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Lyndon B. Johnson Space Center
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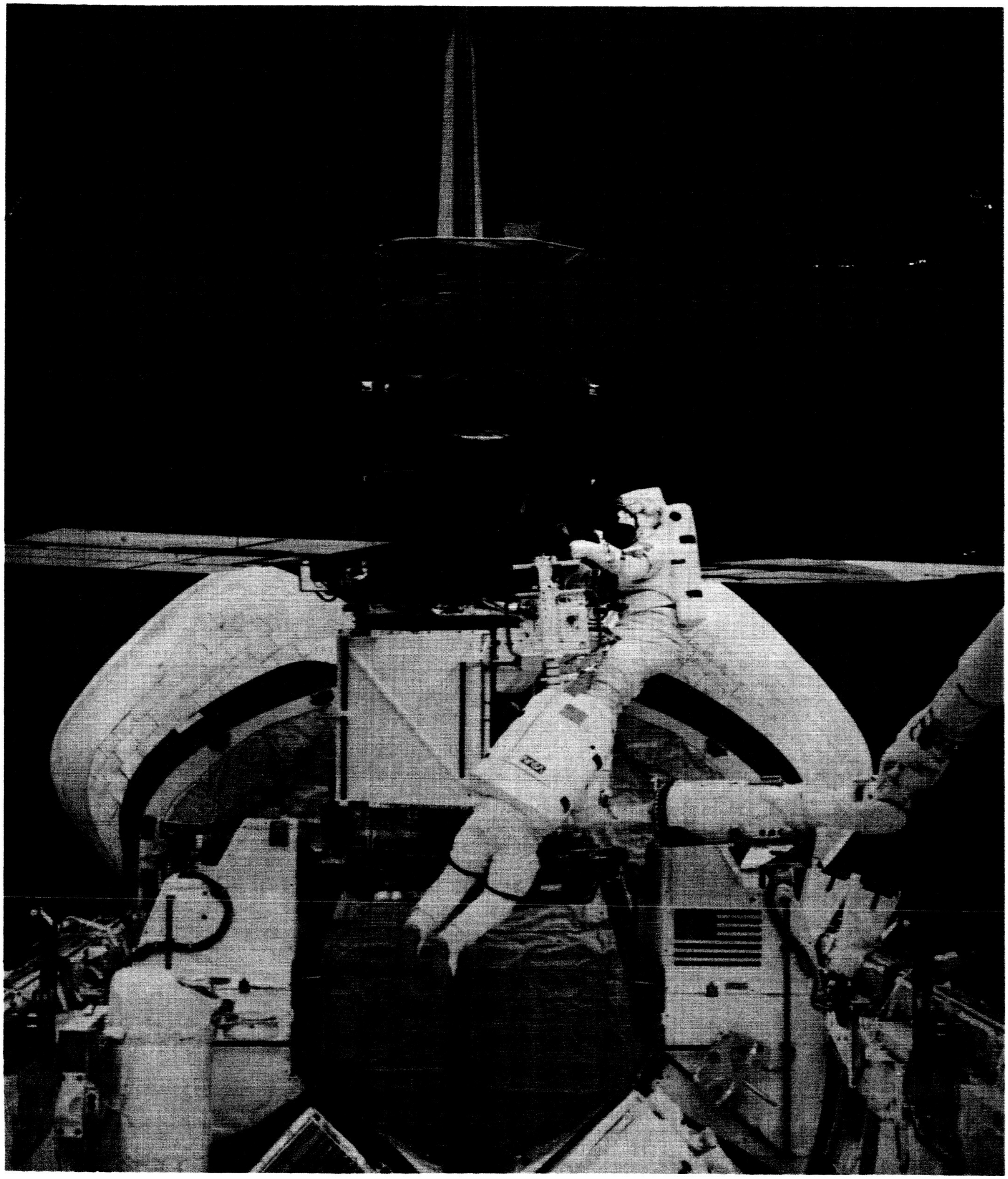
Satellite Services Workshop II

November 6-8, 1985

Volume 2



JSC-20677

SATELLITE SERVICES WORKSHOP II

November 6-8, 1985

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National Aeronautics and Space Administration
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SATELLITE SERVICING ACCOMMODATIONS

By

R.L. Gasteiger, J.A. Schroeder,
S.A. Tice, G. Panos, and B. Thompson

Prepared for presentation at the
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Rockwell International

Space Transportation
Systems Division

SATELLITE SERVICING ACCOMMODATIONS

R.L. Gasteiger, J.A. Schroeder, S.A. Tice,
G. Panos, and B. Thompson

ABSTRACT

As satellite design, fabrication, and deployment costs increase, it becomes imperative to consider the option of recycling, rather than replacing old or malfunctioning spacecraft. For this reason, the concept of on-orbit satellite servicing will become an increasingly significant activity in the U.S. Space Program.

Recent news reports have dramatically depicted the repair of the Solar MAX and Syncom satellites by the Shuttle crew. In the years to come this scenario will be repeated again, as old satellites are refurbished and failed satellites repaired and checked out prior to deployment.

Rockwell's Space Transportation Systems Division, Space Station Division, and Strategic Defense and Electro-Optical Systems Division are working on the development of satellite servicing and checkout concepts and related hardware to support these activities.

This paper presents a few of the satellite servicing hardware concepts Rockwell is currently developing—specifically, a payload berthing system, payload autonomous thermal control system, and satellite checkout equipment. Also included is a discussion of Rockwell's use of computer graphics in the development of satellite servicing hardware and scenarios by providing operation simulation, geometric analysis, and kinematics and general display.

PAYLOAD BERTHING SYSTEM

The Rockwell payload berthing system (PBS) is a versatile, lightweight satellite docking system for maintenance/repair, checkout/verification, or temporary berthing of satellites. The PBS, attached to the orbiter payload bay sidewall by a Rockwell modified extended adaptive payload carrier (MEAPC), is deployed from the bay, as shown in Figure 1, and can be accommodated in the payload bay along with a variety of cargo manifests. During launch and landing, the system is stowed at the bottom of the payload bay. Intrusion into the payload envelope is minimal.

SYSTEM DESCRIPTION

At the heart of the PBS is the berthing ring, fabricated from a 15.2-cm (6-inch) diameter aluminum tube and formed into a 2.5-m (98-inch) diameter ring. This ring serves as the mounting base for the latching mechanisms that hold the satellite/payload to the PBS.

The basic PBS payload attach mechanism is the same one used on the Rockwell-designed flight support system (FSS). This system consists of three motor actuated berthing latches situated on a 1.9-m

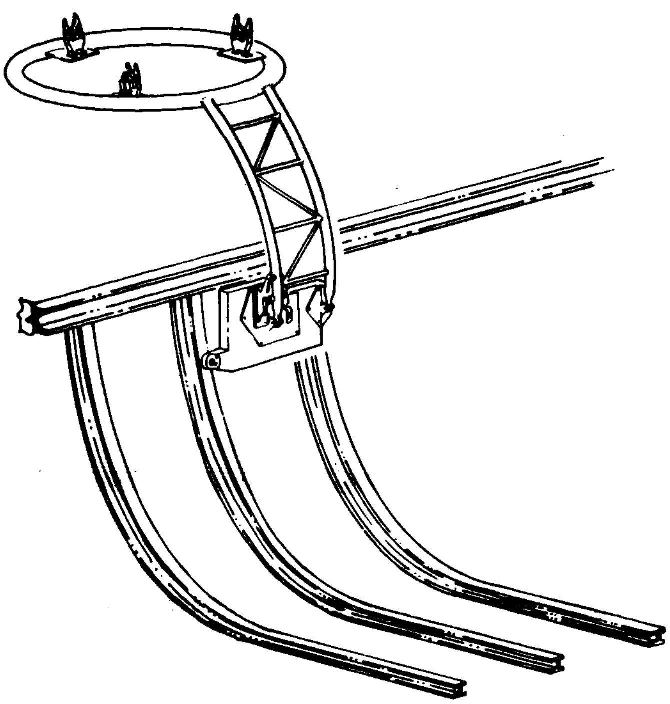


Figure 1. PBS, Deployed Position

(75-inch) diameter circle and located 120 degrees apart (see Figure 2). The locations for electrical umbilical connectors and actuators are also shown in this figure. The PBS provides the capability to mount umbilical connectors adjacent to any one or all three of the berthing latches, thereby allowing for clocking of the satellite/payload in 120-degree increments, depending upon the desired docking orientation

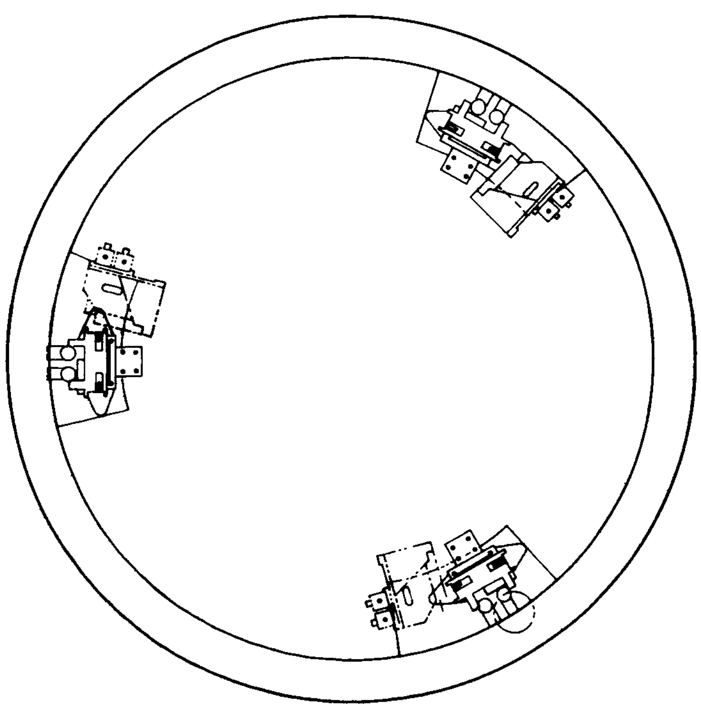


Figure 2. Berthing Ring

on the PBS. Additional clocking capability is available through incremental (30-degree) latch positioning on the berthing ring before orbiter installation.

The berthing ring is attached to the PBS hinge mechanism by a tubular support structure providing a routing path and support for the electrical cabling between the orbiter and the latches and umbilical. The support structure is capable of transporting orbital replacement units (ORU's) and is designed to conform to the payload bay's curved shape in order to minimize intrusion into the payload envelope.

A hinge/latch mechanism allows the PBS to swing out of the payload bay to its operational position over the cargo bay doors. Provisions have been incorporated into the design that allow disengagement of the system from the orbiter by an EV crewman in the event the system must be jettisoned.

While in the stowed position, the PBS uses one of the berthing latches and a special keel bridge/trunnion to provide support during launch and landing, as shown in Figure 3.

PBS DEPLOYMENT/STOWING

The PBS may be deployed by three methods:

Motor

A motor, such as that used on the orbiter latches, is coupled through a gear box and bell crank to the PBS hinge mechanism. The gear box/bell crank is specially designed to provide proper angular rates for PBS deployment. For example, as the PBS begins to move out of the payload bay, the angular rate of motion is relatively high, but as the PBS reaches the fully deployed position, the angular rate slows, reaching zero at full deployment.

Similarly, for stowing, the angular rate is high at the start of the stowing cycle and then slows until it reaches zero at the keel trunnion.

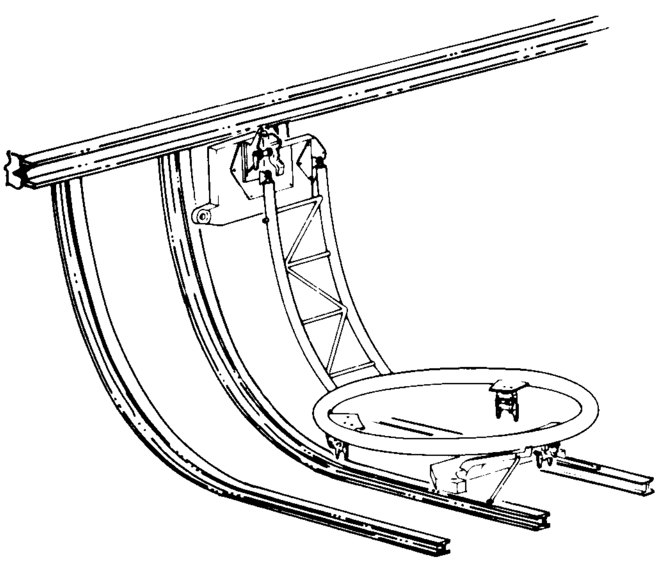


Figure 3. PBS, Stowed Position

Remote Manipulator System (RMS)

For RMS deployment of the berthing system, the ring is equipped with a swiveling fixture that provides the RMS with a grapple point. The berthing system is then deployed by the mission specialist, who operates the RMS from the aft flight deck and maneuvers the PBS from the stowed to deployed position and back to the stowed position.

Extravehicular Activity (EVA)

The PBS can be deployed by a crewman during an extravehicular activity (EVA). In this mode, the PBS keel latch is released for the aft flight deck and the crewman, standing on the manipulator foot restraint (MFR) mounted to the RMS, simply grasps the PBS. The RMS lifts both the crewman and the PBS out of the payload bay. After reaching the fully deployed position the crewman is free to assist in satellite/docking activities, as required.

LOADS ANALYSIS

Stress/loads analyses have been performed on the PBS for (1) lift-off and landing, (2) on-orbit reaction control system (RCS) firings with the PBS deployed and a 11.36-kg (25,000-lb) payload attached and (3) 11.36-kg payload docking at the nominal rate of .03 m (0.1 feet) per second. Analysis results indicate that lift-off and landing loads on the stowed PBS are well within the maximum allowables. Similarly, the loads resulting from the 11.36-kg docking were also well within allowable limits.

The analysis of the RCS firings indicate that the Vernier RCS thrusters yielded loads on the PBS attach mechanism in the range of only 300 kg (660 lb), well below the maximum acceptable values.

OPERATIONS

The PBS is capable of being mounted anywhere in the payload bay from Bay 2 through Bay 13, depending on the orbiter payload compliment or mission operational requirements.

Although the PBS baseline payload attach mechanism is the FSS retention system, the PBS is capable, through the use of several adapter concepts, of accommodating a wide variety of payloads that don't use the FSS retention system.

The primary adapter plate (Figure 4) is a basic adapter with three trunnions that are latched to the PBS retention system. This adapter may then be fitted with payload-unique berthing latches, grapple fixtures, or tiedowns.

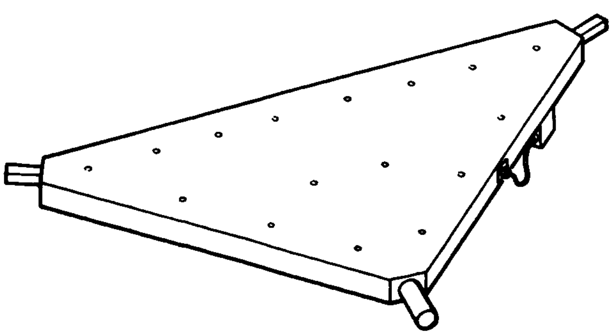


Figure 4. Primary Adapter Plate

Figure 5 shows a standard orbiter keel latch, modified with a bushing/insert. The keel latch is mounted to the adapter plate. Payload docking is accomplished by placing the payload/satellite's keel trunnion into the keel latch, which is then closed, like a vise, holding the payload solidly to the PBS.

Figure 6 presents a concept for a rotating adapter which can be used with the keel latch or other payload-unique system, to rotate the payload while docked on the PBS. The adapter is rotated by an EV crewman who disengages a set pin with the pistol grip, rotates the turntable by moving the handle, and then re-sets the pin with a pistol grip after reaching the desired position.

SUMMARY

The payload berthing system is a lightweight, 272-kg (600-lb), versatile system capable of accommodating a wide variety of existing satellites/payloads, as well as those yet to be designed and deployed. The PBS, with its multiple deployment/stowing methods and payload bay mounting flexibility, provides an excellent addition to the inventory of hardware for satellite on-orbit servicing and docking.

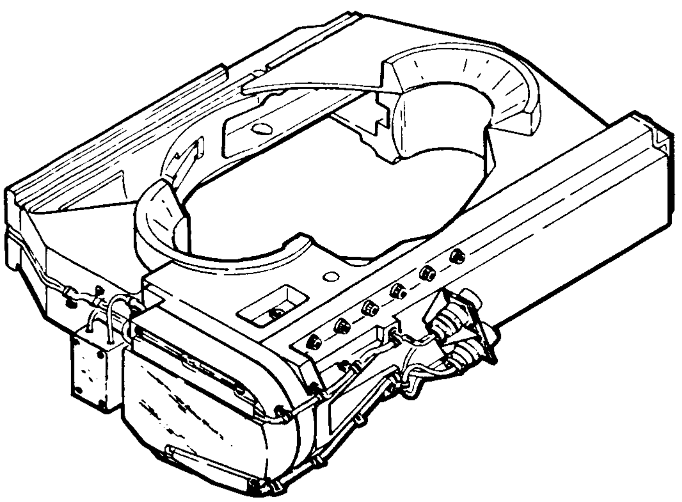


Figure 5. Keel Latch

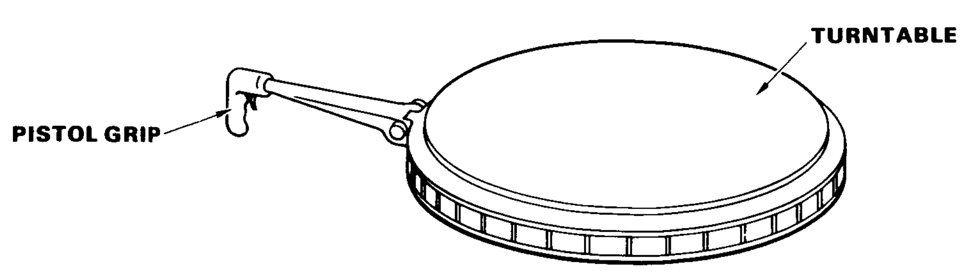


Figure 6. Rotating Adapter

PAYLOAD ACTIVE COOLING/HEATING SYSTEM

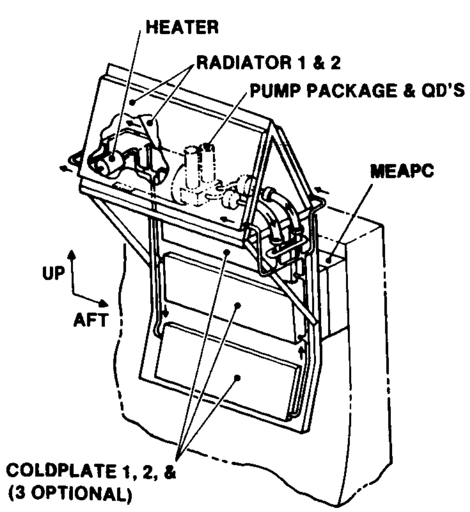
The payload active cooling/heating system (PACS) is an autonomous thermal control system for payloads in the orbiter cargo bay. The PACS was conceived to provide required cooling/heating for relatively lower heat load payloads and orbital replacement units (ORU) requiring thermal control.

SYSTEM DESCRIPTION

The PACS major components consist of a structural framework and shelf that is cantilevered from a modified extended adaptive payload carrier (MEAPC) mounted to the cargo bay sidewall, thermal plates, radiator panels, and a pump package. Figure 7 is an isometric and schematic of the PACS autonomous mode, which is self-contained from a cooling standpoint. However, it does require 28 Vdc, supplied by the orbiter, for the pump and heater package. For this configuration, PACS should be mounted on the starboard side of the cargo bay so that the radiator is not blocked by the remote manipulator system (RMS), which is mounted on the port side.

The modular PACS design allows it to be used in configurations other than the autonomous mode described above. For example, removal of the radiators allows the system to use the orbiter payload heat exchanger for heat rejection. In this mode, called the payload orbiter cooling system (POCS) mode, the pump package and thermal plates remain mounted to the structural framework and the PACS fluid system is plumbed, via a standard active cooling kit (SACK), to the orbiter payload heat exchanger. Figure 8 is a schematic and isometric of the POCS configuration. For this mode of operation, the PACS should be mounted on the port side of the cargo bay in order to interface with the SACK.

A third mode of operation utilizes the pump package alone for use by a payload either with or without the SACK (see Figure 9). The pump package, including its adapter plate, are easily removed from the PACS for mounting on a payload sidewall carrier, such as a getaway special (GAS) beam or an adaptive payload carrier (APC). The pump assembly may also be mounted directly on a payload or payload carrier, as required.



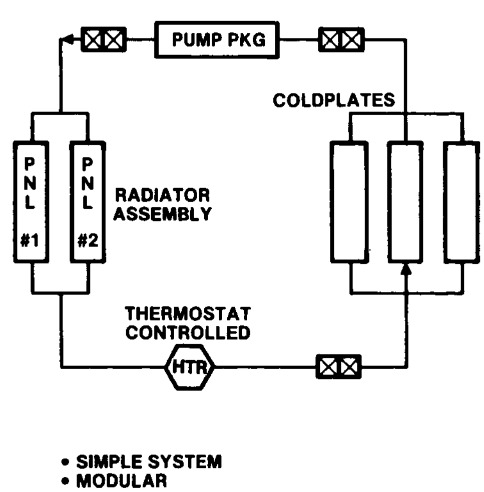


Figure 7. Autonomous Mode

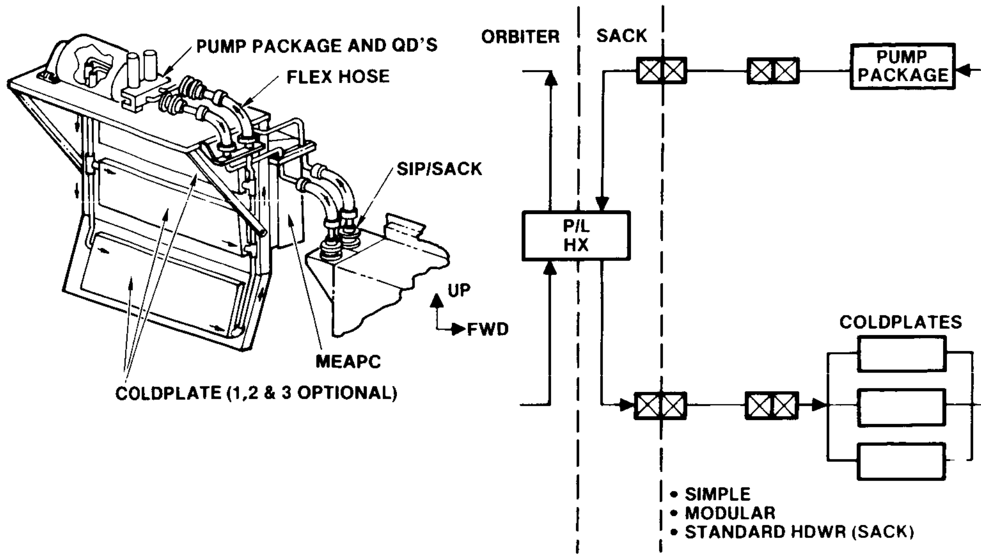


Figure 8. Payload Orbiter Cooling System Mode

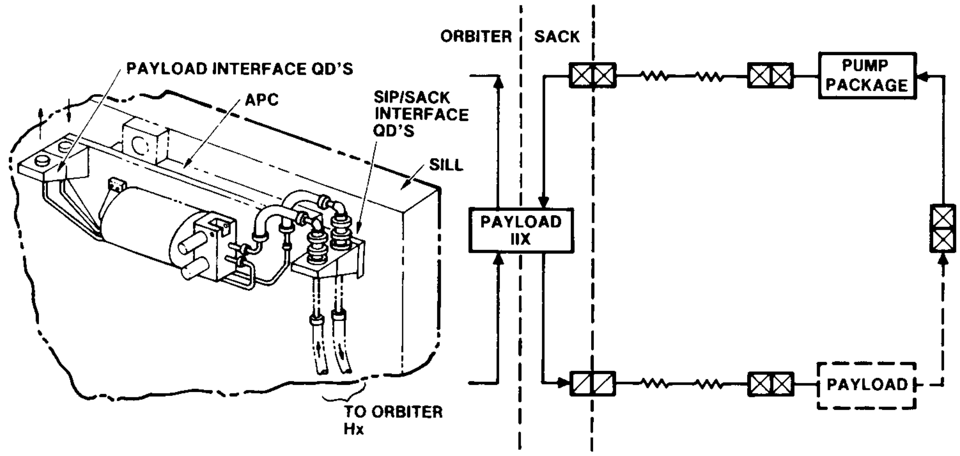


Figure 9. PACS Mode (Pump Only)

THERMAL PLATES

The PACS uses either two or three thermal plates, as required by the payloads. The plates, approximately 50 cm x 100 cm (20 in. x 40 in.), have a multiuse hole pattern of 70mm x 70mm (2.75 in. x 2.75 in.) with blind fasteners. Each cold plate is rated at a load carrying capability of approximately 1,400 kg (640 lb); however, the load capability is limited by the load to be cantilevered from the thermal plates, and is therefore dependent upon both the payload's physical configuration and weight.

As the cargo bay sidewall carrier, the MEAPC, has a load limit of approximately 2,200 kg (1,000 lb), depending upon cargo bay mounting location, the maximum payload capability of the PACS is therefore 1,760 kg (800 lb).

RADIATORS

The radiators consist of two panels, approximately 36 in. x 51 in. (12.5 square feet per panel), mounted in an inverted "V" over the pump package. The panels are constructed from rigid foam with silver Teflon tape on the outboard facing sides.

Coolant fluid flow is through 1/4 in. OD tubing mounted in a serpentine arrangement with a separation of 2.8 in. between flow paths.

PUMP PACKAGE

The pump package is mounted on an adapter plate and consists of two redundant, 28 Vdc fluid pumps with one accumulator of approximately 100 cubic inches in volume, related instrumentation (pressure and temperature transducers), heaters, fluid lines, and quick disconnects. The adapter plate allows for mounting of the pump package on either the PACS structural framework shelf, a cargo bay sidewall carrier, or a payload carrier. Provisions are made for heaters and an adjustable temperature controller, to be mounted on the pump package if heating is required for a particular mission.

The PACS uses either Freon 114 or water as the coolant, with a fluid operating temperature of 0 to 150 degrees F.

SYSTEM PERFORMANCE

In the autonomous mode, with the orbiter cargo bay doors open, the PACS can accommodate low heat loads up to approximately 500 watts. In the POCS mode, using the payload heat exchanger, the system can accommodate heat loads up to 1,500 watts with the cargo bay doors closed, and high heat loads up to 8,500 watts with the cargo bay doors open.

The above heat loads reflect a generalized capability. Heat rejection capability for specific payloads must be determined by using unique parameters for the specific payload, including, but not limited to, size, power, surface area, mission configuration, and orbiter orientation.

SATELLITE CHECKOUT EQUIPMENT

THE NEEDS TO BE ADDRESSED

The number of satellites either in orbit or planned has grown significantly, necessitating new methods of verifying performance and performing on-orbit assembly and servicing. High-altitude satellites far beyond the orbits of the Space Station and orbiter will require design standards and logistics services different from those readily accessible to orbiting support hardware and manpower.

To support on-orbit verification tests, diagnostics testing, and the applications of control signals, Rockwell International is designing on-orbit automatic test equipment, interface standards, and logistics scenarios. This section discusses the automatic satellite checkout equipment (SCE) currently under study and development at Rockwell International in order to meet these needs.

APPLICATIONS

SCE can be used in satellite servicing accommodations to provide checkout and fault isolation capability on the ground and to perform on-orbit checkout of satellites and orbital replacement units (ORU's) attached to the orbiter. The SCE can perform the following functions:

1. Verify satellite performance during on-orbit reassembly prior to deployment.

2. Detect and isolate faults to the ORU level.

3. Verify replacement ORU performance prior to installation.

DESIGN CONSIDERATIONS

A variety of factors must be considered in developing equipment to meet the needs described. The more significant of these are summarized below.

SCE Standards

A methodology is in use by the Air Force for designing, acquiring, and supporting automatic test equipment. The Air Force modular automatic test equipment (MATE) concepts, as defined in the MATE GUIDES and mandated by A/F regulation 800.23, will be used with considerable modification because of the environmental constraints and their implications for the ATE architecture. Software, as mandated by MATE, will consist of ATLAS as the high-order test language under execution of a modified MATE test executive.

Environmental Factors

Electronics in space, including the electronics of automatic test equipment, must deal with cooling problems much different than those on earth, especially if the electronics are not surrounded by a pressurized environment. The dissipation of heat necessitates special packaging techniques for thermal control, as well as the redesign of various standard assemblies. Shielding the electronics from radiation is another environmental problem complicated by the fact that communications equipment produces radiation in addition to that found in space.

Size and Weight Limitations

Space applications require most of the capabilities of a typical five-bay automatic test station containing rack-mounted instruments and computers, but will have to be one order of magnitude smaller and lighter. These design constraints will force the use of new technologies and the inclusion of various built-in test capabilities within satellites and orbital replaceable units (ORU's).

Factors of Location

Satellites having orbital altitudes ranging from 150 to 1,200 miles, for example, will be accessible to the orbiter or Space Station if they are within reach of an orbital maneuvering vehicle (OMV) dispatched to retrieve them for service. High-altitude satellites will be accessible only by way of an orbiting transfer vehicle that remains in a high-altitude orbit and receives resupply equipment and fluids from an expendable orbiting transfer vehicle dispatched from the Space Station. These factors of location will necessitate satellite checkout equipment in varying forms in the orbiter, Space Station, OMV's from the orbiter and Space Station, and transfer vehicle in high-altitude orbit. A ground variant of the spaceborne configuration is also required for development of test programs, as well as a functional test. The factors of environment, size, and weight will vary with location.

Commonality of Hardware and Software

The size and weight of the OMV automatic test equipment will be minimized to the greatest extent, with the orbiter version next, followed by Space Station and ground. However, maximizing the commonality of hardware and software is a design constraint.

Hardware Commonality. The instrumentation set available will be the most complete in the satellite checkout equipment with the least limits on size and weight. The OMV version, with the greatest size and weight limitations, will contain components of the full compliment of instruments, but all SCE will use the same computer, instrument interface, and bus architecture (see Hardware Interfaces below).

Software Commonality. One high-order test language will be used in all SCE, with one test executive containing subsets of each language, used or unused, present in SCE configurations with reduced capabilities due to size and weight constraints.

Communications. All SCE will be linked to telemetry equipment to enable control from lower-altitude manned facilities. This control will include the ability to send, for execution in the SCE diagnostic, test routines, with results to be returned by telemetry. These diagnostic tests will supplement the pass/fail tests resident in SCE memories and may include the application of special stimuli for the purposes of control of the unit under test.

Hardware Interfaces. The hardware interfaces to the satellite checkout equipment will reflect the hardware configuration of the SCE as determined by location (ground, orbiter, Space Station, or OMV). Test connectors for OMV SCE will be smaller, considering the smaller instrument set they ultimately interface with. Connections between the unit under test and the SCE will be performed mechanically with automatic positioning and mating/demating of test connectors. Interface standards will be published for the use of satellite and ORU manufacturers.

Compatibility With Year 2000

The design of all SCE hardware and software will be modular whenever possible to permit substitution of major elements of the SCE as improved designs become available for any of the design elements by the test instruments, controlling processor, software subsystems, etc. Evolution of the SCE is certain, and the design must allow for additions and modifications.

Where We Are Today

Rockwell International's experience in telemetry, automatic test equipment, space electronics, and space transportation has allowed us to expend considerable effort in the design of satellite checkout equipment. Significant milestones have been achieved in developing ATE architecture for space applications that is compatible with the design considerations mentioned earlier in this paper. To date, NASA has not taken a position on its requirements for SCE on the orbiter, but has included test as a payload accommodation to be addressed for the Space Station in Work Package 3.

Summary

Today an obvious need exists to test satellites before deployment from the orbiter. Soon, with on-orbit maintenance and assembly of satellites, functional and diagnostic tests on orbit will be actual elements of satellite support scenarios. Rockwell International is highly involved with satellite servicing concepts, hardware, and the design of automatic test equipment for on-orbit use. The sophisticated and costly payloads already in orbit or planned for the future demand state-of-the-art maintenance facilities to assure their success, including automatic satellite checkout equipment.

INTERACTIVE COMPUTER GRAPHICS AIDS DEVELOPMENT OF ON-ORBIT OPERATIONS

As on-orbit operations become more complex and expensive, increasing reliance is being placed on simulations to provide verification of operations scenarios and hardware designs. Rockwell International has developed specialized software tools to model and simulate these types of operations. This software produces full-color simulations that employ solid-shaded object models observable from multiple viewpoints, and simulation of proposed on-orbit operations can be performed at two distinct levels of complexity:

HIGH-FIDELITY VISUAL SIMULATION

At this level, the simulation operator drives the object models interactively with the aid of a mainframe-based simulation controller. The controller determines relative rates and accelerations of interacting parts by applying the operator's control inputs to dynamical equations of motion representing the modeled system. These equations take into account the masses, forces, and moments of inertia present in the system and, therefore, yield highly accurate system responses.

