS display allows the crew to command to and monitor HST systems and subsystems from the crew spec. The HST SYSTEMS display and a display description are included in Table 10.5-2 for a functional description of commands and telemetry. Fault detection and annunciation are included in Table 10.5-3.

ltem	Type device	Function	Use
ESSENTIAL BUS EXTERNAL POWER switch (S13)	Toggle switch, 3-position DPDT momentary (and should be held for approximately 2 sec)	ON (momentary UP) - Enables Interface Power Control Unit (IPCU) power latching relays K6 and K7 that provide FSS Power Conditioning Unit (PCU) power to HST external essential bus feeds A and B. Relay K6 powers HST external essential bus A and relay K7 powers bus B	Use to provide FSS PCU power to HST external essential buses through FSS umbilicals. Essential buses power the HST DMU CDIs and MA receivers. S13 is powered from S18
		Center position - Normal	Normal switch position when not in use

 Table 10.5-1. Standard switch panel switch and talkback functions

ltem	Type device	Function	Use
		OFF (momentary DOWN) - Disables IPCU power latching relays K6 and K7. This action disconnects FSS PCU power from HST external essential bus A and B feeds and connects HST battery power (internal) to HST essential buses A, B, and C	Use to remove FSS PCU power from HST essential buses. Removal of FSS PCU power from HST essential buses, whether intentional or inadvertent, will automatically activate a fail- over circuit in the HST PCU that will connect HST battery power (internal) to HST essential buses
ESSENTIAL BUS EXTERNAL POWER status (DS13)	Display indicator 3-position	ON - UP OFF - BP DOWN - Indication not used for HST	Status indicator up indicates that at least one external essential bus power service is activated inside the HST PCU. Does indicate continuity through HST umbilicals
ESSENTIAL BUS INTERNAL POWER switch (S14)	3-position DPDT momen- tary (should be held for approximately 1 sec)	ON (momentary UP) - Enables IPCU relay K12 to latch HST PCU power relays K7, K8, and K9 closed, and turns on HST CDI A and B. This action connects battery power from HST diode buses A, B, and C to HST essential buses A, B, and C and ensures that both CDIs are on	Used to configure HST essential bus to internal battery power following PCU changeout
		Center position - Normal	Normal switch position when not in use
		OFF (momentary DOWN) - Enables IPCU relay K13 to unlatch HST PCU power relays K7, K8, and K9. This action disconnects battery power from the HST essential buses A, B, and C	Used to remove internal essential power when external (FSS PCU) power is provided. S14 is powered from S18 Note: For power down in preparation for PCU change out S14 will be held off while External Essential is powered Off to override the fail-over function and prevent unnecessary cycling of K7, K8, and K9 relays

Table 10.5-1. Standard switch panel switch and talkback functions (continued)

ltem	Type device	Function	Use
ESSENTIAL BUS	Display indicator 3-position	ON - UP	Status indicator
INTERNAL POWER		OFF - BP	UP indicates that at least
Status (DOTA)		ON - DOWN indicator not used for HST	one internal essential bus is activated inside the HST PCU. Does indicate continuity through HST umbilicals
FHST SHUTTER switch (S15)	Toggle switch, 2-position DPST	CLOSE (UP) - Provides SSP power (PL CAB3 through CB4) to the HST FHST shutters 1, 2, and 3. Also enables non- latching relay K5 in the IPCU, which provides power from the FSS PCUs to FHST shutters status indicator (DS15)	Used to keep FHST shutter closed during SM to avoid FHST exposure directly to a bright object. Switch must be used to keep shutter closed whenever HST is on the FSS and the HST main bus is powered off unless the FHST covers have been installed
		Note: The IPCU is externally jumpered to provide this capability	
		OPEN (DOWN) – Removes SSP power from the FHST shutters and disables relay K5 in the IPCU, removing FSS PCU power from the HST FHST shutter status indicator	Used following reapplication of main bus power to allow STOCC commanding of FHST shutters. Also used to dead face power to PCU, PDUs 1, 2, and 3, and FHSTs for change out of these units
FHST SHUTTER status	Display indicator 2-position	CLOSE - Gray	Status indicator
(DS15)		OPEN - BP	UP indicates that external FHST shutter power is turned on at the IPCU. FHST shutter close is powered directly from SSP CAB PL3, bypassing IPCU and the SSP talkback is driven by IPCU relay. Does not indicate continuity through HST umbilicals
RSU SURV HTR PWR switch (S16)	Toggle switch, 2-position DPST	ON (UP) - Enables IPCU nonlatching relays K16 and K17. This action provides FSS PCU power to RSU heaters 1, 2, and 3	Used to maintain RSU survival thermal conditioning, not normally used for SM3B

Table 10.5-1. Standard switch panel switch and talkback functions (continued)

ltem	Type device	Function	Use
		OFF (DOWN) - Disables IPCU relays K16 and K17, removing power from the RSU survival heaters	Used to allow STOCC commanding of RSU survival heaters. Also used to dead face power to PCU, PDUs 1, 2, 3, and 4, ECUs, and RSUs for changeout of these units
RSU SURV HTR PWR	Display indicator 2-position	ON - Gray	Status indicator
status (DS16)		OFF - BP	Gray indicates that external RSU survival heater power service is turned on at the IPCU. Does not indicate continuity through HST umbilicals. Monitor is powered from the FSS DPCs
MN BUS INT PWR switch (S17)	Toggle switch, 3-position DPDT momentary (should be held for approximately 2 sec) (Switch cover installed to prevent inadvertent actuation of the OFF command)	ON (momentary UP) - Enables IPCU relay K10, which supplies power from the FSS PCUs to close motorized main buses A and B transfer switches S1 and S2, open main bus C transfer switches S3 and S4, and to close the diode bus interconnect switches S7 through S10. This action connects battery power from diode buses A, B, and C to main buses A and B, and disconnects bus C	The ON command is used to connect HST main bus to HST batteries if the batteries have been switched off-line. During PCU changeout, required switch operations are performed by STOCC via ground command. S17 is powered from S18
		Center position - None	Normal switch position when not in use
		OFF (momentary DOWN) - Enables IPCU relay K11, which supplies power from the FSS PCUs to open motorized transfer switches S1 through S4. This action disconnects battery power from main buses A, B, and C	The OFF command is used to disconnect the HST main bus from HST batteries. This action removes one of two inhibits that could cause a catastrophic HST failure under certain conditions when a SA SPA is connected directly to its respective diode bus

Table 10.5-1. Standard switch panel switch and talkback functions (continued)

Table 10.5-1.	Standard switch	panel switch and	talkback functions	(continued)
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ltem	Type device	Function	Use	
MN BUS PWR ON status (DS17)	Display indicator 2-position	ON - Gray	Gray indicates that at least one main bus is powered. Talkback does not indicate whether power is from internal or external source. Indicates continuity through HST umbilicals	
		OFF - BP	BP indicates main buses are not powered from either internal or external source	
ESS/MN SW ENA switch (S18)	Toggle switch, 2-position DPST	ON (UP) - Provides orbiter power through CB4 (from PL CAB3 bus) to SSP switches S13, S14, S17, and S21	Enables use of SSP control of HST main and essential buses via S13, S14, S17, and S21	
		OFF (DOWN) - Removes orbiter power from SSP switches S13, S14, S17, and S21	Disables use of SSP control of HST main and essential buses	
ESS/MN SW ENA status (DS18)	Display indicator 2-position	ON - Gray OFF - BP	Monitor power is provided from PL CAB3 bus through CB4	
SPACE TEL SW PWR circuit breaker (CB4)	Circuit breaker	OUT - Open	Opened following power down for PCU change out and during umbilical dead facing at HST redeployment	
		IN – Closed	Closed during SSE checkout	
		5-amp circuit breaker, which routes PL CAB3 power to S13, S14, S15, S16, S17, and S18 (UP) concurrent with SSE activation	and prior to PCUR power up	
IPCU RLY CLOSED status (DS20)	Display indicator 2-position	ON - Gray OFF - BP	Display status of IPCU relays. Indicates that IPCU main power switches (S1& S2) or (S3&S4) are closed. Monitor is powered from the FSS DPCs	

ltem	Type device	Function	Use
MAIN BUS EXT PWR switch (S21)	Toggle switch 3-position DPDT, momentary	ON (UP) - Closes motorized transfer switches S1 thru S4 in IPCU that provides orbiter power to HST main buses A and B	Used to provide FSS PCU power to the HST main buses for initial external power activation after berthing and for PCUR power up. S21 is enabled by S18
		Center position - None	Normal switch position when not in usage
		OFF (DOWN) - Opens motorized transfer switches S1 thru S4 in IPCU that provide orbiter power to the HST internal main buses A and B	Used to remove orbiter power from HST prior to PCU changeout and redeployment
MAIN BUS EXT PWR	Display indicator 2-position	ON - Gray	Status indicator.
status (DS21)		OFF - BP	Gray indicates that at least one external main bus is powered inside the HST PCU. Does indicate continuity through HST umbilicals
SSM WK LIGHTS switch (S24)	Toggle switch, 3-position DPDT	ON (UP) - Enables IPCU relay K18 to provide FSS PCU power to all HST site receptacles, plus all SI latch indicator lights	Used to provide power to aft shroud (AS) internal/external work light receptacles and latch indicator lights. For EVA use only
		OFF (CENTER) - Disables IPCU relays K18 and K19, removing power from all internal/ external worklight receptacles and latch indicator lights	Used to disable power to AS internal/ external work light receptacles and latch indicator lights
		ON (DOWN) - Enables IPCU relay K19 to provide HST worklight receptacles, plus all SI latch indicator lights	Used to provide power to aft shroud (AS) internal/external work light receptacles and latch indicator lights. For EVA use only

Table 10.5-1. Standard switch panel switch and talkback functions (concluded)

	1	2		3	4	5
	1 2 3 4 5 6 7 8 9 0 1 2 3 4 5 6 7 8	90123	4567	890123	56789012	3 4 5 6 7 8 9 0 1
1	X X X X / 2 1 0 / X X X	НSТ	SYS		XX X DDD/	HH:MM:SS
2	TL M COUNT XXX	PCS			DDD /	HH:MM:SS
3	СОМ М	7 + M	ODE	XX XXX	(ORU/ORI	DEACT
4	PSPENA 1 ID X	SNPT	DIS	8 + X X	FOC	1 2 + X X X
5	RCVR 1 2	PSEA		ХХХ	K NICMOS	13+ XXX
6	LOCK X X	SOLAR	ARR	ΑY	COSTAR	14+ XXX
7	AGC ±XXX ±XXX	DCE	PWR	AX XXX	RSU 1	2 3
8	NO-OP CMD 2			вх ххх	STATXXX	XXX XXX
9	СМД СТ ХХХ ХХХ	STOP	9 /	OFF 10	MAMPXXX	XXX XXX
10	XMTR 1	X	хххх	sxxxx	s xxx	XXX XXX
1	3 ON X	PDM	хххх	ХХХХ	OFF 15+	16+ 17+
2	SEL 4K 4	SDM	хххх	хххх		
3	5 MOD ON X		хххх	ХХХХ	FGS	BUS V
4	6 SSA OFF	SLE \	N XXX	s xxx	3 18+	2 X X X . X
5	EPS ABC	POS	ХХХ	ХХХ	OCE 19+	4 X X X . X
6	INT ESS BUS XXX	SADE	OFF	1 1 + X X	K EPT	X X . X
7	MAIN BUS V					
8	A XX.XXXX	HGA	+ V 3	- V 3	SA/DIODE	ВОХ
9	B XX.XXXX	HNG	ххх	x xxx	SA DEAC	T 20+
20	C XX.X XXX	LAT	ХХ	x xx	K SA SEC	TS XXXX
1	AMPS XXX	ВАТТ	CURR	ENTS	SURV H	TR XXX
2	UMB SG V INT EXT	1 X X	ХХХ	4 XXXX	SADE O	LD XXX
3	MN BUS XXS XXS	2 X X	ХХХ	5 X X X X	SPA XX	ХХХ
4	ESS BUS XXS XXS	3 X X	ХХХ	6 X X X X	K SA4 CUR	. XX.X
5						
6						

Table 10.5-2. Parameter characteristics: SM 210 HST SYS

			DISPLAY	STATUS INDICATORS					
CRINAME	WSID	HSTSRC	UNITS	RANGE	м	н	L	1	↓ ↓
COMM:									
TLM COUNT	P34U3004A	486/DMU	integer	1 to 120					
PSP ENA ID	V92J1103C		integer						
RCVR 1 LOCK	P34X3050E	DMU	event	blank or '*'					
2 LOCK	P34X3051E	DMU	event	blank or '*'					
RCVR 1 AGC	P34Y3064A	DIU 4	dBMs	-131 to -51					
2 AGC	P34Y3065A	DIU 4	dBMs	-130 to -50.8					
CMD CT 1	P34U3001A	DMU	integer	0 to 255					
2	P34U3002A	DMU	integer	0 to 255					
XMTR 1 ON	P93J1000C		echo fb of						
			item entry						

Table 10.5-2.	Parameter characteristics:	SM 210 HST SYS	(continued)
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	MOID			DISPLAY		STATI	JS INDICA	TORS	
	MSID	HOLOKC	UNITS	RANGE	М	Н	L	1	↓
MOD ON	P93J1001C		echo fb of						
EDQ			item entry						
INT ESS BUS A	P34X3085F		event	blank or '*'					
B	P34X3086E	DIU 2	event	blank or '*'					
С	P34X6087E	DIU 2	event	blank or '*'					
MAIN BUS V									
А	P34V3061A	486/DIU 2	volts	-0.28 to 35.67					
	P34X3046E	DIU 2	event	EXT, INT, ON*					
В	P34V3062A	486/DIU 2	volts	-0.28 to 35.67					
	P34X3047E	DIU 2	event	EXT, INT, ON*					
С	P34V3063A	486/DIU 2	volts	-0.28 to 35.67					
	P34X3048E/ P34X3049E	DIU 2 DIU 2	event	OFF or ON					
AMPS UMB SG V	P34C3060A	486/DIU 2	amps	-1.80 to 151.2					
MN BUS INT	P34V1027V	FMDM	volts	0 to 40	М	н	L		
EXT	P34V1009V	FMDM	volts	0 to 40	М	Н	L		
ESS BUS INT	P34V1026V	FMDM	volts	0 to 40	М	н	L		
EXT	P34V1035V	FMDM	volts	0 to 40	М	Н	L		
PCS:									
MODE	P93J1002C		echo fb of item entry						
	P34X3075E/ P34X3076E	486 486	event	NORM, SNPT, DRFT, blank					
SNPT DIS	P34X3005E	486	event	YES or NO					
PSEA	P34X3284E/P34X308 5E	DIU 4 DIU 5	event	RSET or ACTV					
SOLAR ARRAY:									
DCE PWR A	P34X3011E	DIU 5	event	blank or '*'					
	P34X3033E	DIU 4	event	OFF or ON					
В	P34X3012E	DIU 2	event	blank or '*'					
114/11/10	P34X3034E	DIU 4	event	OFF or ON					
+WING	P34X3035E	DIU 4	event	flash					↓
-WING	P34X3036E	DIU 4	event	flashing or no flash					\downarrow
PDM +WING	P34X3280E/	DIU 4	event	blank, DPLY,					
	P34X3033E/	DIU 4		NDPL					
	P34X3034E	DIU 4							
-WING	P34X3282E/	DIU 4	event	blank, DPLY,					
	P34X3283E/ P34X3033E/	DIU 4 DIU 4		NDPL					
	P34X3034E	DIU 4							
SDM +WING (main)	P34X3077E/	DIU 4	event	blank, DPLY,					
	P34X3078E/	DIU 4		STWD, TRAN					
	P34X3033E	DIU 4							
(red)	P34X3079E/	DIU 4	event	blank, DPLY,					
	P34X3080E/	DIU 4		STWD, TRAN					
	P34X3033E P34X3034E	DIU 4 DIU 4							
-WING (main)	P34X3081E/	DIU 4	event	blank, DPLY.					
	P34X3082E/	DIU 4		STWD, TRAN					
	P34X3033E/	DIU 4							
(red)	F34A3U34E P34X3A83E/		event	blank DDI V					
(ied)	P34X3084E/	DIU 4	Gyoni	STWD, TRAN					
	P34X3033E/	DIU 4							
	P34X3U34E	DIU 4							

Table 10.5-2.	Parameter characteristics:	SM 210 HST SYS	(continued)
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				DISPLAY	STATUS INDICATORS					
	MSID	HST SRC	UNITS	RANGE	М	Н	L	1	↓	
SLEW +WING	P34X3178E/ P34X3180E	486/DIU 4 486/DIU 4	event	blank or +OP (flashing)						
-WING	P34X3179E/ P34X3181E	486/DIU 4 486/DIU 4	event	blank or -OP (flashing)						
POS +WING	P34H3127A	486/DIU 4	degrees	-508 to 504						
-WING	P34H3128A	486/DIU 4	degrees	-508 to 504						
SADE OFF	P34X3129E/P34X313 0E	DIU 2 DIU 2	event	OFF or ON						
HGA HNG +V3	P34X3097E/ P34X3098E/ P34X3067E	DIU 4 DIU 4 DIU 5	event	blank, DPLY, STWD, TRAN						
-V3	P34X3099E/ P34X3100E/ P34X3067E	DIU 4 DIU 4 DIU 5	event	blank, DPLY, STWD, TRAN						
HGA LAT +V3	P34X3093E/ P34X3094E/ P34X3067E	DIU 4 DIU 4 DIU 5	event	blank, TRA, LAT, REL						
-V3	P34X3095E/ P34X3096E/ P34X3067E	DIU 4 DIU 4 DIU 5	event	blank, TRA, LAT, REL						
BATT CURRENTS:										
1	P34C3289A	DIU 2	amps	-25.6 to 25.4						
2	P34C3290A	DIU 2	amps	-25.6 to 25.4						
3	P34C3291A	DIU 2	amps	-25.6 to 25.4						
4	P34C3292A	DIU 2	amps	-25.6 to 25.4						
5	P34C3293A	DIU 2	amps	-25.6 to 25.4						
6	P34C3288A	DIU 2	amps	-25.6 to 25.4						
ORU/ORI DEACT:										
FOC	P34X3013E/ P34X3014E/ P34X3015E/ P34X3016E	DIU 2 DIU 5 DIU 5 DIU 5	event	OFF or ON						
NICMOS	P34X3137E/ P34X3138E/ P34X3139E/ P34X3140E	DIU 5 DIU 2 DIU 5 DIU 2	event	OFF or ON						
COSTAR	P34X3212E/ P34X3213E/ P34X3214E/ P34X3215E	DIU 2 DIU 5 DIU 5 DIU 2	event	OFF or ON						
RSU:										
1 STAT	P34X3120E/ P34X3121E/ P34X3122E/ P34X3123E	DIU 5 DIU 5 DIU 5 DIU 5 DIU 5	event	OFF or ON						
2 STAT	P34X3029E/ P34X3030E/ P34X3031E/ P34X3032E	DIU 5 DIU 5 DIU 5 DIU 5	event	OFF or ON						
3 STAT	P34X3025E/ P34X3026E/ P34X3027E/ P34X3028E	DIU 5 DIU 5 DIU 5 DIU 5 DIU 5	event	OFF or ON						
1 MAMP	P34C3124A	DIU 5	m amps	-8.4 to 705.6						
	P34C3125A	DIU 5	m amps	-8.4 to 705.6						
2 MAMP	P34C3055A	DIU 5	m amps	-8.4 to 705.6						
	P34C3056A	DIU 5	m amps	-8.4 to 705.6						
3 MAMP	P34C3057A P34C3058A	DIU 5 DIU 5	m amps m amps	-8.4 to 705.6 -8.4 to 705.6						

Table 10.5-2. Par	rameter characteristics:	SM 210 HST SYS	(concluded)
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	MOID			DISPLAY	STATUS INDICATORS				
CRINAME	WSID	HSISKC	UNITS	RANGE	м	н	L	1	↓
FGS:									
3 BUS	P34X3023E	DIU 2	event	blank or '*'					
V	P34V3022A	DIU 1	volts	-0.189 to 15.88					
OCE BUS	P34X3024E	DIU 2	event	blank or '*'					
V	P34V3022A	DIU 1	volts	-0.189 to 15.88					
EPT V	P34V3147A	DIU 1	volts	-0.09 to 7.20					
SA/DIODE BOX:									
SA SECTS	P34X3217E/ P34X3294E/ P34X3295E/ P34X3216E/ P34X3296E/ P34X3297E	DIU 4 DIU 4 DIU 4 DIU 4 DIU 4 DIU 4	event	+OFF, -OFF, ON, ****					
SURV HTR	P34X3148E/ P34X3149E/ P34X3150E/ P34X3151E/ P34X3152E	DIU 5 DIU 2 DIU 2 DIU 5 DIU 5	event	OFF or ON					
SADE OLD	P34X3176E/ P34X3177E	DIU 5 DIU 2	event	OFF or ON					
SPA (A)	P34X3155E/ P34X3156E/ P34X3157E/ P34X3158E	DIU 2 DIU 2 DIU 2 DIU 2 DIU 2	event	blank or A					
(B)	P34X3159E/ P34X3160E/ P34X3161E/ P34X3162E	DIU 2 DIU 2 DIU 2 DIU 2	event	blank or B					
(C)	P34X3163E/ P34X3164E/ P34X3165E/ P34X3166E	DIU 2 DIU 2 DIU 2 DIU 2 DIU 2	event	blank or C					
(D)	P34X3167E/ P34X3168E/ P34X3169E/ P34X3170E	DIU 2 DIU 2 DIU 2 DIU 2 DIU 2	event	blank or D					
(E)	P34X3171E/ P34X3172E/ P34X3173E/ P34X3174E	DIU 2 DIU 2 DIU 2 DIU 2 DIU 2	event	blank, E, '*'					
SA4 CUR	P34C3136A	DIU 2	amps	-0.80 to 50.20					

Table 10.5-3. FDA characteristics: SM 210

	Alorm	Naiaa			Sensed parameter				
Msg text class	filter	Rate	SM param ID	Nomenclature	SET	LO (STATE)	н	Preconditions	
210 +WING	3	3	1	P34X3035E	DCE +WING	E +WING 1 BYPASS		BYPASS BYPASS	DCE +WING MOTOR OPER = NO-OP
			MOTOR OPER 2 1	1	1	DCE MAIN FAIL FLAG = *NOFA or DCE RED FAIL FLAG = *NOFA			
210 -WING 3	3	3 3	1	P34X3036E	DCE -WING	1	BYPASS	BYPASS	DCE -WING MOTOR OPER = NO-OP
	MOTOR OPER	2	1		DCE MAIN FAIL FLAG = *NOFA or DCE RED FAIL FLAG = *NOFA				
210 SADE WING	3	3	1	P34X3178E	SADE-1 +WING MOTOR	1		1	
210 SADE WING	3	3	1	P34X3179E	SADE-1 -WING MOTOR	1		1	
210 SADE WING	3	3	1	P34X3180E	SADE-2 +WING MOTOR	1		1	
210 SADE WING	3	3	1	P34X3181E	SADE-2 -WING MOTOR	1		1	

11.0 SERVICING MISSION 4

11.1 INTRODUCTION

11.1.1 Background

The HST was launched on April 24, 1990, during the STS-31 mission onboard the Space Shuttle Discovery and serviced during the STS-61, STS-82, STS-103, and STS-109 missions in December 1993, February 1997, December 1999, and March 2002, respectively. It was deployed on April 25, 1990, at an altitude of 337 n. mi. with a 28.5° inclination. The HST currently sits at approximately 304 n. mi. altitude with a 28.45° inclination. HST anomaly history since the deployment mission is documented in Appendix D.

This will be an 11-day mission, with a crew of seven, utilizing OV-104 (TBC). The current EVA plan is based on a 5 + 2 scenario.

The HST has been designed for on-orbit servicing and refurbishment to meet, and quite possibly exceed, the anticipated 15-year lifetime of the spacecraft.

11.1.2 SI/ORU Replacement Identification

The following list provides the replacement units for the third servicing mission with additional HST servicing components, tools, and aids:

- a. Planned replacement components and tasks (by priority)
 - 1. RSU (3)
 - 2. WFC-3
 - 3. COS
 - 4. Battery Modules (2)
 - 5. FGS 2
 - 6. STIS Repair
 - 7. ACS Repair
 - 8. NOBL (bays 8 and 5)
 - 9. NOBL (bay 7)
 - 10. SCM
 - 11. Reboost

- b. Contingency components TBD
 - 1. ASIPE Safety Bar Assembly (COS)
 - 2. "Y" Harness Top Cable Assembly (COS)
 - 3. Wide Field Handhold (WFC-3)
 - 4. FGS Handhold Assembly (FGS)
 - 5. Stud Adapter Assembly Locking (4) (FGS)
 - 6. Rate Sensor Spare (RSU)
 - 7. RSU Connector Cap Assembly
 - 8. Electronics Control Unit (RSU)
 - 9. Battery Cap Top Assembly (2) (Batteries)
 - 10. Voltage Improvement Kit ORU Assembly (Batteries)
 - 11. Battery Extraction Tool (Batteries)
 - 12. Spare STIS Card (LVPS-2R) (STIS)
 - 13. Axial SI Handhold Assembly (STIS L-Handle)
 - 14. Attachment Plug Assembly Vent Plugs SSM-ES (NOBL)
 - 15. Spare NOBL Vent Plug Soft Caddy (NOBL)
 - 16. Spare OVP-K (OVP)
 - 17. Aft Shroud Standard Door Repair Assembly (5) (ASLR)
 - 18. Aft Shroud Axial Handle Latch Repair Assembly (2) (ASLR)
 - 19. Aft Shroud FGS Handle Latch Repair Assembly Radial Handle (ASLR)
 - 20. Fuse Module Assembly, Tether Equipped (Fuse Module) (2) (Misc)
 - 21. ORU Assembly, Extended Fuse Module (2) (Misc)
 - 22. Continue

For a list of EVA tools and crew aids, see the EVA Support Equipment List (ESEL).

For a complete list of spares and contingency items reference HST SMEL HST-LIST-010004

11.1.3 STS-125 Payload Configuration

The payload complement for the fifth servicing mission (SM4) includes the SLIC, ORUC, FSS, and MULE, with a mission-specific complement of ORUs and ORIs.

For full details of the payload bay configuration, refer to the HST SSE CSM JSC-TBD.