These features make high-fidelity simulators very useful for flight training, and the accurate system responses provide the operator with the same feedback experienced in a real aircraft, spacecraft, or whatever is being simulated. High-fidelity simulators also yield highly accurate time lines, which are vitally important in the development of operations scenarios.

The biggest drawback to high-fidelity simulators is the amount of resources required for their development, maintenance, and operation. These costs vary tremendously but are generally an order of magnitude higher than those of low-fidelity simulators. For this reason, the use of low-fidelity simulators is often a very cost-effective alternative.

LOW-FIDELITY VISUAL SIMULATION

Here, the simulation operator directly drives the object models in real time. No attempt is made to employ dynamic considerations in determining the model motions, or any forces they may impart to each other. Motions and rates of moving objects are explicitly specified, either interactively or by means of kinematic model programming. These simulations are valuable when used as a pre-hardware mock-up evaluation tool.

Low-fidelity simulations are particularly useful for development, verification, and problem identification for future on-orbit servicing scenarios. Many operational and hardware-related problems can be identified at relatively low cost. Previously, these problems could only be identified by means of high-fidelity simulation or neutral buoyancy tank testing. Low-fidelity simulations are also very useful for prescreening scenarios before embarking on a costly high-fidelity simulation.

CONFIGURATION DESIGN ANALYSIS AND SIMULATION ENVIRONMENT

Rockwell's CDAS system is an integrated set of generalized programs designed to produce low-fidelity simulations as well as perform other design and analysis functions. It can be used for real-time as well as non-real-time productions. Real-time simulations involve a man-machine interface for control, with man providing control inputs that the simulator responds to. For non-real-time simulations, the man-in-the-loop is replaced by a program that runs the simulation in some predetermined manner because all display calculations are laboriously done in software. All motions and positions of the object models are calculated ahead of time and can be recalled from memory, frame by frame. Thus non-real-time simulations are very useful for simulations not involving active human control.

Although each frame is produced by the user interacting with CDAS as a result of feedback from previous frames, ability to alter operations is severely hampered. However, non-real-time simulation does have significant importance in the on-orbit operations design cycle. For example, in this mode CDAS allows the user to perform RMS operations using the various driving modes available to the actual RMS. Simulation routines for human model reach and view functions are also available.

During these operations, measurements and cross-sections can be taken to assess operational interferences and feasibility. Various viewing points may be used to assess obscuration for a TV camera or EV crewman. Figure 10 shows how CDAS fits into Rockwell's space vehicle preliminary design cycle.

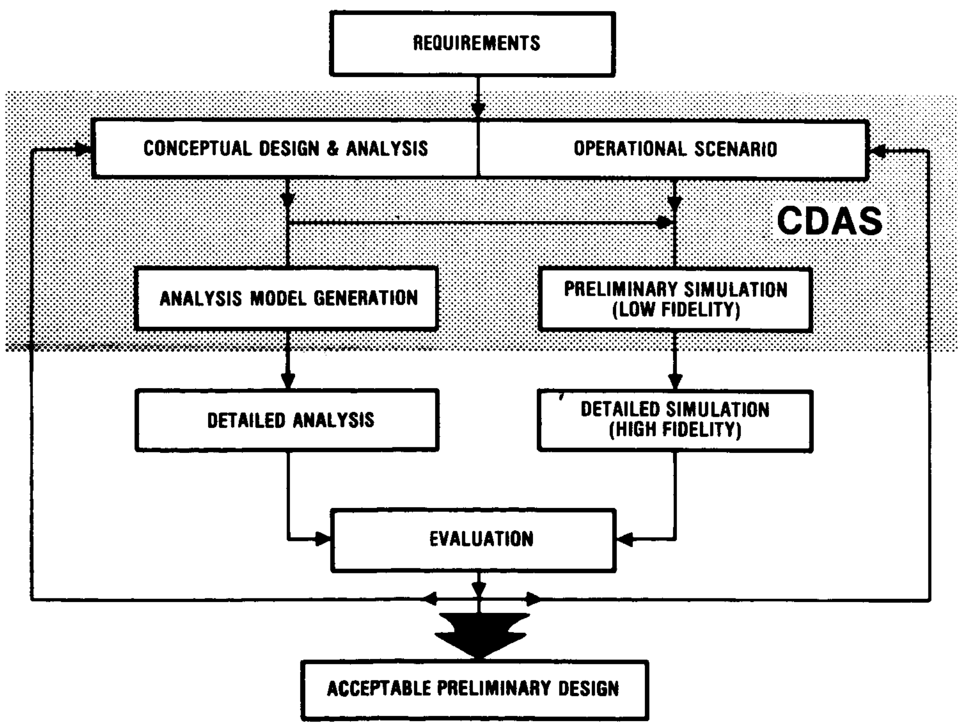


Figure 10. Preliminary Design Flow Develop Concepts and Operational Scenarios

CDAS has also been used to evaluate existing hardware and designs that are further developed. Depending upon the level of information available from the customer, CDAS can fulfill any or all of the following needs:

Design
- Hardware Design
  - Spacecraft
  - Tools and ASE
  - Cradles
- Operations Design-Timeline Est.
  - RMS Ops Schedules
  - EV Procedures

Analysis
- Fit/Feasibility Check
- Design Verification

Simulation
- Operations
  - Design
  - Redesign
  - Feasibility
  - Efficiency

Figure 11 is a schematic of the interrelationship of these functions.

Prior information in any of these areas will either enhance the quality or decrease the analysis time for a CDAS simulation. In cases where hardware and operations have been previously defined, CDAS simulations will provide quick determinations of feasibility and efficiency; however, this may result in highlighting of major design problems that would require redesign.

Table 1 presents a summary of simulations performed to date and the generic types of analyses conducted. The performance and feasibility of these types of analyses are continually upgraded by Rockwell through their ongoing software tool development program. Enhanced analysis capability and generalized EV crewman reach and view functions are currently receiving the greatest amount of attention.

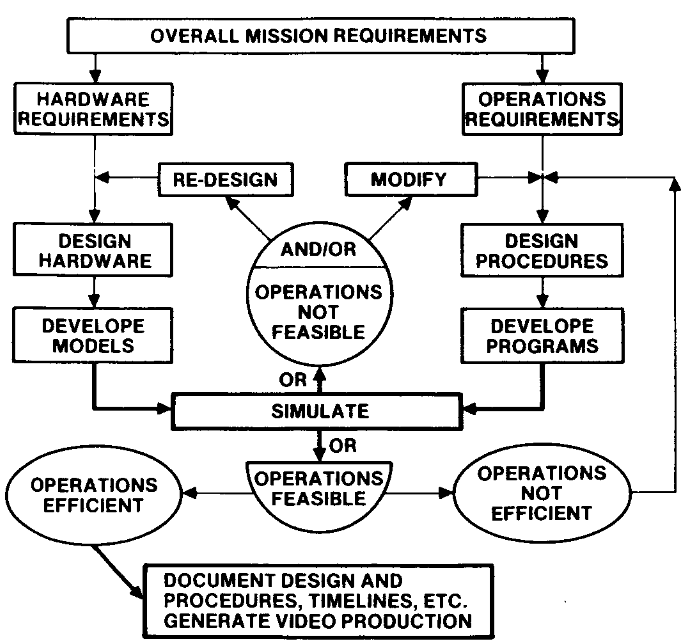


Figure 11. Typical Analysis Schematic

Table 1. CDAS Applications and Operations Simulation Capabilities

	RMS Operations - Retrieval and Deployment	RMS Operations - P/L Maneuvering & EV Crewman Interaction	MRMS Operations Retrieval and Deployment	MRMS Operations P/L Maneuvering and EV Crewman Interaction	Maintainability by EV Crewman - Access/Reach, Obscuration Assessments	Mechanism Operation Simulation	TV Crewman Traffic, Reach, Fit and Obscuration Analysis	Operation Time Line Analysis	Teleoperator Robot Opers
P80 Teal Ruby servicing	●	●			●	●			
Space Telescope servicing	●	●			●	●			
Space Telescope W/OMV servicing	●	●			●	●			
Space Station truss mech deployment - 9 ft and 15 ft concepts			●	●		●			
Space Station habitation module architecture evaluation							●		

SIMULATION ENVIRONMENT

For non-real-time operation simulation, each operation is performed sequentially. CDAS allows independent component rotations and translations as well as mechanical articulations, such as those performed for the RMS and the human arm. Examples of this capability are shown in Figures 12 and 13. During an articulation sequence, multiple viewpoints may be used in order to give the user realistic cues and indications of possible interference problems, as shown in Figure 14.

Cross-sections can be taken through the simulation models to ascertain the appropriateness of a particular set of motions. True measurements may also be made at any point in the simulation cycle.

Simulation sequence storage is available to the user and may be implemented in two ways. One method allows recording of only the visual motions with a tape recorder-type analogy, while the other method stores the specifics of each motion, such as the RMS end effector destinations and joint angles. Either of these may be stored and replayed at a later CDAS session. One advantage of the non-real-time display mode is the ability to render more realistic scenes by using more than one light source and variable model surface reflection parameters, as shown in Figures 15 and 16.

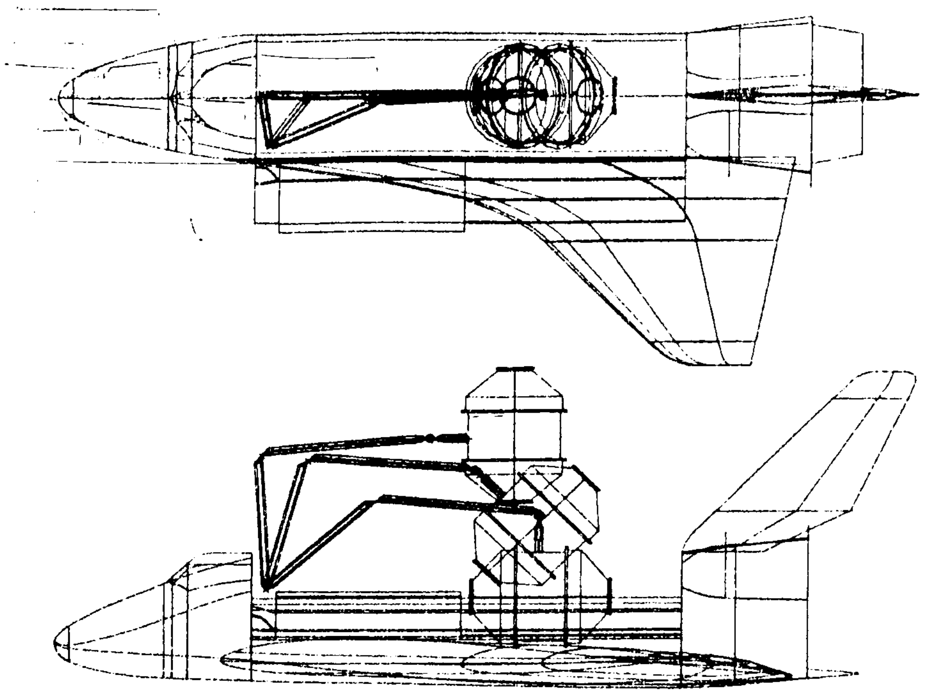


Figure 12. Sample Payload Deployment Simulation

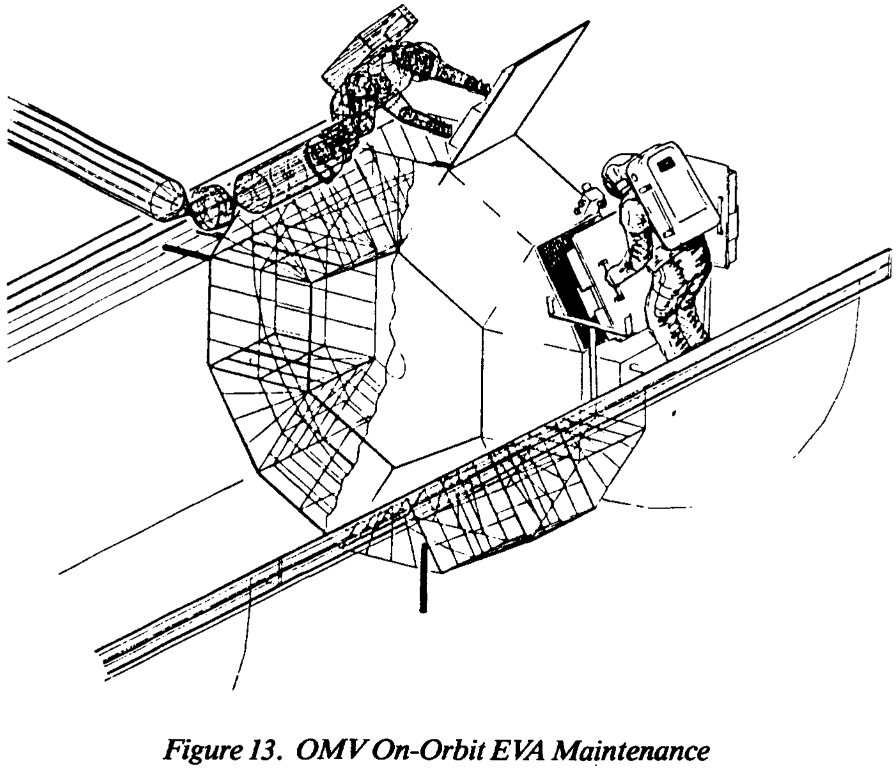


Figure 13. OMV On-Orbit EVA Maintenance

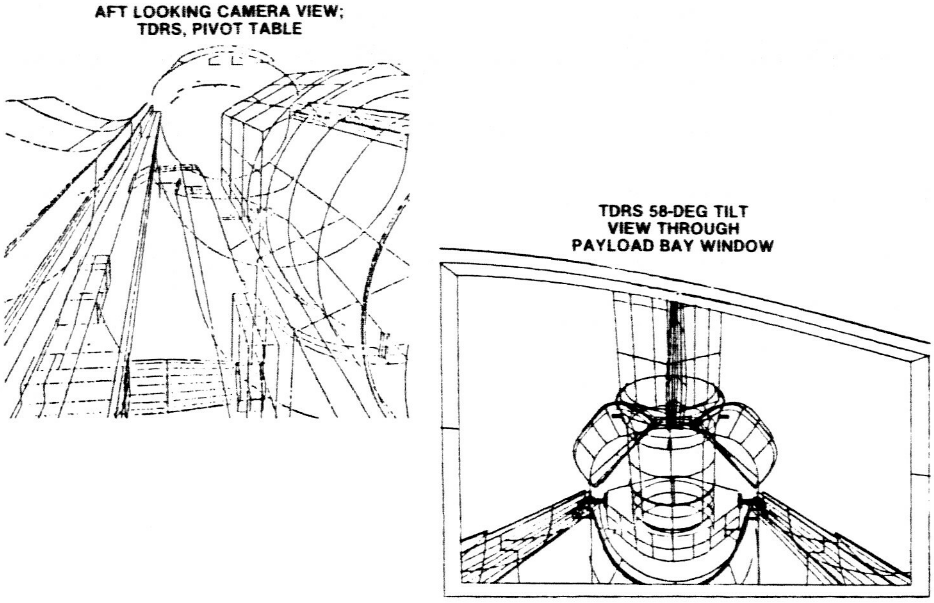


Figure 14. Orbiter Payload Bay Views

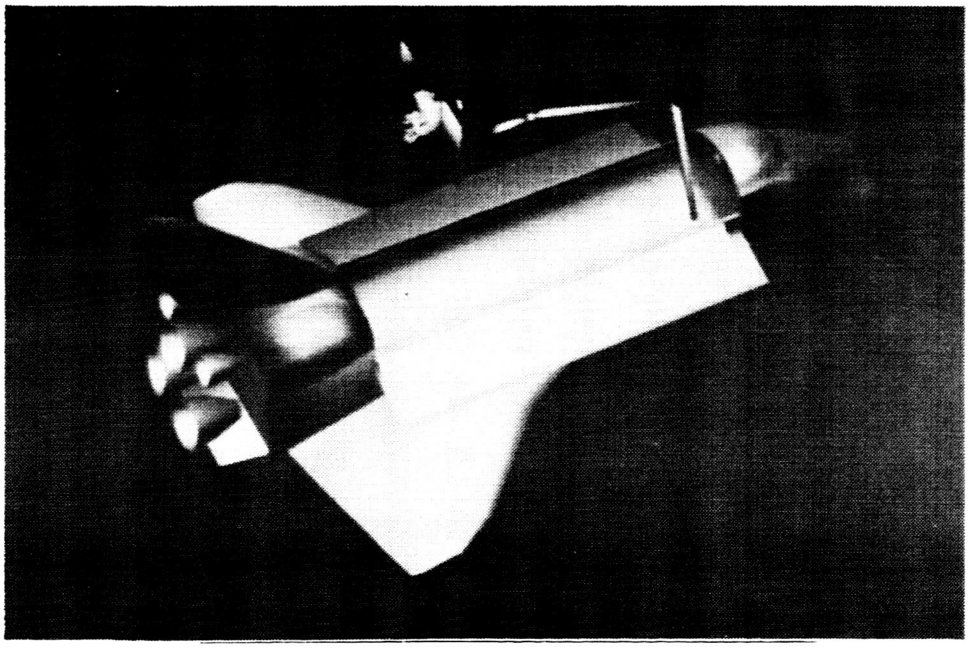


Figure 15. Shaded View of Shuttle Orbiter

Development is currently underway on automatic model collision detection, which will provide the user with quick indications of operational success during a simulation for real and non-real-time display modes.

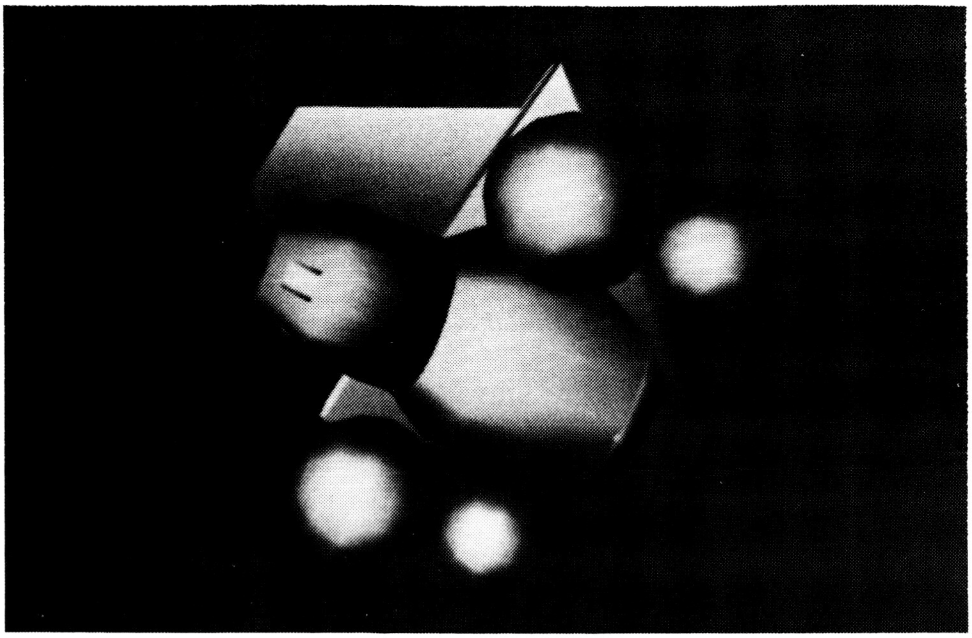


Figure 16. Shaded Version of Orbital Maneuvering Vehicle (OMV)

The CDAS real-time display mode gives the user instant feedback on operational performance of the simulations while displaying realistic views of the operations. Simplified models are created in CDAS for use on the GTI Poly 2000 image generator. This display hardware develops 30 images per second of solid-shaded color objects, as well as generating realistic scene perspectives. Generic simulation programs allow the user to operate an RMS, an EV crewman, and an IV crewman interacting with any hardware required for a particular simulation. For example, the RMS program allows the user to drive the RMS by way of joystick inputs (rotational and translational hand controllers) or through the use of multiple discrete-end effector locations. Space Shuttle closed-circuit TV and aft flight check views are also available during the simulation. Human factors programs allow similar viewing and operations/ capabilities. In addition, it is possible to integrate the human factors and RMS arm operations, allowing the user to simulate on-orbit operations of this type.

MODEL DEVELOPMENT

To perform any operations simulation, models representing the hardware must be developed. For operations involving new hardware, the CDAS system contains a module specifically designed to allow rapid geometry development and/or modification. The "Geom", or geometry creation/modification module, allows the user to create the individual components of a system and then assemble them to form the models needed for an operation simulation.

CDAS geometry entities or components have their own local axis system and orientation with respect to a global axis system. The geometry module allows the designer to individually create these components, which are similar to real vehicles, and later alter their local axis (X, Y, or Z axes) and roll, pitch, and yaw parameters to place the components in their proper orientation with respect to the global axis system.

This module further allows the user to define component geometry with two different approaches. The first approach involves inputting cross sections in a parallel stack to create a component. The designer can directly and numerically enter cross section points, through cross hair input, or create cross sections through primatives that develop conics, circular arcs, reflexed curves, straight lines, or entire section circles or squares. Another way to create cross sections is by superpositioning and scaling circles, ellipses, and squares and then modifying points interactively as desired. The user can also develop a single cross section, copy it into new locations and, subsequently, scale or reshape it as desired.

The second approach to geometry creation available to the user from within the geometry module involves primatives. These primatives require parametric geometry inputs such as those required for the tank routine: volume and end-dome radius over tank radius ratios. The tank routine allows the user to create tanks that are spherical, cylindrical, or torus shaped, with ellipsoid or spherical-shaped end domes. Other primatives include ellipsoid, parabolid, or surface-of-revolution constructs.

For components requiring an accurate surface definition, CDAS uses biquartic path mathematics. For biquartic surfaces, the stored cross section points are used in Bezier-fashion to create an exact mathematic surface. CDAS allows biquartic patch components to be developed with conic cross section inputs by free-fairing to obtain desired cross sections, or by fitting biquartic curves to an input set of surface points. Biquartic components are used for propellant tanks, wings, and other smooth surfaces. The reason CDAS geometry creation is so fast is that once these parallel cross sections are developed, CDAS automatically creates the surface patches between the sections, which the user can alter if desired. For real-time simulations with CDAS, components built using this type of geometric arc are then fitted with polygonal representations through the use of an interactive utility located in the "Geom" module.

Once the designer has defined the basic geometry through these various methods, the geometry module provides interactive commands for shaping a geometry in either top, side, or section-by-section (rear) viewing. Automatic smoothing routines are also available to refine a component's shape.

In addition to CDAS' geometry-building capabilities, it provides the user with functions that quickly access Rockwell's library of existing geometric models. These models can be used intact or as a starting point for new models required for a particular simulation. Some of these models are listed below:

Spacecraft: Orbiter (internal and external), OMV, OTV, various transatmospheric vehicle configurations

Satellites: P80/Teal Ruby, TDRS, GPS Navstar, Space Telescope, IRIS, HRTS, Leasat

Man-Models: EVA or IVA, both male and female, in varying sizes

Servicing Hardware: Various EVA hand tools, FSS*, Spacelab cradles and pallets, PBS*, PACS*, ROEU*

Space Station: NASA baseline, freeflyer platforms, truss sections

*Rockwell-designed hardware

CONCLUSION

The satellite servicing hardware presented in this paper are a small part of those currently under development by Rockwell and are the first in the line of servicing hardware to be simulated and verified using computer graphics technology. As additional hardware are developed they will be modeled, simulated/verified on computer graphics, and added to the Rockwell library of geometric models.

Rockwell is proceeding in a logical path in the development of servicing hardware to meet the needs of both the satellites of the future and existing operational satellites.

DESIGN EXPERIENCES IN THE REPAIR OF THE SOLAR MAXIMUM CORONAGRAPH/POLARIMETER

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ABSTRACT

Seven months after the launch of the Solar Maximum Mission spacecraft, the High Altitude Observatory's Coronagraph/Polarimeter experiment ceased operation. The cause was a time-induced failure of an integrated circuit. This paper discusses the program undertaken to restore operation to an instrument not originally designed for repair-in-space. Included are a discussion of the process to define the appropriate hardware program; the design changes made to the hardware for both ease of service and renewed quality of operation; the problems encountered during the rebuild; the successful removal and reconnection of the experiment electronics by the crew of STS 41-C; and the result of improved instrument performance over the previous SMM I mission.

THE FAILURE

The Solar Maximum Mission spacecraft was launched on February 14, 1980. The High Altitude Observatory's Coronagraph/Polarimeter (C/P) experiment was on the craft. This instrument was the latest spaceborne version of an externally-occulted Lyot coronagraph designed to photograph electronically the solar corona. The instrument is comprised of two hardware subassemblies: 1) the telescope containing the optics, mechanisms and the detector to image and record the corona, and 2) the main electronics box (MEB) containing the necessary electronics to interpret uploaded commands, read out the detector and play the data to the onboard tape recorder.

The C/P instrumentation was built during the years 1975 to 1979. The High Altitude Observatory (HAO), under contract to the Goddard Space Flight Center (GSFC), selected Ball Aerospace System Division (BASD) as the the prime instrument subcontractor to perform the design, analysis, fabrication and functional testing of the C/P instrumentation. HAO supported functional testing in the software development area and calibrated the experiment.

Following launch, the instrument performed well returning high-quality images of the corona. Operation of the C/P was routine until July 9, 1980 when the image quality began to degrade and the image readout to the tape recorder became intermittent.

The intermittent condition was thought to be the failure of a four-bit counter IC (P/N MM54C161 CMOS synchronous binary counter). This intermittent condition worsened until September 23, 1980, when the instrument completely stopped transmitting images. Continued analysis pointed to a hard failure in the address counter in the sweep

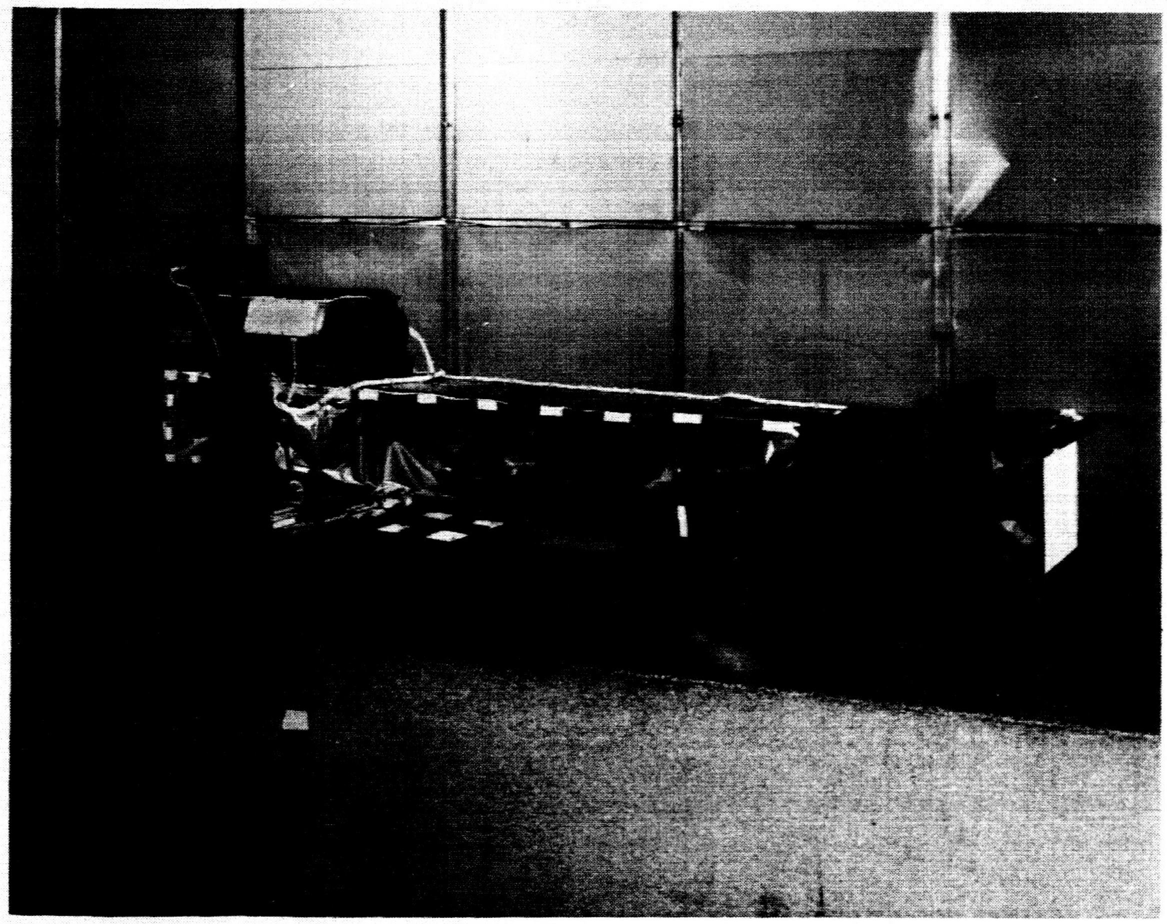

Figure 1
SMM CORONAGRAPH/POLARIMETER INSTRUMENT

The C/P instrumentation is comprised of two assemblies: The telescope (in the background) and the main electronics box (MEB).

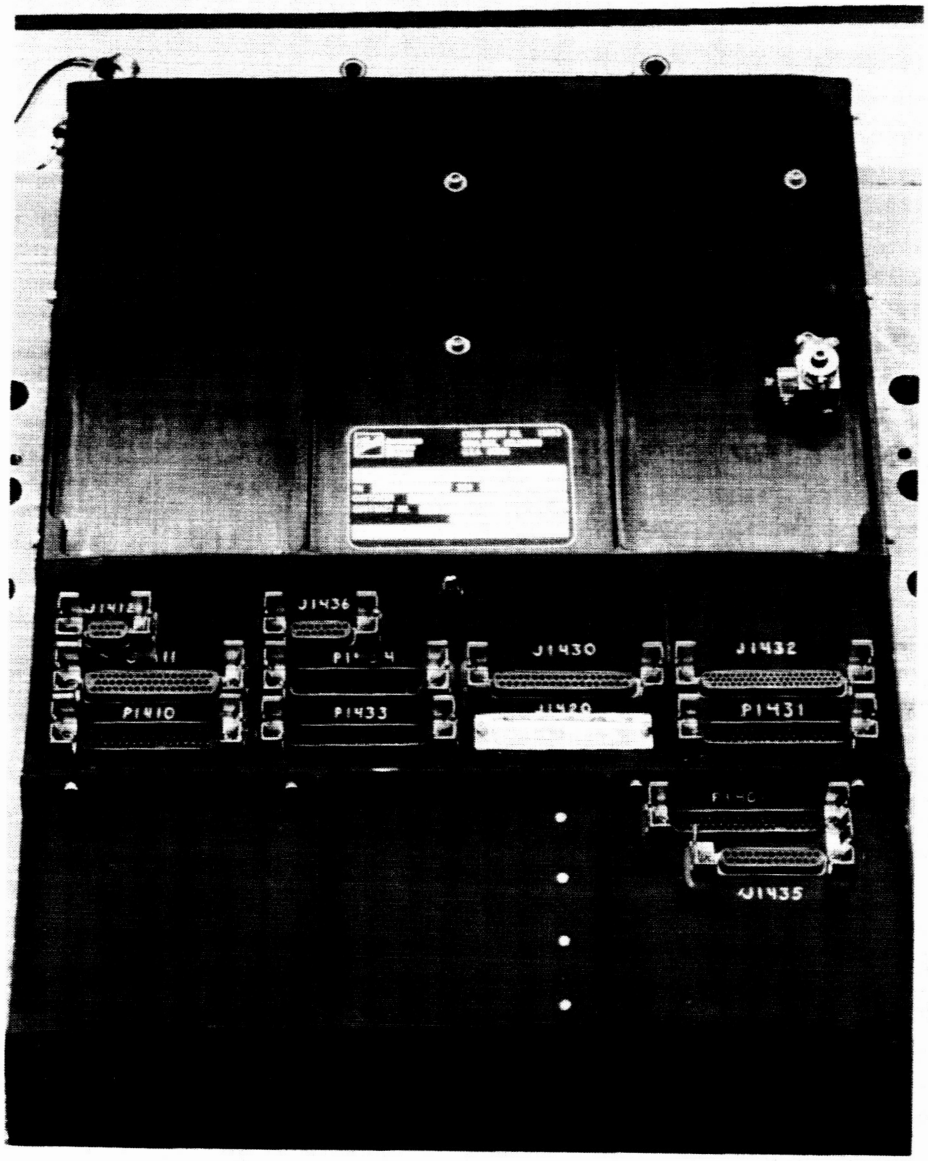

Figure 2
MAIN ELECTRONICS BOX WITH CONNECTOR CLIPS

control sequencer. This failure would not allow the controller of the detector (an SEC Vidicon) to complete the read mode; picture information was not being sent to the onboard tape recorder. Over the following months, repeated unsuccessful attempts were made to revive the instrument.

Between the time of the failure (1980) until the time of the repair (1984) periodic checks were made of the instrument. At three month intervals all mechanisms within the experiment were exercised by command to check for functionality. During that time no additional failures were noted.

After the hardware failure, testing was undertaken by GSFC to determine the susceptibility of the MM54C161 device type to the failure seen in the MEB electronics. After extensive examination GSFC found that 50% of the residual lot of this type device failed under active loading when vacuum-baked at elevated temperature. The conclusion of the GSFC analysis effort, echoing the thoughts of BASD engineers, was that the problems in the C/P instrument could be explained by the failure of this part. This conclusion was important to the possible repair of the instrument because it meant that all of the failed parts were contained in the main electronics box; hence, full instrument operation could be restored with the replacement of the MEB.

STATEMENT OF WORK

The development of an appropriate strategy to rebuild the MEB was the first step in the C/P repair program. For proposal purposes the rebuild process was divided into four categories: analysis of the MEB design, parts procurement, assembly and test. GSFC established one guideline--only a minimum level of analysis was to be done on the original design, primarily in the area of the suspected failed part. GSFC insisted on the importance of keeping the program cost commensurate with the risk of the repair.

Using this as a guideline, four separate strategies were proposed to GSFC, each slightly different in terms of scope of work and cost. The differences centered around two points: 1) Given the inherent risks associated with rebuilding a functionally-identical MEB and the uncertainty of repairing the hardware in space, what was the appropriate program approach to rebuild the MEB? and 2) How should the program tasks and the corresponding risk be best divided among GSFC, HAO and the original designer BASD?

The effort to define the strategy was time consuming, but from an initial standpoint was important; all the roles among the respective organizations were defined, and lines of communication were well established. The risk in the program was understood by all, and, more importantly, the three organizations shared the risk. This sharing helped establish cooperation that was vital to the rebuild success.

The final strategy gave each of the organizations a role in which their respective strengths contributed to the program. GSFC was responsible for parts procurement and screening. BASD fabricated and assembled the replacement main electronics box using the GSFC-supplied parts. HAO accomplishedthe board level and system functional tests in parallel with the ongoing SMM I data analysis.

DESIGN CHANGES

Following a program philosophy that a minimum of funds would be spent on the analysis of the present design (except for the study of the failed component) only a cursory look was given to the design of the MEB. A high-reliability counter chip was substituted for the failed part in the MEB circuitry. Additionally, BASD recommended replacement of a driver IC with a similar part having wider operating tolerances.

Three design changes were proposed for the experiment: two to improve the image quality, and one to make the instrument TDRSS-compatible. To accommodate the anticipated switch from GSTDN to TDRSS during the mission, and to allow for varying lockup time with the TDRSS satellite, additional electronics were added to the MEB to provide adjustable-length headers and trailers on the data. This change was straightforward. The second modification involved reprogramming the firmware in the MEB to alter the Vidicon read out. This modification caused the blanked beam to retrace the line on the Vidicon that was just read, rather than advance to the next line before the retrace. (It was believed that the data on the unread line were being altered by this beam sweep.) Implementing this change required altering the PROM program; additional circuit modification was not required. Although this type of modification was risky, the promise of higher fidelity data convinced us to proceed. The third change, not pursued because of unacceptable risk, was altering the rest position of the Vidicon beam. This was proposed to reduce and/or eliminate the large artifact that had developed on the target of the Vidicon. Incorporation of this change required a degree of software and hardware modification that we were not certain would be successful without having the C/P instrument to verify the operation.

BASD suggested that we examine the power supply construction, particularly the transformer assembly. BASD fabrication practices had changed since the original construction of the instrument. This effort was not undertaken because of time and funding limitations.

Mechanically, the power supply was the weakest area in the main electronics box. Given the reduced vibrational loading of the shuttle launch compared to the original Delta launch, we proceeded knowing that the power supply could be modified for higher reliability, but we were willing to accept the design as it was in order to move into construction.

CONSTRUCTION OF THE MAIN ELECTRONICS BOX

Using a certified parts list supplied by BASD, GSFC worked quickly to procure and screen parts. In cases where original parts were no longer in production, HAO and BASD advised GSFC on appropriate alternative choices. Some parts were available from the original SMM inventory at BASD; some were in inventory at BASD and GSFC in other programs and were "borrowed" until replacements could be obtained.

BASD began construction of the replacement MEB. Parts were let to the machine shop; the electronic boards (stitch-weld technology) were sent to the original manufacturer for assembly. As parts were supplied by GSFC, the first of the fifteen electronic boards were assembled, checked for continuity, and carried to HAO for testing.

TESTING

The work done by the engineers at BASD in establishing the original design and test philosophy for the SMM C/P hardware program was essential to the success of the repair program. The original MEB had been constructed from the breadboard level, with one significant difference--the breadboard was designed physically and functionally identical with the flight version, and the boards were pin-for-pin compatible with the flight versions. In the original program, BASD constructed a flight-equivalent card cage, and built each of the fifteen breadboard electronic boards to flight-ready completion level. All documentation was complete before building the flight boards. No changes were made to the flight versions unless the same changes were made to the breadboard versions. This approach had two positive program impacts: 1) All the documentation was complete prior to the flight build up, and the flight versions of the boards were essentially just copies of the breadboards; and 2) A flight board could be checked for operation by substituting it for its breadboard counterpart. Thus the breadboard hardware that existed at the end of the original hardware program was functionally identical to the one onboard the Solar Maximum Spacecraft; and, importantly, we were able to test the repair mission boards in the same way as the flight versions in space.

Having the breadboard card cage proved invaluable in the rebuild effort. Because of the quick pace with which the original program had been finished, there was some concern for the exact match of documentation with the flight MEB. In cases where the documentation differed from the breadboards, we relied on the latter. Board testing was done on an individual card replacement basis, substituting the flight cards as they became available into the breadboard card cage. Using the original test software with appropriate modifications, all testing was repeated in the same fashion as the original program.

We were confident of our ability to build a replacement MEB using this test philosophy, except for one concern: Could there be a buildup of timing signal error in the replacement boards undetectable in the breadboard card cage, but when assembled in space with the flight coronagraph would be slightly out of limits? To all the participants in the program, this concern was always present. No fiscal or programmatic changes, short of bringing back the C/P from space for compatibility checks with the replacement MEB, would reduce this risk.

As a side note, an area of uncertainty prior to the start of the program was the time required to refurbish and make operational the ground support equipment (GSE). This hardware had been in warehouse storage since 1980, and documentation on the design and operation of the equipment was minimal. HAO subcontracted directly with the original designer of the equipment to produce documentation for the GSE; this method proved very satisfactory in restoring the GSE hardware to operational condition.

PROGRAM NARRATIVE

During the first few months of the rebuild program no serious problems were encountered. Although the contract start date for the rebuild of the main electronics box was November 1982, all program staff were not available until mid-January, 1983. HAO requested from BASD that all of the original program personnel be assigned to the rebuild program; but because of ongoing projects at BASD, this was not possible. Only in the power supply buildup was there experience from the original program. In areas where expertise was vital, HAO used direct subcontracts as described above. In general, parts procurement went smoothly and fabrication of the MEB mechanical components progressed satisfactorily. The ground support equipment was operational and ready for test, and HAO had completed the design and fabrication of an instrument simulator and Vidicon assembly to use during the testing phase. In general, the program was on schedule and costs were within estimates.

Six months later, in mid-July, difficulties began to increase. Much of the slack had disappeared in the BASD schedule, which meant increased pressure on the checkout phase of the schedule. The difficulties were caused by the large amount of paperwork required to test the boards. The program requirement for signed, released test procedures was overwhelming the small technical staff. Only two of the fifteen flight boards had been tested, and the power supply fabrication was behind schedule. Changes in the paperwork aspect of the program were necessary to return the project to the original schedule.

A compromise solution was reached between the technical staff and the quality assurance personnel overseeing the program. A "laboratory notebook" was substituted for formal released procedures to record the test results. Supplementing this notebook were copies of the computer code and computer-recorded output. In exchange for the reduction of formality, the quality control personnel interacted directly with the project staff on a near-daily basis. Both of these changes had positive results.

The next month, August, was pivotal to the program. With the testing process streamlined, the bottleneck holding up completion of the individual board tests was eliminated. Individual board tests were completed on August 22. The power supply was also nearly complete--a transformer that had failed during the buildup testing was being replaced. The mechanical housing and the MEB harness were complete and ready for final assembly.

By the end of September--ten months after the program start--all the electronic boards were checked both individually and by groups, and BASD subcontracted the electronics soldering to Gulton in New Mexico. After return to BASD, the boards were cleaned, potted, conformal-coated, vacuum-baked and assembled into the flight mechanical housing. The MEB was complete and ready for final testing on October 20, one year from the start of the rebuild activity.

PROBLEMS

Three major problems were encountered during the qualification testing, each requiring major rework of the MEB: 1) a failure in the transformer in the power supply; 2) mechanical fatigue failure in a lead of a power supply transistor, and 3) mechanical fatigue of an EMI filter in the power supply.

The first functional acceptance test was run under control of the PDP 11/34 computer at HAO on October 21. The engineering data from this test compared very favorably to the original MEB test results. This first test was the baseline to which successive tests would be compared.

The first vibration (3 axis random, 13.09 Grms) was done on October 24 and 25 at BASD, and the box was returned to HAO for the post-vibration testing. A problem was noted immediately. The MEB was disassembled the following day, and a failure was discovered in the transformer on the 54476 power supply board. The transformer was replaced with a new assembly incorporating the latest BASD design. This effort was complete on November 10.

Before a revibration of the MEB, GSFC recommended a thermal cycle burn-in test. This was done on November 11 and 12. Functional checks were performed throughout this testing, and it was noted during the hot temperature (45° C) soaks that the operation of the box was intermittent. A second test was run to verify that the box would operate in the expected environment of 0° to 40° C; no problems were discovered.

A second vibration was done on November 17 and 18 (3 axis random, 13.09 Grms). The post-vibration functional test on the following day indicated a second power supply problem--a transistor base lead had fatigued. The cause of this was a structural bridge of conformal coat between the transistor can and the adjacent transformer case. As this board flexed during vibration, the motion of the transformer caused the transistor lead to bend and ultimately to break. This base lead failure put additional electrical load on a similar transistor, which failed soon after power was applied for the post-vibration functional check. The project did not have spare flight transistors and had to screen similar parts borrowed from another BASD program. In addition to the failed transistors, cracks were developing in the potting of the inductor assemblies. Mechanical engineering at BASD recommended adding stiffeners to the power supply boards; these items were fabricated and bonded to the sides of each supply board. A final dip-dab of all suspect components was done to prevent further fatigue. Modifications were complete on November 30, and a successful functional test was run that evening.

The MEB was vibrated for a third time. This time in three axes at a reduced level (8.6 Grms) on November 30 and December 1. A health check run that evening indicated a problem in the 5 volt line, but a functional test the next day was problem-free. Suspecting an intermittent electrical connection, a tap was made to the side of the box; the 5 volt problem reappeared. The power supply was disassembled on-site, and the intermittent connection was found in an EMI filter. Examination of the filter revealed that extended vibration had weakened the encapsulation material inside the filter. Consultation among GSFC, BASD and HAO pointed to only one solution--replacement of all seventeen one-amp filters. This was finished by December 7, and the MEB successfully tested on December 8.

The MEB went to the shaker a fourth time (one axis, 8.6 Grms); the functional test run the next day indicated that the MEB was fully functional and ready for delivery to KSC.

CONNECTOR CLIPS

The main electronics box is electrically connected to the coronagraph telescope and the SMM satellite data and power bus through twelve cables ranging in size from from 9 to 50 conductors. These cables end in Cannon 'D' type electrical connectors. The flight assembly of the connectors was done using standard Cannon screw lock hardware to mate the connectors securely. Concern over the astronaut's use of these screw lock assemblies--owing to their very small size--was raised early in the program by GSFC.

GSFC proposed eliminating the use of the screw lock assemblies in the repair. It was expected that during the repair, two events could render the screw lock assemblies useless. First, some of the screw lock assemblies could have cold welded and using a power screwdriver to unfasten the connections would shear the screws. The broken assemblies would be useless for refastening. The second problem was a tendency for the screws to float out of the retainers, and retrieval and handling of floating screws would be impossible with space gloves.

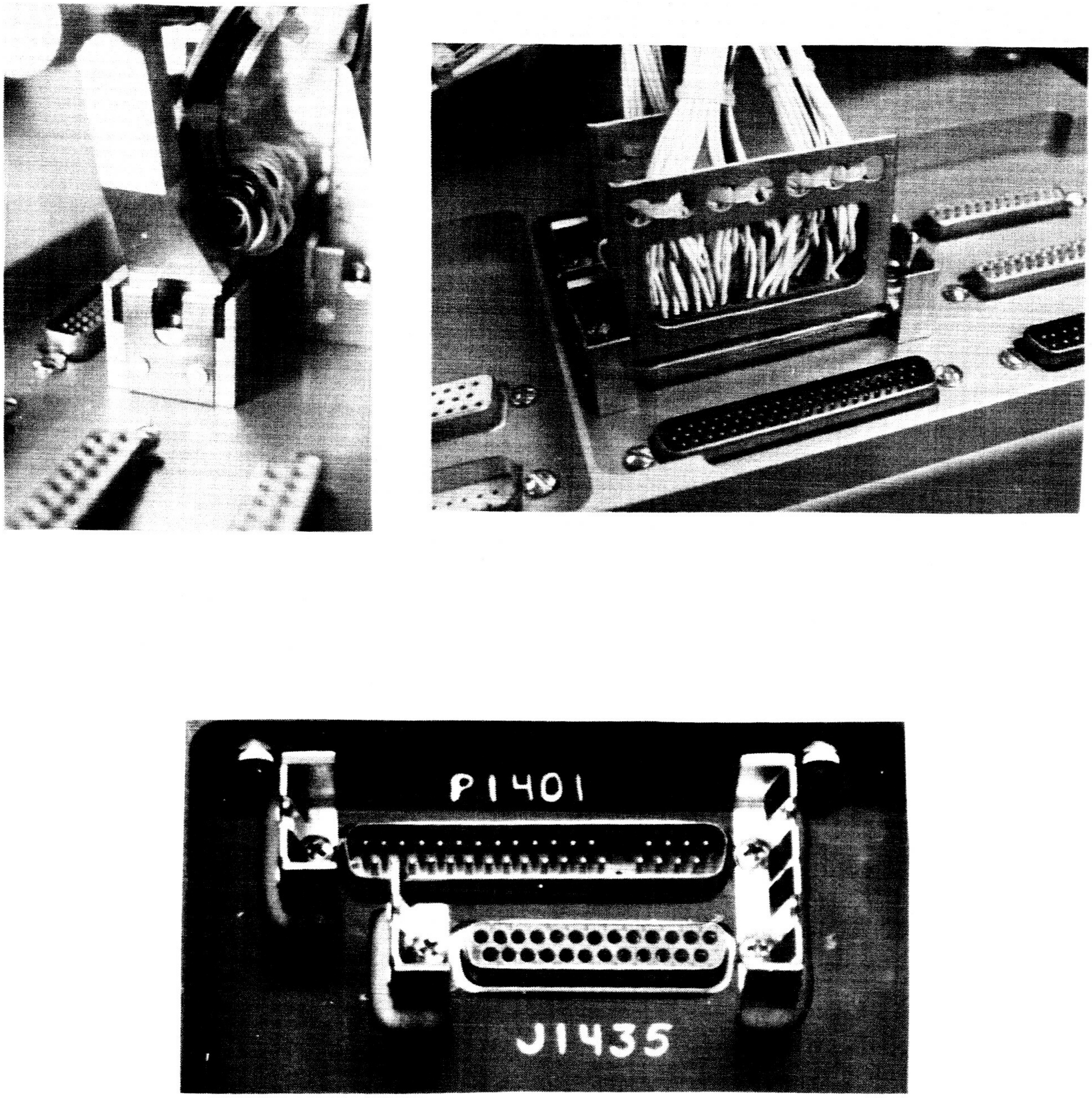

Figure 3
CONNECTOR CLIPS

Close up of connector clip with extraction tool (left), the mated connector (right), and the connector clip mounted on the MEB (bottom).

Figure 4
MAIN ELECTRONICS BOX INTEGRATED WITH THE SMM SPACECRAFT

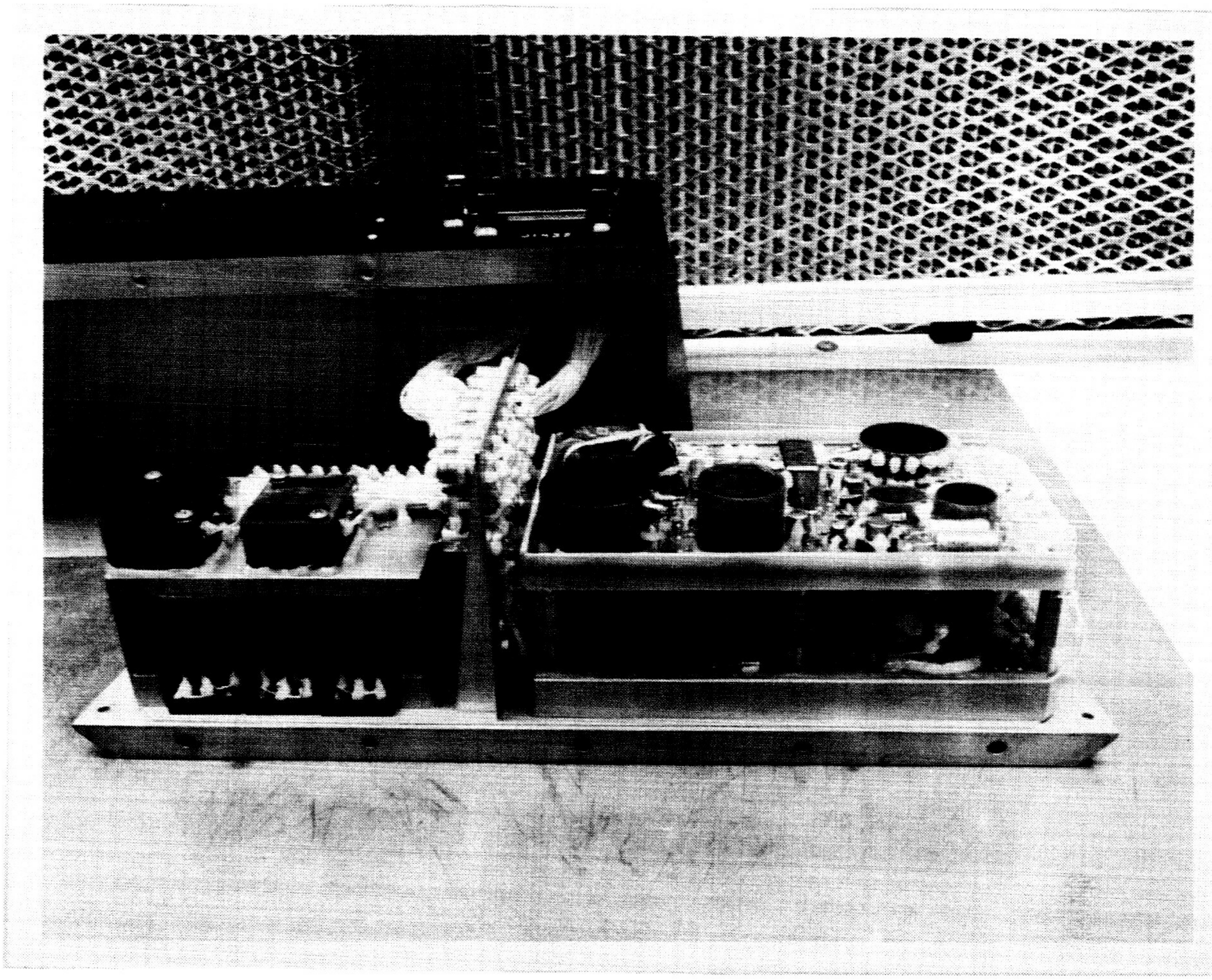

Figure 5
MAIN ELECTRONICS BOX POWER SUPPLY

The power supply section of the main electronics box. The inductor assemblies are top left. The EMI filters are mounted to the center vertical plate. The board stiffeners are attached to the top right PC board, on which the circular transformers are mounted.

GSFC designed and fabricated a set of gold-plated beryllium copper clips to replace the screw lock assemblies. These clips, when attached to the box on both sides of the connector, held the connectors together by spring action. Tests performed on the clips during simulations by the astronauts revealed some shortcomings with the design. Mating effort varied from connector to connector depending on the clip placement. In particular, the rough edges on the connector flange tended to "bite" into the clip surface, producing a stick-slip effect. Aligning the connectors was difficult; the connector contacted the clip before connector shells mated, and a strong push was required to mate them. The result of this was connector pins bending due to the tendency of the connector to mate one side before the other. Similar difficulties were experienced in the lab at HAO, and both HAO and BASD were concerned about connector mating problems during the repair. HAO recommended replacing the clips with a second version, and GSFC agreed to test a second design.

Using the GSFC design as a starting point, the second clip incorporated some additional features. The beryllium copper spring was retained, this time unplated because the gold abraded during the mating process. An aluminum bracket, onto which the spring was riveted, was attached to the top of the box. This bracket provided a piloting action during the mating process; one only had to get the connector into the aluminum brackets to be assured of alignment. The spring action was changed to lock the connector only after the connector pins had engaged. Testing conducted on the prototype clips at HAO by some of the flight crew--Nelson, Van Hoften, Scobee and Ross--was positive. The crew recommended to GSFC that the new design be tested in the simulator, the results of which were equally positive. GSFC authorized HAO to install the clips on the electronic box. Mating the connectors during the repair was done quickly and successfully.

RESULTS

There was concern over possible particulate contamination of the C/P telescope during shuttle rendezvous with the spacecraft and the servicing of the instrument. Coronagraphs are very sensitive to particulates that move into the field of view because they scatter intense sunlight into the instrument. Light from the corona is one-billionth the strength of light from the solar disk; any sunlight that scatters off particles can overwhelm the coronal signal. We have not seen any contamination effects caused by servicing.

Operation of the Coronagraph/Polarimeter experiment has been restored, with certain areas of performance enhanced over the previous SMM I mission. The instrument is fully operational. All mechanisms are functional; all engineering data being received from the C/P are comparable with the data received from SMM I before the intermittent condition began.

There is an improvement in the photometric stability of the Vidicon. Changing the beam sweep retrace firmware during the rebuild, and altering the the flood/erase procedure before taking an image with the C/P, stabilized the response of the detector. We can now see the gradual fluctuation in brightness that one would expect to see as the incident solar flux changes during the earth's elliptic orbit about the sun. This change was not observed during the first mission because of the intermittent operation of the read electronics, and the changing sensitivity of the Vidicon target.

Sampling images from the second mission has revealed peak-to-peak fluctuations in brightness of "green" calibration exposures of $\pm$ 2.5% of all data received; this compared to peak-to-peak fluctuations during the SMM I mission of $\pm$ 4.8% of selected images. This improved performance is important scientifically. SMM II images can now be photometrically corrected for intercomparison. This will allow quantitative examination of changes in the corona over the life of SMM II.

At present SMM II has returned 50,000 images of the corona, nearly double that of SMM I. The repair-in-space of the C/P main electronics box quadrupled the operational life of the experiment; this continuation of observations has been at a cost 15% over the original SMM I program funding.

CONCLUSIONS

(1) Using original program personnel in rebuilding the MEB hardware was important to the success of the repair. Use of consulting subcontracts yielded excellent results.

(2) The expenditure of a minimum amount of time and funds to reduce risk proved to be satisfactory. In retrospect it was less expensive to repair failures during testing than to engage in risk analysis.

(3) Considerably more time than usual was invested defining the scope of the project and the statement of work. This had two effects: 1) a contingency fund was not required to cover uncertainties, and 2) the plan for rebuilding the hardware was understood by all participants.

(4) The existence of a breadboard version of the MEB to use for comparison and testing was essential to the rebuild effort. Using the test philosophy developed for the original program greatly aided our efforts to rebuild the MEB as an exact replica of the original.

(5) Modifying the procedures and paperwork requirements to suit the program did not reduce the quality of the final effort.

(6) Involving the quality control people in the decision making process encouraged cooperative participation in the program, conserved schedule time, and aided the effort.

ACKNOWLEDGEMENTS

The success of this effort was due to both good fortune and the very hard work of a dedicated group of the technical personnel on the Solar Maximum Repair Mission C/P team. Special thanks to F. Cepollina, M. Logan, R. Davis, H . Doyle, J. Henegar and V. Peric of GSFC; also to D. Lloyd, C. Anderson, J. Tracy and W. Cebula of BASD; and to L. House, R. Lee, L. Lacey, H. Hull, R. Reynolds and A. Stanger of HAO. All worked unceasingly to contribute successfully to a new era in space endeavors. Lastly, thanks to the crew of STS 41C for a truly remarkable effort.

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ON-ORBIT SERVICING - HUBBLE SPACE TELESCOPE

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ABSTRACT

The Hubble Space Telescope (HST) is a very unique satellite not only for the science mission but also because of the design features which fully utilize on-orbit servicing.

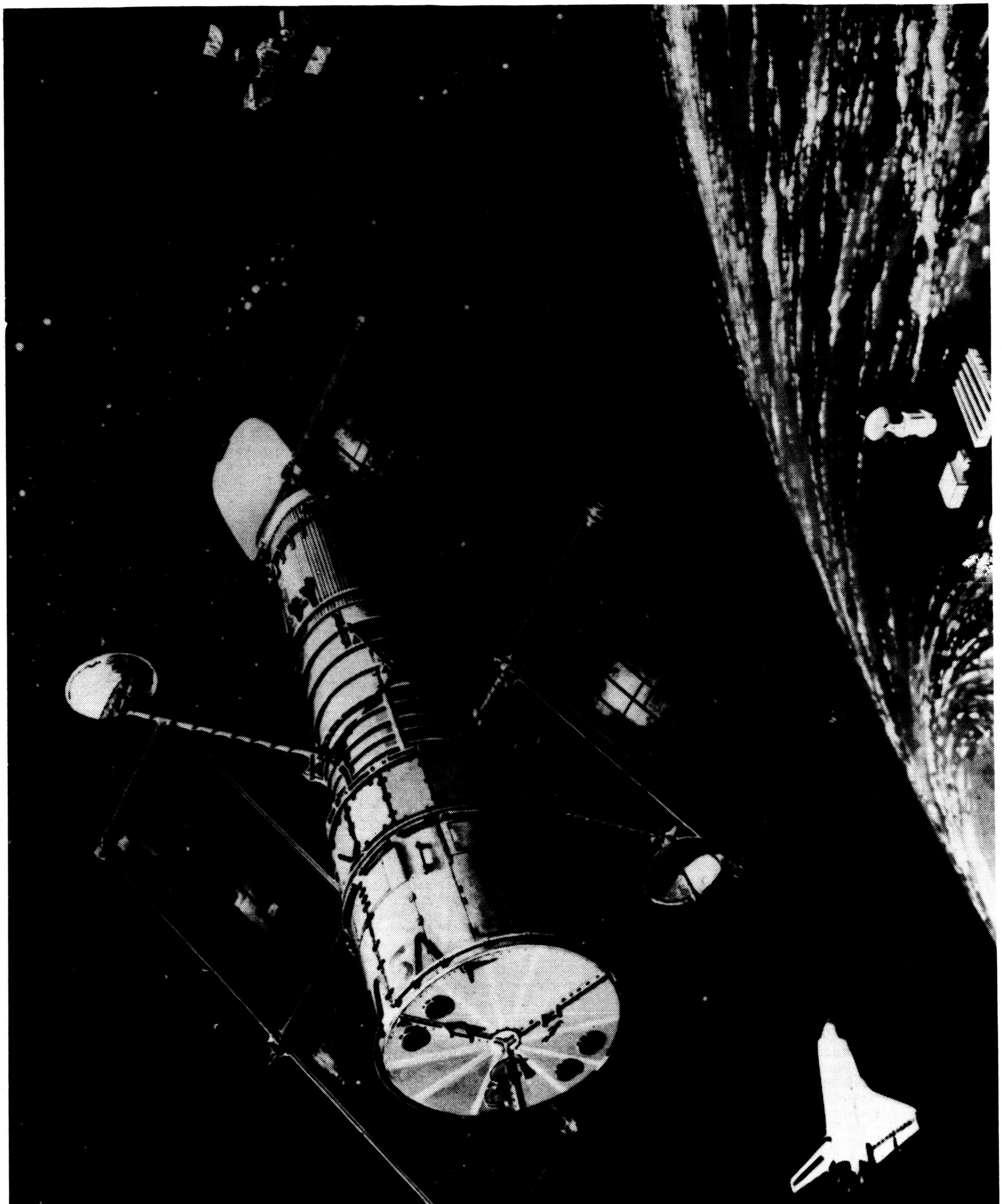
This paper will describe the HST system concentrating on the design and planning for on-orbit maintenance.

HUBBLE SPACE TELESCOPE SYSTEM

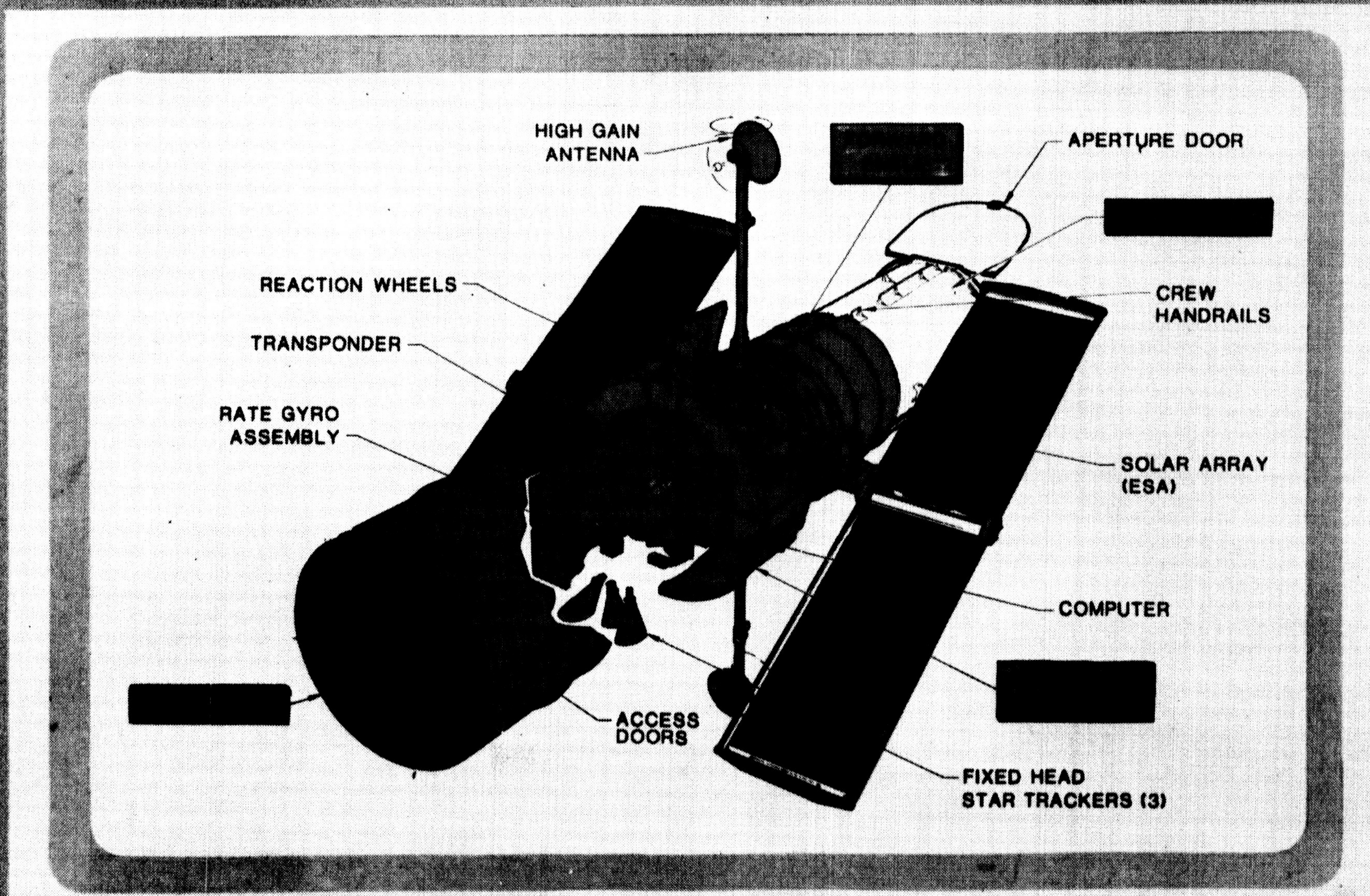
The Hubble Space Telescope (Figure 1) is an unmanned orbiting astronomical observatory which will have an unparalled scientific capability. From a 320-nautical mile orbit astronomers expect to see seven times farther into space than any ground based optical telescope; seeing objects with ten times better resolution and fifty times fainter. Operating on a 24-hour basis, the HST has the ability to lock onto targets for up to 48 hours.

The HST (Figure 2) is divided into three major structural elements consisting of appendages, System Support Module (SSM), and the Optical Telescope Assembly (OTA).