Figures 11.1-1 and 11.2 show the payload bay configuration. The HST centerline coordinates with respect to the orbiter are at Xo = 1149.20 TBD and Yo = 0.0 TBD while berthed to the FSS. The face of the aft shroud is located at Zo = 496.594. **TBD**.

The HST interfaces to the orbiter payload primary power bus and SSP via the IPCU on the FSS and the remotely mateable/demateable umbilical on the BAPS. The IPCU receives regulated orbiter dc power from two FSS Enhanced Power Distribution and Switching Units (EPDSUs) that feed six DPCs each. The outputs of the IPCU are routed directly to the HST through the FSS main umbilical (J101) nominally or through the deploy/return umbilicals (J107 and J108) in a contingency where the maintenance umbilical cannot be mated.

Drawing 11.1 provides an overview of the power, command, and data systems and their respective interfaces to the orbiter.


Figure 11.1-1. Payload Bay Launch Configuration

11.1-4



Figure 11.1-2. Payload bay launch configuration

11.1.4 Mission Overview

For the SM4, the orbiter will launch on a direct insertion trajectory to the HST orbit, estimated to be approximately 324 n. mi. TBD, with a 28.45° inclination. Once the spacecraft has been properly configured, the orbiter will rendezvous with the HST, grapple and berth the HST to the FSS, connect external orbiter power to the HST via umbilical, and service the spacecraft accordingly.

STS-125 will carry a crew of seven for an 11-day mission, with 5 scheduled and 1 unscheduled EVA days to accomplish mission objectives. On FD1, the SSE will be activated, and during FD2 the SSE and orbiter RMS will be checked out. The orbiter will rendezvous, grapple, and berth the HST on FD3. FD4 through FD8 will be used for performing five EVAs for HST servicing. An overview of the EVA sequence can be seen in Figure 11.1-3. HST deployment activities will nominally take place on FD9, with FD10 reserved as an unscheduled EVA/deploy day. The mission also allows for 1 additional on-orbit day for the shuttle to protect against weather conditions or shuttle contingency delays before landing **TBD**.

Two EVA teams, consisting of two crewmembers each, provide alternating support for each EVA day. One team performs EVAs on FD4, FD6, and FD8, and the other team performs EVAs on FD5 and FD7. FD9 is reserved in case an unscheduled EVA is required to complete the primary servicing mission objectives and to properly configure the HST before deployment. Nominal deployment will take place on FD9. The EVA teams are cross-trained for all servicing activities to accommodate off-nominal situations requiring shuffling of the EVA teams and crewmembers.

The objectives of the STS-125 mission are to replace the COSTAR with COS, replace both battery modules, replace all three RSUs, replace WF/PC2 with WFC-3, replace one FGS, repair STIS, repair the ACS, replace the COSTAR with COS, install SCM, and install NOBL on bays 5, 7 and 8. These upgrades will dramatically increase and restore HST science capability and extend the EPS lifetime.



Figure 11.1-3. Proposed HST SM-4 EVA scenario

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11.1-8



SIGNATURE	DATE	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION LYNDON B. JOHNSON SPACE CENTER HOUSTON, TEXAS			
DSGN B. BOWEN	4/7/93				
DR		STS/HST CMD/DATA/PWR OVERVIEW - SERVICING MISSION			
ENGR P. RIECK					
ENGR					
APP					
APP		HST CSM	DWG NO		
CADS		ALL	11.1		
LTR PCN		93.5 X 34	SHEET 1 OF 5		

11.2 MISSION OPERATIONS

11.2.1 Rendezvous and Capture

Rendezvous and Proximity Operations leading to capture of the HST on FD3 will occur at a time based on HST-orbiter phasing.

Primary prerendezvous activities will include checkout of the RMS, EMUs, the orbiter Ku-band antenna, the PGSC TBC (which will be used to enhance HST PSP bypass communications), the HST SSE, and the ground command system. SSE activation procedures will take place within 2 hours of PLBD opening. The procedures include the activation of the FSS, SLIC, ORUC, and Multi-Use Lightweight Equipment (MULE) FMDMs, as well as the heaters for the MULE, FSS, ORUC, and SLIC. The orbiter MDMs are used for monitoring SLIC thermal telemetry.

HST activities prior to rendezvous include dumping of the onboard recorders, configuration of the 486 flight software, closure of the AD, configuration of the communications system to LGA-direct-TDRS, HGA deactivation and retraction, and FHST and FGS deactivation. Should the AD not close as expected, HST will maneuver to a -V1 VV at orbit-noon attitude, and the orbiter will perform an inertial approach.

The STOCC will configure the HST for rendezvous operations with the orbiter prior to and during the terminal phase of the rendezvous with all operations except load shedding, roll-to-grapple attitude, and PSP-bypass communications configuration performed prior to the orbiter TI burn. The STOCC will have maneuvered the HST to the rendezvous attitude. This attitude is designed to maintain HST power position and minimize plume loading on the SAs at the 1500-foot normal Z braking gate. The rendezvous attitude is determined by these conditions. The Sun in the V1-V3 plane and the +V1 axis pointing down the R-bar at an orbit angle of 59° before orbital noon.

Following the orbiter TI burn, the HST will initiate a nonessential load reduction to minimize the power-negative effects of the roll-to-capture attitude. S-band communications between the orbiter and HST will be established via the orbiter payload communications system in the PSP-bypass mode augmented with the HST *BSA*. Contingency HST direct-TDRSS communications links will remain available throughout the rendezvous and capture. The Ku-band is utilized for radar during rendezvous activities.

Prior to grapple, HST performs a roll maneuver to the capture attitude to align the -V3/+V2 grapple fixture on HST with the orbiter RMS. The roll maneuver will place the -V3 52° out of plane north as measured on the anti-Sun side of HST. The magnitude of the roll maneuver is approximately V1= β -52°, resulting in a roll that ranges from 0° to -104°. HST EPS and SA safe-mode tests are disabled. Shortly after the roll has been completed, the PCS safe-mode tests will be disabled. At that time, a 165-minute backup sunpoint timer will be activated to protect the HST in the event of a rendezvous breakoff. The PSEA keep-alive monitoring will also be disabled and the SADEs turned off.

As shown in Figure 11.2-1, the orbiter approaches the HST using the R-bar approach technique that will minimize propellant usage and potential contamination. The capability exists to perform proximity operations, grapple, and berthing with all of the appendages deployed. However, the HGAs will have to be retracted to maximize clearance for RMS grapple. The flight crew will nominally grapple the HST via the -V2/-V3 grapple fixture and deactivate the PCS by commanding the HST into drift mode.

HST-orbiter communications are reconfigured from the aft LGA to the forward LGA to allow for communications during berthed operations. The HST is berthed to the FSS using the orbiter-mounted CCTV system and the FSS top-mounted CCTV with a berthing target on the aft end of the HST. The three HST berthing pins are engaged by three corresponding FSS berthing latches as the HST is secured to the FSS BAPS. The remotely operated umbilical on the FSS is then connected, utilizing the orbiter MEDS and the A6U panel, and external power is applied to the HST via the orbiter SSP. The orbiter is then maneuvered to the battery charge attitude to top off the batteries. After berthing, HST will require approximately 60 minutes of sunlight on the SAs to recharge batteries to full capacity.

The orbiter GPCs and the ground will receive telemetry from the forward HST LGA through the orbiter PI and PDI via the PSP-bypass mode. Ku-band bent-pipe telemetry will be utilized as a backup in the event that the PDI telemetry is lost. Ku-band bent-pipe telemetry can be displayed only at the STOCC.

The HST SSM and OTA heaters will be activated. At the conclusion of rendezvous day activities the FGEs will be turned back on, and the SIC&DH, WF/PC, STIS, NICMOS, and FOC will be placed in warmup mode.

The orbiter will support HST telemetry at 32 kbps via the PDI (500 kbps, 4 kbps, and 32 kbps when augmented by the HST PGSC) and orbiter operational downlink. The HST will be commanded through the PSP/PI. Ku-band bent pipe service for relaying HST telemetry (0.5, 4, 32, or 1024 kbps) to the ground will be supported, as needed, whenever Ku-band TDRS coverage is available.



(a) R-bar approach and inertial grapple

Figure 11.2-1. HST R-bar approach



Figure 11.2-1. HST R-bar approach (concluded)

11.2.2 EVA Operations

This section describes the EVA operations necessary to accomplish the mission objectives. Detailed steps are provided in the STS-125 EVA checklist.

All commanding activity will be done from the STOCC unless otherwise identified. The IV crew will have the capability to send certain commands from the onboard GPC SPEC and to monitor the status of the certain HST systems during EVA operations and the rest of the mission.

All EVA activities are presented with the assumption that the EV crew has already performed the payload bay setup procedures appropriate to the specific task.

SM4 is currently manifested as STS-125 (TBR) aboard the Space Shuttle Discovery (TBR) to be launched in *August* 2008 (TBR) to a rendezvous altitude of approximately 305 (TBR) n. mi. SM4 is scheduled as an 11-day (TBR) mission, during which the orbiter will rendezvous with, capture, and berth the HST to the Flight Support System (FSS) for commencement of the servicing activities. Following the servicing activities, the orbiter will then unberth and redeploy HST to its mission orbit.

The HST SM4 mission has allocated 5 days (TBR) for scheduled EVA tasks and 2 additional days (TBR) for unscheduled EVA tasks. A scheduled EVA is defined as an EVA planned prior to launch and included in the nominal mission timeline. An unscheduled EVA is defined as an EVA not included in the nominal mission timeline, but which may be required to achieve payload operation success or advance overall mission accomplishments or safe the orbiter. The first unscheduled EVA day has been allocated for enhancing mission success, and the second unscheduled EVA day has been allocated for any payload requirements on the HST redeployment day. None of the scheduled SM4 EVA days require a "Quick Response EVA" capability.

11.2.2.1 Battery Module Changeout

One of the primary objectives of SM4 is the replacement of the two original NiH₂ battery modules. One battery module is located on the inside of each of the SSM bay 2 and 3 doors. The NiH₂ battery modules were designed to be EVA-replaceable. They are aligned with four shear pins, and mounted to each door using fourteen 7/16-inch hex bolts with J-Hook interfaces, and have two EVA handles. One battery module is located on the inside of each of the SSM bay 2 and 3 doors.

Each battery module contains three separate multicell batteries mounted to a single base plate. Each multicell battery is electrically connected to the HST Electrical Power System (EPS) through one power connector (J1, sockets) and one telemetry connector (J2, pins), which is used for heaters, instrumentation, and charge control. During SM3A, six Voltage Improvement Kits (VIKs) were installed—one VIK between each battery telemetry connector and the telemetry harnesses to adjust the charge-control system. Because some telemetry connector contacts pins are tied directly to the battery cells, Gender Changers (GCs) (sockets/sockets) were installed between each telemetery connector and its VIK in the process to reduce the chance that the energized J2 pins could be inadvertently shorted during a future battery module changeout. The VIKs remain in place on the HST harness during the SM4 battery module changeout, but the GCs are left on the old battery module telemetry connectors.

The replacement battery modules are to be charged prior to payload bay closeout at the launch pad and are stowed on top of the SLIC pallet, just aft of the WSIPE. Each are mounted on temperature-controlled mounting plates (Battery Plate Assemblies (BPA)) using the 14 J-Hook interfaces (12 J-hooks and 2 shear pins) and are covered with soft Velcro-secured thermal covers. There are no electrical connections to the battery modules that will be dead faced in the launch configuration, but each J2 telemetry connector will have a gender changer mated to it to convert the pins to sockets. Both the J1 power and J2 telemetry contacts connectors are isolated from the battery voltage by three EVA-operable Battery Isolation Switches (BISs) on the top front of each battery module.

Just prior to each battery module changeout, the STOCC will configure the HST EPS to isolate spacecraft power from the battery harnesses and to ensure that no current flows across the battery connectors and **TBD**. At this point, the IVA crew must also turn on the FSS PCU (DPC) hard-line control to ensure that HST external power remains on

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throughout the battery module changeout. This minimizes the possibility of the loss of external power due to an FMDM failure that would result in current being drawn from the HST battery modules during connector operations.

Removing the old battery modules involves opening the SSM bay door, demating and capping the battery power and telemetry connectors (leaving the gender changers on the battery J2 connectors), releasing the 14 J-Hooks, and removing the battery module from the door. Use of the manual door stay or equipment tethers is desirable to rigidize the bay door because the integral door stay tends to release with minor contact.

Retrieving the new battery modules involves opening the thermal cover on the BPA, releasing 12 of the 14 J-Hooks (the forward-most two of the center J-Hook interfaces are used for passive shear interfaces), and removing the battery module from the BPA.

Installing the new battery module on HST is the reverse of removal of the old battery module, except that the Battery Isolation Switches must be closed after the connectors are mated. Stowing the old battery module is the reverse of retrieving the new battery module from the SLIC.

Both the old and new battery modules are approximately 36.0 by 31.8 by 14.2 inches and weigh 465 pounds (new battery modules) and 463 pounds (old battery modules). The only intended EVA handling interfaces are on the two EVA handles on each battery module.

11.2.2.2 COS/COSTAR Changeout

During SM4, the COSTAR is planned to be replaced with the Cosmic Origins Spectrograph (COS). The COSTAR is located in position 4 (i.e., the -V2/+V3 quadrant) of the HST AS. The COSTAR is approximately 35 by 35 by 87 inches and weighs approximately 671 pounds. The COS is approximately 35 by 35 by 87 inches and weighs approximately 915 pounds.



Figure 11.2-2. Top view (post SM4)

Replacing the COSTAR requires the EVA crewmembers to do the following:

- 1. Open the -V2 AS doors.
- 2. Demate four COSTAR Y-Harness electrical harnesses and one groundstrap from the COSTAR.
- 3. Reposition the COSTAR Y-Harness off the SI guiderail.
- 4. Remove the COSTAR.
- 5. Install the COS.
- 6. Remate the groundstrap and four electrical harnesses to the COS.
- 7. Close the -V2 AS doors.
- 8. Stow COSTAR.

Operationally, the AS doors should not be opened until after the orbiter and FSS have been oriented to prevent direct sunlight from entering the AS and also from impinging on the new and old SI. The demating of the four electrical harnesses will not be done until electrical power has been removed from the COSTAR.

Note: The COSTAR Deployable Optical Bench (DOB) was retracted prior to SM3B. The Solar Arrays will be slewed to the 0° position for EVA access to the -V2 worksite.

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11.2.2.3 FGS-2R/FGS-2 Changeout and OCE-EK Connection

There are three FGSs installed radially within the HST AS: FGS-1 is designated for the -V2 FGS, FGS-2 is designated for the +V3 FGS, and FGS-3 is designated for the +V2 FGS. To improve reliability due to hardware degradation, FGS-1 was replaced during SM2, FGS-2 was replaced during SM3A, and FGS-2 is planned for replacement during SM4. Following FGS-2 replacement, the Optical Control Electronics Enhancement Kit (OCE-EK) will be reconfigured to permit the on-orbit alignment optimization for the FGS-3 adjustable articulated fold flat no. 3 mirror. The OCE-EK was installed on the OCE during SM2 and is located within the OTA Equipment Section (ES) bay C. Each FGS is approximately 67 by 46 by 22 inches and weighs approximately 470 pounds.

- 1. Replacing FGS-2 requires the EVA crewmembers to do the following:
- 2. Rotate the +V2 scuff plate
- 3. Open the +V2 AS FGS doors
- 4. Demate eight electrical harnesses and one groundstrap
- 5. Remove the FGS-3 from HST
- 6. Install the FGS-2R (i.e., the replacement FGS-3)
- 7. Remate the ground strap and eight electrical harnesses
- 8. Close the +V2 AS FGS doors
- 9. Stow the FGS-2 for Earth return

Operationally, the AS doors shall not be opened until after the orbiter and FSS have been oriented to prevent direct sunlight entering the AS and impinging on the new and old FGS. The demating of the eight electrical harnesses is not done until electrical power has been removed from FGS-3. After the FGS-2 changeout and its associated FGS aliveness test, FGS-2 must have its associated OCE-EK connector mated. This entails opening the Bay C door, mating a third set of OCE-EK harness connectors to the OCE at P12/J12, and closing the bay C door. Demating of the OCE-EK electrical harness will not be done until electrical power has been removed from the OCE.

11.2.2.4 NOBL: SSM Bays 5, 7 and 8 Installations

NOBLs are planned to be installed on SSM bays 5, 7, and 8 to prevent further MLI deterioration, to eliminate the MLI as a potential source of contamination, and to reduce the risk of thermal degradation of the observatory. Each NOBL was designed to fit a specific SSM bay door. For SSM bays 5, 7, and 8, each NOBL is "flat," secured to the SSM bay door with four NOBL vent plugs, and grounded to HST via a grounding lug. Each of the NOBLs is approximately 60 by 48 by 1.5 inches and weighs

approximately 9 pounds. Each of the four NOBL vent plugs will be used to secure each bay 5, 7, and 8 NOBL to the HST. Each NOBL vent plug is approximately 5 by 2.5 by 2 inches and weighs 0.3 pounds.

Installing a NOBL on SSM bays 5, 7, and 8 requires the EVA crewmembers to remove two NOBL vent plugs left engaged in the SSM bay 5 door during SM3B; to remove the existing MLI from the door; to retrieve the bay 5 NOBL from the NOBL Protective Enclosure (NPE); and to install it onto the bay door. Installing a NOBL on SSM bays 7 and 8 requires the EVA crewmembers to remove the SM2 temporary MLI repair wire assemblies, attachments, and patches from SSM bays 7, 8, and 9 that were installed during SM2; to remove the existing MLI from the SSM bay 8 door to increase the heat rejection capability in this bay; to remove two NOBL vent plugs and the existing MLI from the bay 5 door to increase the heat rejection capability in this bay; to remove two NOBL vent plugs and the existing MLI from the bay 5 door to increase the heat rejection capability in this bay; to remove the NOBL Protective Enclosure (NPE); and to install them on the appropriate bay doors.

During the NOBL installations, the crew should see generated particulates when handling or contacting the original HST MLI or the SM2 patches and, thus, should try to minimize their handling to minimize the particulate generation. Operationally, the Solar Arrays will be slewed to the 0° position for EVA access to the SSM bays 5, 7, and 8 door worksites.

11.2.2.5 OVP-K Installation

One of the objectives of SM4 is to install an Over-Voltage Protection (OVP) Kit (OVP-K) to address a potential HST main bus over-voltage concern. In the unlikely event that HST entered hardware safe mode with a portion of the Solar Panel Assemblies directly connected to the HST main bus (i.e., in battery bypass mode), the HST bus voltage could rise significantly above the 32.7 volt operating limit. Installing the OVP-K during SM4, combined with operational OVP-K programming, will provide the main bus over-voltage protection (without additional contingency loads) and will protect against selected EPS single-point failures as HST ages.

The OVP-K consists of an Electronics Unit (EU), a P1A Jumper Plug, a P601A Jumper Plug, and a replacement P601 MLI Tent. The combined weight of the OVP-K is approximately 21 pounds (TBR). The EU is approximately 9 by 9 by 1.75 inches (TBR); the P1A Jumper Plug is a 4-inch cylinder, approximately 6 inches long (TBR); the P601A Jumper Plug is a 4-inch cylinder, approximately 8 inches long (TBR); and the replacement P601 MLI Tent is approximately TBD.

Installing the OVP-K requires the EVA crewmembers to do the following:

- 1. Install the P1A Jumper Plug between the CSH P1 connector and the +V2 Solar Array (SA) Diode Box Assembly 2 (DBA2) J1 receptacle.
- 2. Attach the OVP-K Electronics Unit (EU) to HST handrail HR21.

- 3. Mate the OVP-K EU harness to the CSH J2 connector.
- 4. Remove (TBR) the existing P601 MLI Tent (over the +V2 trunnion bay EPS Test Panel).
- 5. Install the P601A Jumper Plug between the Cross-Strap Harness (CSH) P601 connector and the HST P601 receptacle.
- 6. Install the replacement P601 MLI Tent (TBR).

Operationally, installing the P1A Jumper Plug and the OVP-K EU harness will not be performed until power is removed. Since all power cannot be removed from the HST P601 receptacle, both the demating of the CSH P601 and the mating of the P601A Jumper Plug to HST will be a "hot" demate/mate operation.

11.2.2.6 RSU Installation/Changeout

Note: The nominal plan is to install the RSUs and the first (of two) battery modules during the same EVA (EVA no. 1).

HST carries three Rate Gyro Assemblies (RGAs); each RGA consisting of one Rate Sensor Unit (RSU) and one Electronics Control Unit (ECU). The three RSUs (each consisting of two mechanical gyros) are six mechanical gyros, which are packaged in pairs known as RSUs. There are three RSUs installed within the HST AS on the -V3 side: RSU-1 is designated for the center RSU, RSU-2 is designated for the right (+V2/-V3) RSU, and RSU-3 is designated for the left (-V2/-V3) RSU. Each RSU is approximately 13 by 11 by 9 inches and weighs approximately 24 pounds. All three RSUs are planned to be replaced during SM4. Each RSU is approximately 13 by 11 by 9 inches and weighs approximately 24 pounds.

Replacing RSU-1, RSU-2, and RSU-3 requires the EVA crewmembers to do the following:

- 1. Open the HST -V3 AS doors.
- 2. Demate two electrical harnesses on the RSU.
- 3. Remove and stow the RSU.
- 4. Install the replacement RSU.
- 5. Remate the two electrical harnesses.
- 6. Repeat these changeout steps for the second and third RSU changeouts.
- 7. Close the -V3 AS doors.

Operationally, the AS doors shall not be opened until after the orbiter and FSS have been oriented to prevent direct sunlight from entering the AS and impinging on the new and old RSUs, and the demating of the RSU electrical harnesses shall not be done until 10 minutes after its gyros have been powered off to allow for spin down.

11.2.2.7 Soft Capture Mechanism Installation

HST has no onboard propulsion system and currently relies upon shuttle servicing missions for reboosting its orbit altitude. Without orbit reboosts, the orbit altitude of HST would decay until HST finally reenters the earth's atmosphere. To prevent a future uncontrolled reentry of HST, a Soft Capture Mechanism (SCM) will be installed onto HST during SM4 to provide an interface for a future rendezvous and capture of HST with the Crew Exploration Vehicle (CEV) or derivative.

The SCM is a cooperative docking system with two independent docking interfaces (a Low Impact Docking System (LIDS) compatible docking interface (passive portion only) and a Flight Releasable Grapple Fixture (FRGF)), which will enable future rendezvous, capture, dock, and undock of HST.

The SCM will be launched on the FSS Berthing and Positioning System (BAPS), and the SCM will not affect the ability of the shuttle to either berth or deploy HST while the SCM is attached to the FSS or affect the ability of the shuttle to either deploy or reberth HST while the SCM is attached to the HST. Since the SCM is a passive system design, it requires no electrical power, thermal control, or Command and Data Handling (C&DH) interface with HST. The SCM is approximately TBD by TBD by TBD inches and weighs approximately 250 pounds.

Installing the SCM requires a free-floating crewmember to do the following:

- 1. Translate to the FSS BAPS.
- 2. Install a battery for the SCM indicator lights.
- 3. Drive a 7/16-inch hex bolt approximately 125 turns.
- 4. Verify (via indicator lights) that the SCM has released from FSS and engaged to HST.
- 5. Remove the battery.

11.2.2.8 STIS Repair

During SM4, STIS Side 1 will not be repaired, but a replacement of the Main Electronics Box 1 (MEB-1) Low Voltage Power Supply-2 (LVPS-2) card will be made to restore functionality to the STIS Side 2; STIS Side 1 will not be repaired. Following a successful replacement, the STIS MEB-1 LVPS-2 card, in addition to the LVPS-2 card replacement, the STIS Cooling System (SCS), consisting of the STIS Thermal Interface Kit (STIK), the STIK Mounting Bracket (SMB), and the STIK Extended Radiator (STIKER), will be installed onto STIS to provide additional cooling of the internal STIS detectors inside the STIS. This will permit STIS to return to scientific productivity on Side 2. The STIS Repair task will be performed on STIS without removing STIS from the HST AS and will permit STIS to return to scientific productivity on Side 2.



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Figure 11.2-3. STIS in HST Bay 1

The original STIS MEB cover is approximately 28 by 27 by 7 inches and weighs approximately 10 (TBR) pounds. The MEB Replacement Cover, including latches, is approximately 28 by 27 by 6 inches (TBD) and weighs approximately 36 pounds (TBR). The STIS MEB-1 LVPS-2 card and its nearly identical replacement, LVPS-2R, are approximately 10 by 16 by 1 inch and weigh approximately 5 pounds. The STIK Mounting Bracket (SMB) is approximately 25 by 2.5 by 6 inches and weighs approximately 7 pounds. The STIK is approximately 40 by 25 by 40 inches in its unfolded position and weighs approximately 53 pounds. The STIKER is approximately 39 by 8 by 35 inches and weighs approximately 15 pounds.

Repairing the STIS requires the EVA crewmembers to do the following:

- 1. Maneuver to the HST +V2 AS doors.
- 2. Open the +V2 AS doors.
- 3. Remove and stow the STIS MEB-1 Cover clamp.
- 4. Remove and stow the STIS Axial SI Handle.
- 5. Retrieve and install the STIS Fastener Capture Plate (FCP) onto the STIS MEB-1 cover.
- 6. Remove and stow the STIS MEB-1 cover (with the FCP attached).
- 7. Remove and stow the STIS LVPS-2 card.
- 8. Retrieve and install the STIS replacement LVPS-2 card.
- 9. Retrieve and install the replacement STIS MEB-1 Cover.
- 10. Retrieve the STIK Mounting Bracket and install the SMB onto the STIS.
- 11. Retrieve the STIK and install it the STIK onto the newly installed SMB.
- 12. Retrieve the STIKER and install the STIKER it onto the STIK saddle plate.
- 13. Close the +V2 AS doors.

Operationally, the AS doors shall not be opened until after the orbiter and FSS have been oriented to prevent direct sunlight from entering the AS or impinging on the SCS internally mounted STIK/STIKER hardware.

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11.2.2.9 WFC3 Changeout/Installation

During SM4, the Wide Field/Planetary Camera (WFPC) 2 is planned to be replaced with the Wide Field Camera 3 (WFC3), which will provide HST with new wide field and wide spectral range imaging capabilities. The WFPC2 is located in the -V3 radial SI bay of the HST AS. The WFPC2 is approximately 94 by 83 by 31 inches and weighs approximately 620 pounds. The WFC3 is also approximately 94 by 83 by 31 inches and weighs approximately 889 pounds.