The appendages are functional elements which will be deployed after launch. The aperture door, located at the front of the spacecraft, is open for viewing but will automatically close if a bright light source appears in the field of view. Two solar arrays provide 4000 watts of electrical power, turning to ensure optimum sun pointing. Two high-gain antennas are the primary link for command, control and data acquisition via the tracking data relay services satellite.



HST SPACECRAFT MODULE CONFIGURATION



The SSM consists of the vehicle's external structural elements and operational subsystems. The external elements provide thermal protection for operational components and structural support for the components and appendages. Power, Communication, Data Management, Pointing Stability and Thermal Control comprise the HST operational systems.

Inside the SSM is the OTA consisting of a forward truss assembly containing the 2.4 meter primary mirror and the 0.3M secondary mirror and an aft truss assembly which supports the fine guidance system and scientific instruments.

Currently the scientific instruments include a wide field/planetary camera, a high speed photometer, a faint object camera, a high resolution spectrograph and a faint object spectrograph. The HST is designed to accept infrared, cryogenically cooled instruments.

HST is designed to utilize the NASA Space Shuttle for deployment, on-orbit servicing, reboost and Earth return. The HST Program is currently developing interface agreements with programs in planning like the Space Station and Orbital Maneuvering Vehicle.

Marshall Space Flight Center is the lead NASA Center for the Design and Development of the HST. Goddard Space Flight Center is responsible for the scientific instruments and HST operations. Lockheed Missiles & Space Company is responsible for SSM Design/Fabrication, HST Assembly/Verification, System Integration and Operations Support. Perkin-Elmer Corporation is responsible for the design of the Optical System and OTA structure. The European Space Agency is participating in the HST Program by providing the solar array systems and the faint object camera in exchange for viewing time.

HST ON-ORBIT MAINTENANCE

The goal of the on-orbit servicing plan is to increase the HST scientific effectiveness by maintaining the operational systems and upgrading the science capabilities for a projected orbital life is fifteen years. In order to reach this goal the following program planning elements must be in place:

- Design for servicibility
- System analysis capability
- Logistics planning
- Design of on-orbit servicing hardware

The basic HST contract requirement called for a system to be totally maintainable in space equating all components designed with orbital replacement features. These features included use of captive fasteners; incorporation of handling features such as handles and tethers; utilizing wind tab or rack/panel electrical connectors; providing tool accessiblity and visibility acceptable to the astronaut suit limitations and paying particular attention to EVA safety requirements. Because the HST Program stressed the use of existing, flight proven components as a cost savings; each subsystem and component design required a unique design approach.

The design of the large axial scientific instruments, weighing up to 750 pounds and as large as phone booths, were driven by a unique alignment to the system boresight requiring special latches to ensure the alignment and react to the launch induced loads. Design frames, including guide rails, portable lights and handles were also required to aid in the changeout process.

The fine guidance electronic box had no unique alignment requirement; however, the thermal transfer requirements drove the number of fasteners retaining the box to structure.

As the program developed and box designs progressed through the preliminary design phase, it became apparent that component reliability and system redundancy could play a big part in reducing design complexity and weight. The program development has led to the definition of nearly half of all HST components as orbital replaceable. Table 1 lists these components. In many cases special tools like connector pliers were developed to aid in the changeout.

HST COMPLEMENT	ORU DESCRIPTION	APPROX. (INCHES) SIZE	APPROX. (POUNDS) WEIGHT	LOCATION
2	SOLAR ARRAY (SA)	172x27x26*	797	EXTERNAL TO HST, ALONG V1 AXIS ON ± V2 SIDES
3	RADIAL BAY MODULE (RBM) (FGS/WFS)	66x46x22	504	IN FPSA, ± V2 AND + V3 RADIAL BAYS
1	WIDE FIELD/PLANETARY CAMERA (WF/PC)	83x31x79	500	IN FPSA, –V3 RADIAL BAY
1	HIGH RESOLUTION SPECTRO-GRAPH (HRS)	36x36x87	700	IN FPSA, AXIAL BAY 1 (+V2, +V3)
1	FAINT OBJECT SPECTROGRAPH (FOS)	36x36x87	700	IN FPSA, AXIAL BAY 2 (+V2, –V3)
1	FAINT OBJECT CAMERA (FOC)	36x36x87	700	IN FPSA, AXIAL BAY 3 (–V2, –V3)
1	HIGH SPEED PHOTOMETER (HSP)	36x36x87	700	IN FPSA, AXIAL BAY 4 (–V2, +V3)
1	DF–224 COMPUTER	24x23x18	112	BAY 1, SSM EQUIP. SECT.
6	BATTERY	24x10x14	137	BAYS 2 AND 3, SSM EQUIP. SECT.
3	FINE GUIDANCE ELECTRONICS (FGE)	23x12x11	52	BAYS D,F,G, OTA EQUIP. SECT.
1	SI CONTROL AND DATA HANDLING (SI C&DH)	34x26x10	136	BAY 10, SSM EQUIP. SECT.
4	REACTION WHEEL ASSEMBLY (RWA)	25 DIA x 21	104	BAYS 6,9, SSM EQUIP. SECT.
3	RATE SENSOR UNIT (RSU)	12x10x9	24	SSM EQUIP. SHELF
3	RATE GYRO ELECTRONICS ELECTRONICS CONTROL UNIT (ECU)	11x9x9	17	BAY 10, SSM EQUIP. SECT.
12	FUSE PLUG	6x5 DIA	0.4	BAY 4, SSM EQUIP. SECT.
2	DIODE BOX	5x6x34	30	EXTERNAL TO HST, FWD FACE OF SSM EQUIP. SECT.
1	DATA MANAGEMENT UNIT (DMU)	26x30x7	83	BAY 1, SSM EQUIP. SECT.
2	MULTIPLE ACCESS TRANSPONDER (MAT)	10x4x2	12	BAY 5, SSM EQUIP. SECT.
2	SOLAR ARRAY DRIVE ELECTRON-ICS (SADE)	14x10x8	18	BAY 7, SSM EQUIP. SECT.
3	TAPE RECORDER (TR)	13x10x7	21	BAYS 5, 8, SSM EQUIP. SECT.
1	ELECTRICAL POWER/THERMAL CONDITIONING ELECTRONICS (EP/TCE)	17x14x8	29	BAY H, OTA EQUIP. SECT.
4	DATA INTERFACE UNIT (DIU)	15x16x7	25	BAY B, OTA EQUIP. SECT. BAYS 3,7,10, SSM EQUIP. SECT.
1	OPTICAL CONTROL ELECTRONICS (OCE)	11x13x7	20	BAY C, OTA EQUIP. SECT.
1	MECHANISM CONTROL UNIT (MCU)	20x12x8	25	BAY 7, SSM EQUIP. SECT.
2	SINGLE ACCESS TRANSMITTER (SAT)	10x8x2	10	BAY 5, SSM EQUIP. SECT.

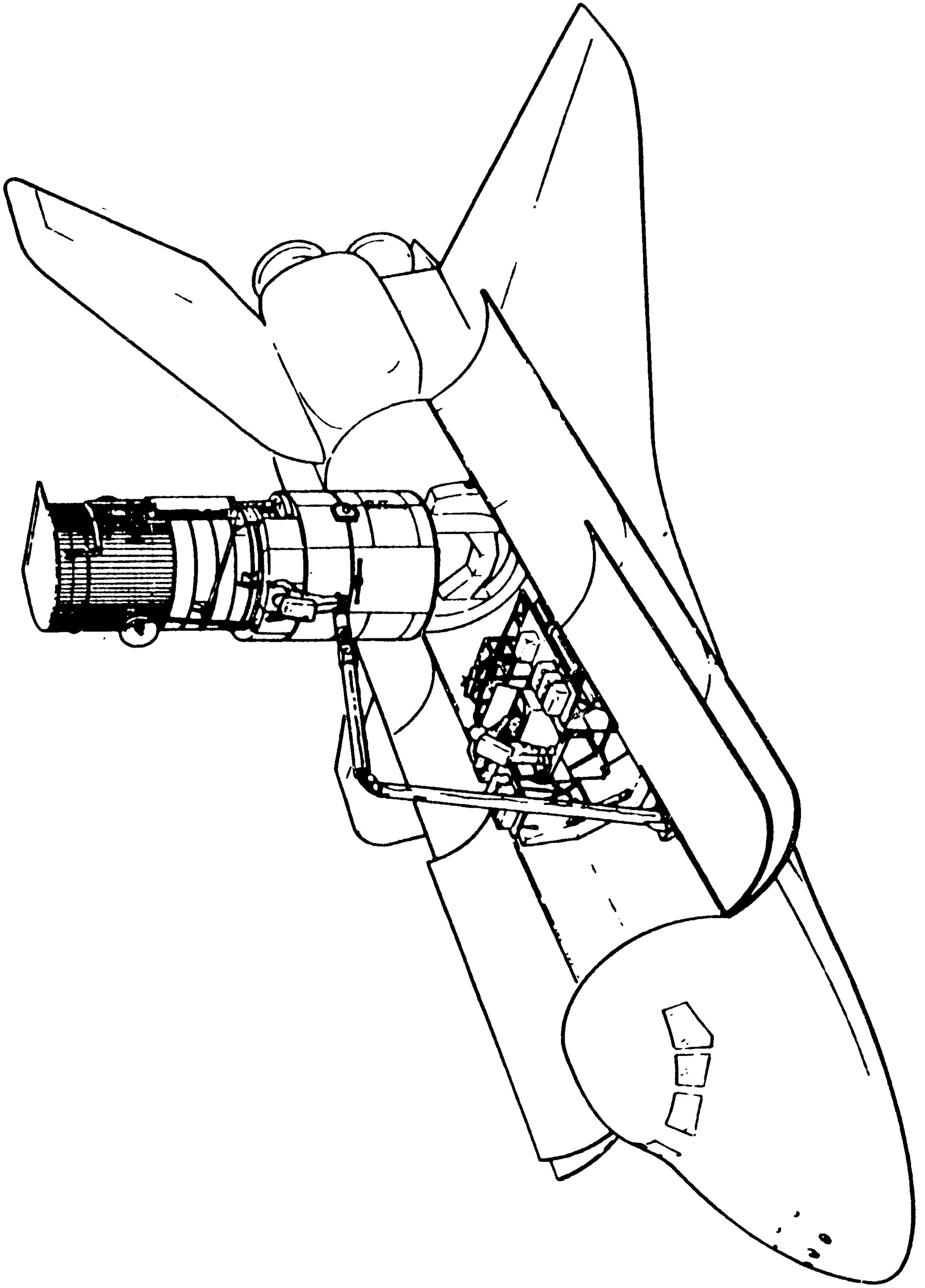
* STOWED DIMENSIONS

Systems analysis plays a critical part in HST servicing. Traditional analysis such as thermal analysis to support operational and non-operational modes must be expanded to include changeout conditions. Dynamic models must include launch and return conditions for components mounted on the satellite and on-servicing hardware. The maintenance of interface documentation and verification capability will play a key role in definition or system compatibility of changeout components as well as in crew training.

A unique problem is the decision to maintain systems to maintain the required data base. A launch-and-leave-it system can rely on short-lived data bases. Long-term programs, like HST, must develop a data base which is compatible to computer hardware/software systems for mooring techniques and maintain an information system for evaluation of future interfaces. HST is developing requirements for such system maintenance.

Logistics planning will ensure that components are available for changeout and ensure the maintenance of an Earth refurbishment capability for the life of HST. Unique problems which will be faced by this activity include long term system maintenance, component retest and shelf life verification/ processing and determination of cost-effective solutions to component unavailability.

The baseline for HST servicing is to utilize the Shuttle, component carrier and the Flight Support System (FSS) (Figure 3) along with a complement of tools. As other space servicing opportunities become available like Space Station, the HST Program will evaluate the interface for system compatibility. HST is in the process of establishing an Interface Requirements Document with the Orbital Maneuvering Vehicle (OMV) Program.



DESIGNING FOR SERVICEABILITY - SOLAR MAX REPAIR LESSONS APPLIED

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Project Engineering Manager
On-Orbit Maintenance Systems
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INTRODUCTION

Satellite servicing has become more popular in the past few years with the successfull Solar Max Repair Mission, Palapa/Westar rescue, Orbital Refueling System Demonstration, and the recent Syncom Salvage missions. However, designing for serviceability has not gained wide spread acceptance. **Current spacecraft, especially spacecraft payloads, are not being designed with** serviceabiity in mind. The chief reason cited for not designing for serviceability is that it is too expensive.

One of the key lessons learned from the Solar Max Repair Mission (SMRM), performed by the STS 41-C Crew in April 1984, is that it need not be unduly expensive to design a spacecraft that can be serviced on-orbit. The SMRM showed that the Modular Attitude Control System (MACS) could be exchanged on-orbit. The MACS is one of the Multimission Modular Spacecraft (MMS) Orbital Replacement Units (ORU). A little known fact is that there were no Extra Vehicular Activity (EVA) design specifications, no special design engineers, no Weightless Environment Test Facility (WETF) simulations or 1g simulations during the MMS design process. These items are most often cited as raising the non-recurring costs of designing serviceable hardware. The Main Electronics Box (MEB) of the Coronograph Polarimeter was also replaced during the SMRM. The MEB was never envisioned to be replaced on-orbit and therefore was never designed for on-orbit replacement.

Throughout the comming decade it will become increasingly important to design spaceflight hardware for serviceability. With the forcast restriced budgets, together with the large expenditures associated with the Space Station, it will become important for NASA to use its ingenuity to reuse or extend the life of its spacecraft in order to maintain its science programs. A spacecraft's life can be extended through unscheduled repair as in the case of the Solar Maximum Mission (SMM) or by scheduled maintenance as in the case of the Hubble Space Telescope.

Although not attributable to servicing, NASA's ingenuity has been demonstrated through the reuse of the ISEE 3 spacecraft as the International Cometary Explorer (ICE). ICE is the first spacecraft to intercept a comet. This mission was performed at a fraction of the cost and schedule of building a dedicated cometary explorer spacecraft.

There are also conceptual plans for the reuse of the SMM and Landsat 4 spacecraft by replacing their current instrument modules with different scientific payloads. In these cases NASA is reuseing the MMS spacecraft which is anticipated to be more cost effective than building new spacecraft.

With space insurance becomming prohibitively expensive or impossible to obtain as a result of numerous failures, commercial users of space could benefit from serviceable spacecraft. Hopefully, the space insurance industry recogizes the lower risk of shuttle launched serviceable spacecraft as evidenced by the retrieval of Palapa and Westar and the successfull repair of Syncom. (As of this writing Syncom has not yet been boosted to its operational altitude.) If commercial users can demonstrate that their payloads are serviceable and within shuttle or space station reach, insurance brokers should recognize that a total loss would be relatively rare.

Commercial space ventures would also realize revenue benefits from serviceability as long as repair or upgrade remains cost effective. STS pricing policy, as it relates to servicing equipment and on-orbit operations, is still a driver and servicing cost trades are likely until pricing policy is set.

This paper addresses the SMRM lessons learned, the design features of the MMS/SMM hardware, and how its most important features can be applied to spacecraft Orbital Replacement Unit (ORU) design. The conclusions are based on practical experience that could possibly lower the cost of designing for serviceability.

SERVICEABILITY DESIGN

Recently, a spacecraft systems engineer stated that it was too expensive to design a spacecraft for serviceablity. When asked what made the design process more expensive than the current spacecraft design process the following points were mentioned:

a. Mechanical design engineers have not been trained to be cognizant of suit mobility, EVA reach, crew access, sharp edge and corner radius, swept volume and tool utilization requirements. Design standards and specifications have to be collected and disseminated to the engineers.

b. Systems engineers and reliability engineers have to determine which components are to be ORUs and which should not. This determnation takes extensive analysis into relative failure rates and Mean time to failure during the course of the mission.

c. Models and mockups have to be built to verify the design concepts before flight hardware is built. Simulations must be performed in 1 g and the WETF to verify accessibility and exchange procedures. The WETF simulations require a set of waterproof mockups. These simulations can bring about unpredicted design refinements which can not be estimated ahead of time.

d. Servicing support equipment has to be designed and tested. The servicing equipment requires its own specifications, mockups, simulations, etc.

The above listed design process for serviceability sparked two questions: (1) How did the MMS module concept of ORUs evolve without any of the above mentioned steps, and what enabled the Main Electronics Box to be replaced? (2) Is it possible to apply the lessons learned from the SMRM hardware to ORU design?

MMS SERVICEABILITY DESIGN

The intent of the MMS spacecraft design was to lower the life cycle cost of spacecraft programs. This was achieved through three goals.

a. Design a standard spacecraft which could fulfill the requirements of a variety of programs.
b. Extend the design life of the spacecraft by providing for on-orbit ~~repair and~~ **upgrade through remote or automatic means.**
c. Lower the development and production cost of the spacecraft through the use of modularization.

The MMS was designed as a multipurpose low cost "standard" spacecraft that could supply basic housekeeping needs for a variety of scientific missions in the STS era. The requirement for the design of the MMS ORUs or modules was that they be serviceable through on-orbit changeout by automated or remote means. EVA changeout was not an original requirement.

During the time of MMS design, the early 1970s, robotic technology was just evolving. This necessitated an extemely simple mechanical and electrical design such that remote changeout could be effected. Each MMS module was designed to be thermally self sufficient, provide a simplified (two bolt, self aligning) mechanical attachment system and self aligning blind mate electrical connectors. Provisions for attaching and aligning a tool for module replacement were also incorporated. Figure 1 shows an exploded view of the SMM Observatory.

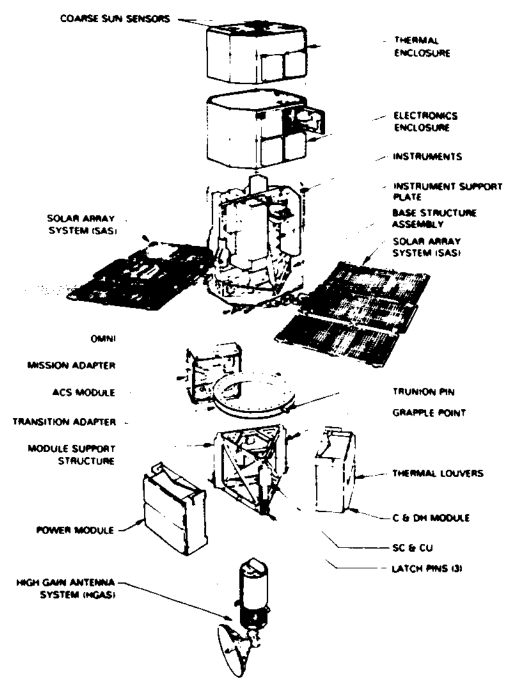

Figure 1 SMM Exploded View

During the SMRM the MACS was changed out by EVA in a little over 30 minutes with a battery powered tool operated by a crewman. The Remote Manipulator System (RMS) and its operator, the Manipulator Foot Restraint (MFR) and the crewman with the tool, functionally served as the "automated" module exchange mechanism as envisioned in the early 1970s. It turned out that the spacecraft system that was originally designed for automated exchange doubled as an ideal EVA changeout system. In other words what was designed to be simple and fool proof for a robot turned out to be acceptable, if not ideal, for an EVA changeout scenario. Figure 2 shows the Module Service Tool in use by a crewman on the MFR.

ORU DESIGN TRADEOFFS

For new spacecraft designs the initial question revolves around the size and complexity of the ORUs. How big should the ORUs be? How should functional systems be separated? What design criteria should be used? And the biggest question, how much will this system cost? These questions can be answered through classical systems engineering techniques such as trade studies, which can involve a considerable investment of time and money.

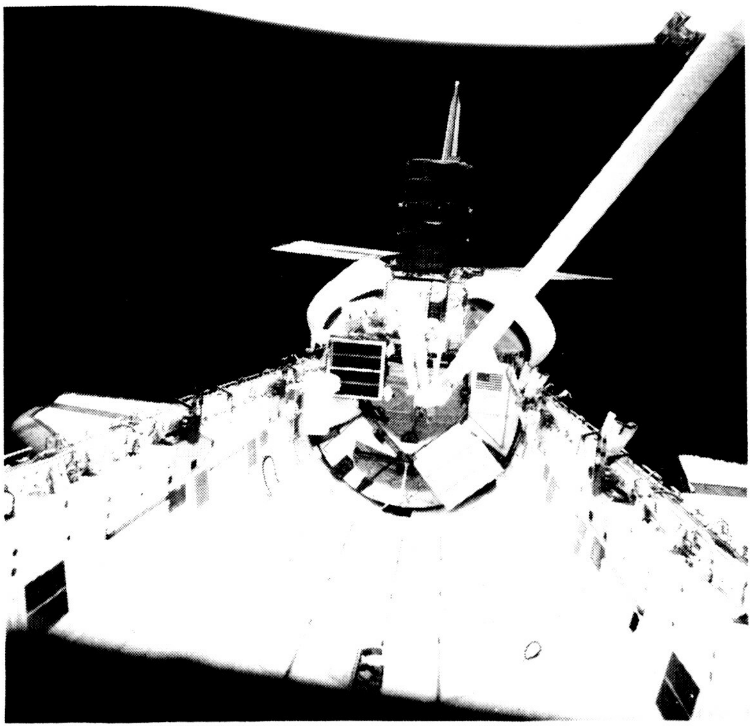

Figure 2 MACS Exchange

The following are examples of the different choices for functional partitioning of ORUs:

a. Items that are single point failures such as some power system components.
b. Components that a reliabiity analysis shows will not meet the mission life requirements or whose failure rate fall below some level.
c. Mechanical or electrical wearout items such as tape recorders, gyros, wheels, batteries, etc.
d. ORUs whose failure can be pinpointed or identified prior to a repair mission and whose replacement unit can be verified as operating normally once it is installed on-orbit.

The MMS approach was to include all components as a part of an ORU. This eliminated the trade studies and reliability analyses which otherwise would have been required. There is a major benefit to including all components as ORUs. This benefit is realized when an analysis performed during the conceptual stage of a program does not prove 100% correct after the design has matured or hardware fabricated. It can be extremely expensive to modify a non-accessible, non-ORU into an on-orbit replaceable unit.

Choosing the size of ORUs goes hand in hand with choosing the functional partitioning of the spacecraft systems. Two Examples to illustrate the size and complexity of ORUs follow:

a. Components which are roughly 1 to 2 cubic feet in size. These components are usually electronics boxes, tape recoders or reaction wheels.
b. Modules, subsystems or instruments which are 3 to 20 cubic feet in size. These modules are typically a collection of electronics boxes, electro/mechanical devices and detectors which function together.

The MMS uses only modules or subsystems as ORUs. The MMS modules do use identical mechanical interfaces between the ORU and the spacecraft. The Hubble Space Telescope (HST) uses both components, such as batteries and the DF224 computer, as well as subsystems, such as the Scientific Communication and Data Handling module (SIC&DH) and the Fine Guidance Sensor (FGS). Generally the HST does not have a standard ORU to spacecraft interface. The four axial science instruments do have a common interface however. Figure 3 shows the standard MMS module to structure interface.

Choosing a standard mechanical/electrical interface for all ORUs offers the advantage of having only to design, test and train for one attachment system. The attachment system should include a standard electrical interface. Ideally the electrical connections should be of a fool proof type that automatically mate when the ORU is mounted on the structure. This eliminates the chance of damaging pins on-orbit, shortens the changeout timeline, simplifies design for crew access and simplifies training.

There appears to be an advantage to building ORUs with standard sizes and interfaces. However, there are some volumetric inefficiencies and weight inefficiencies when modules are not filled to their capacity. In the past, few spacecraft could tolerate weight or volumetric ineffiencies due to the limitations of expendable launch vehicles. In the STS era, payload weight and volume to low earth orbit are not as critical, and some weight and volume sacrifices can be accepted in order to accomodate servicing.

Geosyncronus payloads present a different problem. They normally cannot afford even the weight penality associated with a grapple fixture, let alone other servicing weight ineffiencies. On the other hand, they could utilize the repair services of the STS until they are boosted out of range.

There are disadvantages in placing a known wear-out item in the same ORU package with hardware that does not wear out. For example, the MMS Modular Power Subsystem (MPS) batteries, which are known wear-out items, are in the same module as other electronics. In replacing the MPS, not only are the batteries replaced, but also all the spacecraft power conditioning electronics. Additional costs are incurred in having to spare a complete MPS solely to replace the batteries. The HST has overcome this logistical inefficiency by providing the capability for replacing individual batteries. Enhancements to the MPS module are currently being investigated that would allow on-orbit removal of batteries.

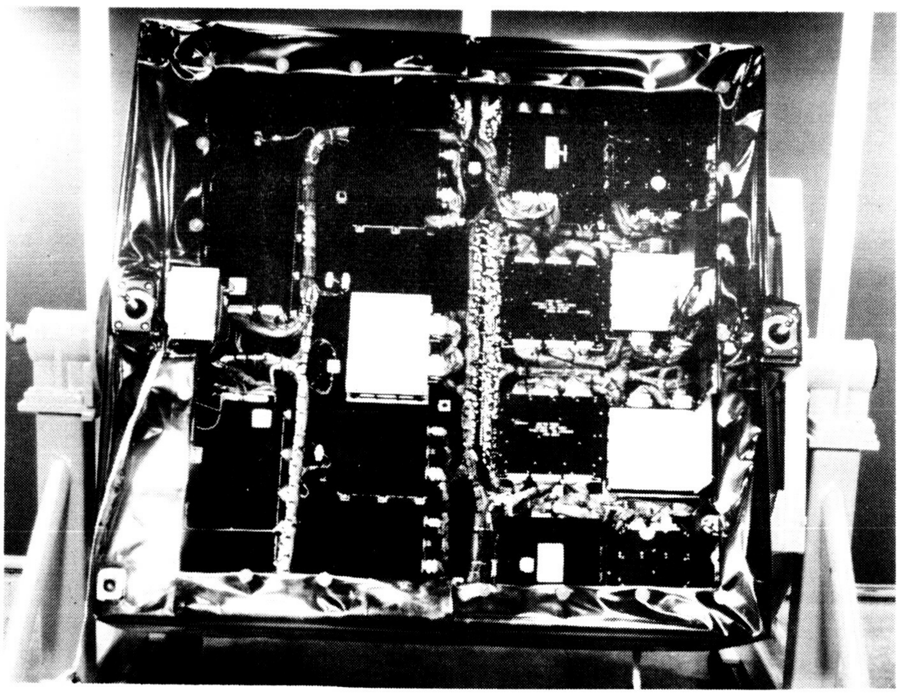

Figure 3 Typical MMS ORU (C&DH)

ORU DESIGN CRITERIA

Examining the lessons learned from SMRM, two major points come to light. The first, when designing for serviceability, keep the ORU thermally self-sufficient, provide a simple module to structure attachment system, and a tool attachment fitting in-line with each bolt and self-aligning, blind mate electrical connectors. The tool, once it is attached, serves as a handhold and as a tether point. These design features permit a short, efficient EVA change out.

The second point in designing for serviceability is that the components or subsystem which are not designed as primary ORUs be "accessible." If components are mounted such that disassembly and replacement can be accomplished within allocated EVA time periods, then the component can be exchanged even though it was not specifically designed for EVA replacement. The associated penalty, of course, is the extension of the EVA timeline as evidenced by SMRM; 30 minutes for the MACS versus 120 minutes for the MEB. Figure 4 shows the MEB on its hinged panel which allowed it to be accessible on-orbit.

Figure 4 MEB Accessible on Hinged Panel

A summary of key features present during SMRM that should be imposed as design requirements during future ORU design efforts are discussed below.

a. The mechanical attachment of the ORU to the flight spacecraft should also be used to attach the ORU to a support structure during maintenance mission launch and landing. This same attach mechanism should be used to mount the ORU to a temporary mounting bracket during the changeout procedure. The attachment bolts should be captive to the ORU and the nuts captive to the mounting structure.

b. A power tool should be used to loosen and tighten the ORU fasteners and latch itself to the ORU so that it reacts the generated torque. The power tool serves several purposes. Primarily, it serves to shorten the EVA time line. Typically, a job can be performed with a power tool in 1/3 of the time required to perform the same lossening or tightening operation manually. The tool serves as a handle and tether attachment point for the ORU. Figure 5 shows the MST in use as a handle and tether.

Figure 5 MST Used as Handle and Tether

c. The ORUs should either be thermally self sufficient or provide for the attachment of a thermal blanket around them to enhance their survival unpowered in the STS cargo bay. The MACS is self sufficient as it has thermal blankets on five sides and thermal louvers on the sixth. The MACS can survive for hours without power. The MEB was not thermally selfsufficient. During exposures to the cargo bay, the MEB was covered with a thermal blanket that was attached by velcro strips. The EVA crew removed this blanket after the MEB was installed on the SMM structure and just prior to the mating of the electrical connectors.

d. ORUs should be designed with electrical connectors that mate automatically as the structural attachment is made. The automatic mate greatly simplifies the EVA tasks. It also simplifies the design, test and simulation requirements that go hand in hand with complex EVA tasks. The automatic mate is also an aid during spacecraft integration and test. The integration flows quicker and there is less risk of damaging or incorrectly mating the connectors.

In summary, the following guidelines could lower the cost of developing serviceable spacecraft:

a. A simple, fool proof mechanical and electrical interface between the ORU and its mounting structure eliminates the need for special EVA specifications, mockups and simulations during the design process.

b. Using a standard mechanical and electrical attachment system for the ORUs lowers the cost of ORU design.

c. Providing for the replacement of all spacecraft components minimizes the need for performing tradeoffs that determine which components are to be ORUs and which should not.

d. Designing spacecraft for on-orbit serviceability through modularity benefits manufacture and test operations.

CONCLUSIONS

The experience gained during the MMS Spacecraft and SMRM programs indicates that a serviceable spacecraft need not result in undue increases in program costs if certain ground rules are set early in the program and some weight and volume increase can be traded for the servicing feature.

In order to realize the full potential benefits of servicing, NASA should take the lead with the participation of the aerospace industry in standardizing simple interfaces between ORUs and structural elements.

In the upcomming, budget constrained decade, satellite servicing can be a significant tool for attaining the most science return for the budget dollar. If satellite servicing is perceived as expensive, and as a result is not implemented, NASA's science return will diminish along with the budget.

STANDARDIZED HYDRAZINE COUPLING DEVELOPMENT

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PROPULSION AND POWER DIVISION

SATELLITE SERVICES WORKSHOP II

NOVEMBER 7, 1985

INTRODUCTION

The advent of the Space Transportation System means that changes have to be made in the way that the spacecraft manufacturers have been designing satellites. This redesign is necessary because of a radical change in the satellite's operating environment - man's presence. Already, this has had an impact - the repair of the Solar Maximum Satellite and the Leasat 3, and the retrieval of the Palapa and Westar spacecraft. These show that services which would have been unheard of only five years ago can now be accomplished. But these are just examples of the capablitities that will be available in the future.

Another example of the capability that will exist was demonstrated during the 41-G mission. An experiment, the Orbital Refueling System (ORS), was designed to simulate the task of refueling a satellite. During the mission, two astronauts, David Leestma and Kathy Sullivan, went out into the Shuttle payload bay and connected a refueling line to a simulated Landsat satellite. After the hook-up, hydrazine was successfully transferred between propellant tanks in the ORS through the refueling line. While the correct quantity of hydrazine was not simulated, the task of refueling a satellite was sucessfully performed - from the connection of the fluid transfer line to the actual transfer of propellant - in the payload bay.

The ORS was designed to refuel an existing satellite, therefore, an existing ground servicing coupling was used in the experiment. In order to meet the payload bay safety requirements, the coupling could not be disengaged during the flight. A review of existing servicing couplings indicated that there was not a coupling available which could meet these requirements, so a coupling development program was initiated. This paper deals with the fluid coupling which is being developed for the Propulsion and Power Division of the NASA-Johnson Space Center by Fairchild Control Systems Company under the contract NAS9-17333. The first use of the coupling will be on the Gamma Ray Observatory, a satellite that is being built by TRW for NASA-Goddard. This satellite, scheduled for launch in May 1988, will contain approximately 4000 pounds of hydrazine at the beginning of its life and will require resupply in May 1990. In order to refuel it, the fluid coupling must be designed, developed, and certified before it is launched because the spacecraft half of the coupling is an integral part of the propulsion system of the satellite. This coupling must have the capability to transfer hydrazine in the payload bay and endure the environments to which it is exposed.

REQUIREMENTS

Because of the environments and potential hazards associated with refueling a satellite, a hydrazine refueling coupling must satisfy a number of requirements. These include the physical requirements - maximum pressure capability, long-term exposure to vacuum and hydrazine, the temperature range within which the coupling must operate - and the safety requirements associated with working with hydrazine in the payload bay.

The Payload Bay Safety Requirements document NHB 1700.7A states that any catastrophic hazard must have three independent "inhibits" to prelude any combination of two failures, errors, or inadvertant operations from causing injury to personnel or damage to the Orbiter, facilities or equipment. Because of the concern about liquid hydrazine coming into contact with an astronaut during an EVA, any spill of this propellant is considered a catastrophic hazard.

Therefore, in any hydrazine coupling for use in the payload bay, three independent seals are required through-out the entire refueling sequence. To satisfy this requirement, three independent valves have to be in each of the coupling halves, and there must be three seals at the interface between the fluid path and the astronaut during the refueling operation. In addition, there must be interlocks built into the coupling to prevent the valves from being opened while the coupling halves are disengaged and also to prevent the coupling from being disengaged with the valves in the open position.

The physical requirements of the coupling are complex because the two halves of the coupling will have very different operating lifes. The spacecraft half will be launched and then exposed to hydrazine and to the hard vacuum of space for up to 20 years. During that time the valves will be cycled only when the satellite requires refueling. The tanker half of the coupling will be attached to a refueling ~~system, which will~~ see up to one hundred launches. The tanker coupling valves will be cycled during ground check-out procedures and during every refueling operation, being exposed to hydrazine for relatively short periods of time. The requirements for the coupling when connected, however, have to be the same. ~~This leads to a complex design.~~ The physical requirements of the coupling are summarized in Table 1. This table includes the maximum operating pressure, the temperature limits, pressure differential across the coupling during the refueling operation, leakage requirements, etc.

One additional requirement, and this may be the most important requirement, is that the coupling be delivered to the GRO by the end of March, 1986. It must be understood that the coupling contract was awarded only December 1984, and that the entire design, development and producion must be performed in the 15 months seperating these two dates. This is a considerable effort in a very short time, and the technical problems that have arisen have had to be, and must continue to be, dealt with quickly and effectively.

COUPLING DESCRIPTION

The coupling consists of four elements, i.e.:

The spacecraft half coupling.
The tanker half coupling.
The spacecraft half protective cap.
The tanker half protective cap.

A general veiw of the spacecraft half coupling and the tanker half coupling is shown in Figure 1, photograph. The photograph is of the functional mock-up that was used in the WETF at JSC in conjunction with the mock-up of the GRO. This mock-up was used to verify all of the interfaces which a crewmember has to access, hence, the spacecraft half was not fully simulated.

A schematic of the spacecraft half coupling and the tanker half coupling is shown in Figure 2.

The spacecraft half coupling, shown in Figure 3, contains three inhibits in series. Two of these inhibits are manually actuated. The third inhibit is opened by the motion of the companion valve on the tanker half coupling. Each manually actuated inhibit contains a colored position indictor to indicate open and closed positions. Relief valves are positioned across the inhibits to relieve pressure buildup of the propellant trapped between the inhibits. Pressure buildup may occur from thermal expansion or chemical decomposition of the hydrazine. Three pressure transducers are used to detect inhibit leakage. The transducers are piezo-resistive and operate at 28 Vdc. The spacecraft half coupling contains the sealing lands for three primary interface seals. Two additional seals are used for leakage checkout of the three primary interface seals. A propellant filter is located at the outlet.

The tanker half coupling, shown in Figure 4, contains three manually actuated inhibits, one of which also actuates one of the spacecraft half coupling inhibits. Position indicators, relief valves and a filter are provided, as in the spacecraft half coupling.

The tanker half coupling is connected to the spacecraft half coupling by a bayonet latch. The bayonet is manually engaged and rotated virtually to the final position by a pair of handles. The bayonet is further driven into the final position and loads the interface seals by a screw-driven lockpin. Rotation of the lockpin releases the interlock mechanisms. The interlock mechanisms prevent opening of the inhibits if the two halves are not engaged and locked in place. A manual override of the interlock mechanism is provided. The bayonet is driven out of the final position by a similar screwdriven lockpin. A single tool, mounted on the tanker half is used to rotate the lockpin, and release the pin and the inhibits.

The tanker half contains five interface seals (three primary and two for checkout). Passages between the seals provide either pressure for leak check or a vacuum source for collecting leakage. The passages terminate in four 1/8-inch diameter lines.

The protective caps provide mechanical protection for the spacecraft half coupling interface sealing surfaces and the tanker half coupling interface seals. The tanker half cap also provides the structural mount for the coupling half. Note that the coupling is launched either with the two halves engaged or with each half engaged with its respective cap.

Provided with the coupling are a tanker half test adapter and a spacecraft half test adapter. These adapters provide capability to check leakage of the three primary interface seals on the tanker half coupling and the three interface sealing surfaces on the spacecraft half coupling. Additionally, the adapters can be used for pressurizing each half for inhibit leak checks, relief valve checkout and transducer checkout.

COUPLING DEVELOPMENT

The coupling development involves testing of breadboard components, as well as complete development testing of two sets of couplings, caps and test adapters. Additionally, there will be a full qualification program.

The breadboard items consist of two types of inhibit assemblies, the relief valve, bayonets and transducer. The inhibits will be subjected to leak, functional, temperature and life cycling tests. The relief valve will be subjected to crack, reseat, leakage and life cycling tests. The bayonets will be subjected to axial force, torque and interface leakage tests. The transducer will be subjected to leakage tests and electrical input/output tests. At the completion of breadboard tests at FCSC, the breadboard test assemblies will be forwarded to NASA-JSC for propellant compatibility tests.

Two complete sets of coupling halves, caps and test adapters will be subjected to full development tests. These tests will include all of the tests that are included in qualification requirements. This is necessary because of the required shipment of the GRO flight coupling from FCSC prior to completion of qualification. The development tests consist of proof pressure, leakage, functional, life cycle, pressure drop, pressure surge, external loads, salt fog, vibration, shock, propellant compatibility and burst.

The developments tests are scheduled for completion by February 1, 1986. The GRO coupling is scheduled for shipment by March 31, 1986.

One complete coupling will be subjected to qualification tests, which are the same tests to which the two development units were subjected. The qualification test is scheduled for completion by July 31, 1986, which is the same date as for completion of two additional production units. One of the production units will be used as a spare. The other unit will be used by NASA-JSC for additional tests that are beyond the current specification requirements.

COUPLING AUTOMATION EFFORT

A study was made to evaluate various possible means of providing an automatic coupling. Three kinds of power were evaluated, i.e., electrical, pneumatic and hydraulic. Fully automated power, with and without manual backup systems, was also evaluated. The power systems were further broken down by single motor with individual gear drives, individual motors, linear actuator and solenoid actuators. Latching and nonlatching couplings were also evaluated. The evaluation was on a system basis rather than on a individual coupling basis.

It was determined that the hydraulic power was the least desirable of the three. Weight and envelope were much more than for a pneumatic power system. Additionally, a safety hazard would exist when the coupling is connected to nitrogen tetroxide service.

The second least desirable power system was the pneumatic system. When compared to the electrical power system, the pneumatic system was heavier and more cumbersome than the electrical system. One of the larger penalties for the pneumatic system was the weight and envelope required for a pneumatic supply tank and solenoid-operated control valves.

The electrical power system proved to be the optimum automated coupling system. The electrical power system provides the most reliable, lightest weight and least cumbersome coupling and coupling system.

The alternate use of a manual backup system was evaluated. It was determined that the use of dual coupling was a superior method. This method also provides the solution to the problem of how to deal with the possiblity of one inhibit failing closed and the possibility of damage to the single interface area.

The question of how to connect the two coupling halves was evaluated. The coupling halves may either be indivicually latched with mechanisms, such as bayonets, fingers, balls, etc., or the couplings may be connected by means of carrier plates. The use of carrier plates greatly simplifies the design and operation of the coupling and is inherently more reliable. This method is also more compatible with anticipated general future space usages.

FUTURE PLANS

This coupling is the first of a series of couplings which must be developed in order to work with fluids in space. With a few modifications, this coupling will be able to be used to resupply a number of fluids - monomethalhydrazine, ammonia, water, nitrogen tetroxide (the seals need to be changed from EPR to Kalrez because EPR is not compatible with NTO), and many other fluids which are or will be used on orbit. As these couplings are developed and systems are designed to be refueled, the interfaces between the refueling system and the satellite have to be standardized. This is especially true when automation of the refueling operation is considered.

The next coupling to be developed is one to resupply high pressure gases, such as helium and nitrogen. Both of these are used in pressure regulated propulsion systems. The gas coupling will not have the strenuous requirements to maintain three seals between the fluid and the payload bay because the fluids are not considered hazardous. The redesign of an existing ground servicing coupling to make it compatible with an astronaut in an EVA suit could be all that is necessary.

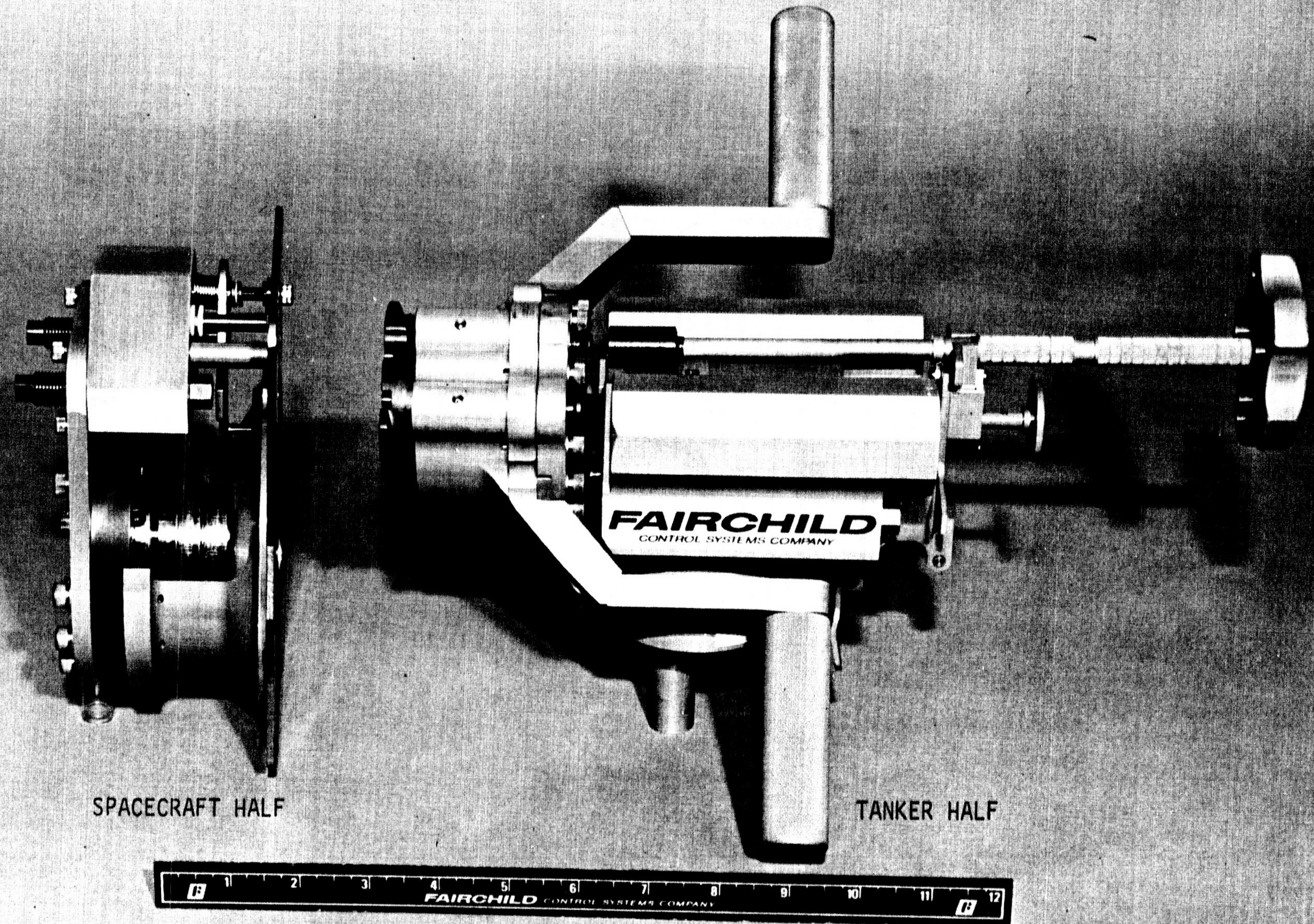
In the near future there is going to be a need for a certified cryogenic coupling. This coupling may not have all of the rigorous safety requirements of the hydrazine coupling, but, especially for hydrogen, the hazards associated with working with cryogens have not been extensively studied and identified. Another point which should be stressed,is that starting with this coupling, we believe that all couplings should be automatic. Therefore, considerable amount of work must be performed to standardize the refueling interfaces. With this automation, the requirements which make this coupling so complex (i.e. the interlocks and the necessity to verify the interface seals and the inhibits) will no longer be required, because the hazard associated with contact with propellant will be reduced.

In summary, this coupling is the first in a series of couplings to be developed to safely work with hazardous fluids in space. It seems complex because of the variety of requirements which drive its deisgn. The coupling will be rigorously tested to verify that it will meet all of these requirements, and, in the Spring of 1988, the first half of the Standardized Refueling Coupling will be launched as part of the Gamma Ray Observatory. Then, in the summer of 1990, the coupling will be connected and the first refueling of a functional satellite will take place.

OPERATIONS CONDITIONS AND REQUIREMENTS

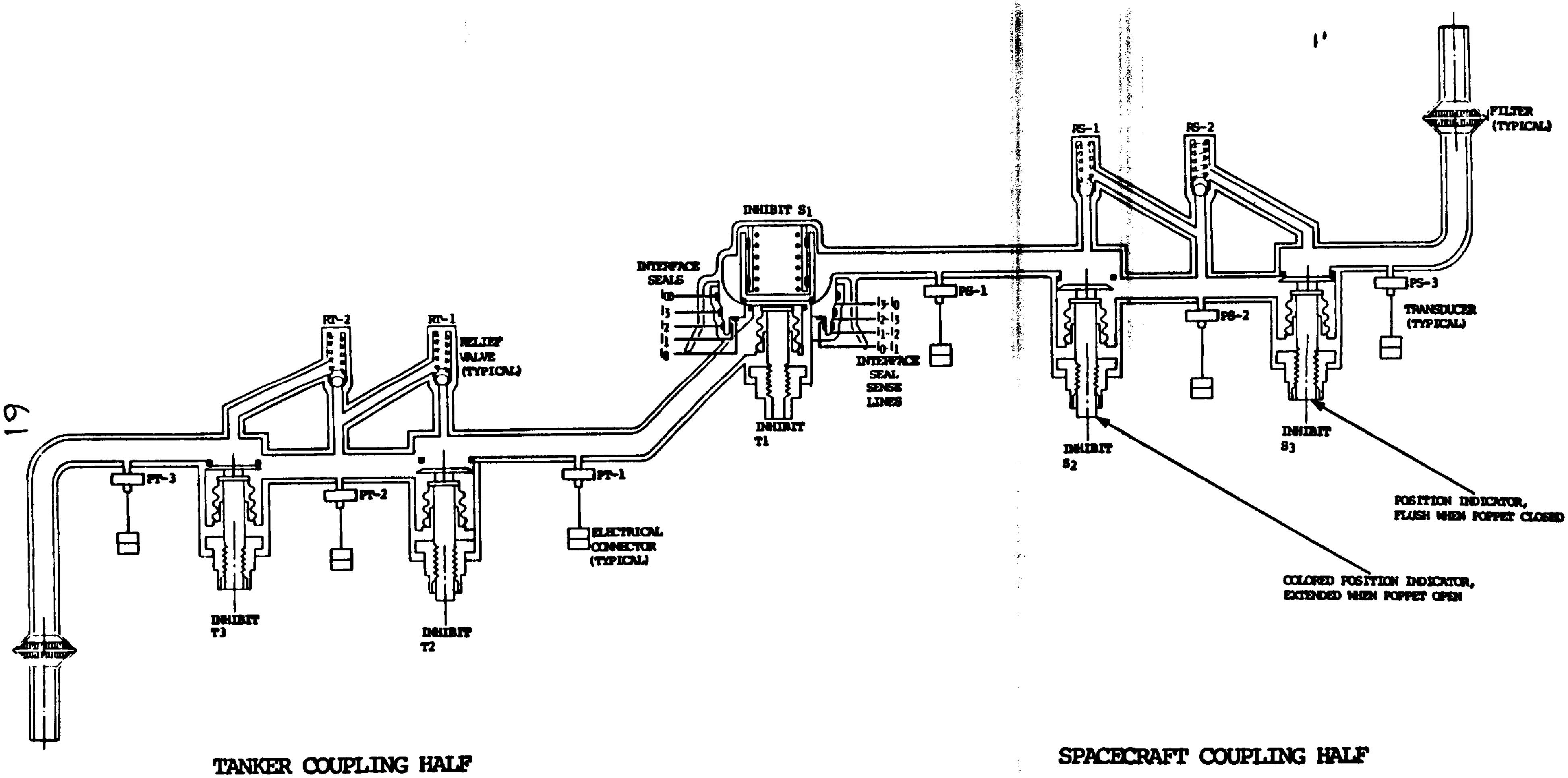
PARAMETER	UNITS	MIN	NOM	MAX	REMARKS
Operating Pressure	(psig)	0	500	600	-
Proof Pressure	(psig)	3000	-	-	-
Burst Pressure	(psig)	5000	-	-	-
Surge Pressure	(psig)	-	-	2000	Spike 20 ms. Duration
Pressure Drop-Normal & Reverse Flow Direction	(psig)	-	-	50	At Maximum Flow Rate
Operating Temperature	°F	40	70	120	-
Flowrates	(gpm)	0	10	20	-
External Helium Leakage	(SCCS)	-	-	1.4×10^{-4}	-
Internal Helium Leakage	(SCCS)	-	-	1.4×10^{-4}	-

Table 1



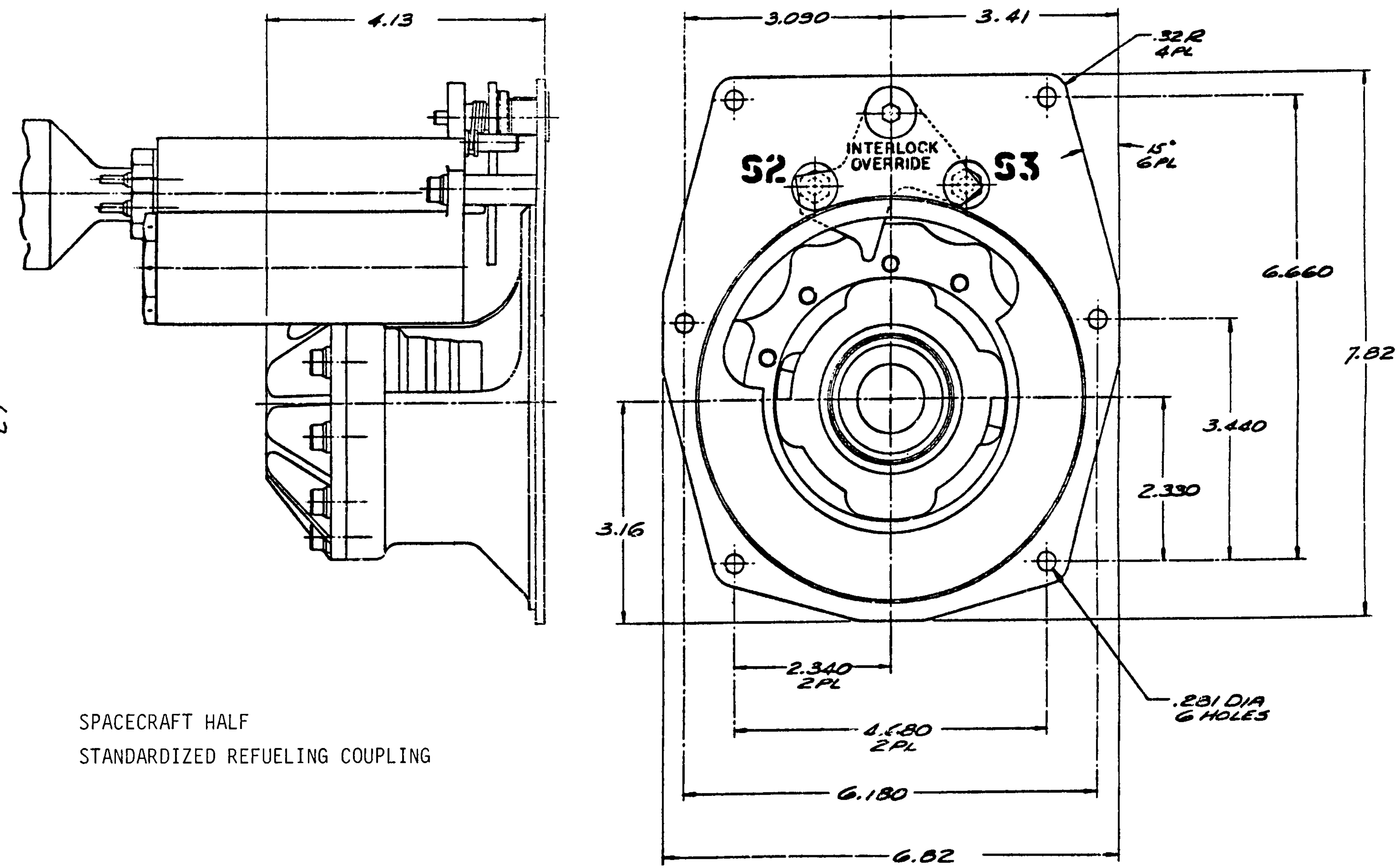
STANDARDIZED REFUELING COUPLING WETF MOCK-UP

FIGURE 1



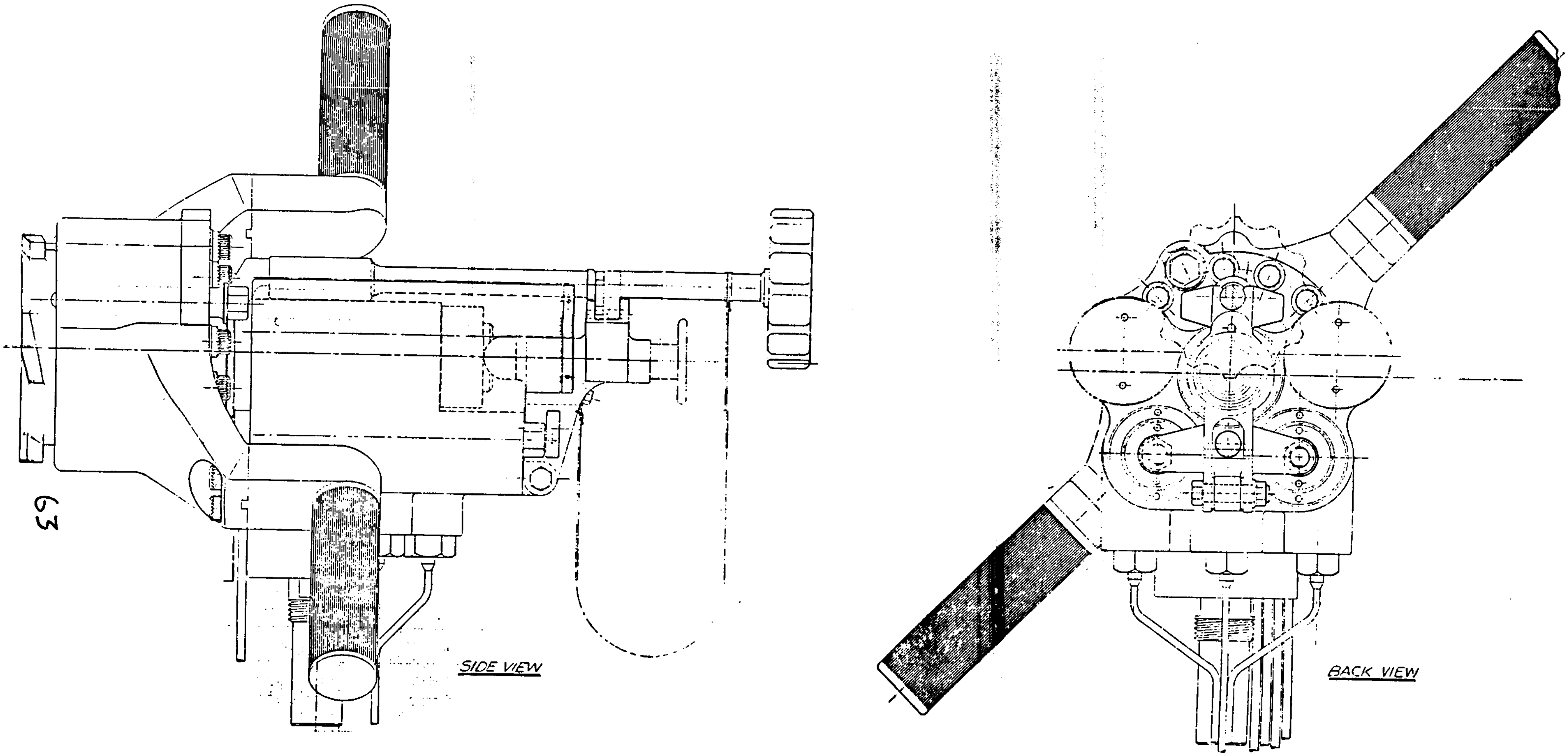
STANDARDIZED REFUELING COUPLING SCHEMATIC

FIGURE 2



SPACECRAFT HALF
STANDARDIZED REFUELING COUPLING

FIGURE 3



TANKER HALF - STANDARDIZED REFUELING COUPLING

FIGURE 4

REMOTELY OPERATED ELECTRICAL UMBILICAL

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ABSTRACT

The Remotely Operated Electrical Umbilical (ROEU) is designed to meet the needs of deployable payloads which require electrical power and/or instrument monitoring while latched into the Shuttle Orbiter Payload Bay. At the present time the only electrical interface available to deployable payloads is a solenoid releaseable disconnect that is not remateable. This device has been flown successfully on several oribter missions but is limited in size (19 - 12 gauge or 128 - 20 gauge pins) and does not provide for remating after initial separation. The ROEU is designed to be remotely operated from the orbiter flight deck in both mate and demate modes of operation. The mechanism provides adequate compliance in all axes to accommodate the maximum relative motions between the Shuttle Orbiter and Payload under all flight conditions. The connectors in the ROEU will accommodate 270 pins which includes 12 - 8 gauge, 6 - 12 gauge with the remainder being 20 gauge pins. This is approximately one quarter of the standard mixed cargo harness (SMCH).

Electrical Power. The ROEU latch drive and latch actuator assemblies will use standard orbiter 115 volt, 400-cycle, AC power. This power source, which is used for all orbiter deployable payload latches, is available at numerous outlets on the longeron sill. Control of the AC power as applied to the payload latches or the ROEU is from the orbiter aft flight deck via the payload deployment (A6A1) panel or a dedicated control panel. Control and indicator circuits are 28 volt DC power.

Electrical Harness Provisions. Electrical harnesses between the longeron AC power connectors and activation devices like latch or ROEU motors are available in flight kit inventories. Harnesses between standard interfaces like a standard interface panel (SIP) can be either standard interface hardware or payload unique. Harness selection will be a function of the ROEU to orbiter interface definition.

Umbilical Mounting Provisions. The core ICD does not provide for any structural interfaces between the orbiter and the payload or primary structure; however, interfaces are available on orbiter sidewall carriers like the payload bridge beam, adaptive payload carrier (APC), extended adaptive payload carrier (EAPC), or the getaway special (GAS) beam. Restrictions apply to these carriers with regard to weight carrying capability, center-of-gravity location and mounting provisions for the attachment of payload integration hardware. Mounting of the ROEU on a payload bridge beam will be the standard configuration so as to provide the most universal attachment with other mountings available by contract negotiation.

Worst Case Relative Misalignments and Displacements. The ROEU will be capable of accommodating the maximum relative misalignments and displacements between the orbiter and payload caused by the thermal and loading conditions. Maximum displacements identified in the core ICD are +3.00, -2.48 inches in the Yo direction and a 6-degree half-cone deflection angle about the Yo axis. Plus or minus 6-degrees of roll deflection about the Yo axis will also be accommodated by the ROEU.

Maximum Intrusion into Payload Envelope. Intrusion of the ROEU umbilical mechanism into the 180-inch diameter dynamic envelope will not occur except at the Xo location of the interface units. A relief in the payload volume will have to be provided at the 84.1 Yo plane, extending below the orbiter interface unit, to provide clearance with the payload during deployment operations. This is depicted in Figures 9 and 10.

STS Redundancy/EVA Requirements. The STS requires payload and payload integration hardware to meet the safety requirements imposed by NHB 1700.7A. This may be different from mission success requirements desired by the payload developer. Guidelines for Extravehicular Activity (EVA) compatable design are documented in CER 76034 and will be interpreted for inclusion in the Systems Requirement Document (SRD) by a member of Rockwell's EVA Design Group.

Umbilical Systems Design

In addition to meeting the orbiter interface requirements of the core ICD (ICD 2-19001), the ROEU is designed to meet the following functional and physical requirements:

1. The ROEU is designed for repeated mate and demate operations.

2. Loads transferred between the orbiter and payload through the ROEU are limited to the centering spring forces.

3. The active portion of the ROEU is on the orbiter side and the passive part is on the Payload side and designed to minimize the weight impact to the payload. The payload disconnect assembly (PDA) does incorporate the Xo compliance feature.

4. The ROEU is capable of accommodating the maximum relative displacements and misalignments between the orbiter and the payload which have been identified.

5. Control of the ROEU will be from a control panel in the aft flight deck.

6. The ROEU will operate with standard orbiter electrical power.

7. Redundant power sources, control and status functions are provided.

8. The ROEU does allow back-up operation for disconnect and reconnect by a crew member during EVA operations.

9. The ROEU is designed to operate under the thermal conditions which may occur during payload deployment and retrieval missions. As a minimum, the mechanism will operate at plus 250 to minus 70 degrees F. Also, the mechanism will operate with a temperature differential of 300 degrees F between units.

10. The ROEU is designed for loads and vibration conditions valid for the longeron area of the orbiter as defined by the core ICD.

SYSTEM OVERVIEW

The ROEU mechanism consists of two separate assemblies. One mechanism which mounts on the payload and contains the electrical receptacle (pins) with the other mechanism, an arm linkage, which mounts to the standard longeron bridge fittings and contains the electrical plug (sockets). This configuration was selected so as to minimize the intrusion into the payload volume (dynamic envelope) and provide accessibility during ground operation or extravehicular activity. The stowed configuration utilizes the same orbiter cross-sectional volume along the upper surface of the longeron as the active payload latches. Attachment to the standard longeron bridge fittings provides incremental positioning along the orbiter payload bay without the requirement for special support structures. Each payload configuration (desired Xo placement) will require a selection of mounting for the payload disconnect assembly which will position the orbiter mechanism in a location which does not conflict with payload latches, payload-bay door mechanisms, electrical wire cross-over linkages or radiator connections. The manipulator positioning mechanisms occupy some of the available cross-sectional volume if the Remote Manipulator System is installed on the same side of the orbiter as the ROEU. The relative large predicted Yo displacements of the Orbiter longeron relative to the payloads, +3.00 or -2.48 inches, helped establish the Yo axis as the most desirable motion direction for mate and demate operation of the electrical connectors. By utilizing a parallel arm drive system to position the orbiter disconnect to engage the payload disconnect, the best overall utilization of available space is achieved while providing the necessary compliance for system operation.

SYSTEM DETAIL

Payload Disconnect Assembly (PDA) consists of a mounting base fitting, bearings with support fittings, receptacle support, guide fitting and the electrical receptacle with its pigtail. The guide fitting provides initial alignment with the orbiter disconnect assembly by means of four conical petals spaced 90 degrees apart. These guide petals are designed to interleave with similar guide petals on the orbiter disconnect. The guide fitting, receptacle, support fitting and bearing support fittings form an assembly which can translate plus or minus 1.60 inches along the Xo axis. Centering springs are incorporated to keep the receptacle/guide assembly at the nominal center-travel position prior to contact with the mating unit. The electrical pigtail is formed into a 90 degree service loop and clamped to an extension of the base fitting to assure adequate Xo compliance travel. The guide fitting has two integral lugs located at the Zo plane, between the two pair of guide petals, which will interface with the latch hooks on the Orbiter disconnect assembly.

Orbiter Disconnect/Arm Mechanism (ODM) consists of two major mechanical assemblies and an electrical panel to provide mounting for 15 interface connectors plus four motor control connectors. The Orbiter Disconnect Assembly (ODA) is bolt mounted on the swing end of the arm drive mechanism. The base fitting of the arm drive mechanism provides the interface for attachment of the assembly onto the longeron bridge fitting. The electrical conductors, from the orbiter plug, are segregated into ten individual bundles (.56 dia. approx.) and clamped at discrete locations to optimize the operational forces and cable routing along the arm to the connector panel. (REF. Figure 1).