Replacing WFPC2 with WFC3 requires the EVA crewmembers to do the following:

- 1. Install an EVA handhold onto WFPC2.
- 2. Release a Blind-Mate connector mechanism, ground strap, and A-Latch.
- 3. Remove the WFPC2 from HST.
- 4. Install another second EVA handhold onto WFC3.
- 5. Install the WFC3 into the HST.
- 6. Re-engage the A-Latch, ground strap, and Blind-Mate connector mechanism.
- 7. Open two WFC3 vent valves.
- 8. Stow WFPC2.
- 9. Remove and stow both EVA handholds.

Operationally, the WFPC2 shall not be removed from the HST until after the orbiter and FSS have been oriented to prevent direct sunlight from entering the AS and also from impinging on the WFPC2 and WFC3; and the Blind-Mate connector mechanism shall not be retracted until electrical power has been removed from the WFPC2.



Figure 11.2-4. WFC3 install location (replaces WF/PC2)

11.2.2.10 ACS Installation

Note: The nominal plan is to install the ACS and ESM and perform PCU cleanup, respectively, during the same EVA.

For more detailed information concerning latching, go to the generic axial SI changeout operation in Section 3.4.2 of this document. The ACS EVA worksite is shown in Figure **TBS**.

Prior to the beginning of the EVA changeout, the BAPS is rotated to the FOC/ACS changeout position (-V2 Forward) using the switch on the A6U panel. Proper position is verified by lining up the AS markings on the HST with that on the FSS BAPS. The STOCC will thermally condition the FOC the night before, and the COSTAR deployable optical bench will have been already retracted (~ 1 month prior to launch).

The EVA begins by deploying the aft fixture on the SAC. The IV crewmember activates the worklights and SI indicator from the SSP. Next, the -V2 AS doors are opened, if not yet completed. Figure **TBS** reveals the ACS detailed worksite.

As the -V2 AS doors are being opened, the IV calls the MCC to power down the FOC. The STOCC normally performs the powerdown; however, the crew has the capability to issue these commands, if necessary. In addition to the powerdown of FOC, the STOCC will also reconfigure the NASA Standard Spacecraft Model-1 (NSSC1) Flight Software (FSW) to support the new ORI.

Once the doors are opened to the fully opened position at the bump stop and tethered open, and upon verification of FOC powerdown, the four FOC power (J1, J2) and signal data (J3, J4) connectors are disconnected.

Next, the FOC purge line will be demated from the FOC by gently squeezing on the wire-locking clip to release it and pulling the fitting away from the FOC. The purge line will then be mated to the purge line stowage fitting on the HST aft bulkhead.

Then, the FOC ground strap will be disconnected and temporarily stowed. Figure TBD shows the floor overview and detail of the AS for FOC changeout.

Next, the A latch is released and the A LATCHED indicator lights verified. After the A latch is released, the B latch cover is removed and the B latch is released. The B latch is shown in Figures TBD and TBD. The crew should verify that the A SEATED indicator lights illuminate when the FOC A latch is unseated. Next, the FOC is lowered until the bottom guiderail is contacted to position the FOC for removal along the OTA/SSM guiderails. The B READY indicator lights should be monitored because they will illuminate during the extraction from the B ready-to-latch position. The FOC is then pulled out of the axial cavity via the guiderail system and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with the non-RMS crewmember providing guidance support. This allows for any repositioning necessary to ensure a level extraction of the FOC. Once removed, the FOC is stowed on the SAC aft fixture and secured with one pip pin. Figure TBD shows the HST -V2 AS (empty bay) overview.

The SAC ASIPE door is opened, and the ASIPE and NCC ESM groundstraps are disconnected and stowed. The A latch is disengaged, and the A LATCHED indicator lights should illuminate. Then, the B latch MLI cover is opened, and the B latch is released. The A SEATED light should be monitored for ACS and is then translated back to the HST worksite for installation. ACS is inserted into place along the OTA/SSM guiderails until the B READY indicator lights are extinguished. The B latch is then engaged and torqued. ACS rises as the B latch is engaged, and the A and C latches are then seated. The A SEATED indicator lights are extinguished when the B latch is torqued. The B latch cover is then closed. Next, the A latch is closed, torqued, and verified to be latched by noting that the A LATCHED indicator lights have extinguished. Next, the ACS groundstrap is attached, and the four power and data connectors are remated using the identification and clocking markings on the connectors.

Upon completion of the ACS installation, the IV gives MCC a GO for ACS aliveness test. The STOCC performs the aliveness test to check the electrical path to the instrument.

The FOC is removed from the aft fixture and installed into the ASIPE until B READY indicator lights extinguish. The A SEATED lights will be extinguished when B latch torquing is complete. After the B latch is fully engaged, the A latch is engaged and torqued. The A latch is verified latched with the extinguishing of the A LATCHED lights. Finally, the ASIPE lid is closed and latched.

11.3 NEW HARDWARE FOR SM4

11.3.1 Cosmic Origins Spectrograph (COS)

11.3.1.1 Overview

The COS shall replace the COSTAR in the (-V2, +V3) axial bay of the HST aft shroud. COS is a fourth generation HST axial science instrument. COS is an optical sensing instrument designed for high throughput, medium resolution (R = 20,000) spectroscopy of point sources, allowing the efficient observation of numerous faint extragalactic and galactic ultraviolet (1150 to 3000) targets. The primary science objectives of the mission are the study of the origins of large-scale structure in the universe, the formation and evolution of galaxies, and the origin of stellar and planetary systems and the cold interstellar medium. The instrument has been designed with maximum effective area as the primary constraint and will provide more than an order of magnitude gain in sensitivity over previous HST instruments in this wavelength region. COS will complement and extend the suite of HST instruments, ensuring that HST maintains a powerful UV spectroscopic capability until the end of its mission.

COS has been designed and tested for launch, orbital replacement, and return on the Shuttle. COS will be installed as one of the four axial science instruments on the HST during the Fourth Servicing Mission (SM4) by the space shuttle astronauts tentatively scheduled for September 2008. On board the shuttle, the COS instrument will be carried in an Axial Scientific Instrument Protective Enclosure (ASIPE) mounted to the ORUC. COS will have an axial SI Handhold that will remain on the instrument after installation into the COSTAR bay. The COS will be unpowered in the payload bay as there are no electrical provisions in the SIPE other than grounding. The mechanical interfaces between the COS and SIPE will be the same type latching system used to hold the COSTAR instrument during the HST first servicing mission and its cousins, NICMOS, STIS, and ACS during the second and third servicing missions.

COS leverages the design and technology of previous instruments. The optical bench is the GHRS instrument bench with modifications for use on COS. The NUV detector is a spare MAMA from the STIS program. The FUV detector is an evolutionary improvement of the detectors flown on FUSE. The main electronics and mechanisms have heritage going back to COSTAR, STIS and NICMOS.

The total weight of the instrument in the orbiter bay is 851 pounds. The instrument envelope is a volume measuring 86.1 by 34.8 by 34.8 inches.

The prime contractors for COS are Ball Aerospace and the University of Colorado/CASA. The principal investigator for COS is Dr. James Green of the University of Colorado.



Weight: 852 pounds; Dimensions: 3 feet high by 3 feet deep by 7 feet wide

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Figure 11.3-1. Two views of the Cosmic Origins Spectrograph (COS)

11.3.1.2 Structural Description

The COS structural subsystem consists of two major components: fixed truss optical bench and enclosure. COS is installed into the HST using the A, B, and C fittings. This interface is common to all the Axial SIs. These fittings support the COS enclosure or outer skin of the instrument. The enclosure provides the interfaces for the COS, including the connector panel, astronaut handles, and other necessary guides, rails, and handling points. Inside the enclosure, the electronics and thermal shelf are supported from enclosure structural members. The truss assembly is kinematically mounted to the A, B, and C fittings, using X, Y, and Z fittings.

Note: The truss assembly is often referred to as the optical bench and these two terms are used interchangeably.

The mechanical interfaces between the COS and SIPE will be the same type latching system used to hold the COSTAR instrument during the HST first servicing mission and its cousins, NICMOS, STIS, and ACS during the second and third servicing missions.

COS Fixed Truss Optical Bench

The optical bench is the primary load bearing structure in the instrument. The bench is a graphite-epoxy truss structure consisting of trapezoidal tubing longerons, square tubing cross members, and plate bulkheads. Truss members are joined with bonded finger plates. Bonded-in metal fittings provide load points for interfacing with other components of the instrument. The X, Y, and Z fittings are the structural interface between the enclosure and the bench. The X fitting provides three-axis constraint. The Y axis fitting constrains lateral motion but allows axial motion through a sliding spherical bearing. The Z fitting constrains rotation about the X fitting.

These fittings are not to be considered mechanisms but are structural mounts or joints that kinematically constrain the bench to the enclosure's SI latch hard points. When installed in the HST, these fittings "damp out" any thermal distortions to maintain the precision alignment of the optics. Bulkheads of the bench also support the optical elements and detectors as well as providing light isolation barriers.

COS Enclosure

The COS enclosure is a rectangular box honeycomb structure with one longitudinal edge chamfered and consists of forward and aft bulkheads with joining panels. The enclosure provides the mechanical interface to the SIPE/HST at the A, B, and C fittings, internal supports for avionics boxes, and astronaut handles, as well as being a contamination and light shield for the sensitive instrument.

The electrical connector interface panel is on the top panel, and the enclosure is vented through openings in the aft bulkhead. These vents are sized to the pressure profiles of the orbiter ascent and descent rates. The enclosure is fabricated from aluminum honeycomb with graphite-epoxy face sheets and is designed to withstand launch and landing loads and to have axial stiffness of 75,000 lb/in. The panels are not vented but are designed to withstand the pressure differentials of ascent/descent. The Main Electronics Box (MEB) is supported by an integral aluminum structural "saddle" that bolts into the enclosure. Two remote interface units (RIUs) attach to the MEB aft side plates. The FUV Detector Electronics Box (DEB) attaches to the enclosure top plate.



Figure 11.3-2. COS access panels

SI Latch Fittings

The instrument half of the A, B, and C latch fittings bolt respectively to the X, Y, and Z fittings on the enclosure bulkheads. Optical bench flight loads are carried directly into the SIPE latches. The enclosure carries only the flight loads from its own weight and the weight of the MEB. A static compressive preload is applied by the SIPE latches between the A and B fittings.



11.3.1.3 Optical Subsystem

Figure 11.3-3. TBD

The COS optical subsystem provides two spectral channels, the NUV (Near Ultraviolet), and the FUV (Far Ultraviolet). Each channel consists of a detector and the optical elements necessary to process the input image received from the HST and focus the processed image onto one of the detector apertures. The NUV channel uses one Multi-Anode Micro channel Array (MAMA) detector. The NUV detector system is a spare from the STIS program. The spare STIS detector is a solar blind, 1K x 1K pixel (2K x 2K in high resolution mode), MAMA detector that can cover a band of 1650 to 3100 Angstroms. The FUV channel uses a large format, high-resolution windowless, crossed delay line microchannel plate detector system, and is sensitive to ultraviolet wavelengths from 400 to 2000 Angstroms. The FUV detector system consists of a Detector Vacuum Assembly (DVA) and a Detector Electronics Box (DEB) that contains all the support electronics. The DVA contains the light sensitive microchannel plates inside of a self-contained vacuum system, sealed by a flight operable door mechanism. The NUV and FUV detectors convert light energy to electronic signals that are processed and transmitted to Earth.

The COS uses five electromechanical assemblies in the optical train to focus, position, and condition incident light energy to the optical detectors when operating in the HST. The mechanisms will be unpowered and nonoperational during prelaunch, launch, and on-orbit operations of the shuttle. Their functions are strictly for alignment purposes and are not safety critical. In this manner they are treated as any other structural mass.

FUV Detector Door Mechanism (1) – Maintains vacuum on FUV detector during ground processing. Door is opened on orbit after ambient pressure is reduced to required level.

An ion gauge is used to provide this measurement (mass of this mechanism included in detector vacuum assembly mass (45 lb)).

External Shutter Mechanism (1) – Developed by Swales based on the STIS external shutter, this mechanism is designed to seal the opening in the enclosure that allows light into the instrument during operation. (2.6 lb)

Aperture Mechanism (1) – Moves the science slit and the Bright Object Protect (BOP) slit into the optical path. This mechanism also contains the mirrors to relay the light from the Dt or PtNe calibration lamps into the optical path. (3.5 lb)

Optics Select Mechanisms (2) – The OSM mechanisms are used to position optics to select either the FUV or NUV channels, wavelength regions, and to send the observed light to the detectors. These mechanisms are mounted on the main optical bench.

OSM-1: Moves a one mirror and three gratings (NCM-1, G130M, G140L, G160M) about one axis of rotational motion rotational motion (rotary stage assembly) and along one axis of linear motion (linear stage assembly) to provide a focus adjustment. (58 lb)

OSM-2: Moves a one mirror and four gratings (TA-1, G185M, G230L, G225M, G285M) about one axis of rotational motion rotational motion. This mechanism is the same as the rotary stage assembly of OSM-1. (18.3 lb)

Light entering the instrument passes through one of the slits in the aperture block, which is positioned by the Aperture Mechanism. The light is then corrected for spherical aberration by the OSM-1 optics and sent either to the FUV detector by the gratings or to the NUV channel by the NCM-1 optic. In the NUV channel, the light is then collimated by the NCM-2 optic. The collimated light strikes one of the plane gratings on OSM2. The dispersed light is interrupted by three concave mirrors (NCM-3a, NCM-3b, and NCM-3c) that focus the spectrum onto the flat surface of the NUV detector. The OSM-2 TA-1 mirror is used to provide rapid target acquisition.

Light from calibration lamps is directed by fold mirrors to the entrance slit in the aperture block in place of the HST input image. There are two types of calibration lamps used in the COS instrument optical system:

- 1. Deuterium lamps (2 each)
- 2. Platinum Neon lamps (2 each)

11.3.1.4 COS Electrical Subsystem

The electronics will not be powered in the shuttle payload bay as there are no electrical provisions in the SIPE other than grounding.

Power is provided by the HST once installed.

Electronic Subsystem Component Characteristics						
Component name (quantity)	Size (in.) H by W by L	Weight (lb)	Mounting location			
Main electronics box (2) with support structure	H by W by L (each)	156.9	Saddle bracket part of enclosure			
Remote interface unit (2)	25 by 7 by 8 (each)	4.9	Aft surface of MEB			
Heater Power Distribution Unit (HPDU) w/support structure	H by W by L	7.6	Saddle bracket part of enclosure			
MAMA	H by W by L	41.5	Bulkhead 6 of optical bench			
Cal Platform Assembly	H by W by L	28.9	On "top" of bulkheads 2 and 3			
Ion Gauge	H by W by L	2.0	"Bottom" side of aft thermal shelf			
Detector Vacuum Assembly (DVA) with support structure	13.2 by 15.5 by 13.9	51.8	Bulkhead 2 of optical bench			
Detector Interface Box (DIB) with support structure	15.3 by 16 by 13.9	54.4	Forward			



Figure 11.3-4. COS electrical block diagram

11.3.1.5 COS Thermal Subsystem

The COS uses passive and active thermal control to maintain proper on-orbit temperatures. In the payload bay, the active controls will be nonoperational. The thermal environment will be established by the SIPE. The COS passive thermal control system consists of MLI and selected surface finishes. The external surfaces of the instrument are graphite-epoxy. Tape has been added to the inboard panels. All MLI will be in the interior of the enclosure. All MLI is vented and grounded. The MLI design will be assessed in accordance with NSTS 22648 for compliance to flammability requirements. There will be no energized systems in the COS while it is installed in the payload bay or during installation in the HST.

After installation into the HST, the COS has an active and passive thermal control subsystem to maintain the various components within acceptable operating limits. Active heaters, thermal straps, thermal blankets, surface finishes, heat pipes, and special radiators are utilized in this subsystem. The optical bench area is maintained at $23 \pm 1^{\circ}$ C to minimize thermally induced structural deformations that would degrade optical performance. The electronic compartment temperature range is from +15° to +22° C, depending on spacecraft and instrument operating conditions.

The heat dissipated by the NUV detector, FUV detector and its electronics box is transported to a transfer plate mounted on an external panel of the enclosure by three heat pipes. A thermal strap is used to provide a conductive thermal path between the FUV detector and its heat pipe. Two additional heat pipes transport the heat from the transfer plate to a radiator on the outer panel of the enclosure.

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The main electronics and DEB have aluminum radiator plates that are black anodized for operating temperature control. The instrument exterior finish is the natural black of the graphite-epoxy except for radiator surfaces.



Figure 11.3-5. COS Thermal Control System Components

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11.3.1.6 COS Installation

The COS installation will begin by opening the -V2 AS doors, during the door opening the EV crew will inspect the handrails and attachment fittings for flaking paint and install handrail covers as needed. The EV crew will begin opening the -V2 AS doors by releasing the latches and finally the handle latch of the door to keep the doors closed and prevent direct sunlight from entering. During actuation of the Hatch latches the EV crew will apply braycote to any exposed threads to ease crew usage of the latches during COS installation. The EV crew will open the doors to their full open position and may tether them off at their discretion.

After opening the doors the EV crew will begin by demating the COSTAR Y-harness and connectors and repositioning the Y-harness out of the COS/COSTAR Translation path. Once the Y-harness has been disconnected and repositioned the EV crew will disconnect the HST ground strap, and then release of the A latch followed by the B latch. Then the COSTAR is removed from HST and temporarily stowed on the ORUC Aft Fixture.

Prior to removing the COS from the ORUC the crew will inspect the COSTAR bay to ensure there is no MLI or contaminants. The crew will only open and remove COS from the ORUC ASIPE when it will not be exposed to direct sunlight and is ready to be installed. The EV crew will then remove the ASIPE Ground strap and release the A and B latches respectively. The EV crew will then remove the COS from the ASIPE and close the lid to preserve the ASIPE thermal environment. COS will be translated to the HST and installed in the guiderails until the "B READY" indicator light verifies it is position for engagement of the B latch. If needed the crew is able to deploy a B-Latch alignment aid to ensure the B latch is properly captured. After the B latch is engaged the crew will engage the A latch, and attach the HST Ground strap to the COS. Then repositioning the COSTAR harness as needed and attaching the appropriate connectors to the COS.

After all connectors are mated the crew will inspect the AS area for any contaminants and document the COS installation. The crew will then close the -V2 AS doors by first closing the right door first and then the left door to ensure the door shear plates are mated and the doors are fully closed. The E crew will the latch the doors closed by first actuating the handle latch first to ensure the doors remain closed. Then they will engage the remaining door latches to seal the AS.

Once the AS is closed the crew will open the ASIPE and seat the COSTAR for return by engaging the B-latch and then the A-latch. Then attaching the COSTAR ground strap, and finally closing and latching the ASIPE lid.



Figure 11.3-6. COS changeout

11.3.2 Wide Field Camera 3

11.3.2.1 Overview

Wide Field Camera 3 (WFC3) is a "facility class" fourth-generation instrument for HST, replacing WF/PC-2 in the -V3 radial bay. It was designed and funded to ensure that HST has a high-quality imaging capability throughout the rest of the HST mission. It is built on a low-cost paradigm that reuses existing designs and parts wherever possible. WFC3 is configured as a two-channel instrument. The incoming beam from the HST is directed into the instrument using a pickoff mirror. It is then corrected for HST's primary spherical aberration by using a two-element optical system to reimage and correct the pupil plane. The corrected beam is then sent to either a near-UV/visible UV-VIS channel or a near-IR channel.



Views of WFC3 Instrument Assembly

Figure 11.3-7. WFC3

Scientific areas that are expected to benefit the most from WFC3 include searches for galaxies at redshift up to z~10; the study of the physics of star formation in distant and nearby galaxies; accurate determination of the baryonic mass by detecting stars at the limit of the hydrogen-burning sequence and extra solar Jupiter-like planets; and study of planetary objects in the Solar System.

WFC3 was developed and built via an Integrated Product Team. That team consists of GSFC and Ball Aerospace as the prime partners, with numerous subcontractors. The instrument manager for WFC3 is Jackie Townsend and the instrument scientist is Dr. Randy Kimble of GSFC (both as of this writing).

11.3.2.2 Structural Description

In the orbiter payload bay, the WFC3 instrument will be carried in the Super Lightweight Interchangeable Carrier (SLIC) mounted in the WSIPE. The mechanical interfaces between WFC3 and the WSIPE will consist of two guiderails and three latches (A, B, C). Total weight of the instrument in the orbiter bay is approximately 900 pounds. The instrument envelope is a volume measuring 25.99 by 74.520 by 87.875 inches.

The WFC3 mechanism subsystem consists of six major components: UVIS Shutter Mechanism, Selectable Optical Filter Assembly (SOFA) Mechanism, Channel Select Mechanism, UVIS Corrector Mechanism, IR Filter Wheel Mechanism, and IR Corrector Mechanism.



Figure 11.3-8. WFC3



Figure 11.3-9. Wide Field Camera 3 (WFC3) - interior configuration

Wide Field Camera 3

UVIS Shutter Mechanism

The shutter mechanism is designed to provide controlled exposure durations for UVIS detector observations. The mechanical drive for the shutter is identical to the ACS shutter drive. Consequently, it includes a DC torque motor with redundant windings and redundant resolvers. The exposure uniformity of the shutter is 10 msec with exposure times of 500 msec and 700 msec up to 60 minutes.

SOFA Mechanism

The SOFA mechanism is a refurbished WF/PC-I mechanism. The SOFA contains 12 filter wheels with 4 filters and 1 open slot in each wheel. The step size for the SOFA drive motors is 0.5 degree per step giving an alignment accuracy of the filters to within 0.031 inches.

Channel Select Mechanism

The CSM has two functions: to position a mirror into the optical path to reflect light into the IR channel (the mirror is out of the path for UVIS operation), and to position a diffuser in the IR path to reflect light from the tungsten calibration source, through the IR optical path, into the IR detector. The nominal optical positions, IR and UVIS, are spaced 55° apart. The CSM has two branches on one arm that rotate in synchronization with each other via a 30° stepper motor with a 180:1 ratio. There are redundant micro switches used to limit the motion if the optic overshoots the nominal

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positions. These limit switches turn off the power of the mechanism if the optic rotates past the nominal position by a few degrees. There are also software stops in between the nominal positions and the limit switch positions. For the rotation to hit the hard stops at either position, the mechanism arm would have to pass the nominal commanded position, the software stops, and the limit switches. CSM positioning is repeatable to within +3 motor steps. The resolution of the movement is better than 10 arcmin per motor step. There are redundant resolvers on the motor providing position feedback. The CSM is capable of switching between the UVIS and IR positions in less than 60 seconds.

UVIS and IR Corrector Mechanisms

The corrector mechanisms are designed to provide tip, tilt, and focus adjustment of the exit pupil onto the primary corrective optic for each optical channel. Other than the optics (UVIS M1 and IR M2) attached to their output shafts, each corrector mechanism is identical to each other. All mechanism drives (Inner Wobbler Motor, Outer Wobbler Motor, and Linear Actuator) are proven technology from ACS and have redundant windings. There are two mechanism states, nutating (tip/tilt) and linear (focus). The nutating mode systems have an output resolution of 2.2 arcsec, and the linear mode has 0.001mm output resolution.

IR Filter Wheel Mechanism

The filter wheel mechanism is designed to provide different observation bands for the IR channel by positioning any one of 16 spectral elements in the optical path.

11.3.2.3 Optical Subsystem - Optical Bench/Structure Subsystem

The WFC3 Optical Bench is used to support all optical elements, including the pickoff mirror. It is kinematically mounted to the instrument enclosure by struts attached to the radial latch system. The WFC3 Optical Bench is a composite material structure composed primarily of honeycomb panels.



Figure 11.3-10. Wide field optical diagram

11.3.2.4 Electrical/Electronic Subsystem

The WFC3 will be unpowered as stowed in the WSIPE with no electrical connections to the SLIC other than a grounding strap. The WFC3-to-WSIPE ground strap becomes the WFC3-to-HST ground strap when WFC3 is installed onto HST. The HST provides power and commands to WFC3 on mission control's GO.

Electronic Subsystem Component Characteristics				
Component name (quantity)	Size (in.) H by W by L	Weight (Ib)	Mounting location	
Low Voltage Power Supply (LVPS) (2)		29	Behind the radiator	
Remote Interface Unit (RIU) (2)	25 by 7 by 8		Behind Enclosure panel	
UVIS CCD Electronics Box (CEB)		26	On the side of the instrument	
Main Electronic Box (MEB) (2)		47	On the side of the instrument	
Detector Electronics Box (DEB)		24	On the side of the instrument	
Ancillary Boxes: SOFA Relay Heater Power Distribution Box Cal Lamp Controller Thermal Controllers (4)			Enclosure panel	

11.3.2.5 Thermal Subsystem

The WFC3 uses passive and active thermal control to maintain proper on-orbit temperatures. In the WSIPE, the active controls will be nonoperational. The required thermal environment will be provided and maintained by the WSIPE. The WFC3 passive thermal control system consists of the Detector Radiator, Electronics Radiator, and MLI Blanketing.

During the Servicing Mission, the entire instrument is unpowered and maintained nearly isothermal by the WSIPE enclosure. Once removed from the WSIPE enclosure, the instrument can survive 2 to 4 hours in the orbital environment before survival limits are reached.

After installation into the HST the WFC3 has an active thermal control capability that joins the passive component in thermally maintaining the various components within acceptable operating limits: <30° C for MEB, <30° C for LVPS, and <30° C for CEB/DEB. The WFC3 active thermal control system consists of heat pipes (containing anhydrous ammonia), the Thermo-Electric Cooler for IR Electronics (TEC FIRE), thermo-electric coolers (TECs), and heaters. The optical bench area is maintained at $0^{\circ} \pm 1^{\circ}$ C for stability while the cold plate is maintained at $-5^{\circ} \pm 2^{\circ}$ C.



Figure 11.3-11. WFC3 schematic

11.3.2.6 WFC3 Installation

The EV crew will begin the removal of the WFPC2 by FGS Handhold onto the guide studs used to install the WFPC2 and tightening the handhold zip nuts onto the guide studs. The EV crew will then release the blind-mate connector by turning the 7/16-inch hex head interface CCW 5.3 turns and releasing the radial ground strap from the WFPC2. Finally crew will release the A-latch about 22.5 turns CCW to free the camera. The crew will then use the FGS Handhold to remove the camera via the HST guiderails, and ensuring that nothing contacts the camera radiator or the pickoff mirror housing. The crew then temp stow the WFPC2 on the ORUC Aft Fixture.