Orbiter Disconnect Assembly (ODA) consists of a guide housing, latching mechanism, EVA disconnect device, limit switch module, electric motor driven rotary actuator, EVA disconnect access cover and electrical plug with a pigtail that is terminated in bulkhead mount connectors. These parts are supported on a spherical bearing which has an angular compliance of 6 degrees half-cone angle and plus or minus 6 degrees roll about the centerline of the connector (Yo axis). An interface support bracket is bolted to the inner part of the spherical bearing to provide mounting at the end of the drive arm assembly. Centering spring devices are incorporated into the spherical bearing race to position the guide housing assembly at its nominal center of angular travel. The guide housing has four integral conical petals spaced 90 degrees apart (centered on Xo and Zo planes) which interleave with similar petals on the payload disconnect assembly. These guides provide initial alignment of the electrical connectors at the payload/orbiter interface during the mating operation. Two opposing hooks (latches) are operated through slots in the two horizontal guides (in Zo plane). These latches have an extended reach of approximately .70 inch with straight-in pull after engagement with the lugs on the payload disconnect assembly. The latching system is driven by means of a rotary actuator, with redundant electrical motors, through a common bellcrank attached to each hook's drive rod system. An interfacing ball-lock device, between the rotary actuator and the drive-gear bellcrank, provides the means for disconnecting the actuator and driving the latch system manually during an emergency.

The ball-lock engages an inline spur gear/bellcrank which is meshed with an off-set spur gear that provides a wrench input (7/16 external, 1/4 internal hex) for EVA driving of the system. A hinged cover over the EVA drive shaft is connected to the ball-lock trigger so that when the cover is opened the rotary actuator is disengaged from the system. The cover also provides a positive position-lock for the holding ratchet which is required for manual operation of the system. A cam on the bellcrank drive gear actuates the redundant sets of limit switches which control the actuator travel and provide talk-back indication. (REF. Figure 2).

The hook (latch) system is a key element of this design in that it provides the forces required to fully engage and disengage the electrical contacts in the connectors. The hook mechanism consists of a pair of opposing hooks with a pin attachment to their respective sliding pistons. The pistons contain springs which apply a force to the pins which attach the hooks to their actuating links. This provides a moment between each hook and piston tending to "engage" the hooks. The pistons have an extension at one end which provides the means of extracting the electrical pins during the disconnect cycle. The hook actuating the electrical links are connected to drive bellcranks which are driven by drive rods connected to a common bellcrank. This bellcrank is driven by a redundant motor rotary actuator. In the fully latched position the drive rods, attached to the two hook-drive bellcranks, are in an over-center position relative to the actuator output bellcrank. The housing guides which support the hooks and pistons are slotted to accommodate the hook kinematics and allow the piston extensions to contact the mating lugs on the payload disconnect assembly. Within each slot is mounted a guide roller which controls the hook kinematics if the springs within the pistons were to fail or there are opposing forces being applied from the payload side of the system. The latch system is depicted in different kinematic positions in Figures 5 through 8.

Arm Drive Mechanism (ADM) consists of two parallel arms (one above the other in Zo planes) which are interconnected at their "free-end" by horizontal (Yo axis) hinge pins in a fitting which connects to the end-plate with a vertical (Zo axis) hinge pin. The drive end of the parallel arms are connected to a drive bellcrank by means of horizontal (Yo axis) hinge pins with the bellcrank providing the vertical (Zo axis) pivot at the base support fitting. The end-plate is also interconnected to the base fitting by means of a rod parallel to the vertical pivots of the two drive arms. In the stowed position the arms run near parallel to the Xo axis above the longeron. The drive arms are held in a nominal Zo position by means of a double-acting spring cartridge mounted to the drive bellcrank. A load-control spring cartridge attaches the arm-drive bellcrank to an intermediate crank which in turn is attached to the rotary actuator bellcrank by means of a drive rod. In the stowed position the drive rod is at an over-center position relative to the actuator bellcrank with the spring cartridge solid in the tension direction. The drive arms are preloaded into a vee-block mounted to an extension on the base fitting. During the arm drive mode (engagement) the spring cartridge controls the interface forces between the Orbiter disconnect assembly and payload disconnect assembly. Sufficient freeplay is incorporated into the spring cartridge so that when the units are mated and the arm-drive actuator is in an intermediate position the spring does not impose any force onto the payload disconnect assembly. An EVA drive mechanism is incorporated into the actuator drive system which is identical to the one described for the Orbiter disconnect assembly except for the nomenclature engraved on the handle and support plate. (REF. Figure 1).

SYSTEM OPERATION

With the payload positioned and latched into the orbiter and the ROEU arm mechanism installed at the proper location on a longeron bridge beam, arm stowed, latches open, a normal operational sequence would be as follows:

Mating Sequence would start by the actuation of a switch that powers the arm drive actuator in the "mate" direction. The initial actuator rotation moves the bellcrank/link off-center and starts rotation of the intermediate bellcrank. The arm mechanism, with the orbiter disconnect assembly attached, does not necessarily move until the freeplay in the spring cartridge (bungee) is taken up by the bellcrank travel. The arm mechanism is then driven inboard from its stowed position above the longeron until the guide petals of the orbiter disconnect and the payload disconnect assemblies come into contact with each other. The arm drive mechanism continues to be driven inboard against the preload of the bungee until the guide surfaces of the units have interleaved by forcing compression of the appropriate centering springs. This positional compliance allows the lugs on the payload disconnect assembly to come into contact with the extensions on the pistons in the orbiter disconnect assembly. The actuator continues to drive and compress the bungee until the "mate" limit switch actuates at full travel of the intermediate bellcrank. The electrical connector pins do not contact the mating socket at this point of operation.

The second operation for mating is to switch power to the latch drive actuator, which is part of the orbiter disconnect assembly, in the "latch" direction. The opposing latch hooks rotate inward until contact is made with the payload disconnect lugs and then the units are pulled into full engagement. The actuator continues to rotate until the drive bellcrank and link are over-center and the "latched" limit switch is actuated.

The third and final operation to complete the "mate sequence" is to apply power to the arm drive actuator in the "stow" direction. This will drive the intermediate bellcrank in the direction to completely relax the force on the spring cartridge and actuate an intermediate limit switch to stop the actuator. This position is at the midtravel of the bungee free-play that is provided so that no Yo-axis spring force is imposed on the payload at its maximum Yo excursions. (REF. Figure 4).

Demate Sequence would be started with the ROEU mechanism in the final mate position described in the "mate sequence" above. The first operation would be to actuate a switch that powers the latch drive actuator in the "unlatch" direction. The initial rotation of the actuator moves the drive bellcrank and links off-center and starts motion of the piston assemblies toward the lugs on the payload disconnect lugs. As the actuator continues its rotation the extensions on the pistons apply the necessary separating force against the payload disconnect until pin separation is complete. Continued actuator rotation moves the hooks back cam surfaces past the restraining roller so that when the pistons bottom in their respective bores the hooks move outward. When the actuator has rotated the hooks clear of the guide petal surfaces a limit switch is actuated to stop actuator rotation.

The second and final operation for "demate" is to switch power to the arm drive actuator in the "stow" direction. As the actuator rotates the intermediate bellcrank toward its stow position the spring cartridge is extended until all freeplay is removed. At this point the arm mechanism begins rotating toward its stowed position and the disconnect assemblies are pulled free. The actuator continues to rotate the bellcrank until the arm mechanism is preloaded into a vee-notch support on the main mechanism support fitting. As the drive bellcrank moves over-center with the drive link a limit switch is actuated to stop actuator rotation. The payload disconnect and orbiter disconnect assemblies are now separated a sufficient distance, 3.20 inches nominal, to allow payload deployment operations.

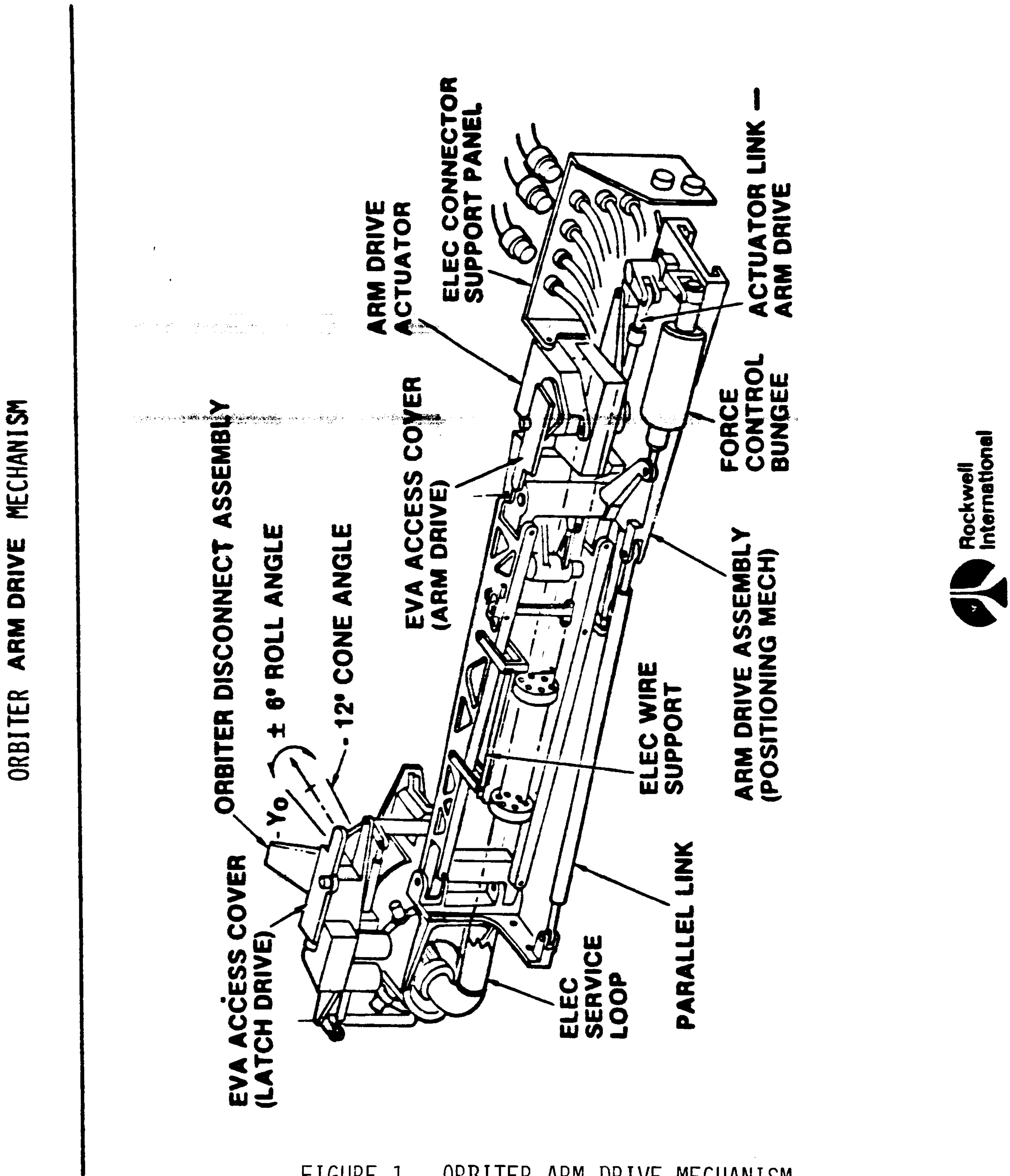

FIGURE 1. ORBITER ARM DRIVE MECHANISM

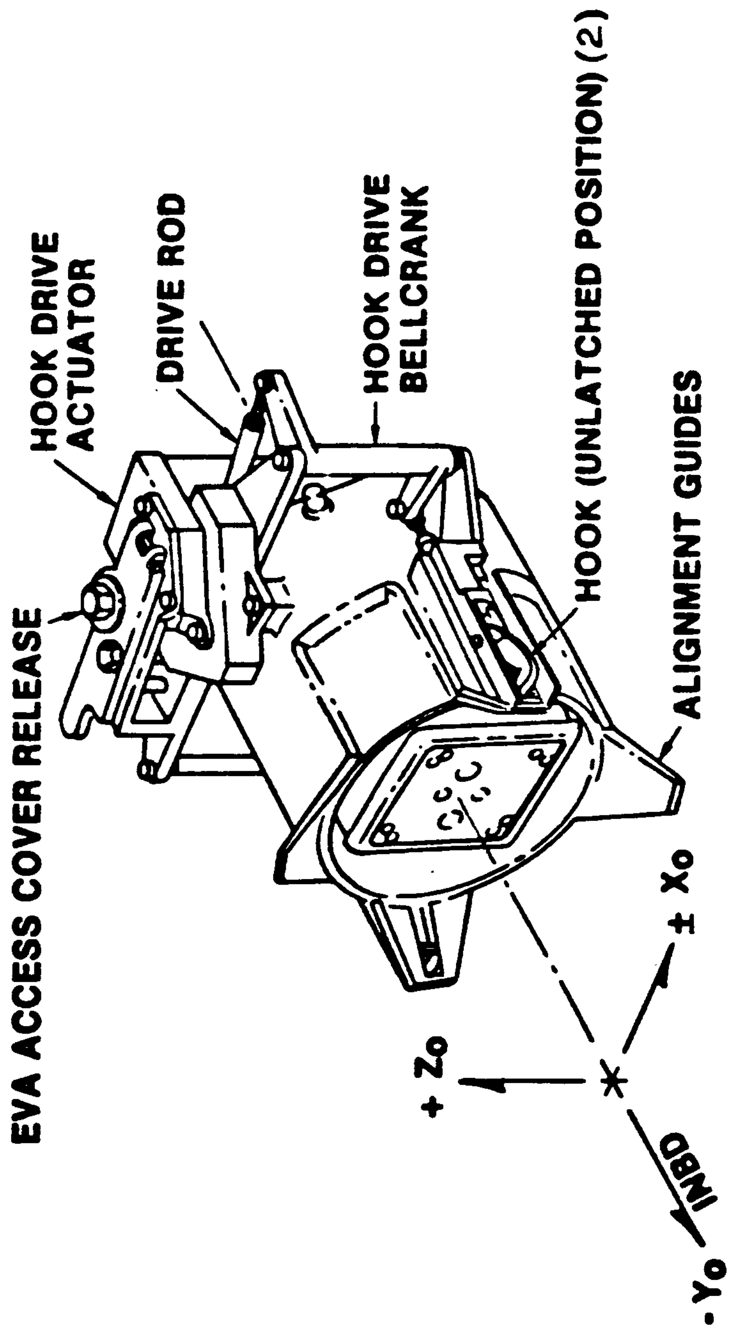

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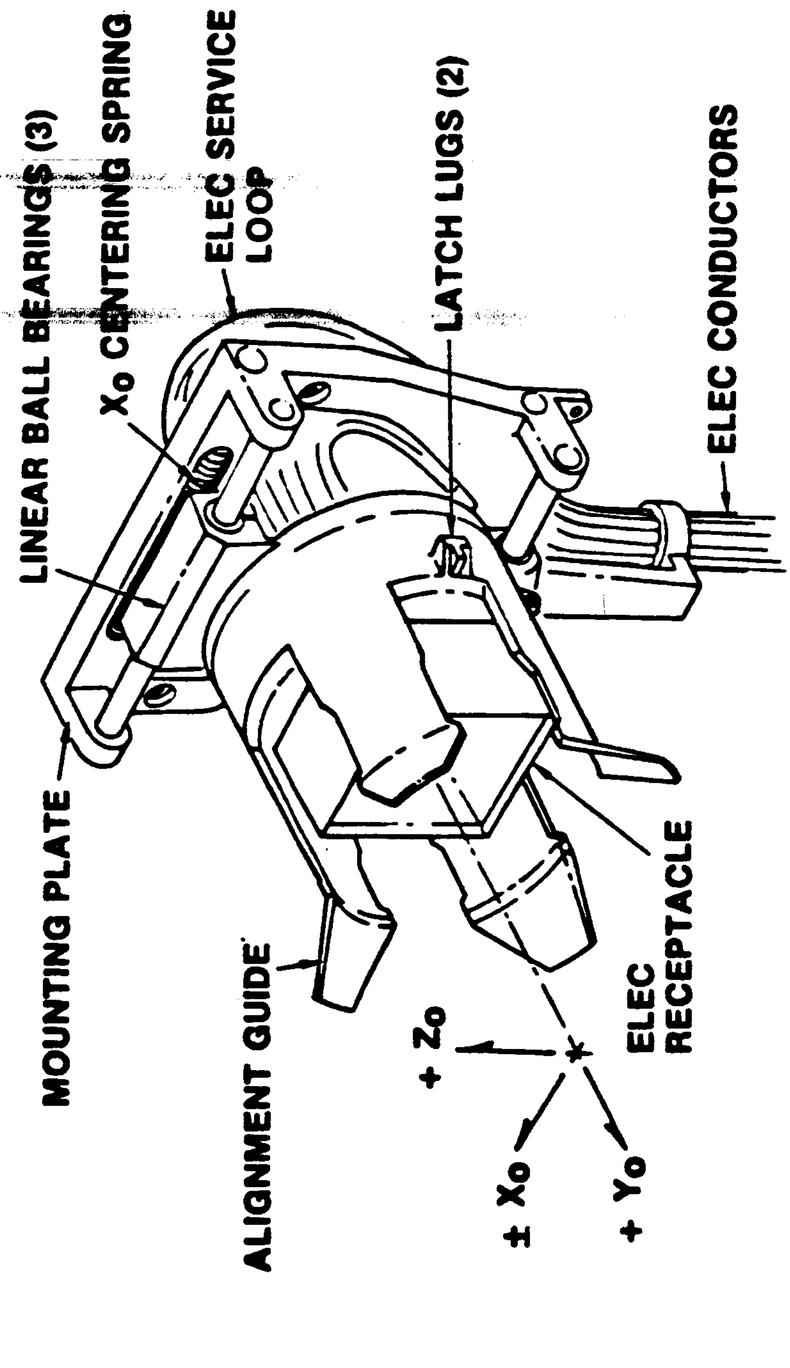
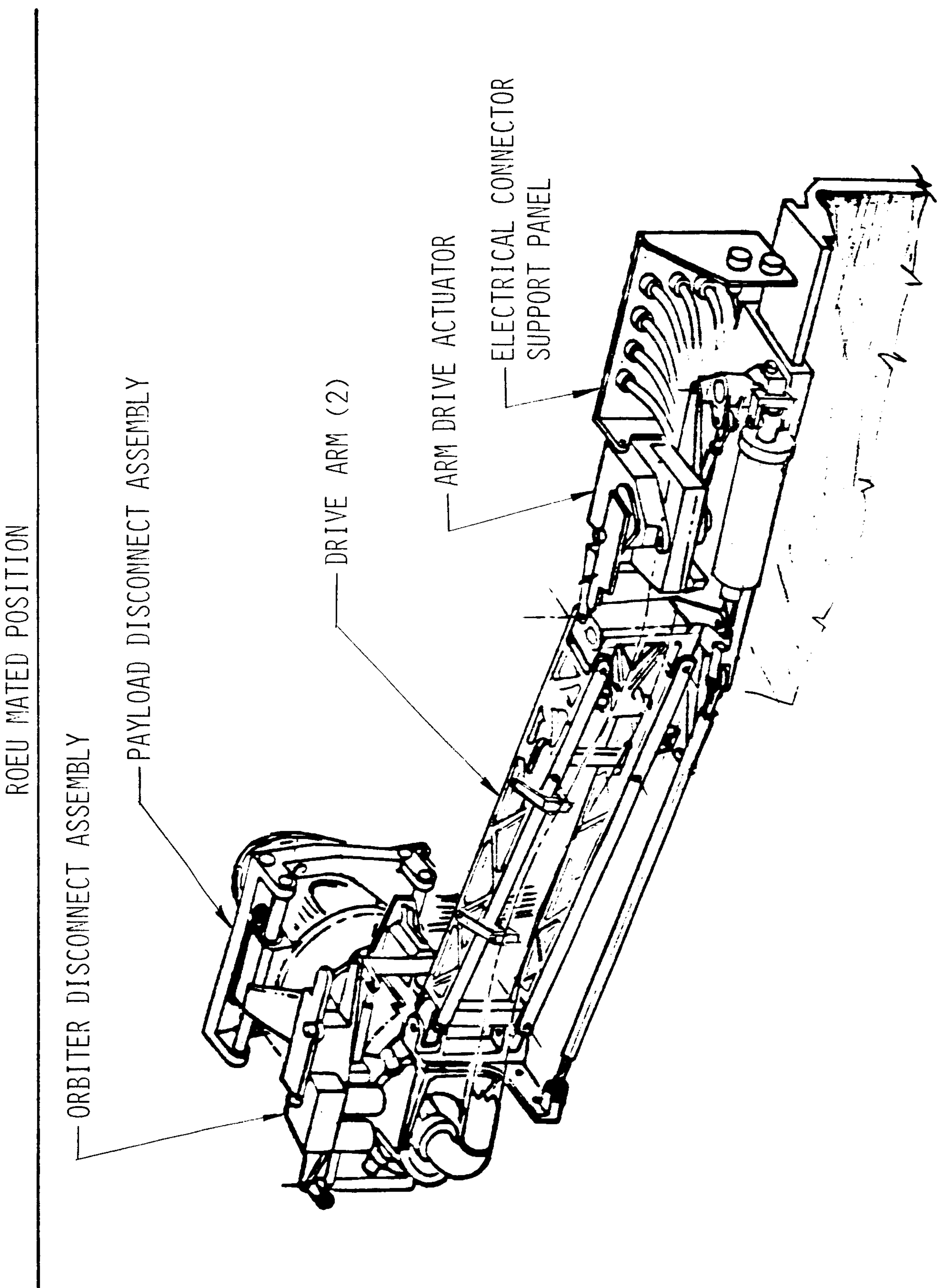
FIGURE 2. ORBITER DISCONNECT ASSEMBLY