Prior to removing the FC3 from the SLIC the crew will inspect the radial port to ensure that the bay is ready to receive the WFC3 and that no contaminants are present. The crew will then open the SLIC WSIPE lid by releasing the three lid latches. They EV crew will then retrieve the WF Handhold from the forward fixture and installing it on the WFC3 guide studs and hand tightening the zipnuts. Additionally the crew will open WFC3 detector vent valves, the volume behind the vent valves was evacuated on the ground and this ensures that the detector volume is maintained at vacuum.

The EV crew then removes the WSIPE ground strap and releases the A-latch to remove the WFC3. During the Removal and installation of the WFC3 the crew ensures that no inadvertent contact is made with the WFC3 radiator and pick-off mirror. The crew then slides the WFC3 from the WSIPE and closes the Lid to maintain its thermal environment. Next the crew installs the camera into the radial bay by engaging the camera on the guiderails and sliding the camera into position and seating it at the end of travel. The crew will then partially engage the A-latch past 15 turns to ensure the camera is being drawn into the bay, and then rotating the A-latch to hard stop. The EV crew will then attach the ground strap and engage the blindmate mechanism. The crew will complete the installation of the WFC3 by removing the WF Handhold and stowing it on the ORUC Forward fixture for return.

To cleanup after the installation of the WFC3 the crew will remove the WFPC2 from the Aft Fixture and insert it into the WSIPE. They will engage the WFPC2 guiderails and fully seat it at the end of travel, prior to actuating the A-latch and installing the WSIPE ground strap. They will then remove the FGS Handhold from the camera, and close and lock the WSIPE lid. Finally they will stow the FGS Handhold on the forward fixture and then stow the aft stowage fixture.



Figure 11.3-12. WFC3 install

11.3.3 Space Telescope Imaging Spectrograph Repair

11.3.3.1 Overview

The complete STIS Repair is broken into two primary tasks, replacement of the failed LVPS-2 board in MEB-1 and the installation of a cooling system.

LVPS-2 Replacement

In 2001 an electrical short occurred in the STIS Side 1 Hold Bus. The STIS Side 2 Mechanism +5 V DC-DC converter failed on August 3, 2004. STIS functionality will be restored during SM4 by the replacement of the low voltage power supply board (LVPS-2) that contains the failed Side 2 Mechanism converter. Side 1 will not be repaired.

Addition of Cooling System for MAMA Detectors

The STIS instrument that was installed on HST on a previous servicing mission has a copper radiator plate on the STIS aft bulkhead to provide cooling to the MAMA detectors. This radiator views the HST aft bulkhead and therefore provided marginal but acceptable cooling for the MAMA detectors prior to SM4. The SM4 mission includes

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replacing 2 HST science instruments (COSTAR with COS and WFC2 with WFC3). The new instruments are higher-power and as a result will raise the temperature of the Aft Shroud environment for STIS. The temperature increase will result in the STIS MAMA detectors approaching their high temperature limits, resulting in

- a. Degraded Signal/Noise ratio
- b. Reduced lifetime of the detectors
- c. Possible need to cycle between the two MAMA detectors during hot orbit/attitude periods to prevent reaching the action limit

To improve the cooling of the ACS, COS, and STIS instrument, an HST extended radiator named the Aft Shroud Cooling System (ASCS) was designed and manufactured for SM4 installation. The ASCS included a STIS Thermal Interface Kit (STIK) to transport heat from the STIS aft bulkhead to a saddle to which the ASCS CPL pump would mate.

Due to limitations of EVA time and launch mass of SM4 and evaluation that the ACS and COS instruments are not in need of supplemental cooling, it was decided that the ASCS will not be flown on SM4. To provide desired cooling of the STIS instrument, a new cooling system will be developed for the STIS that would require fewer mission critical resources.

11.3.3.2 Structural Description

The structural is incorporated as part of the Thermal Subsystem (see below)

11.3.3.3 Optical Subsystem

N/A

11.3.3.4 Electrical Subsystem

STIS Repair hardware shall not require power, command, or telemetry interfaces to the HST or SSE with the exception of the LVPS-2R card when installed in the STIS MEB.

11.3.3.5 Thermal Subsystem

The HST Aft Shroud thermal environment will become several degrees warmer following the COSTAR changeout to the new, higher-power COS instrument during SM4. A cooling system for the ACS, COS and STIS instruments had been developed for installation on SM3B but was not installed due to insufficient EVA time. Improved thermal modeling and on-orbit data have shown the ACS and COS instruments do not require additional cooling, but that the STIS instrument would have improved science performance with at least 7° C cooling.

The SCS is designed to conduct heat from a radiator on the STIS aft bulkhead and radiate it to the +V2 aft shroud door, which is an unblanketed thermal sink. To improve the EVA installation of the SCS, it will be installed in three pieces:

	Volume	Weight (Ib)
STIK Mounting Bracket (SMB)	24.9 by 2.5 by 5.6 inches	6.0
STIS Thermal Interface Kit (STIK)	38.8 by 7.9 by 35.0 inches	44.3
STIK Extended Radiator (STIKER)	35.8 by 7.8 by 35.65 inches	12.5

STIS Cooling System Components

The STIK Mounting Bracket (SMB) is an L-bracket installed onto the two brackets already installed on the bottom +V2 corner of the STIS instrument. The SMB is installed with four captive wing bolts.





The STIK is launched in a folded orientation and unfolded prior to installation. The STIK is inserted under the STIS instrument with guide interfaces on the SMB and latched to the SMB. A pressure plate presses against the STIS aft bulkhead, and heat pipes conduct out to a saddle plate.



Figure 11.3-14. STIK in folded configuration

STIKER attaches to the STIK pressure plate. It consists of L-shaped heat pipes that conduct up to a 4-foot 2-inch radiator containing two spreader heat pipes.



Figure 11.3-15. STIK extended radiator



Figure 11.3-16. STIK mounted onto SMB before and after STIKER installation

All three pieces of the STIS Cooling System will be stowed in enclosures on the aft bulkhead of the ORUC.



Figure 11.3-17. SMB, STIK, and STIKER mounted on ORUC

11.3.3.6 STIS repair installation/operations

Prior to the start of the STIS repair the crew will rotate the HST to allow EV crew access to the +V2 AS doors and prevent direct sunlight from intruding into the AS. As the EV crew approaches the AS doors they will inspect the handrails, handles, and door knobs

to ensure that no paint flacks are introduced into the AS. The crew will then start opening the +V2 AS doors by disengaging the ASLR-S via hex head bolt activation and PiP Pin removal/reinstallation. Then disengaging the remaining three latches and opening the left door to its stops and the right door to its bump stop. The crew will retrieve the STIS Tool Disposal Bag (TDB) from the ORUC aft ATM. The crew will then begin the removal of the MEB-1 cover by attaching the Clamp Removal Tool (CRT) by ratcheting the locking handle and removing the non-captive fasteners holding the cover in place. It is important to note that the CRT is designed to capture the fasteners and their washers to prevent loose fasteners within the AS. Once the fasteners are releases the MEB-1 Cover and CRT will be temp stowed in the TDB. Next the crew will remove the STIS Axial SI Handlebar by installing the Handle Removal Tools (HRTs) on the upper and lower brackets and releasing the non-captive fasteners into the HRTs. Once the fasteners are released the Handlebar will be temp stowed within the TDB.

After the MEB-1 cover and Handlebar has been removed from the STIS, the crew will retrieve the STIS Fastener Capture plate from the ORUC shelf and install it on the STIS. The crew will first install anchors into the counterbores used by the MEB-1 cover. Once the anchors are in place the crew will fit the fastener capture cover onto the anchors and then secure it in place with Zip nuts while verify the visual indicators show the fastener capture plate is properly seated. After the capture plate is installed the crew will stow the TDB on the ORUC Starboard aft ATM for their return to earth. Finally the crew will complete the installation of the cover by removing the CG label via a scrapper installed on the plate to expose 2 fastener holes.

Following the completion of the fastener capture plate installation, the EV crew will disengage 108 non-captive fasteners from the MEB-1 cover. Once all fasteners have been removed from the MEB-1 cover, the crew will use three jack screws to slowly free the cover from the EMI gasket. During the removal of the cover the crew will monitor the EMI gasket to ensure it does not become unseated from the STIS. The crew will continue to remove the MEB-1 cover until they are able to cut the thermister harness using the TDB cutting tool. After the Thermister harness is cut the crew will reposition the harness and the EMI gasket so that they will not interfere with the removal of the LVPS-2 card. Once the gasket and harness has been rearranged the crew will stow the Fastener capture plate and MEB-1 in the TDB, and then photo document the STIS worksite.

The crew will then retrieve the LVPS-2 Soft Card Transport Enclosure (CTE) from the ORUC and install the STIS Ground Strap Extension Harness by clamping to the current ground strap and extending the harness to the Axial SI Handle stanchion. The ground strap extension will be used through out the removal and replacement activities to provide a ground path for the individual tools and cards; this will ensure no inadvertent static discharge that might damage the STIS hardware. Next they will remove the LVPS-2 Card by using the Cardlock Manipulation Tool (CMT) to release and capture to non-captive fasteners, and then installing the Card Extraction Tool (CET). After the CET is installed and locked to the MEB, the crew will actuate the elevator blocks to begin unseating the LVPS-2 card from the STIS. The crew will finish the removal by

pulling the card from the STIS, and then inserting it and the attached tool into the LVPS-2 Soft CTE.

Next the crew will retrieve the Hard CTE and Card Insertion Tool (CIT) from the SOPE and transporting it to the STIS Worksite. The crew will begin the installation of LVSP-2 card by installing the CIT onto the new card and extracting it from the Hard CTE. The crew will then inspect the new LVSP-2 card for damage, and then insert the car into the MEB-1 until it is mated in the connector receptacle. Once the card is inserted with about 22lb-force into the MEB-1, the crew will disengage the CIT from the new LVSP-2 card and stow the CIT with the Hard CTE. The crew will then retrieve the STIS MEB-1 cover from the ORUC Shelf, and install it onto the STIS MEB-1 using the alignment guides and securing it with the two EVA latches.

After the installation of the STIS Cover the crew will retrieve the STIK Mounting Bracket (SMB) from the ORUC and install it on the STIS. The SMB will be installed on the -V1 edge of the STIS, the crew will install engage the SMB mounting interface with two alignment pins prior to engaging the four spring loaded wing tab fasteners. After installing the MSB the crew will retrieve the STIK from its enclosure, and ensuring that it is not exposed to direct sunlight or inadvertent contact is not made with the heatpipes. The STIK is installed on the SMB by deploying the actuator arm 90 degrees and maneuvering the STIK into the mounting brackets and engages the soft docking Jhooks. After engaging the J-hooks, the crew will turn the EVA actuator knob about 13 turn to seat the thermal base plate onto the aft surface of the STIS, then applying the final pressure needed to mate the STIK to the STIS by pivoting the EVA actuator handle down approximately 100 degrees. The crew will also secure the NCS flex line and harness to the center guiderail to ensure that it does not contact the STIK. Next the crew will retrieve the STIKER from its enclosure, and transport it to the AS for installation. The STIKER is installed by seating the STIKER onto the STIK saddle sliding it along the saddle to the right to align and engage the 6 key way bolts. The crew will then close and secure the +V2 AS doors.



Figure 11.3-18. NBL run showing STIS fastener capture plate install

11.3.4 Rate Sensing Unit and Electronic Control Unit

11.3.4.1 Overview

The HST utilizes three Rate Gyro Assembly packages that sense vehicle motion and, in turn, provide digital and analog rate data. The digital data is provided to the Data Management Unit (DMU) dedicated Rate Gyro Assembly Interface (RGAI) module for use by the HST 486 software in determining angular rate during normal vehicle operations of fine pointing and spacecraft maneuvers. The analog data is provided to the Pointing and Safe-mode Electronics Assembly (PSEA) in determining vehicle angular rates during safe-mode vehicle operations, provided they are not the cause of entrance into safe-mode.

Three RGAs provide a total of six independent gyro channels. Two of six gyro channels are now normally used. Each gyro is mounted in a skewed configuration relative to the vehicle axis.

A single RGA is composed of a Rate Sensing Unit (RSU), which houses the Rate Integrating Gyros (RIGs), and Electronics Control Unit (ECU). The ECU provides the RSU electronic interface to the vehicle, with each RSU/ECU set providing two electronic channels (channels 1 and 2) and RIGs.

Each RSU contains two electronics cards and two cylindrical Rate Sensors (R/Ss) with output axes arranged parallel to one another within a rectangular magnetically shielded housing. Each Rate Sensor comprises a RIG, a normalizer, and Pulse Rebalance Loop electronics.

Inside each RIG is a wheel that spins at a constant rate (nominally 19,200 rpm). The wheel is mounted in a hermetically sealed cylindrical float with the rotor spin axis

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oriented 90° from the float's output axis. The float, in turn, is arranged inside the gyro in such a way as to allow rotation about the output axis. At each end of the float is a magnetic suspension rotor which interacts with two corresponding magnetic stators which are integral to the stationary end housings of the RIG to keep the float suspended within the flotation fluid. One end of the float has a signal generator to detect the rotation of the float caused by a rotation around the gyro input axis. This signal is processed to provide the digital attitude and analog rate information. The other end of the float contains a torquer coil that is used to restore the float to its null position. The entire assembly is enclosed in a sealed sleeve.

For SM-4, three primary and one spare RSU and one spare ECU are manifested. The ECU may be replaced if a new RSU fails and the troubleshooting implicates the ECU.

11.3.4.2 Structural Description

Three RSUs are mounted below the Support Systems Module (SSM) Equipment Shelf along with the Fixed Head Star Trackers (FHSTs) accessible via the -V3 Aft Shroud Bay Doors. The main mechanical units of the RSU consist of a housing containing the two RIGs, a connector panel, an alignment cube, an astronaut handle and an electronics housing containing two electronics cards. The RSU is approximately 12.8 inches long (including connectors) by 10.5 inches wide by 9 inches high and weighs approximately 24.5 pounds.

The RSU housing is a hog-out that is machined from 6061-T6 aluminum bar stock per specification QQ-A-225/8. The electronics module and hog-out covers are made of 6061-T4 dip-brazed stock. The material of the hold-down bolts is titanium 6AI-4V procured to AMS-4965. Each bolt receives penetrant inspection per MIL-I-6866, Type 1, Method "B", and is proof tested to $4,500 \pm 50$ pounds tensile load. The electronics cards are fabricated from epoxy fiberglass.

The structural design of the RSU exceeds the 1.4 safety factor requirement for all normal mission phases.

Two 5-micron filter vents are provided on the connector panel for venting and pressurization environments.



Figure 11.3-20. Typical ECU



Within the aft shroud, three captive ORU bolts retain the RSU to its mounting platform, aligned with diamond guide pins in the base of the mounting platform and slotted bushings in the base of the RSU. System design precludes mechanical latch interfacing to the HST.

Each RSU contains two very delicate rate gyros and three very delicate Mykroy mounting feet. The three Mykroy mounting pads should never contact the alignment pins or the coarse alignment guides (L-shaped brackets) of the mounting surface.

Note: This is the most critical EVA handling operation of an RSU.

The wheel within an RSU gyro can be damaged if the RSU is manually moved within 5 minutes after sending the OFF command to the ECU. The gyro wheel will spin down within this time from its nominal speed of 19,200 rpm to a complete stop to ensure that the air-bearing wheel assembly will not incur damage. The air-bearing wheel is susceptible to damage if it is subjected to a disturbance when at a low speed (just prior to coming to a complete stop).

11.3.4.3 Optical Subsystem

11.3.4.4 Electrical Subsystem

Each channel of an RSU is connected by one dedicated cable to the corresponding ECU channel. This cable carries all power, telemetry, command and support electronics required by the RSU.

RSU housing temperature excitation voltage originates in a PDU and is then routed through the Instrumentation Control Unit (ICU), where the power could be switched off through a relay. Although ICU A and B may be commanded off from the DIU, the disconnected current is less than 0.3 milliamperes. These low power connections are not a safety hazard to the EVA crew. See Figure 2.3.5-3, RGA Power Inhibits for EVA Changeout.

Survival and Control Heater circuits are nominally enabled prior to turning on the RGAs. Control Heater circuits are nominally disabled for an RGA that is powered off.

The Temperature Controller provides for both normal and backup temperature sensor control. Survival heaters are provided for off-mode protection, with control being provided from thermostatic switches.

Flight Software (FSW), which operates in either a high rate mode (1,800 deg/hr maximum) or a low rate mode (20 deg/hour maximum), depending on the PCS mode of operation normally commands the gyros.

The RGA does not contain any material, piece part, or circuit capable of generating any ionizing or electromagnetic radiation.

11.3.4.5 Thermal Subsystem

Each RIG of an RSU includes an internal heater that maintains the gyro at 135° F during normal operation. The RSU also contains two survival heaters bonded to the RSU housing, which maintain the gyro at or above its 40° F survival temperatures during periods when the RSU is powered off. Survival heaters will not cycle ON when at least one gyro in an RSU is powered because of control heat from the other gyro. Each survival heater element may be separately switched on by command to its corresponding PDU relay.

An external power source is provided to the RSU survival heaters which are controlled at the orbiter's SSP or at the HST Astronaut Control Panel (ACP). The RSU survival heater external power is controlled by SSP switch S16 command to IPCU relays K16 and K17.

Heat transfer from the RSU is minimized by use of titanium mounting bolts and Mykroy isolators for conductive isolation from the OTA mounting shelf. Three sides and the bottom of the RSU are covered with an MLI blanket and a portion of the top surface of the RSU is covered with reflective tape to control radiant heat losses.

The RSUs are one of the most thermally time-critical items on the HST vehicle, in that the unit cools down quickly if not provided with operating or survival heater power. During RSU changeout operations on a servicing mission, the RSUs will be without power for an extended period of time. Prior to changeout activities, the RSUs must be thermally conditioned to their operating temperatures. Once power is removed a minimum time of 5 minutes must be allowed for the gyro spin motors to spin down prior to removing an RSU from its mount. The changeout activities must be accomplished before the RSUs (new and old) temperature falls below their 40° F survival limit. RSU survival heater power should be restored immediately after installation of the new units, or as soon as practical to avoid damage to the RIGs.

11.3.4.6 RSU Installation

The RSU is furnished with handles containing integral tether loops. Three sides and the bottom of the RSU are covered with a Multi-Layer Insulation (MLI) Blanket. A Velcro strip is located on the bottom of the RSU and at the corresponding vehicle interface side to provide soft-dock restraint of the RSU during installation.

During EVAs the RSUs must be kept out of direct sunlight to prevent contamination of the external surfaces. To meet this requirement as well as thermally protect the RSUs, the EVA crew will utilize a transport bag for both the new and old RSUs during transport.

The RSU change out operation is identical for each RSU. For specific hardware details and latching information, see Section 2.7.1.5.

The IV crewmember rotates the BAPS to the RSU change out position using the switch on the A6U panel. Proper position is verified by lining up the markings on the HST with those on the FSS BAPS (180° for all of the RSUs).

Prior to the beginning of the EVA change out, the FHST shutters are verified closed from SSP tb DS15, and the RSU is deactivated by commanding OFF the RSU control and survival heaters as well as the rate sensor and support electronics. RSU commands are nominally sent from the DIU and DMU from the STOCC, although the crew has the capability to issue these commands from the spec, if necessary, or the SSP for power to the RSU survival heaters.

RSU-2R is retrieved from the COPE, stowed in an ORU transfer bag, and maneuvered to the -V3 aft shroud. At the -V3 aft shroud worksite, the FHST 2 and 3 shade seals are retracted, the latches on the -V3 aft shroud doors and the handle latch are disengaged, and the doors were opened and tethered at the padded stop position. At least 5 minutes prior to the -V3 doors opening, all three RSUs are powered off.

After a GO was received from the IV crewmember, the two wingtabbed RSU power connectors are disconnected, the three captive blots disengaged, and the original RSU removed. The replacement RSU is tilted slightly for easier installation. Then, the three captive bolts are reengaged, and the two power connectors reconnected. The original RSU is maneuvered to the COPE and stowed for landing.

After RSU-2R installation is complete, EV notified MCC GO for RSU-2R aliveness test. The STOCC then performs an aliveness test to verify the integrity of the RSU-2R connections.

This sequence is repeated for RSU-3R and then for RSU-1R. Next the -V3 doors are closed with the change out procedure, concludes with reengaging the FHST 2 and 3 shade seals.

11.3.5 Fine Guidance Sensor

11.3.5.1 Overview

The Fine Guidance Sensor/Radial Bay Module (FGS/RBM) designed and manufactured by Goodrich Space and Electro-Optics Systems provides fine pointing error signals to the HST pointing control subsystem, allowing it to stabilize the HST within fractions of an arc-second about a selected line of sight. A secondary scientific functional mode (Astrometry) is used to determine the angular distance between stars. Three FGS/RBMs are located in the Optical Telescope Assembly (OTA) and occupy three of the four radial bay instrument positions (WF/PC-II currently occupies the fourth). Only two FGS/RBMs are required for precision guidance of the HST, the third is used to conduct astrometric observations.

The Fine Guidance Sensor-3/Radial Bay Module (FGS/RBM) is scheduled to be replaced during the STS-125, Fourth Servicing Mission (SM4) of the HST. The

replacement FGS/RBM has been modified to enhance the on-orbit operation. The FGS will be transported in the FSIPE on the ORUC for SM4.

The replacement FGS/RBM incorporates external zeolite absorbers for contamination control. An EVA handle will be installed before launch for use in removing the FGS from the Carrier and installation into the HST. This requires an addition of studs to the existing attach points previously used for ground handling. These studs will remain on the FGS/RBM after installation.



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Figure 11.3-21. FGS/RBM



Figure 11.3-22. FGS/RBM ORU configuration (MLI not shown)

11.3.5.2 Structural Description

The elements of the FGS/RBM include

- a. Star Selector Servos A and B (SSA, SSB)
- b. Filter Wheel (FW)
- c. Field Stops (FS)
- d. Photomultiplier Tubes (PMTs)
- e. The Actuator Mechanism Subsystem
 - 1. Actuator Mechanism Assembly (AMA)
 - 2. Fold Flat 3 Mirror (FF3)
 - 3. Actuator Mechanism Electronics (AME)
 - 4. Sensor Electronics Assembly (SEA)
- f. Optical Control Electronics (OCE located in Bay C of the OTA equipment section)
- g. Remote Servo Units A and B (RSUA, RSUB)

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- h. Dual Servo Unit (DSU located in Bay C of the OTA equipment section)
- i. Fine Guidance Electronics (FGE-3 located in Bay G of the OTA equipment section)

The OCE Enhancement Kit Harness located in Bay C is used to support command and data interfacing with the FGS/RBM Actuator Mechanism Assembly. This kit provides high-level discrete commands to the replacement FGS/RBM.

Three FGS/RBMs are mounted radially in the Aft Shroud of the HST, just aft of the Space Support Module Equipment Section (SSM-ES) within the focal plane deck assembly of the OTA focal plane structure. Access to the FGS/RBM is via Aft Shroud FGS/RBM doors. The doors for FGS/RBM-3, the current FGS/RBM planned for replacement, are located on the +V2 side of the HST. The FGS/RBM measures 54 by 47 by 22 inches and weighs approximately 503 pounds, including EVA handle.

The FGS/RBM's structure consists of the Keel, Optical Bench, Exostructure and Dome. The Keel is the primary load carrying structure. The Keel supports the four latches ("A," "B," "C," and "D") that are used to install the FGS/RBM into the focal plane structure. Kinematic mounts (6) attach the Optical Bench to the Keel. The Optical Bench provides structure for optical elements.

The Exostructure serves as an environmental enclosure and structural support for the FGS/RBM's electronics thermal radiator. It is made up of several large fiberglass panels. Each panel is constructed as a "window frame" with aluminum edges completely surrounding each fiberglass panel. Uniaxial graphite epoxy is used to form stiffening struts in two locations on the Pickoff Panel, and four small titanium plates are used to stiffen shear paths on the B and C side panels. The Exostructure mounts to the FGS/RBM Keel Assembly at the Keel's perimeter. All the Exostructure loads are reacted through the Keel. It encloses the Optical Bench and its components and provides required support for the Dome. The Dome provides required support for electronics boxes and cables.

11.3.5.3 Optical Subsystem

The FGS/RBM's are a system of photomultiplier tubes and white-light amplitude interferometers (Koesters' prism) that are used for fine guidance of the HST. The FGS/RBM is not an imaging device like a camera; instead it is an assembly of two interferometers that determine the tilt of the wavefront relative to the full HST primary mirror (the entrance pupil of the interferometer). A plane pickoff mirror samples the light converging toward the focal plane of the main optics. The light is diverted into a direction perpendicular to the HST V1 axis. This light comes to a real focus before diverging to the off axis aspheric collimating mirror from which emerges nearly parallel but essentially reversed in direction. The light encounters the "A" and "B" star selectors, which, by independent adjustments, transforms the direction of light from any star in the FGS total field-of-view into a direction normal to the Koester prism interferometer. The replacement FGS/RBM replaces a fixed mirror internal to the FGS/RBM with

commandable articulating capability. Positioning the articulating mirror via ground command optimizes the modulation transfer function of the interferometer.

11.3.5.4 Electrical Subsystem

The FGS/RBM and its associated mechanisms will be unpowered while stowed in the Orbiter and during EVA operations. The FGS/RBMs receive power from their respective FGEs and the OCE after the HST is deployed. The FGS consumes 39 watts during normal operations.

11.3.5.5 Thermal Subsystem

Primary and redundant heater circuits throughout each OTA radial bay cavity maintain active thermal control of the FGS/RBM. The FGS operational temperature range is 68° to 72° F, and survival temperature range is from 50° to 90° F. The Optical Bench is controlled to 0.2° F. Low thermal coefficient of expansion materials are used to minimize distortions due to thermal gradients. Most of the external FGS surface is covered with black multilayered insulation except for the thermal radiators.

11.3.5.6 FGS Installation

The FGS will be removed from the ORUC protective enclosure using the EVA handhold that is permanently installed on the FGS prior to launch. The handhold remains in place after installation of the FGS/RBM into its aft shroud location. When installed, it is completely enclosed in the aft shroud.