PAYLOAD DISCONNECT ASSEMBLY



Rockwell International

FIGURE 3. PAYLOAD DISCONNECT ASSEMBLY



F. Holman 11 Sep 85

FIGURE 4. ROEU MATED POSITION

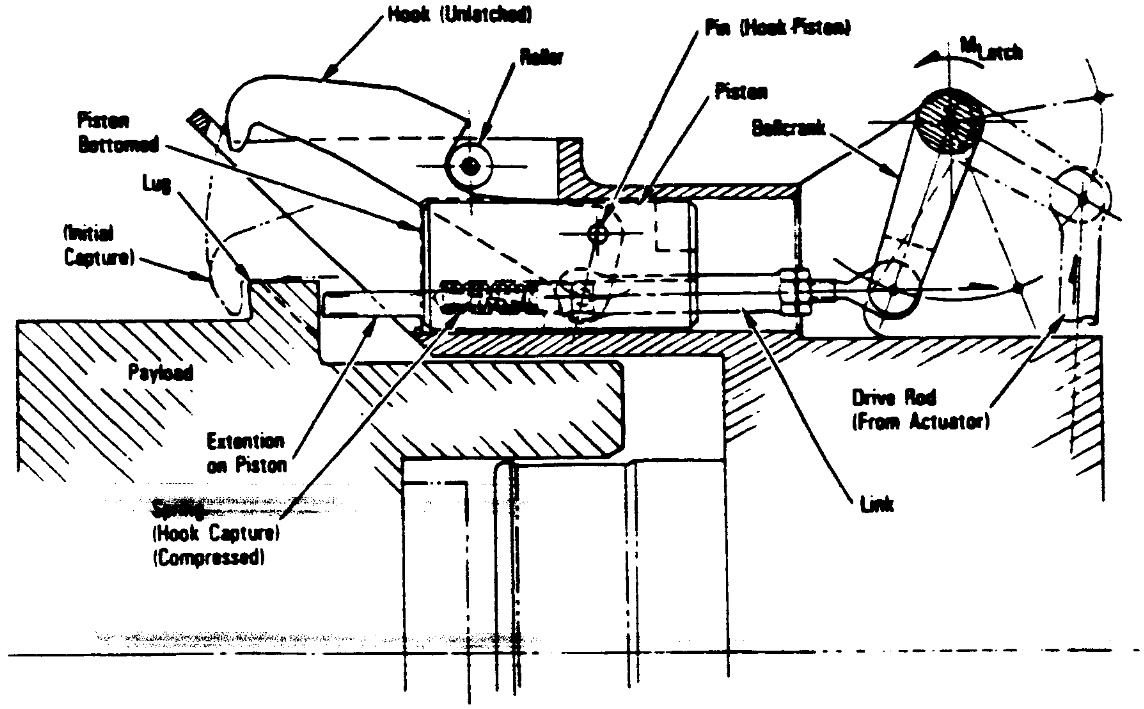


Figure 5. —— Ready to Latch Sequence

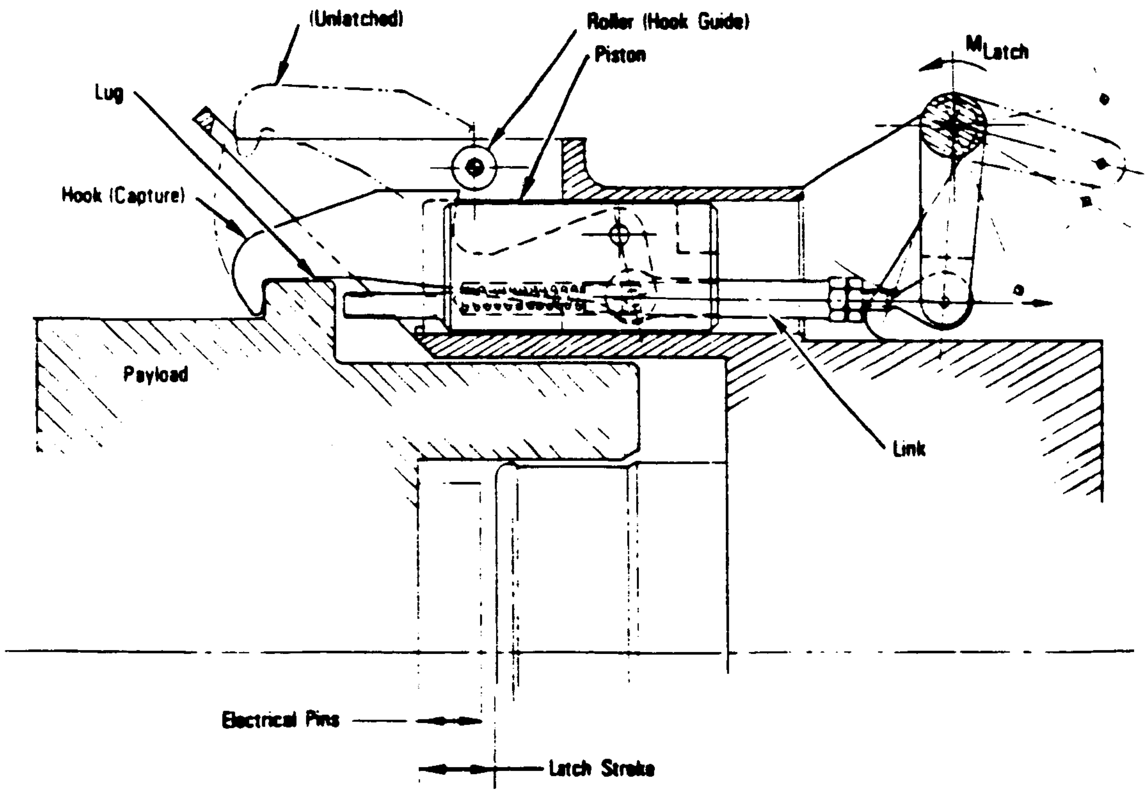


Figure 6. -·-·-— Hook Capture

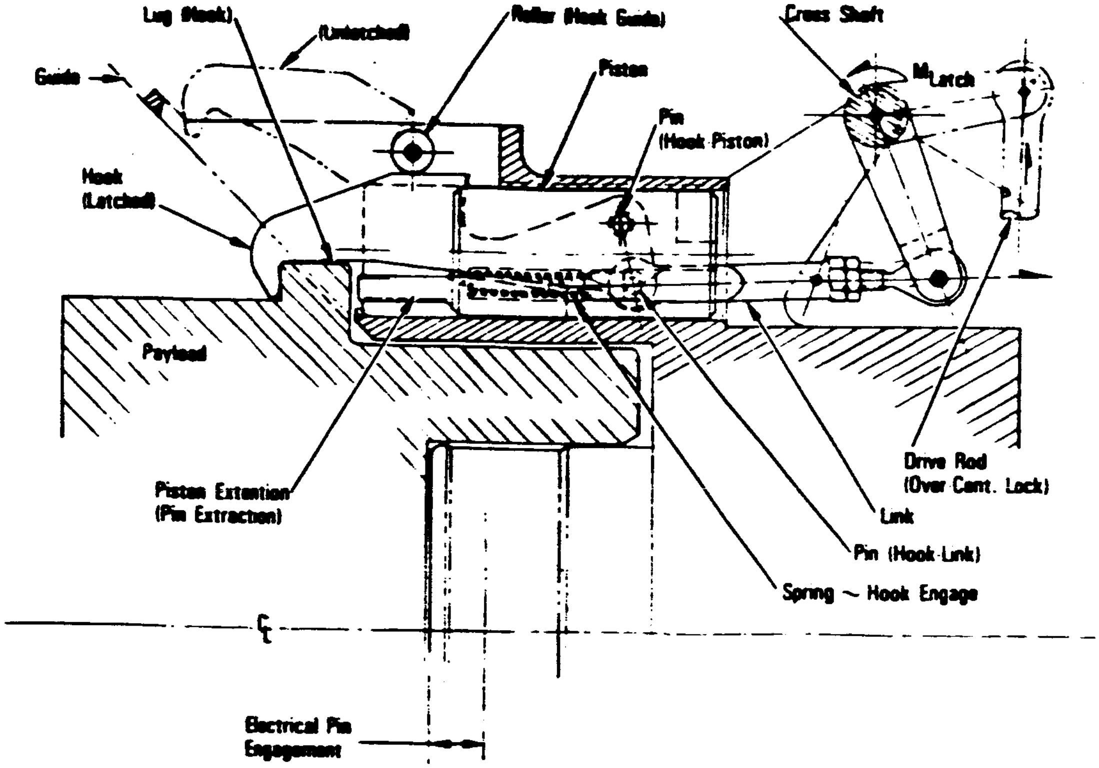


Figure 7. —— Latched Position

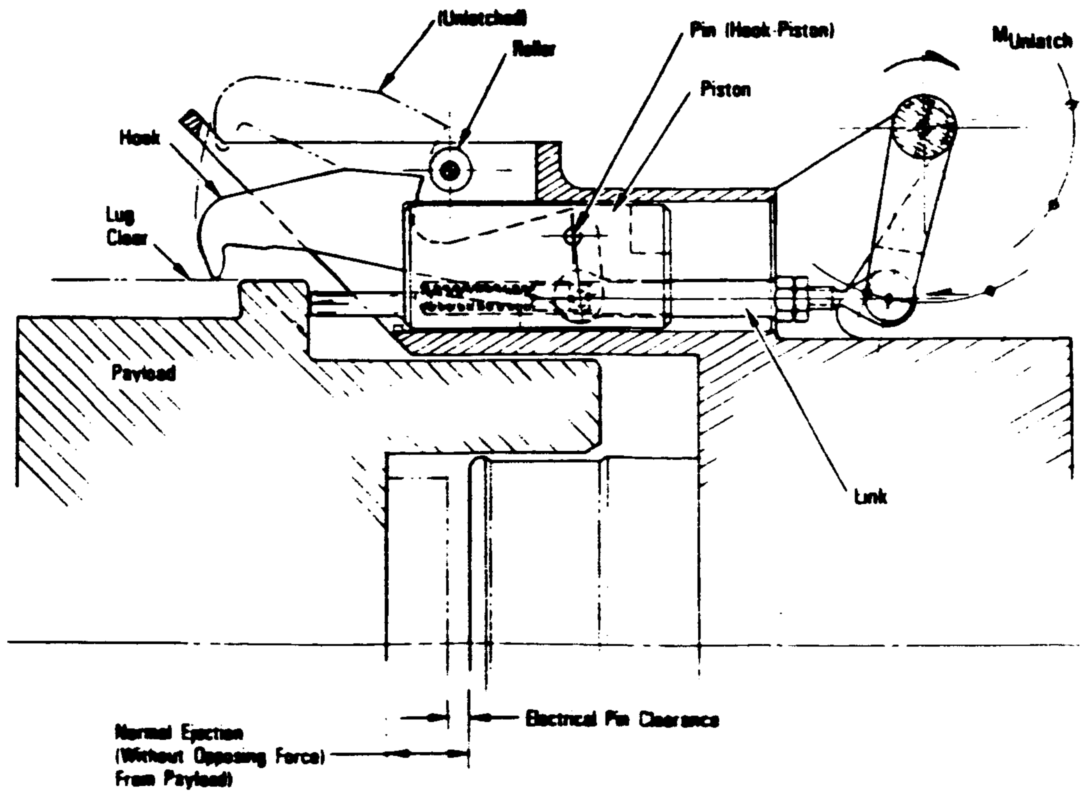


Figure 8. —— Unlatch Sequence

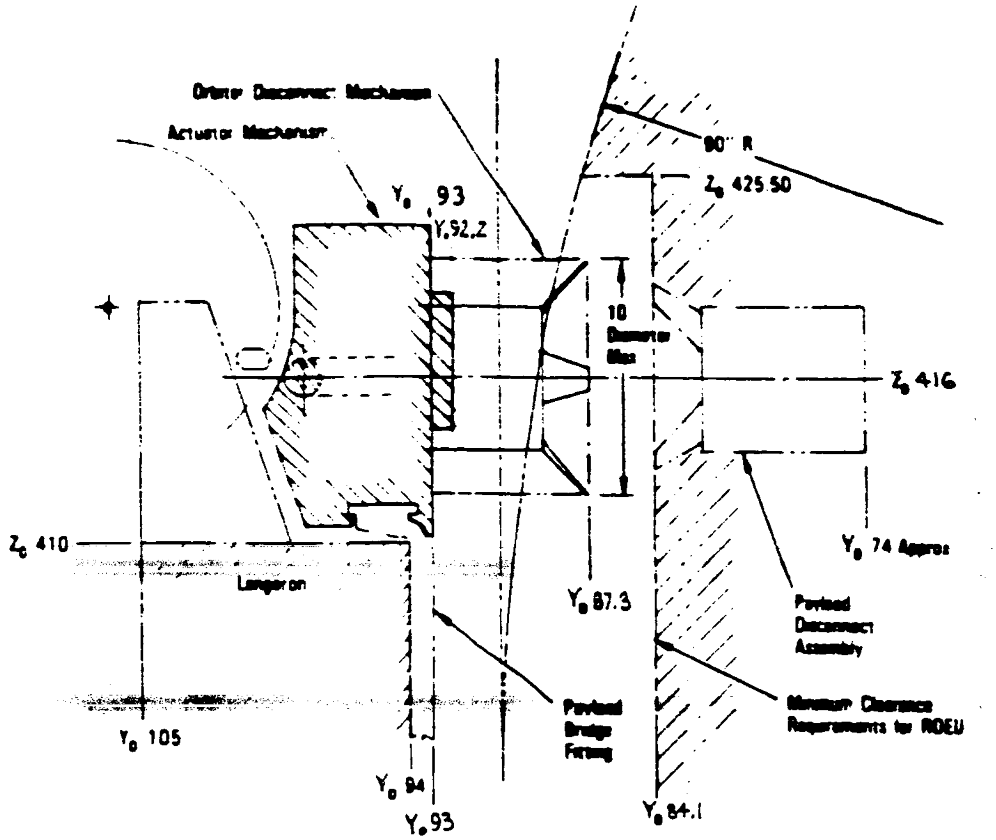


Figure 9. ROEU ENVELOPE

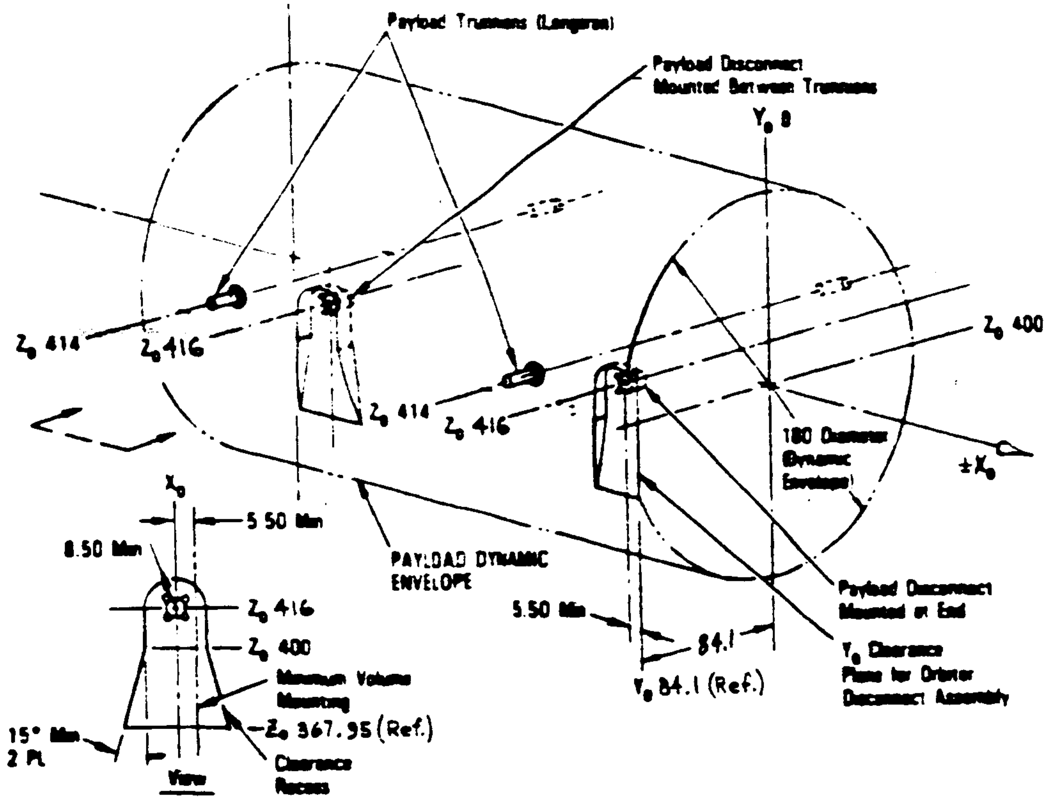


FIGURE 10. ROEU/PAYLOAD DYNAMIC ENVELOPE

EVA TOOLS

SATELLITE SERVICING ROLE

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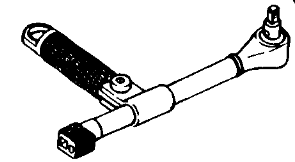
NASA JOHNSON SPACE CENTER

NOVEMBER 6-8, 1985



EVA TOOLS - SATELLITE SERVICING ROLE
TYPICAL TOOL DEVELOPMENT

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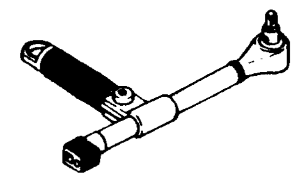
ILC SPACE SYSTEMS

EVA TOOLS - SATELLITE SERVICING ROLE
TYPICAL TOOL DEVELOPMENT

MICHAEL WITHEY



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NOVEMBER 6-8, 1985

ABSTRACT

THE SOLUTION OF A PROBLEM COMES AFTER ITS DEFINITION AND IDENTIFICATION. THIS IS TRUE FOR THE DEVELOPMENT AND MODIFICATION OF EVA TOOLS. BASED ON THE COMPLEXITY OF THE PROBLEM, EVA TOOLS ARE EITHER DEVELOPED OR EXISTING ONES MODIFIED. CONCEPTUAL DEVELOPMENT, DESIGN AND ANALYSIS OF SPECIAL TOOLS FOR USE IN EXTREME ENVIRONMENTS, AND ZERO-G CONDITIONS ARE VERY IMPORTANT. HUMAN FACTORS ENGINEERING IS EFFICIENTLY UTILIZED IN THE DESIGN OF TOOLS SO THAT THE ASTRONAUTS CAN USE THE TOOLS EASILY AND EFFICIENTLY. SPECIAL TOOLS WERE DEVELOPED AND UTILIZED DURING THE RETRIEVAL OF SOLAR MAX, PALAPA, WESTAR AND LEASAT. IN ORDER TO DEVELOP A CONCEPT OF A TOOL FOR A PARTICULAR MISSION, A CLEAR UNDERSTANDING OF THE SATELLITE'S MECHANICAL AND STRUCTURAL DESIGN IS OF UTMOST IMPORTANCE. HENCE, A DETAILED DOCUMENTATION OF THE SATELLITE'S DESIGN DRAWINGS AND RELATED DOCUMENTATION SHOULD BE KEPT AND MADE AVAILABLE TO THE ENGINEERING PERSONNEL. POST-MANUFACTURING ADDITIONS AND MODIFICATIONS ALSO SHOULD BE WELL DOCUMENTED AND MADE AVAILABLE. A SMALL OMISSION OF AN IMPORTANT ITEM CAN BE VERY COSTLY CAUSING A MISSION FAILURE.

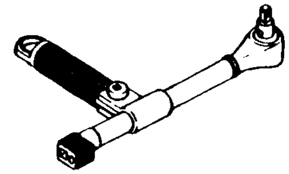
ILC SPACE SYSTEMS

EVA TOOLS - SATELLITE SERVICING ROLE
TYPICAL TOOL DEVELOPMENT

MICHAEL WITHEY

SATELLITE SERVICES WORKSHOP II
NASA JOHNSON SPACE CENTER

NOVEMBER 6-8, 1985



INTRODUCTION

- WHEN SATELLITE SERVICING IS REQUIRED, SPECIAL TOOLS ARE REQUIRED.
- SUCCESS OF THE SERVICING DEPENDS ON PROPER TOOL DESIGN AND DEVELOPMENT.

PROBLEM DEFINITION

- IDENTIFICATION OF THE PROBLEM.
- NATURE AND COMPLEXITY OF THE PROBLEM.

NECESSARY DOCUMENTATION

- MECHANICAL, ELECTRICAL AND STRUCTURAL DESIGN DETAIL OF THE SATELLITE.
- DOCUMENTATION ON THE DETAIL DESIGN AND DRAWING OF THE SATELLITE (ICD).
- DOCUMENTATION ON THE POST-MANUFACTURE AND MODIFICATION OF THE SATELLITE.

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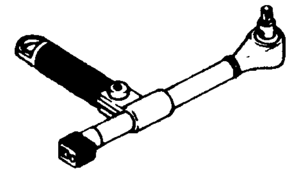

METHODOLOGY OF TOOL DEVELOPMENT

- CONCEPT DEVELOPMENT
- DESIGN ACTIVITY
- ANALYSIS
- FABRICATION
- TESTING
- CERTIFICATION
- DOCUMENTATION

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EVA TOOLS - SATELLITE SERVICING ROLE
TYPICAL TOOL DEVELOPMENT

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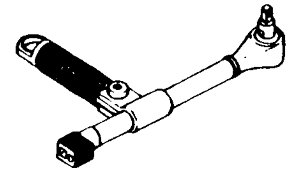


CONCEPT DEVELOPMENT

THE COMPLEXITY OF THE PROBLEM DICTATES THE CONCEPT DEVELOPMENT TO A GREAT DEGREE. THERE MAY BE NUMEROUS POSSIBLE SOLUTIONS TO THE PROBLEM. THE BEST SOLUTIONS WILL BE SELECTED BASED ON THE SIZE, WEIGHT, AND STOWAGE CONSIDERATIONS. IN DEVELOPING THE EVA TOOLS, HUMAN FACTORS ARE CRITICALLY IMPORTANT. THE CREW SHOULD NOT BE FATIGUED FROM THE TOOL USE. THE TOOLS SHOULD BE EASY TO OPERATE.

DESIGN ACTIVITY

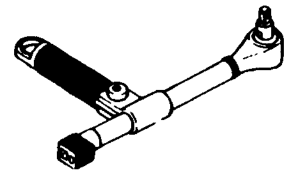
THE MOST IMPORTANT STEP IN DEVELOPING A TOOL IS TO GENERATE SPECIFICATIONS AND DRAWINGS. THE INTERFACE CONTROL DRAWINGS SHOW THE TOLERANCES AND CLEARANCES AROUND WHICH THE TOOL IS BEING DESIGNED. THE FINAL CONFIGURATION OF THE TOOL IS ESTABLISHED THROUGH PRELIMINARY DESIGN REVIEW (PDR) AND CRITICAL DESIGN REVIEW (CDR).

ANALYSES

AFTER THE FINAL CONFIGURATION HAS BEEN ESTABLISHED, STRESS, THERMAL AND FRACTURE ANALYSES SHOULD BE PERFORMED. THE STRESS ANALYSES SHOULD SHOW THAT THE TOOL IS STRUCTURALLY SAFE FOR THE OPERATION FOR WHICH IT HAS BEEN DESIGNED. THE THERMAL ANALYSES SHOULD SHOW THAT THE TOOL IS OPERABLE AT EXTREME THERMAL ENVIRONMENT CONDITIONS. THE FRACTURE ANALYSES SHOULD BE PERFORMED WHEN REQUIRED.

EVA TOOLS - SATELLITE SERVICING ROLE
TYPICAL TOOL DEVELOPMENT

MICHAEL WITHEY



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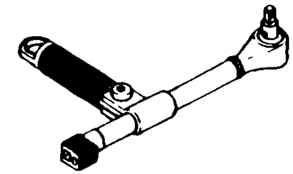
FABRICATION

THE MOCKUP UNIT COULD BE FABRICATED FROM ANY MATERIAL JUST TO SHOW THE PHYSICAL SHAPE OF THE TOOL. THE PROTOTYPE UNIT SHOULD SHOW THE ACTUAL CONFIGURATION AND FUNCTION OF THE TOOL. THE WEIGHTLESS ENVIRONMENT TEST FACILITY (WETF) UNIT IS THE TRAINING UNIT. THE CERTIFICATION UNIT IS FUNCTIONAL AND THIS IS THE UNIT THAT SHOULD BE SUBJECTED TO THE REQUIRED TESTING TO PROVE FLIGHT WORTHINESS. THE FLIGHT UNIT IS THE SAME AS THE CERTIFICATION UNIT AND THIS IS NOT SUBJECTED TO THE CERTIFICATION TESTINGS.

TESTING

TESTING SHOULD BE CONDUCTED TO DETERMINE THAT A TOOL IS CAPABLE OF MEETING PERFORMANCE REQUIREMENTS TO MAKE IT FLIGHT WORTHY. THE VIBRATION TESTS AND THE THERMAL TESTS VERIFY THE ABILITY OF THE TOOL TO WITHSTAND A DEFINED VIBRATION TEST SPECTRUM AND THERMAL TEST SPECTRUM, RESPECTIVELY. THE THERMAL/VACUUM TEST IS CONDUCTED TO VERIFY THE PERFORMANCE DURING THERMAL/VACUUM TEST. THIS TEST SHOULD BE CONDUCTED IN A MINIMUM VACUUM OF 10^{-5} TORR AND THIS MAY BE MANNED OR UNMANNED. THE OPERATION TEST SHOULD BE CONDUCTED TO VERIFY LIFE CYCLE PERFORMANCE, AS AN EXAMPLE, IN CASE OF A SWITCH. THE ELECTROMAGNETIC INTERFERENCE (EMI) AND ELECTROMAGNETIC CONDUCTANCE (EMC) TESTS SHOULD BE PERFORMED TO VERIFY THE OPERATION OF AN ELECTRONIC TOOL IN PRESENCE OF OTHER ELECTRICAL/ELECTRONIC DEVICES. THE FINAL TEST IS PERFORMED WITH THE TEST ARTICLE, SUITED CREW MEMBER AND FLIGHT-LIKE INTERFACES.



CERTIFICATION

THE CERTIFICATION IS THE MOST IMPORTANT STEP FOR THE HARDWARE TO BE QUALIFIED AS FLIGHT WORTHY. THE CERTIFICATION MAY BE BASED ON ANALYSIS, OR ANALYSIS AND TESTS, OR SIMILARITY. THE FIT-CHECK IS ALSO AN IMPORTANT CRITERIA FOR CERTIFICATION.

DOCUMENTATION

THE DOCUMENTATION IS THE FINAL PHASE OF ANY DEVELOPMENT. THE DRAWINGS, ASSEMBLY PROCEDURES, PDA PROCEDURES, PIA PROCEDURE, ANALYSIS, SAFETY AND CERTIFICATION REPORTS ARE CONSIDERED AS THE INTEGRAL PART OF THE DOCUMENTATION.

EVA TOOLS - SATELLITE SERVICING ROLE
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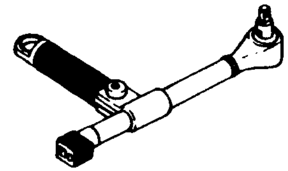

CASE STUDY #1
TOOL DEVELOPMENT

PFR WORK STATION

FOLLOWING THE METHODOLOGY DISCUSSED EARLIER, PORTABLE FOOT RESTRAINT (PFR) WORK STATION WAS DEVELOPED. IT PROVIDES STATIONARY FOOT RESTRAINT AND TOOL HOLDING CAPABILITY FOR CREWMEMBER WORKING IN CARGO BAY DURING SATELLITE REPAIR OPERATIONS. IT CONSISTS OF THE FOLLOWING:

- PORTABLE FOOT RESTRAINT
- WORK STATION STANCHION

CASE STUDY #2
MODIFICATION OF OFF-THE-SHELF TOOL

ELECTRICAL CONNECTOR TOOL (VISE-GRIP)

IT IS USED TO REMOVE AND INSTALL ELECTRICAL CONNECTORS ON THE SATELLITE. IT HAS ADJUSTABLE GRIPS TO ACCOMMODATE ELECTRICAL CONNECTORS OF VARIOUS DIAMETERS. THIS TOOL WAS MODIFIED TO PERFORM A SPECIFIC FUNCTION.

ILC SPACE SYSTEMS

EVA TOOLS - SATELLITE SERVICING ROLE
TYPICAL TOOL DEVELOPMENT

MICHAEL WITHEY

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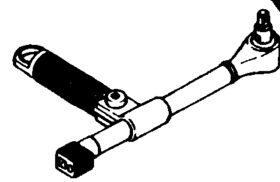
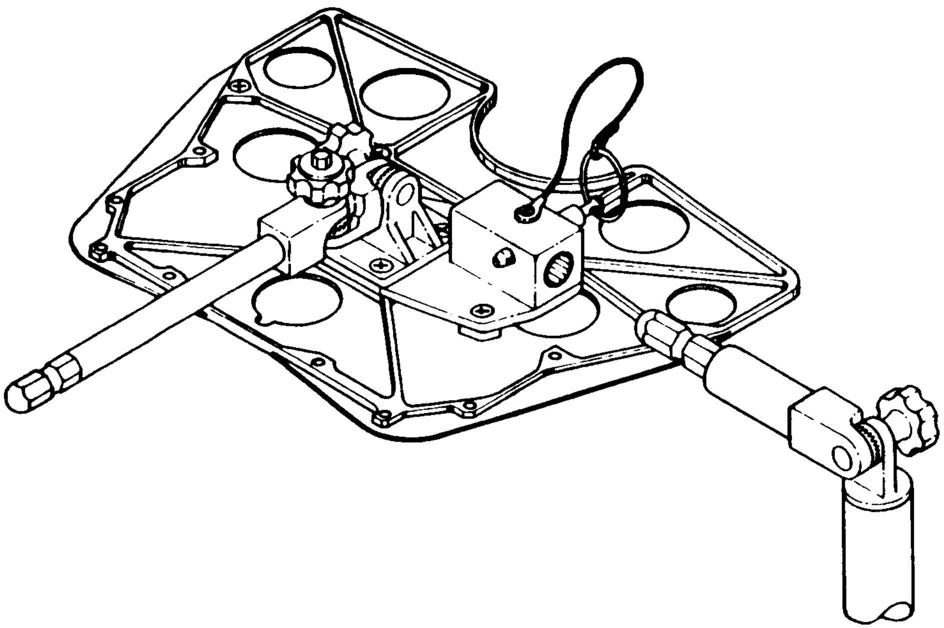
NOVEMBER 6-8, 1985

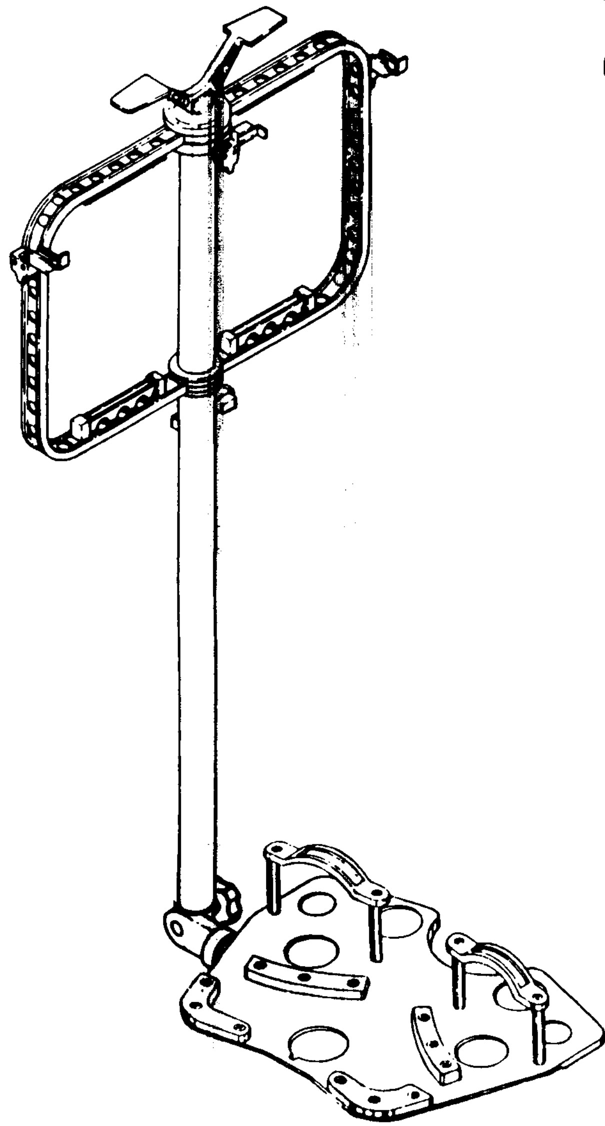
CASE STUDY #1

TOOL DEVELOPMENT





PORTABLE FOOT RESTRAINT



PFR WORK STATION

ILC SPACE SYSTEMS

EVA TOOLS - SATELLITE SERVICING ROLE
TYPICAL TOOL DEVELOPMENT

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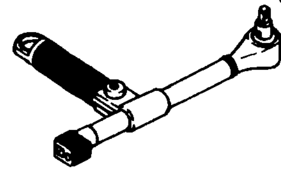
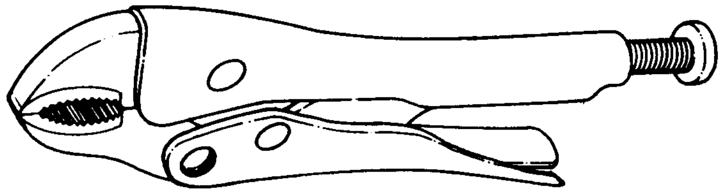
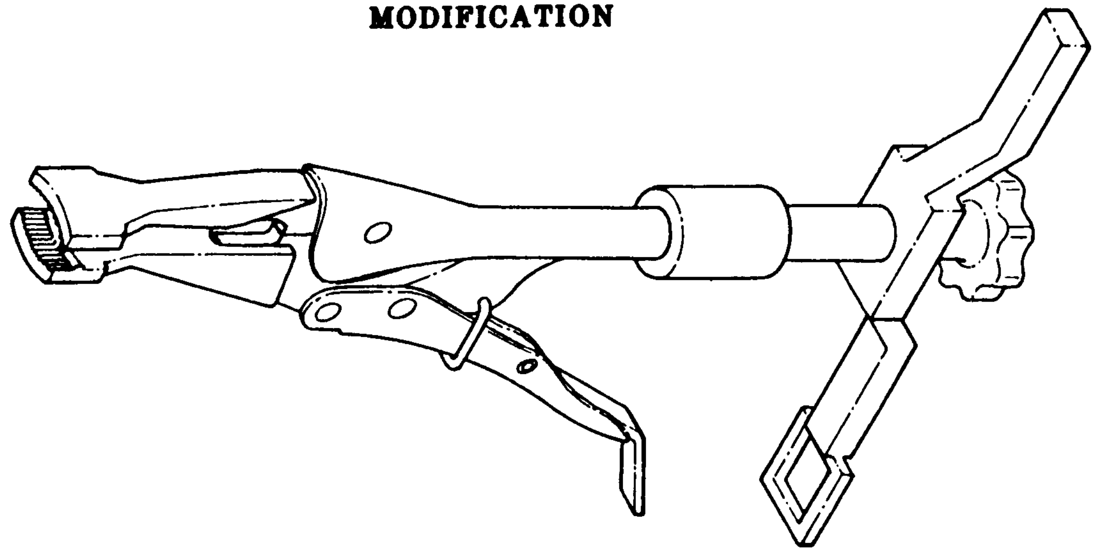


CASE STUDY #2

MODIFICATION OF OFF-THE-SHELF TOOL



ORIGINAL TOOL BEFORE MODIFICATION



ELECTRICAL CONNECTOR TOOL (POST MODIFICATION)

ILC SPACE SYSTEMS

SATELLITE RETRIEVAL - CASE STUDIES

Ralph J. Marak
Crew Systems Division
Engineering Directorate
NASA Johnson Space Center

Satellite Services Workshop II
Lyndon B. Johnson Space Center
November 6-8, 1985

SATELLITE RETRIEVAL - CASE STUDIES

- CASE #1 - SOLAR MAX

  PLANNED PROCEDURE:

  - RETRIEVE SATELLITE BY ATTACHING GRAPPLE FIXTURE TO TRUNNION PIN
  - TRUNNION PIN ATTACHMENT DEVICE-
    - SOFT DOCK- AUTOMATIC BY MEANS OF TRIGGER
    - HARD DOCK- UTILIZING SCREW THREADS IN TRUNNION PIN

  FLIGHT ANAMOLIES:

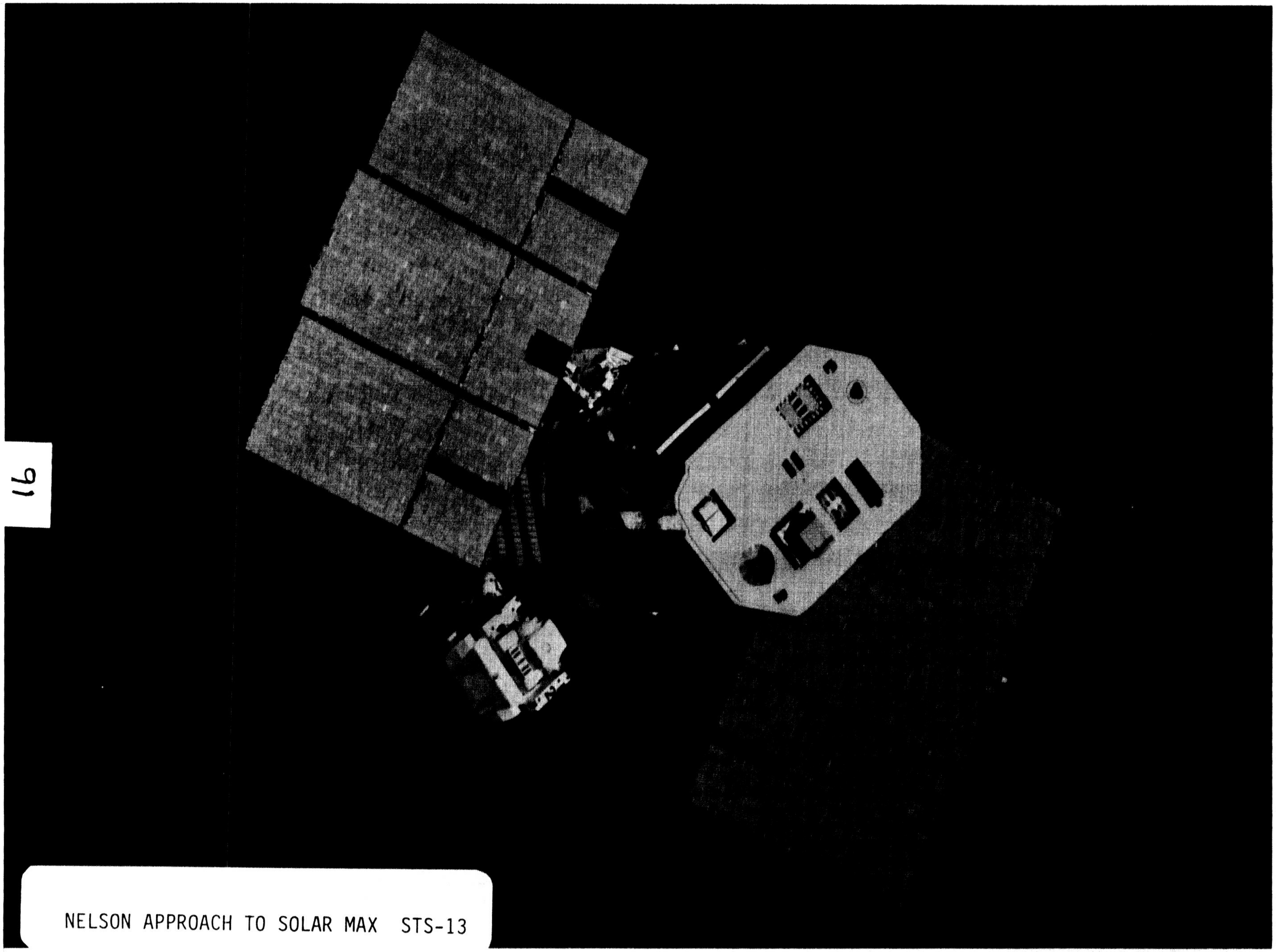
  - UNABLE TO GRAPPLE SATELLITE
    - CAUSE- FIBERGLASS PIN INSTALLED DURING THERMAL BLANKET INSTALLLATION PROHIBITED TPAD OPERATIONS.
      - PIN LOCATIONS NOT DOCUMENTED ON ICD OR DRAWINGS
      - LIMITED CLOSE-OUT PHOTOGRAPHS
      - NO PROVISIONS FOR MANUAL FIRING OF TPAD

  ACTUAL PROCEDURE:

  - UTILIZE RMS TO GRAPLE BY MEANS OF EXISTING GRAPPLE FIXTURE MOUNTED ON SATELLITE.