For more detailed information concerning latching, go to the generic radial SI change out operation in Section 3.4.1 of this document.

Before the change out, the IV crewmember rotates the BAPS to the FGS-3 change out position using the switch on the A6U panel. Proper position was verified by lining up the markings on the HST with those on the FSS BAPS (270° or the +V2 fwd).

IV activates S24 SSM WK LIGHTS to activate the FGS latch indicator lights and the HST electrical receptacles.

The change out begins by deploying the aft fixture. The +V2 scuff plate is rotated 90 degrees by disengaging four fasteners. The +V2 FGS doors are opened by disengaging the two latch bolts and releasing the handle.

Then IV verifies that MCC is GO for FGS-3 connector demate, after the STOCC performs the powerdown. The crew has the capability to issue the poweroff commands, if necessary. After receiving verification that the FGS blind-mate connector has been deadfaced, four left wingtab connectors and groundstrap are demated and tethered back to the left tether loop. Next, the four right wingtab connectors and groundstrap wingtab connectors are demated and tethered back to the right tether loop.

Four FGS guide studs are installed on the FGS. The FGS portable handhold is then retrieved from the forward fixture and installed onto the FGS guide studs via the four

zipnuts with the spacers and MLI A-latch cutout on the FGS handhold at the bottom (-V1).

The FGS A-latch is then disengaged with the A LATCHED indicator lights illuminating.

The FGS is pulled out of the FGS cavity using the portable handhold and the guiderail system and a coordinated EV/IV effort between the RMS-mounted crewmember and the RMS operator, with the non-RMS crewmember providing guidance support. This allows for any repositioning necessary to ensure a level extraction of the FGS. The FGS may need to be pitched up (+V1) to avoid A latch contact with the coplanar D-latch fitting. The FGS can only be touched at the portable handhold, ground support equipment brackets, and FGS guiderails.

FGS-3 is translated to and temporarily stowed on the aft fixture.

The FSIPE door is opened. The FSIPE groundstrap is disconnected and stowed in the FSIPE groundstrap stowage receptacle. The FGS-3R A-latch in the FSIPE is then disengaged 1/2 revolution to break the seating torque followed by the remaining 22+ revolutions (until the A-latch is fully disengaged). The A-latch indicators are verified activated after the first 2.5 revolutions CCW of the A-latch, indicating a not fully latched condition. The A-latch indicators are verified with the A LATCHED lights illuminated.

The FGS-3R is removed from the FSIPE guiderails. The FGS-3R may need to be pitched up (orbiter forward) to avoid A-latch contact with the coplanar D-latch fitting. During removal, the FGS-3R exterior adsorber module mirror, just above the pickoff mirror, cannot be touched because of sharp edge and contamination concerns. The FSIPE door is then temporarily closed and one of the latches latched. Next, the FSIPE lid on-orbit latches were verified armed.

The FGS-3R is installed into the HST FGS-3R cavity using a coordinated EV/IV effort with an EV crewmember on the RMS and another EV crewmember providing guidance. This allows for any repositioning necessary to ensure a level insertion of FGS-3R. However, the FGS-3R may need to be pitched up (+V1) to avoid A-latch contact with the coplanar D-latch fitting. The FGS-3R is inserted into the cavity and repositioned approximately to engage the guiderails. It is then fully inserted into its seated position. The A-latch is engaged and torqued and the A LATCHED indicator lights are extinguished.

Next, the EV crewmember mates the FGS-3R groundstrap wingtab connector to the replacement FGS-3R. The eight wingtab connectors are mated to the FGS-3R.

After completing the FGS-3R installation, IV gives MCC a GO for FGS-3R aliveness test. The STOCC performs the aliveness test to verify electrical integrity of the instrument connectors.

With both doors fully closed, the handle latch and two door latches are reinstalled and torqued. Then, the three FGS-3R door latches are retorqued. Next, the FGS-3R doors

were visually inspected for closure including the resilient door light seal. Finally the crew rotated the +V2 scuff plate into its original position.

The FSIPE door s opened, and the FGS was removed from the aft fixture and inserted into the FSIPE guiderails. The FGS may have needed to be pitched up (orbiter forward) to avoid A-latch contact with the coplanar D-latch fitting. The FGS was inserted into the FSIPE and repositioned appropriately to engage the guiderails. It was then fully inserted into its seated position. The A-latch was engaged and torqued with the FGS A LATCHED indicator lights extinguishing.

The FSIPE groundstrap wingtab connector is mated to the FGS-3 and the four FGS-3 handhold zipnuts and the handhold on the FGS-3 are removed, leaving only the four guide studs on the FGS-3. The handhold is installed on the forward fixture.

After the worksite is clear, the FSIPE lid is closed and all three lid latches latched and verified. The aft fixture is also stowed.

Next the crew performs the OCE EK connection, the HST bay C door is opened by releasing the three J-hooks and opening the door. The door is secured via either the OTA door stay or tethering. The bay C door opening angle cannot exceed 118°.

Next, IV verifies MCC GO for OCE connector demate, after the STOCC performs the powerdown. The crew has the capability to issue the poweroff commands, if necessary. After receiving a GO from IV, the EV crewmember demates OCE connector P12.

Once OCE-EK P12A/J12A is connected, EV informs IV GO for OCE aliveness test. When the aliveness test is complete, the bay C door is closed and latched.

11.3.6 Battery Module Assemblies

11.3.6.1 Overview

On SM4 the NiH₂ batteries currently installed on HST will be replaced with new NiH₂ batteries. These batteries are essentially the same design as the current batteries with some minor modifications to improve battery performance and increase safety for the crew during battery changeout. The batteries will be transported on the SLIC.

In 1995 procurement began for the purpose of producing replacement HST batteries using the existing Manufacturing Control Document at Eagle Picher Technologies (EPT). The cell internal stack manufacturing process was changed from dry sintered plaque to wet slurry plaque. The new battery cells were manufactured, assembled and then placed in dry storage in 1995. The cells remained in dry storage for 4 years.

In June 2000, the battery module procurement for SM4 was restarted; 205 cells were removed from dry storage and began cell activation in August 2000. A Cell Acceptance Data Review was held on December 5, 2000. All cells were developed, produced, and qualified under the previous program and were acceptance tested only for SM4.

The Battery Cell Spec was revised and reissued as LMSC/P106652, Hubble Space Telescope Specification for Nickel-Hydrogen Battery Cell, dated 2/1/96. The reissue document specified the wet slurry sinter substrate process rather than previous dry sinter process, updated document references, implemented nickel precharge, modified capacity values as appropriate and qualified cell electrical characteristics by similarity.

A battery is composed of 22 cells wired in series to the J1 power connector. Two connectors, one for power (J1), and one for telemetry and heater circuits (J2) are attached to the connector bracket. Each cell is wrapped with 0.015-inch thick cho-therm and then clamped within a machined 6061-T6 aluminum sleeve. A dual heater element strip is bonded at the bottom inside of each sleeve. The sleeve/cell assemblies are then fastened to a battery baseplate made from 0.125-inch aluminum. A support tube, the same diameter as the sleeve, is attached to the baseplate in the middle of the battery.

This efficient design, along with other compact battery packaging allows the BMA to fit within the limited space of the HST Bay. The cell pressure vessel is made out of Inconel 718 stock, 0.040 inch nominal thickness. The cell pressure vessels are cylindrical with dome shaped ends and are approximately 3.5 inches in diameter and 9.3 inches long. The cell terminals are each located at 45° from the vessel's longitudinal axis on one dome end. The cells are activated with potassium hydroxide electrolyte. The electrolyte is an aqueous solution of 31 ± 1 percent (by weight) potassium hydroxide (KOH) in an activation-limited quantity; i.e., when activation is complete there is little or no free liquid electrolyte remaining in the pressure vessel.

The cell and its components are structurally designed in accordance with the requirements of MIL-STD-1522A. They also meet the fracture control requirements of NASA-STD-5003. The cell components are designed to minimum structural factors of safety of 1.25 to yield and 2.0 to ultimate for acceleration loads, and minimum pressure vessel factors of 1.5 proof pressure in air, 1.10 yield pressure in hydrogen and 3.0 burst pressure. MDP is 1200 psi. (3.0 x MDP = 3600 psid).

To account for the potential effects of hydrogen embrittlement of the cell pressure vessel material (Inconel 718), the cell stress and fracture analyses used derated material property values even though all available data obtained indicates that the internal KOH/H₂ gas environment should be inert. Development and production lot qualification cells were burst tested (in a nonhydrogen environment) with test results showing a minimum safety margin of 1000 psi over the ultimate pressure requirement, and typical burst values in excess of 1400 psi over requirement. MDP is 1200 psi. (3.0 x MDP = 3600 psid).

After the cell electrolyte activation is complete, boss (terminal port) and fill tube welding are performed. An incoming visual inspection and an electron beam weld per process control procedures and records are performed. A prebatch weld sample is cross-section evaluated at the beginning of each continuous weld setup or shift with a preweld visual inspection and tracking check. Active tracking, monitoring and adjusting is done during the weld process, and a post-batch weld sample is cross-section evaluated. Pre and post weld visual inspections at 30X magnification are accomplished per welding

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specification. A final special NDE of the entire exterior is performed and then the following is performed: A 10X visual inspection and dye penetrant spot check of the boss/fill welds are performed per MIL-STD 6866 and MIL-I-25135.

One pressure vessel per material/heat treat lot is subjected to an accelerated cycle burst test. Two more vessels per lot are taken for destructive evaluation of weld process quality, and two additional coupon samples per lot are taken for independent metallurgical sectioning for lot acceptance.

A final special NDE of the entire exterior is performed followed by

- a. Radiographically inspecting all cells in two views per cell.
- b. Conducting 100 percent five cycle proof test to 150 percent of MDP (1800 psid).
- c. Conducting 100 percent helium leak (bubble) test after final proof pressure cycle.
- d. Conducting 100 percent hydrogen leak check before acceptance testing.
- e. Conducting 100 percent special NDE fluorescent dye penetrant post proof external inspection of entire vessel. Critical flaw size detection is at the level of 0.05 inches in length.

The physics of the NiH₂ battery technology provides inherent characteristics that eliminate cell reversal and unstable recombination of hazardous gases as safety concerns (although cell reversal from a performance view is still undesirable). During battery overdischarge, hydrogen is generated at the positive nickel electrode and recombines at the negative platinum electrode at the same rate; thus, there is no change in pressure or electrolyte. During charging, oxygen is generated at the positive electrode, and hydrogen is generated at the negative electrode. Oxygen recombination rates at the negative electrode are rapid enough that hazardous combustible mixtures are not formed during overcharge. The amount of platinum catalyst present is in an order of magnitude greater than required to catalyze this recombination. This results in a typical oxygen partial pressure in the hydrogen gas of less than 1 percent.

An updated Battery Module SCN-02 Specification was released on June 16, 2000. The update added a requirement electrically isolating battery connectors (i.e., an isolation switch) resulting from a JSC Safety Panel TIM held in 1997. The JSC Payload Safety Panel stated that the design would not be compliant with their requirements if a power isolation capability was not included in the design.

11.3.6.2 Structural Description

The HST utilizes two Battery Module Assemblies (BMAs) (three batteries and associated electronics within each BMA) on the interior side of the SSM Bay 2 and 3 doors. Batteries 1, 2, and 3 are located in Bay 2 and Batteries 4, 5, and 6 are located in Bay 3. Each BMA contains three 88 Ah NiH₂ batteries. Each battery consists of 22 cells wired in series to a power connector. Each on-orbit battery has one spare cell that is not connected. In the SM4 replacement BMAs (4177755-503), the spare cell in each battery was replaced with a Battery Isolation Switch (BIS). The SM4 replacement BMAs

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also have three extra connectors, one for each battery, that provide individual cell monitoring capability.

All three batteries in a BMA are fastened to a Module Base Plate (P/N 4177760) enclosed by a Module Cover Assembly (P/N 4177761). Each BMA is an Orbital Replacement Unit (ORU). The BMAs are designed to structural flight phase/handling loads factors of safety of 1.25 to yield and 2.0 to ultimate, with sufficient venting for pressure equalization within the BMA housing.

Temperature control of the battery environment is provided by means of SSM Bay Door radiators and active, redundant, heater controllers, thermistors and thermostats in each battery. Normal battery temperature range is between of 28° and 36° F.

During SM-3A, six Battery Voltage/Temperature (V/T) Curve Improvement Kits (VIKs) (one for each of the six batteries) were installed. Each VIK provides modified V/T charge-on and chargeoff points to improve battery thermal stability in vehicle safemodes.

The design capabilities for the spacecraft batteries are (to be updated as necessary)

- a. To provide for continuous orbital power at 26.2 to 34.3 V dc
- b. To provide for up to 5 orbits of up to 30° off-normal roll, with Depth Of Discharge (DOD) limited to 20 percent over any 24-hour period
- c. To provide at least 225 AH for Safe-Mode operations
- d. To provide up to 60-month operational life (original specification), extended to 180 months, and to be extended again due to SM4 delays
- e. To provide maximum orbital power of 1250 watts at the OTA/SI interface at 2 years into the mission with one battery failed

11.3.6.3 Electrical Subsystem

N/A

11.3.6.4 Thermal Subsystem

N/A

11.3.6.5 BMA Installation

It is important to note that during the change out of the HST battery modules it will not be possible to deadface both sides of the connectors since power is provided to the HST via the batteries and the Main busses during change out.

Before the battery change out the IV crewmember rotates the BAPS system to expose the SSM doors using the switch on panel A6U. The EV crew opens and secures the

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SSM bay doors by rotating the 6 J-hooks, it is important to note that only one bay door is allowed to be open at a time to prevent damage from interference between the Bay 2 and Bay 3 thermal louvers.

After receiving a MCC GO, the crew demates the 3 battery power connectors one at a time and immediately caps each connector. Next the EV crew demates the VIK connectors from the Gender Changers and immediately caps the Gender Changer connectors. The crew then removes the batteries from the HST by releasing the 14 J-hooks in any order, as long as hooks 11, 12, 13, 14, and 4 are completed last to maintain thermal contact. The crew then removes the batteries from the 4 shear pins and temporarily stows it. While one EV crewmember is removing the old battery the free float EV crewmember is retrieving the new batteries from the SLIC and disengages the 14 J-hooks in the same manner. Prior to battery removal from the SLIC the EV crewmember verifies that the Battery isolation switch is in the off position.

The EV crewmembers swap the old battery for the new battery and begin installation of the new battery. The new battery is installed by placing the battery onto the 4 shear pins and engaging J-hooks 11, 12, 13, 14, and 4 first then engaging the remaining J-hooks. After all three batteries are installed the crew will make the new power connectors to the battery connectors in any order, and then connect the VIK P2 connectors to the corresponding Gender Changers P2 connector. The EV crewmember will then turn the Battery isolation switches to ON, and close the SSM Bay Doors.

While the new batteries are being installed, the Free Float EV crewmember is installing the old batteries into the SLIC by first engaging J-hooks 11, 12, 13, 14, and 4, then the remaining J-hooks. Finally the EV crew member will close and latch the SLIC battery carrier.

11.3.7 New Outer Blanket Layer

11.3.7.1 Overview

The HST includes active and passive thermal control designed to maintain HST's structure and electronic equipment within operational parameters. Such thermal control is essential to maintain temperature of the structures and equipment within their specified design operating range and to reduce structural deflections/stresses caused by thermal contraction and expansion on orbit. The passive temperature control elements consist of Multi-layer Insulation (MLI) thermal blankets, aluminum radiation shields, and various surface finishes (i.e., paint, tape, Flexible Optical Solar Reflectors (FOSRs), and clear anodized finishes).

During the HST Second Servicing Mission (SM2) it was observed that much of the outer layer of the external MLI on the telescope was degraded or damaged. The most notable damage was to the blankets on the +V3 (sun-facing) side of the HST. Further observation determined that the damage was limited to the external layer of the MLI and that cracks and other deformations were clearly visible.

Utilizing materials available on-orbit and with extensive input from ground controllers, the crew fashioned patches to temporarily repair the MLI, which were then installed over some of the worst damaged areas (SSM Bays 8 and 10, and two patches on the light shield) until a more permanent improvement could be incorporated in SM-3A and SM-3B. (The torn SSM Bay 7 outer layer was held down with wire.)

During the Servicing Mission SM-3A (1999), parts of the MLI portion of the SSM Bay Door passive thermal control system was covered by the New Outer Blanket Layer (NOBL). The NOBLs provided a system able to maintain an acceptable thermal environment (α/ϵ) on HST for 10 years as well as to prevent further degradation of existing MLI.

NOBLs were installed on the SSM bays 9 and 10 during SM-3A. A NOBL was also installed on bay 1 as part of the HST 486 task (for thermal reasons). The installation in support of the HST 486 task was not related to MLI repair. On SM3B, the bay 6 NOBL was installed. Each NOBL covers essentially 100 percent of the current MLI area on each SSM bay.

During SM-4, the NOBLs on bays 5, 7, and 8 are planned to be installed. The NOBLs will be transported on the ORUC in the NPE.

HST DOOR CONFIGURATION

NOBL CONFIGURATION



Figure 11.3-23. NOBL SSM bay 5 (example)

Location	Repair
Bay 1	Completed on SM-3A
Bay 5	Remove existing MLI, add NOBL (SM4)
Bay 6	Completed on SM3B
Bay 7	Remove SM2 temporary wire, add NOBL (SM4)
Bay 8	Remove SM2 patches, remove existing MLI, add NOBL (SM4)
Bay 9	Completed on SM3A
Bay 10	Completed on SM3A

SM3A, SM3B, and SM4 NOBL Task Summary

11.3.7.2 Structural Description

The NOBL hardware is designed to meet all the launch and landing loads as defined in HST-Shuttle ICD-A-14009-SM and GEVS. Landing cases shall include both return on the SSE and installed on the HST. The NOBLs are designed to withstand crew handling and any environmentally induced loads. A waiver to STE-59, the NOBL CEI spec addresses the inability of the NOBL to meet the design requirements for a 125-lb inadvertent kick load. This is not a safety/structural issue but a mission success issue; i.e., a dented NOBL that may not fit onto the HST. The crew has been trained in EVA handling techniques for the NOBLs. NOBLs are listed as "no damage" hardware in the EVA Checklist.

The NOBL covers are 4-mil stainless steel coated with a 4-layer composite coating, SiO₂/Al₂O₃/Ag/Al₂O₃. The stainless steel sheet is framed with .036-inch stainless steel. There are two types of NOBL covers; flat and curved. For SM4, only flat NOBLs (bays 5, 7, and 8) will be carried. Each one has a 1.25-inch vent hole in each corner, and the covers are installed with expanding plugs inserted into the vent holes.

The attachment hardware is two-fault tolerant to inadvertent release and one-fault tolerant for proper installation. The attachment of the NOBL hardware will not interfere with the existing functions of on-orbit hardware (vents, EMI door seals, etc.). The covers are external to the HST structure and so will be exposed to an orbital vacuum of <1E-6 torr. Pressure differential within their stowage volume during ascent is less than 0.1 psi.

The flat NOBLs (bays 5, 7, 8) are attached with attachment plugs (four per door). A plug will be inserted into each vent hole, and the handle will be pivoted down 90°. This operates a cam and locks the plug in place. Titanium fingers are flexed radially outward by a sliding plunger inside the vent plug barrel. The fingers contact into the door vent, providing secure attachment. A flange on the vent plug holds NOBL sheet securely against the door. The fingers will spring back to the original position for easy plug removal, if needed.

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Figure 11.3-24. NOBL attachment plug assembly

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11.3.7.3 Electrical Subsystem

N/A

11.3.7.4 Thermal Subsystem

Each NOBL is designed to cover a particular SSM Bay door. The new covers will not cover any area of SSM bay doors that were not previously covered by MLI blankets (e.g., radiators, FOSR, J-hooks). Thermal requirements for the replacement covers include a low solar absorbance and a high-infrared eminence surface to minimize the sensitivity to HST sun angles as well as to minimize the day/night temperature variation. The order of installation for the NOBL is prioritized by thermal effects, contamination issues, and EVA efficiency.

11.3.7.5 NOBL Installation

The installation of the three NOBLs during the mission may be accomplished at any convenient time and in any order. Thus this installation section is written with that in mind. All MLI removed from the HSt will return in the Middeck.

For NOBL 5 installation, the crew will retrieve the bay 5 MLI Recovery bag from the ASIPE exterior. Next the bay 5 MLI is removed by extracting two NOBL vent plugs from SM3B, peeling off the Velcro, and cutting 2 bay 5 ground straps. After the MLI is removed, it is folded such that the outer surfaces face each other to minimize contaminates. The bay 5 NOBL and vent plugs are retrieved from the NPE, and transported to bay 5. Then the NOBL is placed in the correct orientation and the vent port plugs are installed and latched in any order, and finally connecting the NOBL ground strap to the J-hook hex bolt.

For installation of NOBL 7, the crew will remove the 2 temporary wire assemblies that support the MLI patches and reseat the open J-hook on this bay. The crew will then retrieve the bay 7 NOBL and vent plugs from the NPE, and move them to the bay 7 worksite. Once the NOBL is placed in the correct orientation over the existing MLI, a single vent plug will be installed in the vent port and latched. The three remaining vent port plugs will be installed in any order, and the ground strap installed on the J-hook bolt head.

To install NOBL 8 the crew will begin by retrieving the bay 8 MLI Recovery bag from the ASIPE exterior. The bay 8 temporary patches will be removed with the wire assemblies from the appropriate J-hooks, and then the MLI will be removed and folded with the exterior surfaces touching to prevent contaminants. Once the MLI is stowed for return, the EV crew will retrieve the bay 8 NOBL and four vent port plugs. After transporting it to the installation site, the crew will orient the NOBL and install the vent port plugs in any order. After the plugs are secured, the crew will finish by installing the ground strap on any J-hook bolt head.

11.3.8 Overvoltage Protection Kit

11.3.8.1 Overview

The Overvoltage Protection Kit (OVP-K) provides Main bus overvoltage protection to HST in the event that SPAs are tied directly to the Diode Bus(es) (battery bypass) and HST enters Pointing and Safemode Electronics Assembly (PSEA) safemode. This kit is composed of four Extravehicular Activity (EVA)-installed hardware elements:

- a. One P601A Jumper Plug (installed between the Cross Strap Harness (CSH) P601 connector and HST P601 Test Connector)
- b. One P1A Jumper Plug (installed between the +Diode Box Assembly 2 (+DBA2) J1 connector and the CSH P1+ connector)
- c. One Electronics Unit (EU) with P2 Harness (interfaces to CSH J2)
- d. One replacement Electrical Power System (EPS) Test Panel Multi Layer Insulation (MLI) Cover

The OVP-K is ground-command programmed using currently existing HST DBA2 Diode Box Controller (DBC) Flight Software (FSW).

11.3.8.2 Structural Description

To describe the Overvoltage Protection (OVP) components, the reader must be aware of a change that occurred during Servicing Mission 3B (SM3B). As a part of the Solar Array 3 installation, Diode Box Assemblies, Version 2 (DBA2) (2 each) were installed. In addition to providing SA3 String diode isolation, two types of battery charge control capability were added to the DBA2 design over the previous model. Definitions of these two new charge control modes are provided below as a basis for understanding why the OVP is necessary.

The OVP includes a ground programmable device, which disconnects selected Solar Array 3 (SA3) Solar Panel Assembly (SPA)s in the Diode Box Assembly 2 (DBA2) at Safemode entry (PSEA entry). OVP preserves existing DBA2 Software Charge Control capability and uses currently unused DBA2 Autonomous Charge Mode (ACM) circuits.

The OVP system level single fault tolerant design uses redundant Field Programmable Gate-Arrays (FPGAs) for OVP programming with existing DBA2 486 Flight Software (FSW). The OVP FPGA design is based heavily on the existing DBA2 FPGA design. DBA2 Bypass Relays can be closed to connect failed open Charge Control (CC) relays.

The OVP decodes FSW generated Power Distribution Unit (PDU) relay pulse command durations during programming and off-lines desired SPA(s) at Safemode entry. OVP uses latching relays to store programming configuration. The 18 Teledyne 422 26.5 volts latching relays are common to both the Primary and Redundant FPGA drivers. There are separate relay poles for redundant Main Bus C (MBC) trigger signals.

OVP uses Main Bus C turn-on as Safemode entry trigger. Programmed SPAs are rendered offline by activation of currently unused DBA2 ACM functions. OVP activation of DBA2 ACM has been successfully tested in Vehicle Electrical Simulation and Test (VEST) with brass board unit.

Once programmed, a failure of the FPGA program logic does not inhibit desired SPA removal at Safemode entry. The relay states are nonvolatile, and the FPGA is only powered during programming. The DBA2 CC relays open during OVP program mode to permit ground verification of program configuration memory.

Thirty minutes of EVA time are estimated for installation of the OVP kit **(TBR)**. Installation does not require any connections at -V2 DBA2 location. EVA work is executed on +V2 DBA2 only.

Overvoltage Protection Hardware Elements

The +DBA2 P1A Jumper Plug (installed first on EVA) mates between +DBA2 J1 and CSH +P1. This jumper plug taps redundant CSH PDU and AMD commands and Power Returns into unused CSH battery temperature sense lines on the +CSH leg. Command/Telemetry (CT) buses, DCE, and SADE are to be powered off prior to EVA to deadface contacts.

The OVP Electronics Unit (EU) with P2 harness (installed second on EVA) enclosure contains OVP electronics (see Figure 11.3-25). This box has dimensions of 8 inches by 11 inches by 1.5 inches (TBR). The EU attaches externally to HST on Handrail no. 21. One removable harness from Electronics Module mates to CSH J2. The EU contains an active redundant thermostatic heater control. The EU chassis structural ground is provided by a carrythrough shield from the CSH J2 on the EU harness. The EU and the CSH J2 and +DBA2 P1/J1 interfaces are shown in Figure 11.3-26.

The Cross Strap Harness (CSH) P601A Jumper Plug (installed last on EVA) jumpers MBC primary and redundant telemetry status signals into unused CSH battery temperature sense lines. This Jumper Plug fuses and jumpers MBA, MBB, and MBC into unused CSH battery temperature sense lines. The main buses will not be powered off (current draw less than 3A). The CT buses are to be powered off prior to EVA to dead-face these contacts. The P601A Jumper Plug will be mated to the P601 CSH first, then to HST J601 to reduce hot mate to one occurrence.

P601A Jumper Plug installation requires removal and installation of the P600/P601 Thermal Tent (see Figure 11.3-27). The on-orbit Thermal Tent is held in place with two wire tethers from a MLI Patch Kit due to Velcro failure to retain the tent. The Replacement Tent is a semi-rigid fixture and attaches to HST with two 1/4 turn fasteners (TBR). The old tent will be placed in a trash bag. A Flight Spare CSH (contingency for damaged/pushed pins/sockets) will be flown, although there are no plans to fly a flight spare DBA2.



Figure 11.3-25. OVP hardware configuration


Figure 11.3-26. CSH J2 and +DBA2 P1/J1 interfaces



Figure 11.3-27. P601A concept

11.3.8.3 Electrical Subsystem

The OVP-K electrical systems shall operate with 24.7 to 32.7 V dc on Main Bus A, B, and C. It shall operate with 24.0 to 32.0 V dc on OPE-A, OPE-B, PDU-A, and PDU-B inputs.

The OVP-K peak power consumption, including power for active thermal control, shall be 40 watts (TBR) or less for all operational modes (Standby, Program, and Trigger modes). The peak power consumption shall be (TBD) minutes or less in duration. The orbit average power shall be 10 watts (TBR) or less.

Software Charge Control: The DBA2 receives and decodes Power Distribution Unit (PDU) +28 volts pulse width modulated commands to select Solar Panel Assemblies (SPAs) for off-lining to perform battery charge control. PDU pulse commands are generated in the 486 based on battery voltage and temperature. The selected SPAs rendered offline through the DBA2 Charge Control (CC) relays. In addition, the DBA2 contains contingency Disconnect relays to disconnect the SPAs and Bypass relays which are normally available to counter failed open CC relays.

Autonomous Charge Control (ACM)

This feature was added to the DBA2 to provide battery charge control in Safemode in the event that the Power Control Unit Replacement (PCUR) could not be changed out during SM3B. ACM operation requires the EVA installation of a P2 Jumper Plug onto the DBA2 Cross Strap Harness (CSH) and EVA installation of the VIK Jumper Plugs to first enable ACM. The Jumper Plugs connect battery voltage and temperature signals to the ACM circuits in the DBA2s via the DBA2 Cross Strap Harness (CSH). With the Jumper Plugs installed, ACM is normally inhibited under 486 computer control by the application of the Autonomous Mode Disable (AMD), a +28 volts signal provided by the Deployment Control Electronics (DCE). At Safemode entry, the Pointing and Safemode Electronics Assembly (PSEA) powers off the DCE, which results in loss of the AMD signal. Loss of the AMD signal in the DBA2 activates ACM, which provides lower Voltage/Temperature cutoff thresholds for battery charge control preventing battery thermal instability due to overcharge. Since the PCUR was successfully installed on SM3B, the Jumper Plugs were not installed on SM3B and ACM was not enabled. As a result, the battery voltage and temperature wires in the CSH are not connected to the DBA2s, and the ACM circuits are not used.



Figure 11.3-28. OVP-K EU P2 to CSH J2 interface

Note: This figure is for reference only. Some contacts on J2 test connector are deleted for clarity.



Figure 11.3-29. OVP-K P601A jumper plug interface

Note: This figure is for reference only. Some detail is deleted for clarity.



Figure 11.3-30. OVP-K P1A Jumper Plug Interface

Note: Figure for reference only. Some detail deleted for clarity.

11.3.8.4 Thermal Subsystem

OVP-K thermal environments

Parameter	Hot case	Cold case
Altitude (nmi)	303 to 320	303 to 320
Beta angle	52°	0°
Solar Flux (W/m ²)	1419	1287
Albedo	0.35	0.25
Earth IR Flux (W/m ²)	265	208

Servicing Mission

The OVP-K shall remain within its nonoperate survival temperature range, while being stowed in the SSE throughout every phase of the mission: During nonoperational conditions, the OVP-K thermal design shall limit the component nonoperate temperatures to within -45° to +115° C (TBR). Nonoperational conditions are defined as prior to the successful completion of OVP-K EVA installation onto HST. The P601A and P1A Jumper Plugs shall survive nonoperate temperatures of within -45° to +115° C (TBR). In addition, the OVP-K temperature shall be greater than or equal to +20° C (TBR) prior to removal from the SSE for transport to and installation on the HST.

The OVP-K temperatures shall be maintained above their cold nonoperate survival levels. (During nonoperational conditions, the OVP-K thermal design shall limit the component nonoperate temperatures to within -45° to +115° C (TBR.)) Nonoperational conditions are defined as prior to the successful completion of OVP-K EVA installation onto HST. The P601A and P1A Jumper Plugs shall survive nonoperate temperatures of within -45° to +115° C (TBR) during transport and installation, with a goal of maintaining the component temperatures above their cold power limits to support an aliveness test immediately following installation.

Design Limits

a. Operational Limits

The OVP-K thermal design shall limit the EU radiator temperatures to within -35° to +70° C in standby, program, and trigger modes under worst case hot and cold conditions. The OVP-K shall maintain its operate temperatures for unlimited durations for all modes (standby, program and trigger). The P601A and P1A Jumper Plugs shall remain functional within temperatures of (TBD) and (TBD) for all operational conditions.

b. Nonoperational Limits

During nonoperational conditions, the OVP-K thermal design shall limit the component nonoperate temperatures to within -45° to +115° C (TBR). Nonoperational conditions are defined as prior to the successful completion of OVP-K EVA installation onto HST. The P601A and P1A Jumper Plugs shall survive nonoperate temperatures of within -45° to +115° C (TBR).

Surface Properties

Surface properties and finishes for the OVP-K hardware shall be selected to meet the temperature requirements defined in section "A" and "B" above and shall include the on-orbit degradation effects expected over the OVP-K mission life having a minimum on-orbit life of 5 years with a goal of 10 years. The color, finish, and surface reflectivity of the OVP-K outer surfaces shall be taken into account when choosing finishes.

Active Thermal Control

The OVP-K minimum temperature shall be actively maintained with redundant heaters and thermostats. Maximum temperatures shall be controlled with passive radiation to space.

a. Heater Power

The peak heater power for the OVP shall be a maximum of 21 watts (TBR) for a duration of (TBD) minutes. The orbit average heater power shall be less than 9 watts (TBR) across the operational range: The OVP-K electrical systems shall operate with 24.7 to 32.7 V dc on Main Bus A, B, and C. It shall operate with 24.0 to 32.0 V dc on OPE-A, OPE-B, PDU-A, and PDU-B inputs. The OVP-K peak power consumption, including power for active thermal control, shall be 40 watts (TBR) or less for all operational modes (Standby, Program, and Trigger modes). The peak power consumption shall be (TBD) minutes or less in duration. The orbit average power shall be 10 watts (TBR) or less.

b. Thermal Control Range

The OVP-K thermal control range shall limit the EU radiator temperatures to within -35° to +70° C in Standby, Program, and Trigger modes under worst case hot and cold conditions. The OVP-K shall maintain its operate temperatures for unlimited durations for all modes specified in Section 3.2. The P601A and P1A Jumper Plugs shall remain functional within temperatures of (TBD) and (TBD) for all operational conditions and during nonoperational conditions, the OVP-K thermal design shall limit the component nonoperate temperatures to within -45° to +115° C (TBR). Nonoperational conditions are defined as prior to the successful completion of OVP-K EVA installation onto HST. The P601A and P1A Jumper Plugs shall survive nonoperate temperatures of within -45° to +115° C (TBR).

Thermal Control Hardware

a. Heaters

Heaters shall be sized for minimum bus voltage and shall have margins in compliance of GSFC-STD-7000, GEVS.

b. Multi Layer Insulation (MLI)

Multi Layer Insulation shall meet the grounding requirements of ICD-A-14009-SM, Section 10.7.4.2.3.5.1.

c. Thermostats

Thermostats shall comply with the applicable requirements of STR-147, Section 11.

11.3.8.5 OVP Kit Installation

At the start of the OVP, the crew will retrieve the OVP-K caddy and P601 MLI Tent from the SLIC forward port side. On MCC GO and after the STOCC completes the power inhibits, the crew will install the P1A jumper plug between the DBA 2 J1 receptacle and the CSH P1 connector. The crew will then install the OVP Electronics unit (EU) on Handrail 21 via to EVA clamp knobs. The crew then mates the CSH J2 to EU P2. The crew will then remove the P601 MLI Tent installed on the +V2 trunnion on SM3B, and install the P601A Jumper plug into P601 installing the new P601 MLI Tent. The crew will then stow the OVP Caddy and old P601 MLI Tent.

11.3.9 Soft Capture and Rendezvous System

11.3.9.1 Overview

The Soft Capture Mechanism (SCM) will provide a means of attaching a future propulsion module or a standard interface for capture by a future servicing vehicle. The Relative Navigation Sensors (RNS) reside solely on the MULE and are designed to passively test equipment as a proof-of-technology experiment. The RNS will not be capable of motion, commanding or any other active interaction with surrounding equipment so as to cause a hazard. The RNS will, however, be designed so as not to interfere with EVA activities.

11.3.9.2 Structural Description

The SCM rides on the Flight Support Structure (FSS) for launch from which it is nominally deployed onto HST's aft bulkhead using a single drive point activated by a free-floating astronaut. In the event of a contingency, it would be restowed onto FSS for landing. Astronauts will deploy the SCM during an EVA using the Pistol Grip Tool (PGT) from a single drive point. Foot restraints will not be required for a nominal installation. When deployed, the SCM will be securely attached to the Aft Bulkhead end of the HST structure via HST's berthing pins. It will, however, be capable of providing status lights through a self-contained, battery powered system that uses a GFE EHIP battery.



Figure 11.3-31. TBD

When the STS launches, the SCM is positioned onto the FSS such that, upon capture of the HST, the SCM is configured for direct installation onto HST. Deployment of the SCM is conducted as an EVA task with a free-floater astronaut using a Pistol Grip Tool (PGT). Light Emitting Diode (LED) lights powered by a GFE supplied EHIP battery are used to indicate positive disconnect from the FSS and positive installation onto the HST. Manual verifications by mechanical means are available for establishing safe-to-land status following a failure to insure a safe reattachment to the FSS.



Figure 11.3-32. SCM in launch configuration (side view)

Flex drive shafts are used to disconnect the SCM from the FSS and to attach it to HST via three identical mechanisms. The flex drive will be activated by a single drive mechanism located on the SCM for ease of astronaut access. Figure 11.3-33 shows the routing of the flex drives. P-clamps are used to tie down the flex shaft every 6 inches. A secondary gearbox is used so that each leg of the flex-shaft provides equal torque by virtue of being the same length.



Figure 11.3-33. SCM flex drive system

The SCM is installed onto the HST using the berthing pins. The berthing latches grasp the HST and connect it to the FSS. When stowed on the FSS, the lower two arms clamp below the ball/cup connection, providing a preloaded interface for launch and landing. When the SCM is installed onto HST, the two upper arms clamp onto the HST berthing pins outside the berthing latches. The clamps are lined with RTV to prevent creak, which could interfere with HST science. Independent EVA overrides are provided for each major mechanical function activated using a PGT. The primary functions are as follows:

- a. Disconnecting from the FSS
- b. Connecting to the FSS
- c. Attaching to HST
- d. Disengaging from HST

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Figure 11.3-35. SCM EVA overrides

To prevent inadvertent activation of one or more of the SCM Mechanisms, the gearbox which drives the flex-shaft contains an antirotation device. The antirotation device uses two pawls as a positive lock against inadvertent rotation during launch or landing. One pawl is sufficient to prevent inadvertent activation. This device is disengaged once the PGT is engaged for installation onto the SCM.



Figure 11.3-36. SCM main gear box



Figure 11.3-37. SCM main gear box antirotation device

11.3.9.3 Electrical Subsystem

The SCM will not have any powered connections to HST or the STS.

11.3.9.4 Thermal Subsystem

N/A

11.3.9.5 SCRS Installation

To begin the ACRS Installation the crew will install an EHIP battery into the SCM battery housing. The battery is used to power the indicator lights on the SCM indicator panel. Once the battery is installed the Battery Power On switch is taken to enable, and the indicator lights tested to verify no LEDs are nonfunctional. The crew will then use a PGT to release the SCM from the FSS and engaging it to the HST by turning the SCM Gearbox until the FSS Release indicator lights illuminate. The crew will then take the Battery Power On switch to disable, and remove the EHIP Battery.

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11.3.10 Bit Sync Assembly

TBD

11.3.10.1 Overview

During the HST deploy and SM-1 missions, HST telemetry was downlinked via the Orbiter Ku-band and S-band systems. The S-band system downlinked the 32 kbps "PSP-Bypass" mode telemetry only, while the Ku-band system downlinked the 4, 32, or 1024 kbps telemetry in parallel. There are several known HST-related limitations of these modes.

The S-band limitation was known prior to the HST Deploy and SM-1 missions and is a consequence of the HST I&C hardware configuration. The normal path for a shuttle payload requiring RF processing consists of the following:

- a. Payload
- b. Payload Interrogator (PI)
- c. Payload Signal Processor (PSP)
- d. Payload Data Interleaver (PDI)
- e. S-Band downlink

However, HST does not utilize a PSK modulated subcarrier scheme. To accommodate this hardware limitation, JSC devised the "PSP-Bypass" mode concept. A mission unique cable was built to route the PI baseband output around the PSP, directly to the PDI. The PSP is bypassed because it requires use of a PSK modulated subcarrier input. During proof-of-concept testing prior to the Deploy mission, it was determined that PSP-bypass mode was feasible at the 32 kbps data rate only. For the HST Deploy and SM-1 flights, this limitation was acceptable for nominal on-orbit operations.

The Ku-band limitation was also known prior to the HST Deploy and HST SM -1 missions. The low-end bandpass for the orbiter Ku-band system is 16 kbps for Channel-2. HST telemetry at the 32 and 1024 kbps rates is passed within specification. However, HST 500 bps and 4 kbps telemetry is downlinked on a "best effort" basis. Additionally, Ku-band communications are dependent on maintaining direct line of sight between TDRS and the orbiter Ku-band antenna. Significant periods of interruption can be expected, thus effecting the requirement to provide near-continuous downlink to the STOCC. For the HST Deploy and SM-1 flights, these limitations were acceptable for nominal on-orbit operations.

To overcome the limitations, it was decided to insert a digital bit synchronizer between the output of the orbiter PI and input to the PDI to regenerate the degraded 500 bps and 4 kbps waveforms. The HST Payload General Support Computer Assembly (HPGSCA) hosted the digital bit synchronizer board on SM-2, SM-3A, and SM-3B.

Following SM-3B, concerns about HPGSCA included equipment availability in that only two flight-certified IBM 755 ThinkPad Laptops remain in HST stock (one prime, one spare). If one fails prior to launch, no replacement is available. Also, IBM ThinkPad Laptops are required to run antiquated Windows 3.1 Operating System in order to be compatible with heritage HPGSCA Aydin software (requires flying a stack of Operating System Diskettes and requires additional crew training with Windows 3.1).

The new HPGSCA design concept will be called the Bit Sync Assembly (BSA). This concept is designed to minimize risk by utilizing previously flown, heritage Aydin PC335 ISA cards (HST Project has four flight worthy Aydin PC335 bit sync cards in stock; two of the four have flown on SM2, SM3A, and SM3B).



Figure 11.3-38. BSA configuration

11.3.10.2 Structural Description

New hardware incorporated into the BSA includes an Actel SX72 FPGA-based Controller Card responsible for configuration of the Aydin card following poweron or upon Astronaut request via an externally accessible BSA switch panel interface located on the long thin front face of the BSA. The BSA is more compact than the previous HPGSCA; it is designed as a single all-in-one unit that requires no on orbit assembly other than power and data interface connections.

Design improvements incorporated into the BSA design include Controller Card Single Event Upset (SEU) immunity, a new DC/DC power supply design, passive power dissipation that does not require fan based convection, and a robust fault management algorithm that, when enabled, is capable of managing any and all reset operations required by the heritage Aydin Bit Sync Card due to its know susceptibility to SEUs. The Controller Card SEU immunity coupled with the onboard fault management capability eliminates the need for the crew to power down, uncable, and install a turn-

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around plug at the end of each mission day. The BSA may remain cabled, configured, and operational through the entire mission without requiring crew intervention. In addition, a new feature is a change of the data Y-cable to 1-to-1 cable with easy to mate circular connector (instead of two heritage D connectors).

The BSA incorporates "ease of use" features, including

- Minimal User Interfaces composed of a Power Switch (On/Off), a Reset Switch (Momentary), Data Polarity to PDI (Normal/Inverted), Bit Rate Selection (4 kbps, 32 kbps), and a Watchdog Enable/Disable Switch
- b. Does not require complicated time consuming reboot procedure, using only a Reset button or a Power Cycle if manual crew intervention required



c. Telemetry provided via LEDs

Figure 11.3-39. BSA front panel

11.3.10.3 Electrical Subsystem

Electrical power for the BSA is independent from HST.



Figure 11.3-40. BSA block diagram

11.3.10.4 Thermal Subsystem

11.3.10.5 BSA Installation

Two BSAs will be flown; one will be primary and the other designated as a backup. Only one BSA, the primary unit, will be removed from its launch stowage location and configured for operation during the mission.

The BSA will be transported from its stowage location to connect it with the Payload Data Interface Panel (PDIP). Crew will remove one PDIP turn around plug and mate the data cable from the BSA with the PDIP. The crew should verify that the BSA Power switch is in the OFF position prior to connecting the orbiter supplied power interface cable. With the BSA power switch in the OFF position, the +28 V dc power interface cable will be connected to the back of the BSA, followed by the data interface cable.

Once the BSA power and data cables are connected, the user should verify that the BSA front panel switch settings are configured as follows prior to enabling BSA power:

- a. PWR: OFF
- b. Rate: 32K
- c. Polarity: NORM
- d. Bit Sync Watchdog: DIS

Following verification of the above switch positions, the BSA PWR switch may be transitioned to the ON position. At the time of BSA power on, an LED all on lamp test is provided. The user should verify that all LEDs on the BSA front panel briefly illuminate and are not red.

Once HST has been captured and berthed and nominal HST data flow has been verified, the user should configure the BSA for the remainder of the mission as follows:

- a. PWR: ON
- b. Rate: AUTO
- c. Polarity: NORM
- d. Bit Sync Watchdog: EN
- Note: Reconfiguration can be completed with the BSA unit powered.

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11.3.11 ACS Repair

The Advanced Camera for Surveys (ACS) axial SI replaced the Faint Object Camera (FOC) axial SI during the HST Third Servicing Mission B (SM3B) in March 2002. On June 19, 2006, a Low Voltage Power Supply (LVPS) anomaly occurred on ACS Side 1, resulting in loss of all High-Resolution Channel (HRC) and Wide Field Channel (WFC) CCD Electronics Box (CEB) telemetry. ACS was then switched to Side 2 and full science capability was restored. On September 23, 2006, ACS Side 2, suffered an anomaly in the HRC CEB relay, but a cycling of the relay again restored full science capability. However, on January 27, 2007, ACS Side 2 again sustained an anomaly with a blown fuse, which resulted in loss of the premier ACS science capability—only the Solar Blind Channel is currently delivering ACS science. During SM4, the WFC CEB electronic boards will be replaced with a module (CEB-R) and a new LVPS (LVPS-R) installed without removing ACS from HST. This will restore the WFC science capability and potentially restore the HRC science capability to ACS.

The WFC CEB EMI Grid is approximately 17 by 6 by 0.2 inches (TBR) and weighs approximately 1 (TBD) pound; the Electronics Board Cover is approximately TBD by TBD by TBD inches and weighs approximately TBD pounds; each of the four CEB PWA boards are approximately 15 by 8 by 0.5 inches (TBR) and weigh approximately 1 (TBR) pound; the CEB-R is approximately 15 by 3 by 9 inches and weighs approximately 20 (TBR) pounds; and the new LVPS is approximately 13 by 3 by 7 inches and weighs approximately 23 (TBR) pounds.

Repairing the ACS requires the EVA crewmembers to maneuver to the HST -V2 Aft Shroud doors; open the doors; retrieve and install four guide studs; retrieve and install the EMI Grid Cutter; cut, remove, and stow the EMI Grid with the EMI Grid Cutter; retrieve and install the ACS Fastener Capture Plate (FCP); remove and stow the CEB Access Cover Plate with the ACS FCP; remove and stow the four WFC CEB electronic boards; retrieve and install the CEB Replacement Module (CEB-R); retrieve and install the LVPS (LVPS-R) with connections to the ACS J1/P1 power connectors and to the CEB-R; and close the -V2 Aft Shroud doors.

Operationally, the AS doors shall not be opened until after the orbiter and FSS have been oriented to prevent direct sunlight from entering the AS.

11.3.11.1 Detailed EVA Requirements

The crew will begin the ACS repair by moving the BAPS system via the A6U panel switch to allow the -V2 to be opened. The crew will then disengage the -V2 latches and open the doors to their full open position. The crew will then remove four fasteners from the ACS WFC CEB, and installing the four guide studs into the fastener holes.

Once the guide studs are in place, the crew will retrieve and install the EMI grid cutter and ground strap from the NOPE. The crew will then cut the EMI grid and remove the grid and the cutter and restow in the NOPE. Next the crew will retrieve and install the ACS capture plate on the guide studs. The ACS capture plate allows the crew to remove 32 non-captive fasteners and the CEB cover from the ACS to expose the CEB

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electronics cards. The crew will then release all of the CEB non-captive fasteners and stow the cover for return.

With the cover CEB cover removed the crew will retrieve and install the ACS CEB Card Extraction Tool (CET). With the tool installed they will remove all four of the CEB PWA Cards. If there is a day break in between the removal of the CEB boards and the installation of the boards the crew will install an ESD protective cover to prevent FOD from entering the ACS CEB. After the CEB cards are removed they will be stowed with the CET in the NOPE for return to earth.

The crew will retrieve the CEB replacement module and LVPS-R from the LOPE. The crew will install the CEB replacement module in the ACS, and mount the LVPS-R to the ACS handhold. The crew will then unstow the LVPS-R Power Input Element (PIE) Harness from the LVPS-R. They will then remove the ACS power connector P1 from ACS J1, and install it onto the PIE Harness J1. Like wise PIE Harness P1 will be installed into ACS J1, and finally PIE harness P6 will be connected to LVPS-R J6. The crew also connects Power Output Element (POE) Harness P8 to CEB replacement module J8. The crew will then closes out the ACS worksite, and close and latch the -V2 doors.

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11.4 SAFETY

11.4.1 Introduction

This section provides a scaled down version of the safety requirements to which the HST satellite adheres for the fourth servicing mission. This safety information comes directly from HST hazard reports and the accompanying payload inputs to the Payload Safety Review Panel. It primarily focuses on the operational safety workarounds to hazardous situations and the payload inhibits to prevent those hazards from occurring. For generic payload system safety requirements, see the appropriate system section in this document.

Hazardous subsystems that are inadvertently activated or not properly safed during the rendezvous, retrieval, berthing, EVA servicing, and deployment could result in severe damage to the orbiter or injury to the crew. All requirements for safing and monitoring of HST hazardous systems are called out in individual hazard reports. For a complete list of the hazards for each mission phase, refer to HR-32.

11.4.2 Rapid Safing

For a rapid safing condition, both sides of the FSS CDUs are utilized for mechanism operation to expedite HST deploy and payload safing (dual motor ops). The HST is released via an orbiter backaway maneuver, and the FSS is pivoted down. The FSS must pivot to 45° to clear the payload bay envelop, and 43.8° (with BSP latched) for landing. If the FSS is in the reboost configuration, the FSS has to be pivoted up before HST backaway. If the SCM has not been released from the FSS, then the BAPS support post must be in place. Additionally, any stray or potentially PLBD-offending ORUs/ORIs crew aids or tools will be EVA jettisoned.

For emergency deorbit, the payload is safed in less than 20 minutes, and for the Next Primary Landing Site (NPLS) contingency, the payload is safed within 50 minutes.

Five scenarios were chosen as representative worst-case payload bay configurations and each was analyzed in detail. These included four EVA scenarios and a non-EVA crew-sleep scenario. The chosen scenarios adequately envelop all nominal phases of the Servicing Mission for rapid safing considerations. Note that the pivoted-down reboost position is not a planned configuration for SM4.

The EVA scenario no. 1 initiates near the midway point in a representative Axial Scientific Instrument (COS) changeout. The EVA scenario no. 2 initiates near the midway point in a representative Radial Scientific Instrument (WFC3) Changeout. The EVA scenario no. 3 initiates near the midway point in a representative Fine Guidance Sensor (FGS) changeout. The EVA scenario no. 4 initiates near the midway point in a representative Battery Module changeout. A fifth scenario is a crew-sleep scenario and initiates from the nominal crew-sleep payload bay configuration.

Rapid safing functions for these scenarios depict the maximum number of crew tasks that would be required to safe the payload for door closure in the event of rapid safing at the worst possible time.

Based on current estimates, the 20 minute requirement for initiation of PLBD closure can be met for all mission phases with zero fault tolerance. In all cases, dual motor pivoter operation (considered zero fault tolerant) is required to meet the 20 minute requirement. Per NSTS-JSC, MA2-96-190, Contingency Return and Rapid Safing, the 1:45 (h:mm) requirement for the Next Primary Landing Site can be met with single motor pivoter operation (single fault tolerant). **27:00** (mm:ss) is the approximate NPLS rapid safing time for single motor pivoter operation for all SM4 activities (obtained by doubling the pivot time).

11.4.3 EVA Operations

During most EVA operations, electrical connectors are deadfaced prior to removal/ installation. For SM-4, the exceptions to this are the battery connectors during battery module changeout and for unscheduled operations, Fuse Plug replacement, VIK replacement, and ECU replacement.

The connector design and the method of demating/mating the connectors will not pose a hazardous condition to the astronauts. Procedures with the EVA astronauts identify energized connectors and demonstrate the ability to manipulate energized connectors without grounding, mismating, or shorting. Any special EVA protocol necessary to minimize conductive hardware/material in connector worksite vicinity is identified. Procedures will minimize time the demated connector is exposed while not under positive control of an EVA/IVA astronaut.

All remaining connectors have at least one upstream inhibit in place during EVA connector demating/mating activities. Connectors that are mated/demated by an astronaut are deenergized (<500 mA) by opening a minimum of one controlling power relay. During mating/demating of all connectors, commands to the HST are inhibited by disabling the remote command panel at JSC. EVA connectors that are energized are designed as scoop-proof, remote/mechanically driven, or use connector covers.

Depending on a connector, the types of inhibits vary in kind. In general, a scoop-proof or rack-and-panel connector design is treated as an inhibit. Any switches or relays upstream with verifiable monitor and downstream are considered inhibits. A high impedance line or any line that can provide no more than 0.5 A is considered nonhazardous. (All the data lines belong in this category, and they are considered nonhazardous.) The inhibit-compliant criteria for the connectors are qualifying connector design (scoop-proof or rack-and-panel), upstream inhibit (switches or relays) with verifiable monitor, and downstream inhibit (load removed).

The connectors of the following hardware and/or interfaces have been reviewed for potential hazardous conditions either to the astronaut or to the orbiter during the third servicing mission EVA activities.

At this time, the planned ORU hardware is assumed to be the following:

- a. Replace FGS3
- b. Replace three RSUs
- c. Replace Battery Modules
- d. Repair STIS
- e. Repair ACS
- f. Install NOBLs on bays 7 and 8

In addition to the above hardware connector interfaces, there are receptacles in the ORU carrier and HST that are to be used by the astronauts for powering the portable worklights.

The following hardware components have less than the proper number of inhibits required to ensure the deadfacing of HOT connectors. Waivers have been provided for demating the connectors on these components.

Battery Modules

Battery modules are unique since both sides of the connector are powered. The on-orbit battery does not have an upstream inhibit; however, the replacement battery uses a Battery Isolation Switch (BIS) which deadfaces the connectors on the battery side. Installation of the battery modules requires the demate and mate of three J2 connectors and three J1 connectors. By design, the J2 connector pins are powered; however, a plug was installed on-orbit during SM3A which renders the battery (powered) side as sockets. Replacement batteries have preinstalled plugs to convert the pins to sockets.

ECU Changeout (Unscheduled)

The EVA task would be required only if a gyro ECU failure were experienced prior to or during SM4. ECU connectors V13GXP3 and V13GXP4 are considered high power and are mated/demated hot. However, the ECU connectors are mated using a remote drive.

PDU P15, P16, and P17 Fuse Plug and OTA P27A/P28A Fuse Plug Changeout (Unscheduled)

The above fuse plugs are carried in the event of a contingency that might require a fuse changeout. The fuse plugs have the following design attributes that mitigate the hazard involved during changeout. All plugs have EVA wingtabs with PDU P15, P16, and P17 fuse plugs that have had their wingtabs and tether loops covered or made from a nonconductive material. The following EVA protocol is implemented to increase safety.

An HST PFR is positioned in front of the bay 4 door on which the PDUs are mounted, providing a stable work platform. Any unnecessary conductive hardware within the worksite vicinity is stowed before commencing. The fuse plugs are changed out one at a time.

11.4.4 Touch Temperatures

There are no touch temperature concerns for the HST servicing mission. All HST systems remain within the -180° to +235° F range, even with heaters failed ON or OFF. All surface temperatures are maintained for the nominal EVA orbiter attitudes.

11.5 DISPLAYS, CONTROLS & VISUALS

11.5.1 Standard Switch Panel and A6U

Display and Control (D&C) functions are accomplished using the following crew-controlled equipment:

- Use of the A6U panel is required for the operation of the FSS mechanisms via the FSS mechanism selection box for SMs. The FSS requires a separate A6U switch to control each redundant PRLA power system (one switch for system 1, and one switch for system 2).
- Three sections of the standard switch panel for the SMs (one for FSS, one for HST, one for PSP bypass).
- For SM4, two additional sections of the standard switch panel are required (one for the SLIC and one for the MULE).

11.5.2 Crew Displays and FDA Tables

TBD

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JSC-29029

JSC-2902

APPENDIX A SUPPLEMENTAL DATA

A.1 HST PAYLOAD CONTROL WEIGHTS

Table A-1 is a summary of HST payload weight by individual component for Servicing Mission 3B. For a complete listing, see ST/SE-04, HST SM3B Mass Properties Status Report.

Description	Current basic wt (lb)	
Total ST on orbit (pre-changeout)	24395.0	
SA3 (2)	1320.0	
PCUR	159.8	
ACS	883.0	
ESM	212.0	
NCC	308.0	
NCC RADIATOR	309.0	
NOBL (Bays 5 – 8)	42.9	
DBA2 (2)	96.2	
Changeout complement	3330.9	
SA-II (2)	702.0	
DBA (2)	30.0	
PCU	152.0	
FOC	726.0	
Changeout complement - Earth return	1610	
Total ST on orbit (post-changeout)	26157.0	

Table A-1. HST third SM summary weight statement
(nominal mission)

A.2 NICKEL-HYDROGEN (NIOOH-H₂) BATTERIES

Description

The Nickel-Hydrogen batteries are rechargeable. It uses a nickel hydroxide electrode produced by electrochemical deposition of nickel hydroxide into the pores of a sintered nickel plaque with screen support. The hydrogen electrode comprises a Teflon bonded platinum black on a nickel mesh screen to allow reduction and oxidation of hydrogen. A gas diffusion screen facilitates hydrogen diffusion. The electrolyte is 30 to 35 percent aqueous potassium hydroxide by weight. An asbestos fuel cell bidirectional separator is used. The cell configuration can contain 800 to 1000 psi of hydrogen on full charge. The reaction is given as

disch $2NiOOH + H_2 \leftarrow 2Ni(OH)_2$ chg

A plasma sprayed zirconia oxide is sometimes used to coat the inside of the cylindrical pressure vessel to enhance electrolyte distribution. It has the potential for greater depth of discharge and higher rate capability than nickel-cadmium.

Operation

The cell can be charged at a relatively high rate constant current (~C/2) similar to the nickel-cadmium cell. During charge, hydrogen is evolved at a rate consistent with the charge rate. Pressure builds up to 800 to 1000 psi. During overcharge, the O_2 generated in place of the Ni(OH)₂ oxidation reaction reacts with the hydrogen generated according to the reaction:

 $2H_2 + O_2 \rightarrow 2H_2O$

thus maintaining the high pressure at a constant value. The cells are in a cylindrical pressure vessel container that can withstand up to 2400 psi. On discharge, the hydrogen recombines in the cell. If discharged to 100 percent DOD, the cell pressure returns to its original value.

The nickel-hydrogen is reported to be an improvement over the nickel-cadmium because the cadmium electrode, which degrades with life and depth of discharge, is replaced by the H_2 electrode. However, this has not yet been proven.

Applications

This type of cell has had one major application, that of replacing the nickel-cadmium in GEO spacecraft. The enhanced specific energy (Wh/kg) over nickel-cadmium cells offers a greater opportunity for increasing the spacecraft capability of reducing total mass. GEO (22,000 miles above the Earth) affords only two eclipse periods per year, thus requiring ~100 cycles/year. The nickel-cadmiums are operated from 65 to 75 percent DOD. The nickel-hydrogen cells are projected to reach 85 percent DOD. Small nickel-hydrogen cells have been designed for a number of other specialized applications. However, the nickel-hydrogen offers an improvement in cell sizes greater than 50 Ah while the nickel-cadmium is more efficiently used below this value. There is also interest in their use in *Low Earth Orbit (LEO)* where their capability for greater DOD has a distinct advantage. Although the specific energy of nickel-hydrogen cells is greater than nickel-cadmiums, the volume energy density is less.

Advantages

- a. State of charge directly related to pressure
- b. Specific energy/power greater than nickel-cadmium
- c. Higher depth of discharge than nickel-cadmium projected (not yet proven)
- d. Can tolerate overcharge and reversal at controlled rates

Disadvantages

- a. High cost
- b. Self-discharge proportional to decrease H₂ pressure
- c. Volume energy density and power << nickel-cadmium

Potential Hazards

a. High-rate charging and heat buildup.

Use charge control and thermostatic devices.

b. High-pressure buildup.

Use strain gage to monitor pressure.

A-3

- c. Leakage of strong alkaline electrolyte.
 - 1. Prevent by performing seal and cell testing.
 - 2. White encrustation on seal or case acid indicates a leak.
 - 3. Inspection and welder qualification and calibration necessary.
 - 4. Use rubber gloves when handling electrolyte.
- d. Gas (H₂ or O₂) leakage creates cell imbalance potential of exceeding pressure limit.
 - 1. Perform helium leak testing on each cell.
 - 2. Use high reliability, well-tested seals.
- e. External shorting can result in very high-rate, high-temperature excursion leading to violent venting.

Use fuse and other protective circuitry.

APPENDIX B CONFIGURATION CONTROL

B.1 INTRODUCTION

B.1.1 Purpose

The purpose of *Appendix B* is to delineate configuration control procedures for the Cargo Systems Manual (CSM). This will ensure the proper coordination of changes and provide a record of proposed changes, rationale, and disposition.

B.2 CONFIGURATION CONTROL PROCEDURE

B.2.1 Submission of Changes

Proposed changes are solicited from any individual or any organization having a valid input. Changes should be submitted to the CSM book manager at mail code DO5, NASA/Johnson Space Center, Houston, Texas 77058.

Individuals desiring to submit a change should complete all applicable items on the Change Request Form (see Figure B-1). This form should be typed or neatly printed. Additional pages may be used if the space provided is not adequate. The original completed form should then be forwarded to the CSM book manager.

B.2.2 Disposition of Changes

The CSM book manager will obtain formal concurrence and comments from the necessary personnel, including customer representatives. All proposed modifications to the CR will be coordinated with all necessary personnel prior to disposition.

Upon obtaining the required concurrences, the CSM book manager will present the proposed change to the appropriate section head for final disposition.

A copy of all dispositioned CRs will be retained by the CSM book manager for future reference. A courtesy copy of all approved CRs will be sent to ZC01/J. Woodall, EV12/D. Sykes, and MS3/F. W. Parker.

B.3 CSM REVISIONS/PCNS

B.3.1 Development

The CSM book manager will compile all approved changes and any typographical errors and incorporate them into a revision or PCN to the document.

Pen and ink changes may be used to correct typographical errors if there are no other changes on the page concerned.

B.3.2 Approval

Any revisions/PCNs to the document will be approved by the appropriate section head. Revisions/PCNs to the final versions will also be approved by Space Shuttle Systems Integration Office *manager*.

B.3.3 Publication

Revisions/PCNs will be made on an as-required basis. Revisions/PCNs will be printed and distributed to the standard distribution list.
JSC-29029 Basic, Rev. A

NASA - JOI CHANGE F	NASA - JOHNSON SPACE CENTER CHANGE REQUEST FORM		NUMBER:
SECTION(S) AFFECTED:			
WAS:			
IS:			
RATIONALE FOR CHANGE:			
ORIGINATOR		CONCURRENCE	
ORGANIZATION		LEAD PAYLOAD OFFICER	DATE
TELEPHONE		CUSTOMER REPRESENTATIVE	DATE
BOOK MANAGER		APPROVED	DISAPPROVED
DA	ATE	APPROVE WITH	CHANGES INDICATED
INCORPORATED:	TE	SECTION HEAD	DATE

Figure B-1. Change Request Form

B-3

APPENDIX C ACRONYMS AND ABBREVIATIONS

Å A-h ac ACE ACP ACS AD AFD AFD AFM APS AS ASCS ASLR ATM	Angstrom Ampere-Hour alternating current Actuator Control Electronics Astronaut Control Panel Advanced Camera for Survey Aperture Door Aft Flight Deck Articulated Fold Mirror Antenna Pointing System Aft Shroud Aft Shroud Cooling System Aft Shroud Latch Repair Apollo Telescope Mount
BAPS	Berthing And Positioning System
BCU	Bus Coupler Unit
BOD	Bright Object Detector
BOL	Beginning of Life
BOT	Beginning of Travel
BPRC	Battery Protection and Reconditioning Circuit
bps	Bits Per Second
BPSK	Biphase Shift Keyed
BREC	Body Rate Error Check
BSA	Bit Synchronizer Assembly
C&DH	Control and Data Handling
C/C	Configuration Control
CAH	Charge Ampere-Hours
CASH	Cross Aft Shroud Harness
CC	Cargo Control
CCC	Charge Current Controller
CCD	Charge-Coupled Device
CCTV	Closed-Circuit Television
CCW	Counterclockwise
CDI	Command and Data Interface
CDU	Command Detector Unit
CEB	CCD Electronics Box
CET	Card Extraction Tool
CFRP	Carbon Fiber Reinforced Plastic
CGG	Contingency Gravity Gradient
CIF	Computer Interface
CIT	Card Insertion Tool

CMD COM COPE CORU COS COSTAR CPL CPM CPU CSM CSS CU/SDF CW	Communications Module Contingency ORU Protective Enclosure Candidate Orbital Replacement Unit Cosmic Origins Spectrograph Corrective Optics Space Telescope Axial Replacement Capillary Pumped Loop Central Processor Module Central Processing Unit Cargo Systems Manual Coarse Sun sensor Control Unit/Science Data Formatter Clockwise
D/A DAH DAK dB DBA DBC dc DCE DIH DISP DIU DIUI DIUI DIUI DMS DMU DOB DOD DPC DRAM	Digital to Analog Discharge Ampere-Hours Double-Aluminized Kapton Decibel Diode Box Assembly Diode Box Controller Direct Current Deployment Control Electronics Discrete Input High Display Data Interface Unit DIU Interface Module Data Management Subsystem Data Management Unit Deployable Optics Bench Depth-Of-Discharge Direct Power Converter Dynamic Random Access Memory
ECA ECU ED EDB EMI EOL EOT EP/TCE EPS ES ESA ESA ESEL ESM	Electronics Control Assembly Electronics Control Unit Engineering Data External Data Bus Electromagnetic Interference End Of Life End Of Travel Electrical Power/Thermal Control Electronics Electrical Power System Equipment Section European Space Agency EVA Support Equipment List Electronics Support Module

ESTR	Engineering/Science Tape Recorder
ETR	Engineering Tape Recorder
EU	Electronics Unit
EVA	Extravehicular Activity
F/S FCP FDM FEP FGE FGS FHST FHSTI FIFO FMDM FOB FOC FOD FOD FOM FOS FOV F/S FPA FPDA FPSA FSS FSS FSW FWH	Filter/Shaper Fastener Capture Plate Field Dividing Mirror Fluorinated Ethylene Propylene Fine Guidance Electronics Fine Guidance Sensor Fixed Head Star Tracker Fixed Head Star Tracker Interface First In First Out Flexible Multiplexer/Demultiplexer Fixed Optical Bench Faint Object Camera Flight Operations Directorate Field Offset Mechanism Faint Object Spectrograph Field of View Filter/Shaper Focal Plane Assembly Focal Plane Deck Assembly Focal Plane Deck Assembly Forward Shell Flight Support System Flight Software Flexible Wire Harness
GEA	Gimbal Electronics Assembly
GFE	Government Furnished Equipment
GG	Gravity Gradient
GGM	Gravity Gradient Mode
GHRS	Goddard High Resolution Spectrograph/Spectrometer
GPC	General Purpose Computer
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
GSTDN	Ground Spaceflight Tracking and Data Network
HGA	High-Gain Antenna
HLD	High-Level Discrete
HOSC	Huntsville Operations Support Center
HRC	High Resolution Channel
HRS	High Resolution Spectrograph

HSP	High-Speed Photometer
HST	Hubble Space Telescope
HWL	Hardware Load
I&C	Instrumentation and Communication
ICAPC	Increased Capacity Adaptive Payload Carrier
ICU	Instrumentation Control Unit
IDB	Internal Data Bus
IMU	Inertial Measurement Unit
IOU	Input/Output Unit
IPCU	Interface Power Control Unit
IR	Infrared
IV	Intravehicular
JSC	Lyndon B. Johnson Space Center
KA	Keep Alive
kbps	Kilobits Per Second
KSC	Kennedy Space Center
KUSP	Ku-band signal processor
lb LED LGA LHC LLD LMU LS LMSC	Pound Light Emitting Diode Low Earth Orbit Low-Gain Antenna Left Hand Circular Low-Level Discrete Logical Memory Unit Light Shield Lockheed Missiles and Space Company
MA	Multiaccess
MAMA	Multianode, Microchannel Array
Mbps	Megabits Per Second
MCC	Mission Control Center
MCC-H	MCC Houston
MCE	Monitor and Control Electronics
MCU	Mechanism Control Unit
MDB	Multiplexed Data Bus
ME	Minimum Energy
MEB	Main Electronics Box
MET	Mission Elapsed Time
MHZ	Megahertz
MLD	Manual Locking Device
MLI	Multilayer Insulation
MMS	Multimission Modular Spacecraft

MOD	Mission Operations Directorate
MOPE	Multimission ORU Protective Enclosure
MR	Main Ring
MRA	Main Ring Assembly
MSB	Most Significant Bit
MSFC	Marshall Space Flight Center
MSS	Magnetic Sensing System
MT	Magnetic Torquer
MTE	Magnetic Torquer Electronics
MTP	Master Timing Pulse
MTS	Magnetic Torquing System
MU	Memory Unit
MULE	Multi-Use Lightweight Equipment
N'm	Newton meter
n. mi.	Nautical Mile
NASA	National Aeronautics and Space Administration
NASCOM	NASA Communications
NCC	NICMOS Cyro-Cooler
NCS	NICMOS Cooling System
NGT	NASCOM Ground Terminal
NICMOS	Near Infrared Camera/Multiple Object Spectrometer
NiCd	Nickel-Cadmium
NiH ₂	Nickel-Hydrogen
nm	nanometer
NOBL	New Outer Layer Blanket
NPLS	Next Primary Landing Sight
NSSC-I	NASA Standard Spacecraft Computer, Model I
OAO OCE OCS OCXO OLD OPI ORI ORU ORUC OSO OTA OV OZ	Orbiting Astronomical Observatory Optical Control Electronics Optical Control Subsystem Oven-Controlled Crystal Oscillator Offload Device Orbiter Payload Interrogator Orbital Replacement Instrument Orbital Replaceable Unit Orbital Replaceable Unit Orbital Replacement Unit Carrier Orbiting Solar Observatory Optical Telescope Assembly Orbital Verification Ounce
P-E	Perkin-Elmer Corporation
PACOR-II	Packet Processor II
PAM	Pupil Adjust Mechanism

PC	Planetary Camera
PCE	Power Conversion Electronics
PCM	Pulse-Code Modulation
PCS	Pointing and Control System
PCT	Portable Connector Tray
PCU	Power Control Unit
PDI	Pavload Data Interleaver
PDIP	Payload Data Interface Panel
PDM	Primary Deployment Mechanism
PDSU	Power Distribution and Switching Unit
PDU	Power Distribution Unit
PGSC	Payload General Support Computer
DI	Payload Interrogator
	Payload Interrogator/Ku band Signal Processor
	Processor Interfoce Table
	Processor Interface Table
	Payloau Recoluel
	Primary Mirror Accomply
PIMA	Primary Mirror Assembly
PMB	Primary Mirror Baffle
PMI	Photomultiplier Tube
PMU	Physical Memory Unit
PN	Pseudorandom Noise
POCC	Payload Operations Control Center
PRCS	Primary Reaction Control System
PRD	Payload Retention Device
PRI	Primary
PRIPL	Unregulated Orbiter Power
PRLA	Payload Retention Latch Assembly
PROM	Programmable Read-Only Memory
PSDP	Payload Station Distribution Panel
PSEA	Pointing and Safe-Mode Electronics Assembly
psi	Pounds per square Inch
psia	Pounds per Square Inch Absolute
PSP	Pavload Signal Processor
PSP/PI	Pavload Signal Processor/Pavload Interrogator
-	
QD	Quick Disconnect
QEH	Quantum Efficiency Hysteresis
RBM	Radial Bay Module
RFD	Redundant
RF	Radio Frequency
RFI	Radio Frequency Interference
RGA	Rate Gvro Assembly
RGAI	RGA Interface
	Pate Integrating Cyro
	Nate integrating Gyru

RIU	Remote Interface Unit
RMGA	Retrieval Mode Gyro Assembly
RM	Remote Module
	Reimaging Mirror
RMS	Remote Manipulator System
ROM	Read-Only Memory
RPA	Reaction Plate Assembly
R/R	Recharge Ratio
RSS	Root Sum Square
RSU	Rate Sensor Unit
RTC	Real-Time Command
RWA	Reaction Wheel Assembly
S→	SDM selectable driver switch
SA	Solar Array
	Storage Area
SAC	Solar Array Carrier
SAD	Solar Array Drive
SADA	Solar Array Drive Adapter
SADE	Solar Array Drive Electronics
SADM	Solar Array Drive Mechanism
SD	Scientific Data
SDF	Science Data Formatter
SDM	Secondary Deployment Mechanism
SFOC	Space Flight Operations Contract
SI	Scientific Instrument
SIC&DH	Scientific Instrument Control and Data Handling
SIP	Standard Interface Panel
SIPE	Scientific Instrument Protective Enclosure
SISS	SI Support Structure
SM	Safe-Mode
	Secondary Mirror
SMA	Secondary Mirror Assembly
SMB	Secondary Mirror Baffle
	STIK Mounting Bracket
SMC	Safe-Mode Computer
SMCH	Standard Mixed Cargo Harness
SMSA	Secondary Mirror Subassembly
SMU	Safe-Mode Utility
SOC	State-Of-Charge
SOFA	Selectable Optical Filter Assembly
SPA	Solar Panel Assembly
SPC	Stored Program Command
SPG	Single-Point Ground
SQPSK	Staggered Quadriphase Shift Keyed
SS	Safing System

SSA SSC SSE SSM-ES SSP SSR SSRF SSS STACC STDN STIK STIKER STINT STIS STOCC STSCI	S-band Single Access Science Support Center Space Support Equipment Support Systems Module SSM Equipment Shelf Standard Switch Panel Solid State Recorder Shield/Shelf Replacement Fabric Star Selector Servo Standard Telemetry and Command Components Spaceflight Tracking and Data Network STIS Thermal Interface Kit STIK Extended Radiator Standard Interface Space Telescope Imaging Spectrograph Space Telescope Operations Control Center Space Telescope Science Institute
T	Time
TAH	Total Ampere-Hours
TAG	Two-Axis Gimbal
tb	talkback
TBD	To Be Determined
TCE	Thermal Control Electronics
TCS	Thermal Control Subsystem
TCS	Temperature Controlled Crystal Oscillator
TCXO	Tracking and Data Relay Satellite
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TEC	Thermoelectric Cooler
TFC	Telemetry Format Control
TI	Texas Instrument
TIM	Timing Interface Module
TLM	Telemetry
TM	Turning Mirror
TRI	Tape Recorder Interface
TTL	Transistor-to-Transistor Logic
UDM	Umbilical Disconnect Mechanism
ULE	Ultralow Expansion
URM	Umbilical Retraction Mechanism
USA	United Space Alliance
UV	Ultraviolet
V	Volt
VATA	Vehicle Assembly and Test Area
VCO	Voltage Controlled Oscillator

VDA	Vacuum Deposited Aluminum
V/M	Volts/Meter
VSS	Vehicle Support Software
VSWR	Voltage Standing Wave Radio
V/T	Voltage/Temperature
W	Watt
WFC	Wide Field Camera
WFS	Wave Front Sensors
WF/PC	Wide Field/Planetary Camera
WSC	White Sands Complex
ZOE	Zone Of Exclusion

APPENDIX D HST ANOMALY LOG

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
1	GHRS Low Voltage Power Supply Intermittent Failure and Carousel Operational Anomaly (Pre-SM1)	The GHRS power supply anomaly was attributed to an open solder connection. This failure and the carousel operational anomaly affect only the sci- ence operation of the GHRS and therefore do not pose any safety impacts to the HST or SM. GHRS Repair Kit was installed onto GHRS during SM1. GHRS was replaced by STIS on SM2	No safety impact
2	FGE Bit Flip (Pre-SM1)	This anomaly has no safety impact because the FGE will not be operational during any of the SM phases	No safety impact
3	PCU Relay K16 Failure (Pre-SM1)	The K-16 relay within the Power Control Unit (PCU) is failed open. The failure has been attributed to a loose spring assembly within the relay. The spring assembly is believed to be fixed at its present location. However, an analysis of possible connections that could be made by further movement of the loose spring assembly (if it is still free to move) within the relay has determined that a short from the +DD Solar Panel Assembly (SPA) or Battery-1 to battery return, DIU-2 instrumentation output, relay coils, or chassis/coil return could occur. None of these conditions is a safety impact to the orbiter or crew (*) because	No safety impact
		 Power to the K-16 relay is diode isolated from the HST power bus. 	
		The solar array is an inherently current limited source.	
		 The relay coils and DIU are high impedance devices. 	
		• Ground testing has determined that the battery and its associated cabling have sufficient capacity to clear a short to chassis/coil return within the relay, and damage would be contained within the relay or the PCU.	

Note: Failures with no safety impact are grayed out.

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
		This failure occurred prior to SM1 and has existed with no change of state within the relay.	
		This failure will be eliminated from the system with the replacement of the PCU planned for SM3B.	
		(*) There is no safety impact due to this failure as opposed to the other-side K44 relay (see anomaly #4, which describes the K44 failure impact on DIU changeout)	
4	PCU Relay K-44 Failure (Pre-SM1)	The K-16 relay within the Power Control Unit (PCU) is failed open. The failure has been attributed to a loose spring assembly within the relay. The spring assembly is believed to be fixed at its present location. However, an analysis of possible connections that could be made by further movement of the loose spring assembly (if it is still free to move) within the relay has determined that a short from the +DD Solar Panel Assembly (SPA) or Battery-1 to battery return, DIU-2 instrumentation output, relay coils, or chassis/coil return could occur. None of these conditions is a safety impact to the orbiter or crew (*) because	Requires specific HST electrical system configuration during servicing missions prior to PCU changeout
		 Power to the K-44 relay is diode isolated from the HST power bus. 	
		The solar array is an inherently current limited source.	
		 The relay coils and DIU are high impedance devices. 	
		Ground testing has determined that the battery and its associated cabling have sufficient capacity to clear a short to chassis/coil return within the relay, and damage would be contained within the relay or the PCU.	
		This failure occurred prior to SM1 and has existed with no change of state within the relay.	

Note: Failures with no safety impact are grayed out.

HST ANOMALY LOG (continued)
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#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
		The K-44 failure requires a unique configuration of the HST electrical power system during servicing missions (until PCU replacement) to ensure mission success. When HST is on orbiter power, there is no direct charge control capability for battery-4 with K-44 failed closed since Charge Current Controller (CCC) relay K-42 must not be allowed to open in order to protect DIU-2B from high voltage damage. (See "DIU-2 A Side Failure" anomaly (#TBD).) Therefore, to prevent overcharge of Battery-4 when in the servicing configuration, solar array section-4 will be switched to diode bus-C. To prevent high voltage damage to the essential bus, which could result if diode bus-C were inadvertently unloaded when in this configuration, the internal essential bus will be powered off whenever solar array section-4 is switched to diode bus-C. Thus, the essential bus will be required to be powered by the external service when on orbiter power.	
		The K-44 failure results in a hot connector at the DIU-2. See "DIU-2 A Side Failure" (anomaly #6), which is made safe by solar array shading, blocking diodes, and scoop-proof gender changers, as described in the NCR for applicable servicing missions. This failure will be eliminated from the system with	
		the replacement of the PCU planned for SM3B	
5	The -V3 aft shroud doors did not align easily for closure, delay in closure of the -V3 aft shroud doors. (During SM1)	An Aft Shroud Door Trainer was fabricated using flight materials (including titanium sheet) and flight drawings, simulating the -V3 doors, was used extensively for crew training for STS-103 (HST-SM3A), and will be used for future servicing missions. The -V3 doors are the hardest to close.	No safety impact, mission success concern
		No safety impact; mission success concern	
b	(During SM1)	The A side of the on-orbit DIU-2 was subjected to an overvoltage condition during SM1 because the K-44 relay failure allowed open circuit solar array voltage to reach the DIU when the CCC relay K-42 was open (battery-4 at full charge). After this failure occurred, the CCC K-42 voltage- temperature level was raised to the highest level such that K-42 will never open during normal operations. This prevents a recurrence of the high voltage into the DIU-2 B-side. Changeout of DIU-2 occurred on SM2. The DIU changeout had hot connectors and was not compliant with	impact

Note: Failures with no safety impact are grayed out.

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
		JSC/MA3-94-002, but can be made operationally safe as described in the NCR. NCR HST SM2FNCR1 was submitted 3/96.	
		Reference HR EVA-4 for the safety assessment of EVA handling of hot connectors. DIU-2R has overvoltage protection built in, so open-circuit solar array voltage will not cause damage to the new unit	
7	Solar Array II Short (Post-SM1)	The +E SPA output (5 percent of the total array output) was lost because of a short between the +E SPA output and the +B SPA return. The short is believed to have occurred at the inner buffer assembly at the base of the blanket. The short initially resulted in battery-5 failing to reach full state-of-charge and erroneous solar array section-4 current readings. The state-of-charge imbalance was corrected by raising the CCC K-51 voltage-temperature level. The short cleared itself in 1/95 and has not reoccurred. The CCC K-51 level was lowered back to its normal level. Shorts of this nature are not a safety impact to the orbiter or crew because the short is internal to the solar array and the solar array is an inherently current limited source	No safety impact
8	DF-224 Memory Unit (Post-SM1)	Three of the six DF-224 flight computer memory units have failed and were taken off-line. The DF-224 coprocessor (installed during SM1) was used to support shared memory to replace the failed memory units.	No safety impact
		This anomaly is not a safety impact to the orbiter or crew. Two memory units were failed prior to SM1.	
		The DF-224 and coprocessor were replaced with a 486 computer during STS-103 (SM3A)	
9	SADE 1R Slew Runaways (Post-SM1)	Two runaway slews have occurred during commanded slew operations. The runaways were immediately detected by the onboard safing system and SADE-1R was autonomously commanded off, which terminated the slews. The largest runaway resulted in an overshoot in wing position of approximately 17°. The direct cause of the runaways has not been determined, but it is thought to be EMI or single-event-upset induced. These runaways are attributed to single-event upsets in the SADE. The HST has returned to normal operations on SADE-1R. There is no	No safety impact

Note: Failures with no safety impact are grayed out.

D-4

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
		safety impact to the orbiter or crew during future servicing missions because the safing system will be enabled whenever in-bay solar array slews are performed. Reference HR-12 for a complete description of all controls in place to preclude inadvertent solar array slew	
10	ESTR-1 Tape Speed Anomaly (Post-SM1)	On several occasions, ESTR-1 failed to achieve proper tape speed, which resulted in corrupted or aborted operation. The anomaly was attributed to particulate debris within the unit. This anomaly is not a safety impact to the orbiter or crew. ESTR-1 was replaced with a Solid State Recorder, SSR-1, on SM2. ESTR-2 was replaced with ESTR-2R on SM2, and ESTR-3 was replaced by SSR-3 on SM3A. The ESTR-1 changeout had hot connectors and was not compliant with JSC/MA3-94-002. An NCR is HSTSM3B EVA4 ELECTRICAL NCR. Reference HR EVA-4 for the safety assessment of EVA handling of hot connectors	No safety impact
11	ESTR-2 Bit Slip Anomaly (Pre-SM2)	Ground processing of recorded data from ESTR-2 revealed numerous "bit slips" or missing data bits. The anomaly was attributed to an irregularity in the head gap, which may have been caused by accelerated head wear from faulty tape. This anomaly was not a safety impact to the orbiter or crew. Changeout of ESTR-2 was accomplished on SM2. The ESTR-2 changeout had hot connectors and was not compliant with JSC/MA3-94-002. An NCR is HSTSM3B EVA4 ELECTRICAL NCR. Reference HR EVA-4 for the safety assessment of EVA handling of hot connectors	No safety impact
12	Gyro #5 High Current (Pre-SM2)	The gyro-5 motor current increased from the nominal 98 milliamps to 157 milliamps with an associated shift in rate bias. The gyro continues to function and provide usable data. The increase in current is attributed to a lubrication patch within the gyro wheel bearings, which bridged the gap between stationary and rotating parts, increasing the drag on the gyro wheel. This anomaly is not a safety impact to the orbiter or crew. Changeout of RSU-1, RSU-2, and RSU-3 was accomplished on SM3A. ECU-3 was replaced on SM1 (prior to occurrences of this current fault.) The current fault was isolated to RSU-3. The ECU changeout had hot connectors and was not compliant with	No safety impact

Note: Failures with no safety impact are grayed out.

D-5

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
		JSC/MA3-94-002. An NCR is HSTSM3BFNCR. Reference HR EVA-4 for the safety assessment of EVA handling of hot connectors. Two ECUs were changed out during SM1 with hot connectors	
13	Excessive High Gain Antenna Torque (Pre-SM1)	The High-Gain Antenna (HGA) gimbals exhibited excessively high torque early in the HST mission. The high torque was attributed to interference between the gimbal counterweights and the harness service loop between the mast and gimbals. A software mask was implemented to prevent the gimbals from operating in the region where the interference exists. This anomaly is not a safety impact to the orbiter or crew, and existed prior to SM1	No safety impact
14	Failure of High Gain Antennas to Stow Properly (During SM1)	During SM1, both HGAs failed to stow properly during retraction operations prior to grapple. Both HGA masts were retracted, but the latch fittings of both HGAs failed to seat fully in their respective latches. The HGAs were determined to be in a stable configuration for grapple of HST. A photo/TV survey was conducted after grapple, and the driven and slave latch fittings of both antennas were determined to be within their respective latch capture range. Both HGA latches were commanded closed. The -V3 HGA latch pawls rode up on the latch fittings and no berthing confirmation was received. The -V3 HGA was, however, considered captured and safe for reboost. The +V3 HGA latch closed normally and was considered safe for reboost. The failure of both antennas to berth normally was attributed to interference between loose multilayer insulation and the HGA counterweights. Similar interference is anticipated during HGA retraction planned for future servicing missions. This anomaly is not a safety impact to the orbiter or crew because the HGA is required to be latched only for PRCS reboost. EVA manual overrides are available to provide alignment and close the HGA latches if required before PRCS reboost	No safety impact
15	GHRS Low Voltage Power Supply Failure (Pre-SM1)	The intermittent GHRS side-1 low voltage power supply failure is attributed to an open solder connection. The failure has been prevented from reoccurring by modifying the operational modes of the instrument to avoid low-temperature excursions. The GHRS Repair Kit was installed during the SM1. It allows for cross-strapping the science data routing from the GHRS side-1 and	No safety impact

Note: Failures with no safety impact are grayed out.

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
		side-2 to the HST spacecraft side-1 science data formatter. This anomaly is not a safety impact to the orbiter or crew.	
		The GHRS was replaced with the STIS on SM2	
16	FGS-1 and -2 Star Selector Server Anomaly (Pre-SM1)	The star selector servo bearings in the FGS-1 and FGS-2 exhibited high torque and sticking, which worsened over time and began to adversely impact science operations. The anomaly was attributed to poor lubricant distribution and Teflon debris within the bearings because of extensive coarse track operation. Normal operation of the servos was restored after a recommissioning operation that redistributed the bearing lubricant. The servos are now operated using new guidelines, which will significantly extend the bearing life. This anomaly is not a safety impact to the orbiter or crew.	No safety impact
		The FGS-1 was replaced with FGS-1R on SM2. FGS-2 was replaced with FGS-2R on SM3A	
17	Solar Array II Blanket Temperature Anomaly (Post-SM1)	The +V2 wing blanket T3 temperature monitor is intermittently noisy. The -V2 wing blanket T3 monitor intermittently saturates at entry into orbit day. The anomaly is attributed to an intermittent open in the data lines at the flexible data harness to inner buffer assembly interface located at the base of the solar array blankets. This anomaly is not a safety impact to the orbiter or crew	No safety impact
18	RWA-1 Anomaly (Post-SM1)	An RWA-1 anomaly occurred in software sunpoint safemode entry when the RWA-1 momentum test exceeded its threshold of 2.2 n. mi. for 30 seconds; this event was caused by the commanded RWA torque not resulting in a corre- sponding wheel speed response. The likely source of the RWA anomaly was a polycarbonate capacitor leakage at high voltages in the wheel speed integrator circuit. This is a known failure mechanism of this type of capacitor; it can be cleared through subsequent circuit operations. The RWA- continued to operate properly after being powered back on following the safemode event. This anomaly is not a safety impact to the orbiter or crew. RWA-1 was replaced with RWA-1R on SM2	No safety impact

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
19	Seasonal Battery Heatups (Post-SM1)	The HST batteries overheat during periods of high beta angles; the first occurrence was around Dec. '96. The cause is overcharge from excess trickle charge time, and battery aging is also a contributing factor. During normal operations, the heatups are managed by opening SA trim relays. The primary concern is while HST is in safemode with an associated loss of comm, since comm is required to reconfigure HST to a safe state, and the reduced electrical load in safemode exacerbates the overcharge.	No safety impact
		The Voltage Improvement Kits (VIKs) that were installed during SM3A alleviated this concern. The VIK installation did have hot connectors, which are addressed via NCR	
20	Solar Array Shorts (Post-SM1)	Numerous SA2 shorts have occurred and self- healed over the last 5+ years. Most shorts are "soft" and only become visible when the affected SPAs are off-line. A few shorts have been hard enough to reduce usable power by a small amount. The shorts are generally attributed to burnthrough of the thin Kapton layers, isolating conducting traces within the flexible blankets. The shorts are isolated to the respective SA wing and do not propagate or adversely affect the rest of the power system. This anomaly was not a safety impact to the orbiter or crew during SM3A, but may impact SA3 installation safety issues on SM3B	Solar Array Replacement Impact on SM3B
21	PCU Relay K-74 Failure (During SM2)	During SM2 K-74 failed during solar array checkout following the airlock depress/solar array slew event. The relay was operated in the sunlit portion of an orbit, which caused a sustained arc in the relay, and the relay contacts welded in the "disconnect" or shorted state because of contact overstress. The +EE SPA is now shorted out and unusable. This failed relay will be bypassed with the installation of SA3 during SM3B. This anomaly is not a safety impact to the orbiter or crew. This failure will be eliminated from the system with the changeout of the PCU planned for SM3B	No safety impact on SM3B because failed relay will be bypassed

Note: Failures with no safety impact are grayed out.

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
22	Aft Shroud Door Latch High Running Torque (During SM2)	On SM2 high running torque was encountered on the +V2 Aft Shroud door latches, which were previously cycled on SM1. Light leaks around the AS doors would degrade science, so Aft Shroud Latch Repair (ASLR) kits were flown on SM3A to ensure latches achieved proper engagement torque. The kits were to be used if problems were encountered with the -V3 AS doors (RSU changeout) or +V3 FGS door (FGS-2 changeout). "Latch breaker" tools were flown in case the door latches could not be opened, and the repair kits would override the damaged latches. This anomaly is not a safety impact to the orbiter or crew. Aft Shroud Latch Repair (ASLR) kits were not needed on SM3A. An optional task to install two ASLRs on the lower two +V2 aft shroud door latches was not performed due to the shortening of the SM3A mission from 4 to 3 EVA days. ASLR installation on those same two latches is considered likely for SM3B but will be planned as either a contingency operation on EVA day 5 or a get-ahead task on an earlier day. A lubricant application was developed for the SM3A mission and -V3 aft shroud and FGS-2 (+V2) door latches were lubricated with braycote prior to actuation. The -V2 and +V2 aft shroud door latches are planned to be lubricated on SM3B. The lubricant application results in cautions not to get braycote on the EMU or transfer it to other more sensitive HST surfaces. There is some potential for lubricant getting on the faceplate resulting in reduced visibility for that crewmember	Danger of contamination by latch lubricant and of reduced EVA crewmember visibility
23	MLI Damage	On SM2, significant MLI damage was noticed, primarily on the +V3 "Sun side" of HST. MLI patch repairs were called up, fabricated, and installed during SM2 on SSM bays 7, 8, and 10 and a section of the light shield. Permanent replacements for the SSM bay doors (NOBL) were to be installed on SM3A, and time permitting, the SSRF was also to be installed on the LS/FS sections. This anomaly is not a safety impact to the orbiter or crew. Permanent replacement insulation (NOBLs) for SSM bay doors 9 and 10 were installed on SM3A and NOBLs may be installed on bays 5, 6, 7, and 8 on SM3B. NOBL installation includes removal of the patches that were installed on SM2.	No safety impact

Note: Failures with no safety impact are grayed out.

D-9

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
		Bay 10 patches were removed on SM2. Replacement insulation for the LS/FS, called Shield/Shell Replacement Fabric (SSRF) was flown on SM3A but not installed due to the shortened mission and will not be flown on SM3B. Installation post SM3B is TBD	
24	NICMOS Dewar Thermal Short (Post-SM2)	Due to a thermal short between a cold baffle and the warm filter wheel housing, the solid nitrogen coolant in NICMOS evaporated at a higher than expected rate, resulting in a reduced lifetime. NICMOS reached cryogen exhaustion on 3 January 1999. On 6 January 1999, the temperatures for all three cameras exceeded the limit of 78.1 K and the filter wheel was commanded to the blank position. This anomaly is not a safety impact to the orbiter or crew.	No safety impact
		The NICMOS Cooling System (NCS) has been developed and is planned for installation on SM3B. The fill and drain valves were opened and the fill and drain caps were removed and returned on SM3A in preparation for SM3B. Installation of NCS will involve removing the cryo vent line from NICMOS and the HST aft bulkhead, installation of the NICMOS Cryo-Cooler (NCC) in front of the NICMOS instrument in the aft shroud, installation of the Cross Aft Shroud Harness (CASH) and the COSTAR Y-Harness, and installation of the NCS radiator on the exterior of HST, conduit deployment, and seating of the NCS evaporator onto the NCC. All of the equipment being installed is entirely new to the HST configuration, not a replacement of existing hardware	
25	SADE2 Control Register Bit Upset (Post-SM2)	Shortly after SM2, a safemode entry was caused by an SEU in the SADE-2 Slew Control Register that occurred during a slew. This SADE was the SADE-2R that was installed during SM2 to replace the original SADE-2. The problem has only recurred a few times in hundreds of subsequent slews, and the events have not resulted in profile errors large enough to be acted upon by the safing system. There are no plans to change out SADE-2 again. Safety issues associated with potential SADE	Potential runaway slew
		upsets are	

Note: Failures with no safety impact are grayed out.

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
		 Problem can only manifest itself during a slew Crew monitoring is always required during SA slews, and the crew has the capability to command SADE off if runaway is observed SADE is required to be powered off if the crew is not available to monitor SA position and HST is positioned such that a runaway could contact orbiter structure EVA crew is required to remain clean of SA envelope during slews. 	
		Flight software protection is always in place during slews, which is currently not a formal hazard control	
26	S-Band Single Access Transmitter Failure (Post-SM2)	HST SSA Transmitter 2 failed during a nominally commanded startup March 10, 1998. The failure left HST with one functional high-power TDRSS transmitter to downlink 1 Mbps science or engineering data and is attributed to a failure in the final stage power amplifier transistor. The SSATs are normally not used during servicing missions and are configured to a three-inhibit safety state prior to rendezvous. During SM3A, SSAT-2 (S/N 1001) was changed out with the flight spare SSAT (S/N 1003). Following closeout of SSM bay 5, the STOCC performed an Aliveness Test of the SSAT-2R, which resulted in RF radiation through the -V3 High-Gain Antenna. The HGA boresight was directed to either the orbiter -X axis for a +V3 forward changeout (over the vertical tail), or over the starboard wing (+Y axis). Orbiter and EV crew safety constraints prohibited pointing the HGA boresight to the orbiter upper S-band quad antenna (located atop the crew compartment) or placement of the EMU backpack between or near the HGA reflector and feedhorn. This anomaly does not apply to future servicing missions after SM3A except in the case of an SSAT changeout. This anomaly is not a safety impact to the orbiter or crew	No safety impact

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
27	Rate Gyro #4 Failure (Post-SM2)	On April 9, 1997, a flex lead failure caused a complete failure in the gyro. It is believed that the failure can be attributed to the presence of oxygen in the gyro, which caused higher flex lead corrosion rates. A new method of filling the gyro with float fluid should eliminate the introduction of oxygen in the gyro. This anomaly is not a safety impact to the orbiter or crew. Rate gyros were replaced on SM3A	No safety impact
28	Rate Gyro #6 Failure (Post-SM2)	On June 29, 1998, a flex lead failure in Gyro 6 caused increased current flow in the remaining flex lead. On October 22, the remaining flex lead failed and resulted in a complete failure of the gyro. Anomaly #27 discusses the probable cause of the failure. This anomaly is not a safety impact to the orbiter or crew. Rate gyros were replaced on SM3A	No safety impact
29	Rate Gyro #3 Failure (Post-SM2)	On January 25, 1999, the Phase A flex lead for Gyro 3 failed, causing increased current flow in the remaining flex lead. Later, on April 20, the remaining flex lead failed and resulted in a complete failure of the gyro. Anomaly 27 discusses the probable cause of the failure. This anomaly is not a safety impact to the orbiter or crew. Rate gyros were replaced on SM3A	No safety impact
30	Battery 5 and 6 Load Share Imbalance (Post-SM2)	A battery 5 and 6 load share imbalance has been intermittent since January 1999. A Failure Review Board determined the most likely cause to be the loss of preload in a bolted bus bar connection within the PCU, which has resulted in a 25 m Ω resistance across the connection. The battery 5 and 6 discharge path is through this connection, causing batteries 5 and 6 to give up less charge resulting in an overcharge condition. The overcharge condition exacerbates the battery 5 and 6 heatup tendency, which is currently managed by opening trim relay serving battery 5. The VIKs will counter this problem while HST is in safemode. An on-orbit repair of the PCU was considered but instead the PCU is scheduled to be replaced with the PCU-R on SM3B. This anomaly is not a safety concern to the orbiter or crew	No safety impact

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
31	Rate Gyro #1 Failure (Post-SM2)	On November 13, 1999, Gyro 1 experienced a failure that took it off-line and resulted in HST operating in Zero Gyro Sunpoint Mode. During SM3A HST transitioned to hardware sunpoint during rendezvous to minimize vehicle rates and attitude excursions. The ECU was suspected but determined to not be the cause. Rate gyros were replaced on SM3A. This anomaly is not a safety concern to the orbiter or crew	No safety impact
32	PCU Relay K61 Closed but Failed to Pass Current	On November 30, 1999 the K61 CCC relay in the PCU closed but did not pass current. The relay has subsequently been cycled by the PSEA and is again passing current. The project has elected to open the two trim relays upstream of K61 to preclude current flow through the relay given that the failure mode is not understood. The Anomaly Review Board recommended that up to one SPA worth of current could be safely passed through K61 if needed for battery charge purposes, but that switching of K61 should be avoided. This failure will be eliminated from the system without the planned SM3B changeout of the PCU	
33	Rate Gyro #6 Bias Shifts	On June 17, 2000, Rate Gyro #6 began exhibiting large bias shifts that have been correlated to vehicle maneuvers. Gyro #6 was powered off on March 2, 2001	
34	Rate Gyro #5 Failure	On April 28, 2001, Rate Gyro #5 experienced a lube patch event and then subsequently stalled. Gyro #5 was restarted on May 8, 2001 and then stalled again on the following day. Gyro #5 was subsequently powered off	
35	STIS Power Anomaly	In May 2001, Side-1 of the STIS science instrument experienced an unrecoverable hard short from the +28V power input to return. Side-1 of STIS is unusable, and both A-B and C bus fuses serving STIS Side-1 are blown (the bus C fuse was blown during an attempt to recover Side- 1 using bus C). The STIS was subsequently re- commissioned on Side-2 which continues to operate nominally. PDU-2 P16 fuse will be changeout as part of the installation of PCU-R during HST SM-3B. This will not restore STIS Side-1, but will restore the power service if STIS is replaced or repaired in the future	TBD

#	Anomaly Description and Life Cycle Phase	Safety Impact Evaluation	Status
36	Reaction Wheel Assembly #1 Failure	On date TBD the Reaction Wheel Assembly #1 (RWA-1) tachometer circuit failed. RWA-1 is scheduled to be replaced with RWA-1R on SM-3B	
37	ACS Failure	HST was functioning nominally until it entered inertial hold on Saturday, January 27, 2007 - root cause was an electrical short in ACS, which was operating with its (redundant) Side-2 electronics (Side 1 electronics for the CCDs failed in June 2006). This problem was unrelated to Side 1 failure or September 2006 ACS suspension.	No safety impact
		Low Voltage Power Supply Box or the Auxiliary Power Box, inside ACS, suffered Serious electrical short resulting in:	
		A blown Side 2 Bus A/B 20 amps fuse in Power Distribution Unit	
		Discharge of volatile gas from ACS	
		NSSC-I detected anomalous ACS states and safed the ACS	
		NICMOS Cooling System was safed	
		FGS high voltage was shut off	
		Fixed Head Star Trackers were shut down	
		HST entered inertial hold Box	
		The Wide Field Channel (WFC) and High Resolution Channel (HRC) are unusable, but the SBC continues to operate nominally	
		Testing confirms the short is on Side-2 Hold Power circuitry and the ACS was returned to Side-1 SBC operations after test (No additional on-orbit tests planned)	

HST ANOMALY LOG (concluded)

APPENDIX E HST APPLICABLE DOCUMENTATION

The material presented in this manual was obtained from the following documentation:

- 1. HST Systems Description Handbook. ST/SE-02, LMSC/D974197B, May 1985.
- 2. HST Data Management Subsystem Functional Description. LMSC D712867E, Rev. F, April 1990.
- 3. HST Data Management Unit Detailed Specification. LMSC 4171881F, SCN-5, March 1989.
- 4. SSM HW/SW Interface Document. LMSC 4171989G, Rev. K, June 1988.
- 5. HST System Electrical Schematics, ST/SE-33, all vols. LMSC D978562C, Dec. 1985, Rev. E, March 1990.
- 6. HST Command List DM-01, Oct. 1993.
- 7. HST Telemetry List DM-02, Oct. 1993.
- 8. HST Flight Software Design Desc. DM-04, Vol. III, Part I. LMSC 4173944L, Rev. R, April 1993.
- 9. HST to Space Support Equipment ICD ST/CM-08. ST-ICD-10C, Rev. F, July 1993.
- 10. Solar Array Deployment Control Electronics Description Handbook. TN-SA-K001 Rev. G, March 1993.
- 11. Solar Array Secondary Deployment Mechanism Description Handbook. GL-SA-B001 Rev. E, Feb. 1985.
- 12. Solar Array Primary Deployment Mechanism/Solar Array Drive Adapter Description Handbook. TN-SA-C008 Rev. G, March 1989.
- 13. SSM to OTA ST/CM-08. ST-ICD-01F, Jan. 1985.
- 14. Solar Array to SSM ICD ST/CM-08. ST-ICD-05B, June 1984.

- 15. SSM Systems Procedures (SE-23).
 - Vol. I Instrumentation and Comm. LMSC D889541A, Rev. B, March 1990.
 - Vol. II Structures/Mechanisms and TCS. LMSC D889542A, Rev. B, Nov. 1989.
 - Vol. III Electrical Power Subsystem. LMSC D889543A, Rev. B, Nov. 1989.
 - Vol. IV Data Management Subsystem. LMSC D889544A, Rev. A, April 1990.
 - Vol. V Pointing Control Subsystem. LMSC D889545A, Dec. 1984.
- 16. HST Mass Properties Report (SE-04). LMSC/P564058, Jan. 21, 2002.
- 17. HST SM Flight Safety Compliance Data for JSC Phase III Safety Review. Parts 1 and 2, P440-062, June 8, 2001.
- 18. Orbiter/HST-SM ICD. JSC-ICD-A-14009-SM, Jan 25, 1995.
- 19. Cargo Arrangement Drawing, VA72-200186, Dec. 19, 2001.
- 20. Cargo Interface Drawing, VD72-200186, Nov.15, 2001.
- 21. Integrated Schematics, VS72-200186, May 11, 2001.
- 22. HST Electrical Power Subsystem Description and Operating Manual. LMSC F420548, May 1992.
- 23. Solar Array II System Description Handbook. GL-SA-B202, Rev. B, April 1993.
- 24. HST Thermal Control System Description and Operating Manual. LMSC/F420173A, Oct. 1991.
- 25. HST SM Mass Properties Report, SE-04. LMSC/F440037, April 1993.
- 26. HST Second Solar Array to SSM, Level II, ICD, CM-08, April 1993.
- 27. HST Servicing Mission 4 CDR datapack, April 2007.
- 28. HST PIP, Rev L, NSTS 14009, September 2006.

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