National Aeronautics and Space Administration

Lyndon B. Johnson Space Center
Houston, Texas 77058



NELSON APPROACH TO SOLAR MAX STS-13

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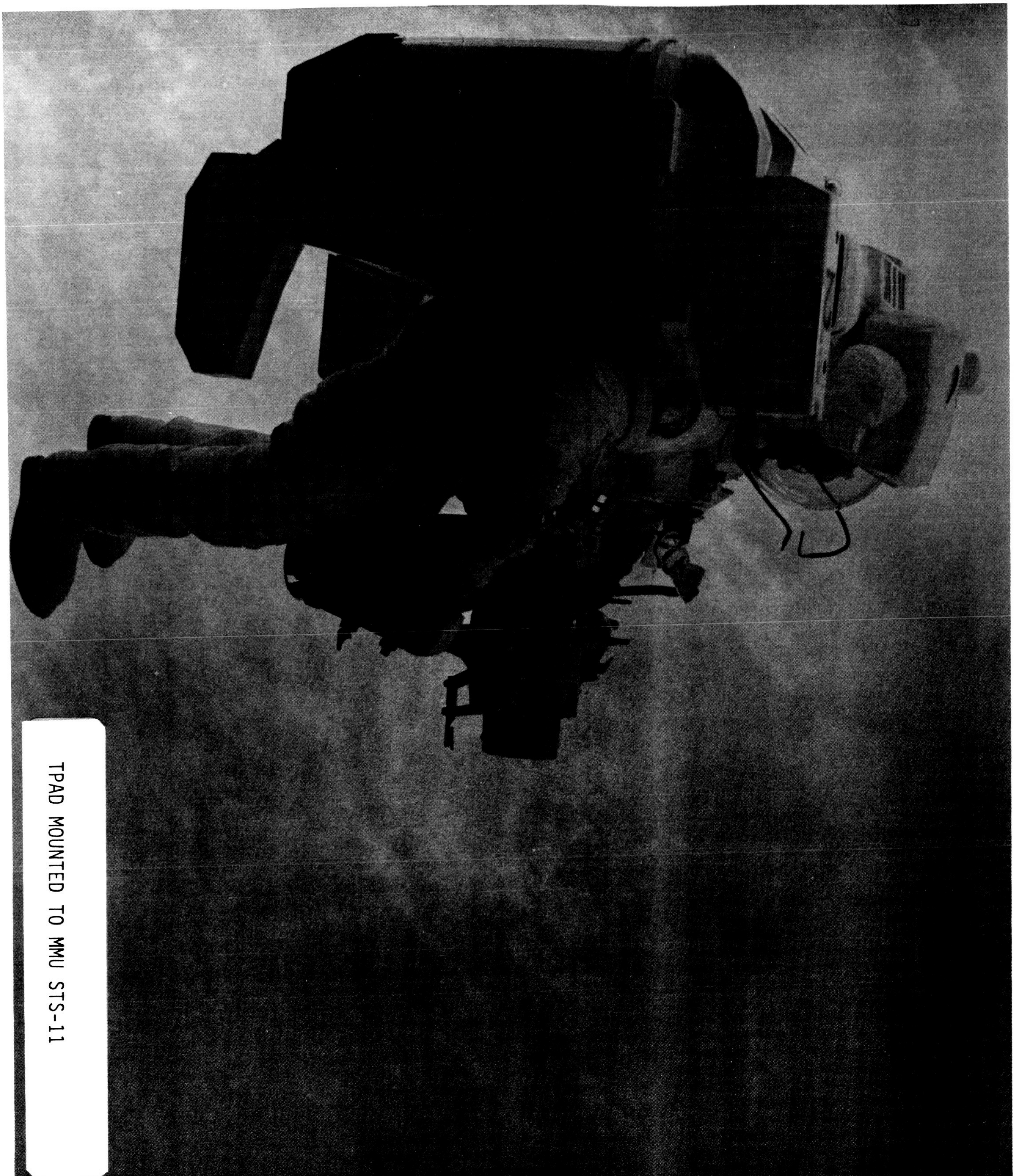
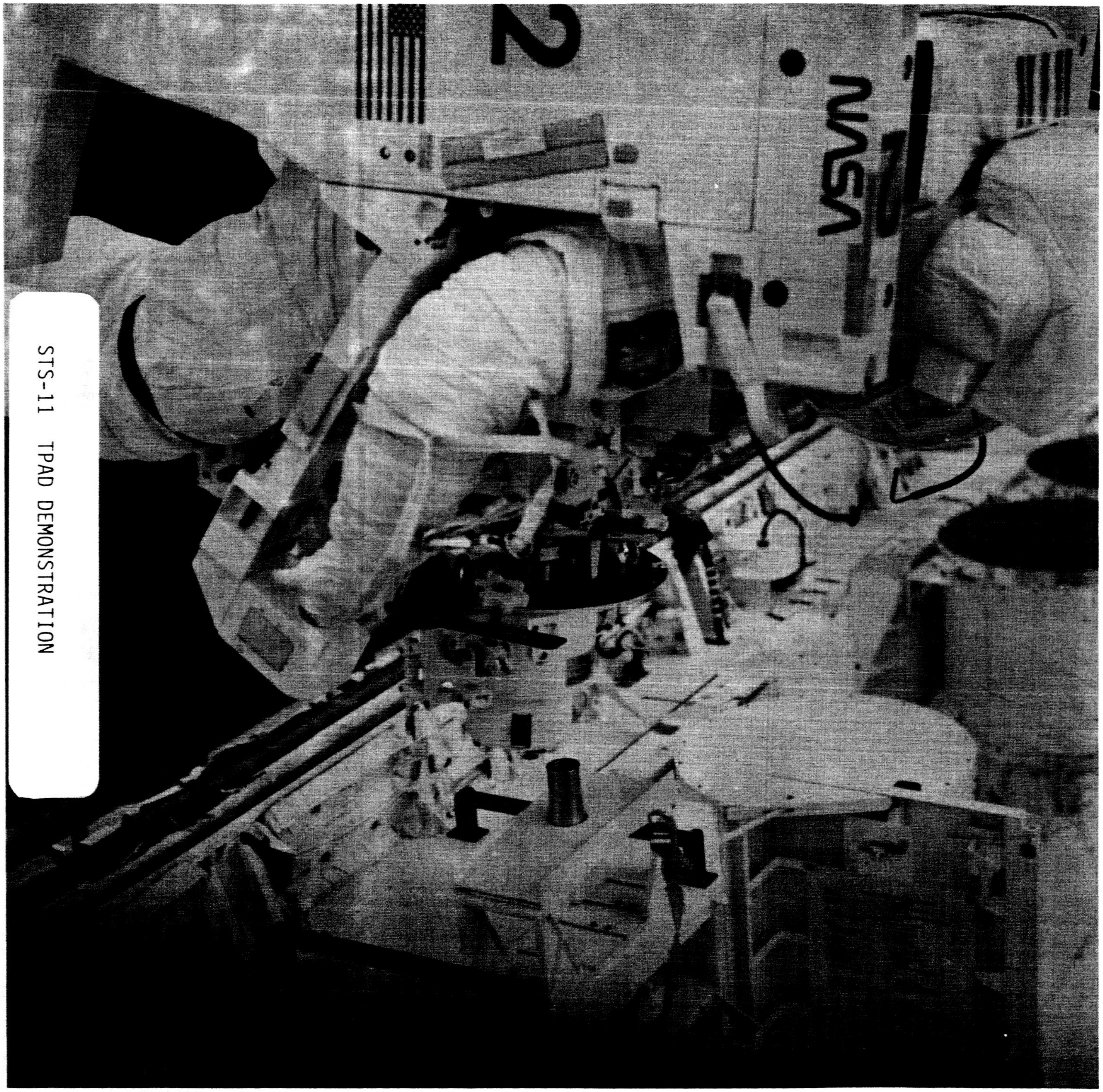


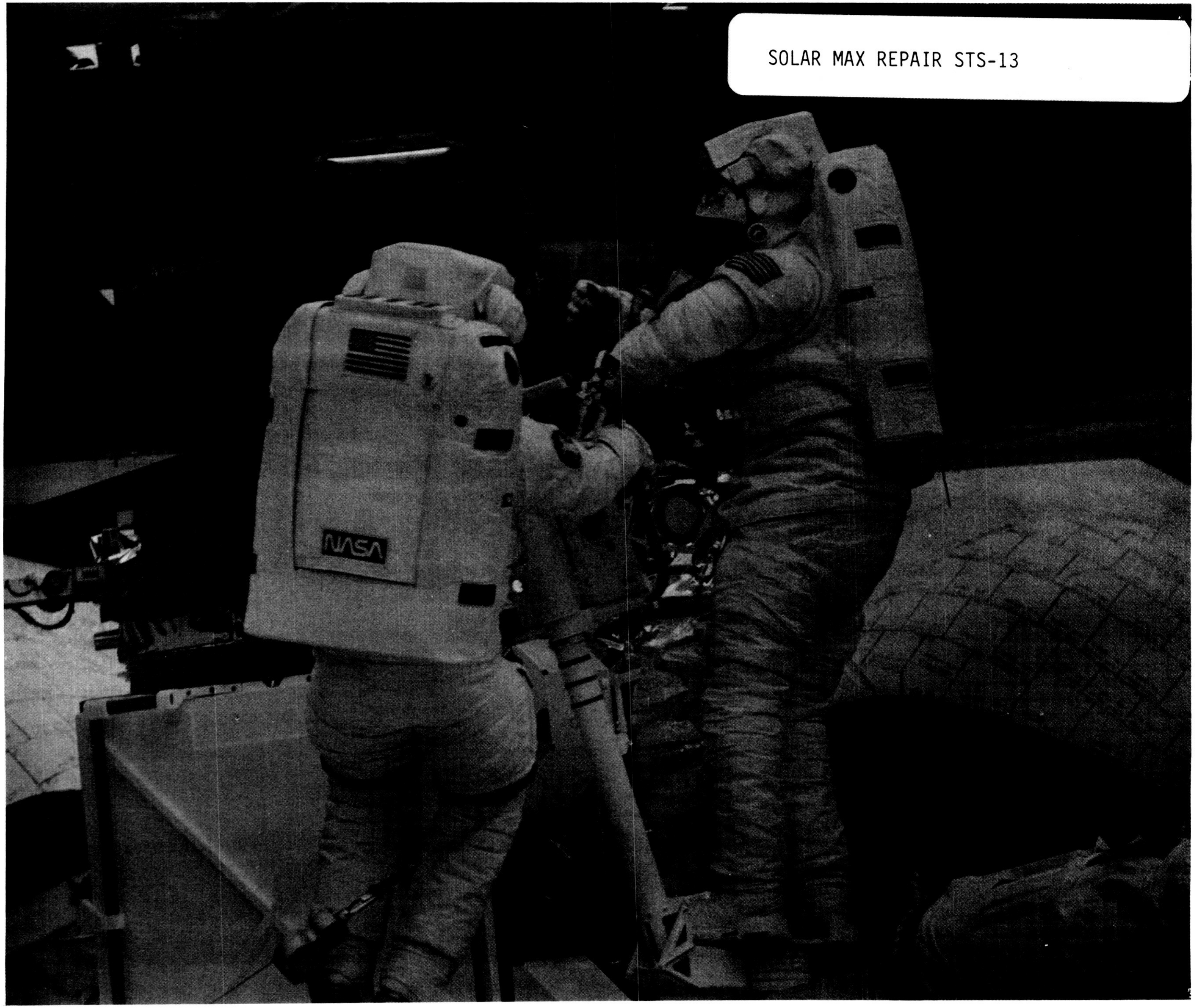
TPAD MOUNTED TO MMU STS-11

National Aeronautics and Space Administration

S84-27996

Lyndon B. Johnson Space Center
Houston, Texas 77058





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1985

- CASE #2 - WESTAR/PALAPA

  PLANNED PROCEDURE:

  - RETRIEVE SATELLITE BY ATTACHING GRAPPLE FIXTURE TO EXPENDED AKM
    - ATTACH SECONDARY HANDLING FIXTURE BY A MEANS OF A-FRAME BETWEEN BUMPER BRACKET AND COMMON BRACKET.

  FLIGHT ANAMOLIES:

  - UNABLE TO ATTACH COMMON BRACKET CLAMP
    - CAUSE- WAVEGUIDE INTERFERENCE NOT PREVIOUSLY INDENTIFIED
      - WAVEGUIDE ADJUSTMENT MADE AFTER FINAL ASSEMBLY
      - NO CLOSE-OUT PHOTOGRAPHS/DRAWINGS AVAILABLE TO DOCUMENT PLACEMENT OF WAVEGUIDE.

  ACTUAL PROCEDURE:

  - CREWMEMBER ABLE TO MANUALLY MANAGE SATELLITE FOR PLACEMENT INTO PAYLOAD BAY.

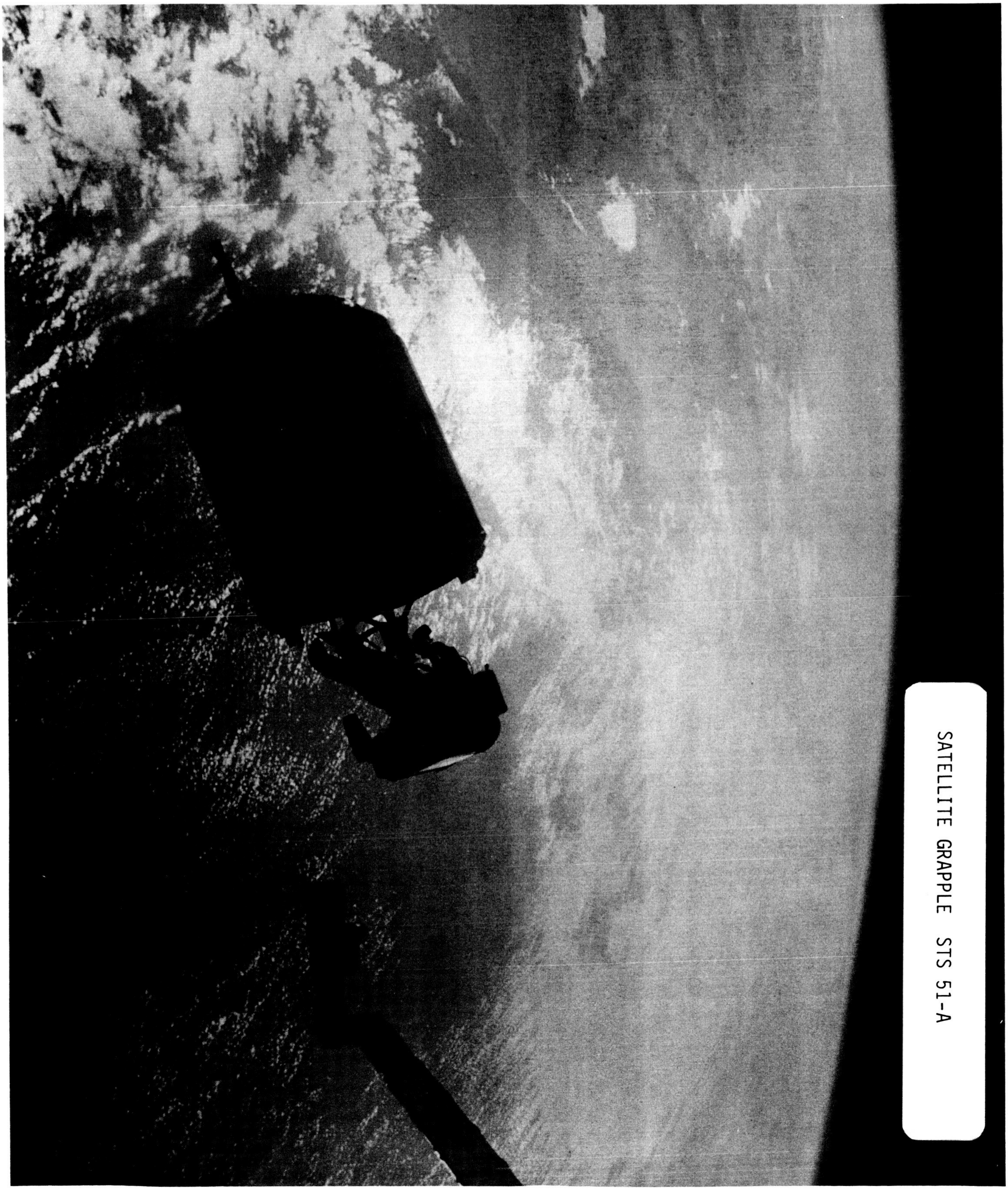
- CASE #3 - LEASAT

  PLANNED PROCEDURE:

  - RETRIEVE BY MANEUVERING SHUTTLE NEAR SATELLITE AND MANUALLY ATTACHING GRAPPLE FIXTURE.
  - SUCCESSFUL MISSION- ABLE TO IDENTIFY CRITICAL INTERFACES ON SPARE SPACECRAFT.
    - EXPERIENCE FROM WESTAR/PALAPA MISSION IN HANDLING LARGE MASSES BY CREWMEMBER PROVIDED CONFIDENCE IN THS PROCEDURE.
    - DEVELOPMENT OF ACCURATE MASS SIMULATOR PERMITTED REALISTIC TRAINING.

  ANAMOLIES: NONE

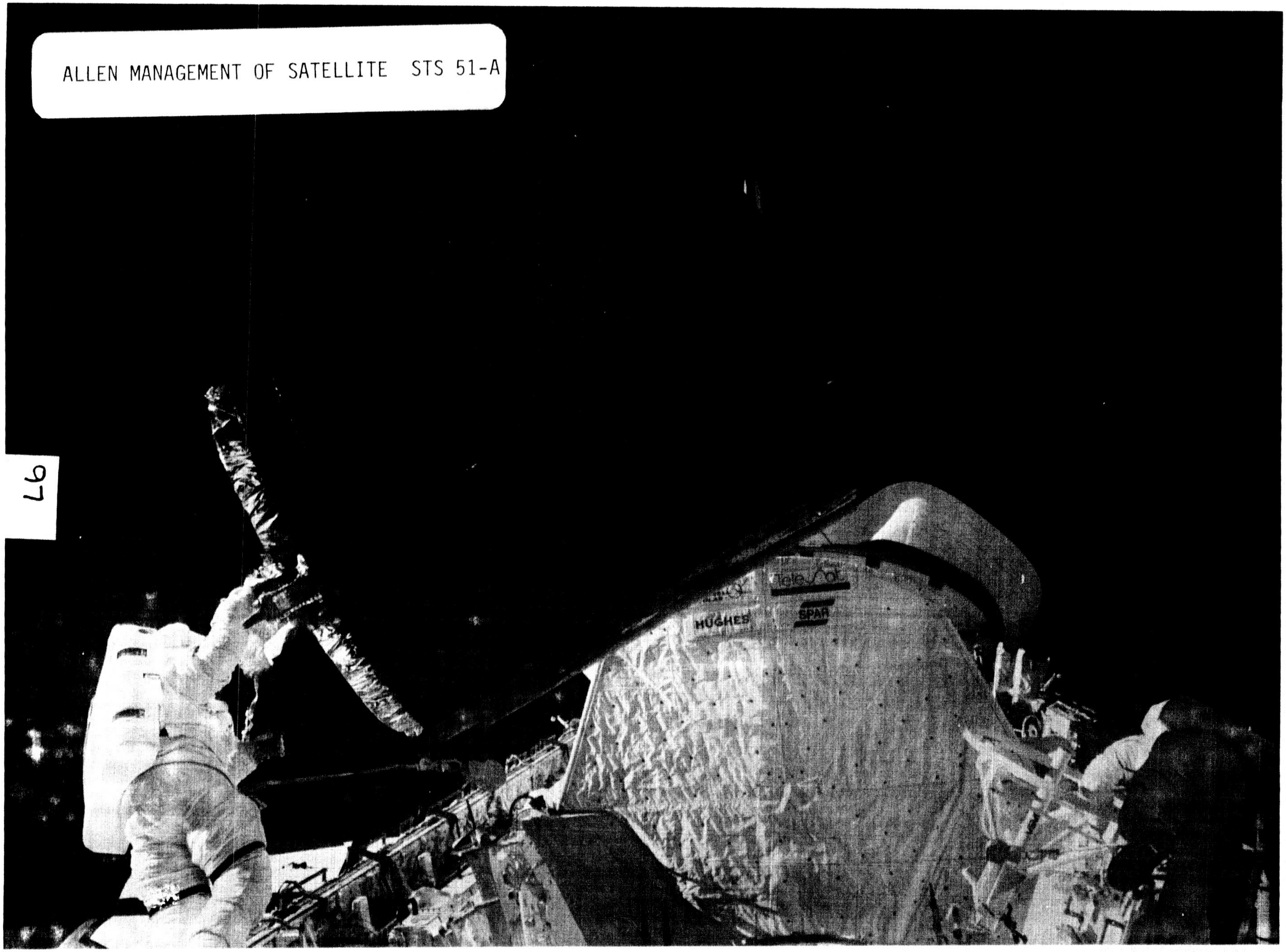
  ACTUAL: AS PLANNED


SATELLITE GRAPPLE STS 51-A

National Aeronautics and
Space Administration

S19-41-050

Lyndon B. Johnson Space Center
Houston, Texas 77058



ALLEN MANAGEMENT OF SATELLITE STS 51-A

National Aeronautics and
Space Administration

86 I-41-009

Lyndon B. Johnson Space Center
Houston, Texas 77058

FISHER USING HANDLING BAR STS 51-I

SUMMARY

0 ADEQUATE DOCUMENTATION IN THE FORM OF DRAWINGS AND CLOSE-OUT PHOTOGRAPHS ENHANCE MISSION SUCCESS.

- o PERMITS DEVELOPMENT OF HI-FI MOCKUPS
- o INDENTIFIES PROBABLE INTERFERENCE LOCATION

EVA TOOL CATALOG

A Designer's and User's Guide

Robert C. Treviño
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Mission Operations Directorate
NASA Johnson Space Center

Satellite Services Workshop II
Lyndon B. Johnson Space Center
November 6-8, 1985

Background

With the successful completion of the Solar Maximum Repair Mission in April, 1984, repairing satellites in space became a reality. There has been a steady increase in the number of EVA missions on the shuttle. Some have been scheduled a long time in advance, like Solar Max, some have been conceived and developed within six months, like the Westar/Palapa Retrieval Mission, and one has been an unplanned EVA mission, like the "Flyswatter Episode" on STS 51-D. Of a total of 20 shuttle launches, seven have had EVA. The Space Station is expected to rely extensively on EVA for assembly, maintenance, and payload servicing.

The problem that exists today is that there are still no clear EVA design standards and no current and complete listing of EVA tools and equipment available to the user. A real weakness in the EVA/mission operations area has been that EVA problems are solved in one program and new EVA tools are developed, but these are not documented and made available to assist another program at a later time. The EVA Tool Catalog is the first genuine attempt to provide the satellite designers, users, and mission operation personnel with a practical and useful document that provides current information and a source of ideas for future EVA tool and equipment development.

EVA Summary Through STS 51-I

Shuttle Mission	Date	Duration h:m EVA 1	Duration h:m EVA 2	Use of MMU?	Remarks
STS - 5	11/14/82	0.0		No	EVA cancelled because of EMU problems
STS - 6	4/7/83	3:54		No	First Shuttle EVA
					Demonstrated basic EMU capability
					Demonstrated translations and limited EVA tasks
STS 41-B	2/7/84	5:35		Yes	Test flight of MMU
	2/9/84		6:02	Yes	Demonstrated MMU docking
					Test of MFR
					Test of refueling tools
					Unplanned repair of SPAS experiment.
STS 41-C	4/8/84	2:59		Yes	First repair mission - Solar Max
	4/11/84		7:07	Yes	First operational use of MMU and MFR
					Demonstrated repair of EVA and non-EVA components
STS 41-G	10/11/84	3:29		No	First demonstration of Orbital Refueling System
					First American woman to go EVA
					Unplanned EVA assistance in antenna stowage
STS 51-A	11/12/84	6:13		Yes	6 months for EVA hardware/procedures development and training
	11/14/84		6:01	Yes	Retrieval of 2 satellites using contingency procedures
					Demonstrated flexibility of Shuttle EVA

EVA Summary Through STS 51-I

Shuttle Mission	Date	Duration h:m		Use of MMU?	Remarks
		EVA 1	EVA 2		
STS 51-D	4/17/85	3:10		No	First unplanned EVA Fabrication of flyswatter IVA and installed on RMS EVA
STS 51-I	8/31/85	7:08		No	4 months for EVA hardware/procedures development and training Use of single-joint RMS for MFR ops
	9/1/85		4:23	No	Manual retrieval of 15,000 lbs spacecraft from MFR Repair of Leasat Replanning from 1 EVA to 2 EVA Manual spinup and deployment of the spacecraft

List of Frequently Used EVA Tools and Equipment

EVA Tool or Equipment	Shuttle Mission						
	6	41-B	41-C	41-G	51-A	51-D	51-I
MMU		X	X		X		
MFR		X	X		X		X
MFR Tool Board		X	X		X		X
PFR	X	X	X	X	X		X
PFR Socket	X	X	X	X	X		X
PFR Universal Socket			X		X		
EVA Power Tool		X	X		X		X
EVA Ratchet	X	X	X		X		X
7/16" Socket		X	X		X		X
EVA Scissors		X	X		X		X
No. 10 Allen Wrench		X	X		X		X
Miniwork Station	X	X	X	X	X	X	X
Small EVA Trash Bag	X	X	X	X	X	X	X
Large EVA Trash Bag		X	X		X		X
Payload Retention Device		X	X		X	X	X
Adjustable Wrist Tether	X	X	X	X	X	X	X
EVA Camera	X	X	X	X	X		X
EMU TV		X	X				

Contents of the EVA Tool Catalog

Tools and Equipment Description Section

Front Page

- Title
- Photograph
- Overview
- Operational Comments
- Contacts - Operational
Technical
Source

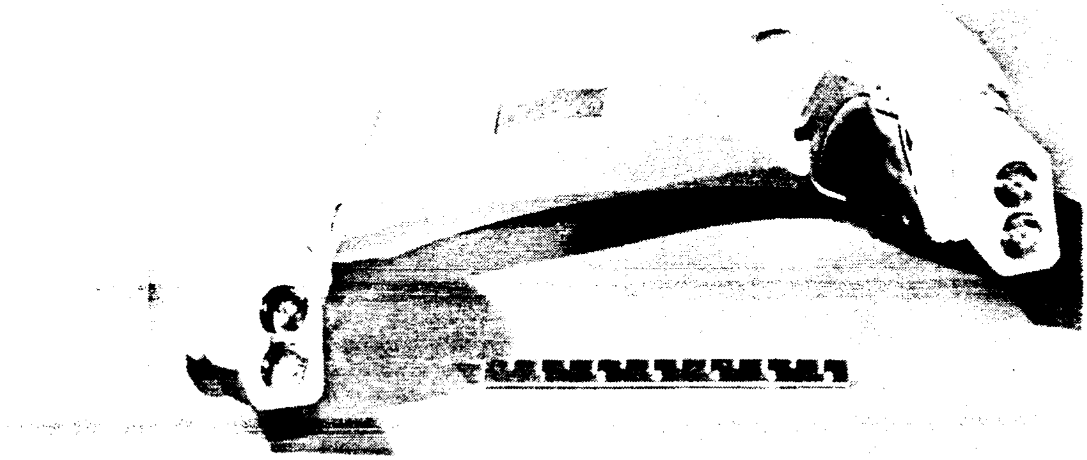
Back Page

- Title
- Table of Technical Information
- Table of Dimensional Data
- Technical Drawing - 2 or 3 views or pictorial

Note: There is a total of 165 items in this section.

Contents of the EVA Tool Catalog (continued)

EMU LIGHTS ASSEMBLY



OVERVIEW

The extravehicular mobility unit (EMU) lights assembly provides a crewmember with portable lighting during an extravehicular activity (EVA). The assembly contains two independent lamp modules connected by a crossmember. Each side contains a battery module, two lamps, a switch, and a sequencing circuit. The EMU lights attach to the helmet with simple latches.

Each lamp module has a left-right swing angle of 85°, 5° towards the helmet and 80° away from the helmet. Each module also has an up-down swing angle of 60°, 30° up and 30° down.

The battery module can supply power to illuminate two lamps for 3 hr or one lamp for 6 hr, with less than 10 percent degradation of light intensity. Each battery module can also be easily replaced in flight, if necessary, after each EVA.

OPERATIONAL COMMENTS

The switch and sequencing circuit provide one-hand operation of the lamp module. Depressing the switch once turns on the upper lamp; the second activation turns on the lower lamp; the third activation turns on both lamps; and the fourth activation turns both lamps off.

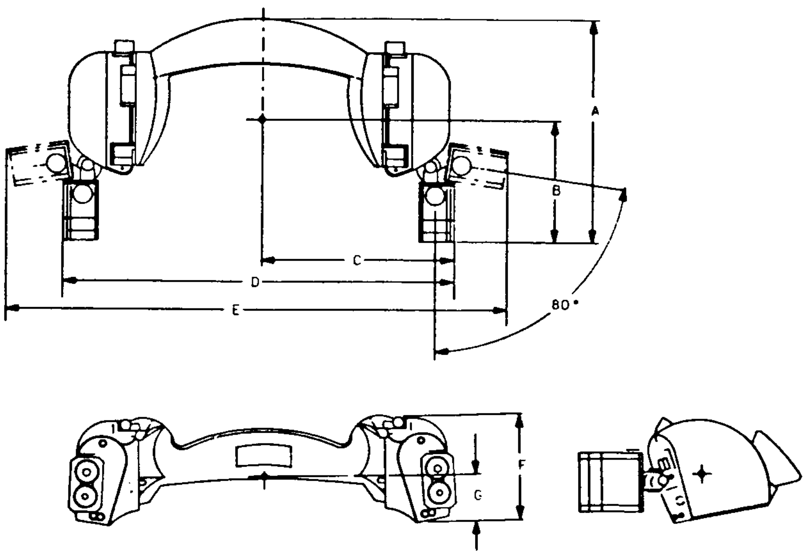
CONTACTS

Operational: Name, Organization, phone number
Technical: Name, Organization, phone number
Source: Name, Organization, phone number

EMU LIGHTS ASSEMBLY

Technical Information	
Part number	10161-10061-04
Power supply	Two independent battery modules (one per side)
Battery	D-size lithium bromine complex 3.5 V, 8 A-h
Battery life	3 hr min. with four lamps operating
Voltage Open circuit Loaded	 3.9 V dc 3.25 V dc
Lamps	Two halogen lamps per side (2.5 Watts each)
Lamp intensity	20 ft-c (min.) per lamp at 3 ft
Lighting pattern	16 by 24 in. at 2 ft (four lamps on pointed forward)
Operation	Momentary switch activated sequencing circuit per side
Weight	5 lb

Dimensional Data	
A	8.81 in.
B	4.66 in.
C	7.76 in.
D	15.52 in.
E	19.78 in.
F	4.30 in.
G	1.88 in.



Concluding Remarks

The EVA Tool Catalog is an attempt to provide the spacecraft designers, users, and operations officers, a useful, cost effective, and practical guide of available and potentially available EVA tools and equipment. When the requirements of a payload are being developed, the tool catalog can be used as a reference of tools that could be used for planned or manual backup tasks. The catalog can also be used to get ideas from previously qualified hardware. Feedback from the user's will be important in order to continue to improve the tool catalog.

In conclusion, the following recommendations are made:

- Utilize the EVA Tool Catalog as a guide to designers and users as a coordinated NASA source of information on EVA tools, equipment, and operations.

- Make your EVA requirements known early to the Program Office.

- Continue to exchange information and ideas on satellite servicing in order to better understand the user's needs and requirements.

EVA TOOLS FUTURE DEVELOPMENT

Ralph J. Marak
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NASA Johnson Space Center

Satellite Services Workshop II
Lyndon B. Johnson Space Center
November 6-8, 1985

EVA TOOLS
FUTURE DEVELOPMENT

TOOLS USED FOR THE RETRIEVAL AND REPAIR OF SATELLITES HAVE TO THE PRESENT BEEN OF SPECIAL DESIGN AND SUITED ONLY FOR PARTICULAR APPLICATIONS. AS SATELLITE SERVICING BECOMES A MATTER OF ROUTINE, AS WILL BE DONE WITH THE SPACE STATION, CERTAIN FEATURES WILL BE NECESSARY TO ASSIST THE CREWMEMBER IN THE MAINTENANCE AND REPAIR TASK. THESE FEATURES MUST BE STANDARDIZED AMONG ALL OF THE VARIOUS SATELLITE MANUFACTURES, WHETHER THE SATELLITES ARE TO BE LAUNCHED BY EXPENDABLE VEHICLE, (INCLUDING THE ARIANE) OR THE SHUTTLE, THE STANDARD FEATURES MUST BE THERE. ONLY THEN, IF A SATELLITE MALFUNCTIONS IN ORBIT OR FOR PLANNED MAINTENANCE, CAN A QUICK AND COST EFFECTIVE RESPONSE BE EFFECTED.

THESE FEATURES SHOULD INCLUDE:

- A) GRAPPLE FIXTURE ATTACHMENT PROVISIONS
- B) SWING-OUT ACCESS PANELS
- C) ELECTRICAL CONNECTOR SPACING
- D) HANDHOLD ATTACHMENT PROVISIONS
- E) TETHER PROVISIONS
- F) CAPTIVE FASTENERS

NASA IS PRESENTLY ESTABLISHING DESIGNS FOR EVA COMPATABLE ELECTRICAL AND FLUID CONNECTORS, EQUIPMENT HANDLING, TRANSFER TETHER ATTACHMENTS, AND VARIOUS HAND TOOLS (SOCKETS, SCREW REMOVAL, TORQUE AMPLIFIERS LIMITERS, POWER TOOLS, PORTABLE HANDHOLDS, ETC.) AND METHODS FOR STOWAGE AND HANDLING. WHEN COMPLETED, PROTOTYPE HARDWARE WILL BE FABRICATED FOR EVALUATION BY EXPERIENCED EVA CREW MEMBERS. UPON COMPLETION OF THE EVALUATION, ANY NECESSARY DESIGN REFINEMENTS WILL BE MADE AND THE MANUFACTURING DRAWINGS WILL BE COMPLETED. THE DESIGNS WILL BE LOGGED, FILED AND AVAILABLE FOR FUTURE USE.

ALSO, UPON COMPLETION OF A EVA MISSION (SUCH AS LSM), ALL THE DATA FROM THE MISSION TAPES AND CREW DEBRIEFING IS SCREENED TO DETERMINE IF ANY OF THE TOOLS ARE CANDIDATES FOR A GENERIC EVA APPLICATION. TOOLS WHICH FALL IN THE CATAGORY WILL BE EVALUATED TO ASSURE THT THERE ARE NO DESIGN OR HUMAN FACTOR DEFICIENCES. WHEN DEFICIENCES ARE FOUND, THEY WILL BE CORRECTED AND THE DRAWINGS UPDATED. THESE DESIGNS WILL ALSO BE LOGGED AND FILED FOR FUTURE USE. THIS WILL ASSURE THAT AN INVENTORY OF TOOLS IS AVAILABLE WHICH CONTAINS ACTUAL HARDWARE AND DESIGNS, AVAILABLE FOR FABRICATION AND CERTIFICATION.

AS CAN BE SEEN IN THE EVA TOOL CATALOG, NASA HAS AMASSED A NUMBER OF TOOLS WHICH CAN BE USED FOR SATELLITE RETRIEVAL AND MAINTENANCE. HOWEVER, UNTIL THE SATELLITE MANUFACTURERS ESTABLISH RETRIEVAL AND MAINTENANCE STANDARDS, THIS TASK WILL BE A COSTLY AND TIME CONSUMING ONE.

A SHUTTLE EXPERIMENT SENSING FORCES AND TORQUES ON THE ORBITER RMS

Robert L. French

ABSTRACT

The Jet Propulsion Laboratory and the Johnson Space Center are conducting a cooperative demonstration project to measure and graphically display the forces and torques that are developed at the base of the end effector of the Remote Manipulator System. The demonstration hardware and software is based upon research conducted over the past seven years and will feature a force-torque sensing unit mounted in series between the last wrist joint and end effector on the mechanical arm. The sensor will consist of four radial spokes or deflection beams with bonded silicon-based semiconductor strain gages. Strain gage data will be collected at the sensor, transmitted to a dedicated computer in the aft flight deck, resolved into forces and torques and displayed on an Orbiter closed circuit TV. Requirements of the Shuttle Orbiter have impacted the sensor system design and created some difficult design trade-offs. The system will fly in 1987.

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INTRODUCTION

The Jet Propulsion Laboratory (JPL) and the Johnson Space Center (JSC) have undertaken the Remote Manipulator System/Force Torque Sensor (RMS/FTS) flight demonstration as a joint and cooperative project. The RMS/FTS will be an add-on system to the Shuttle Remote Manipulator System (RMS) and will measure and graphically display the three orthogonal forces and three orthogonal torques which are developed at the base of the end effector of the RMS. These forces and torques will be displayed to the RMS operator on a closed circuit television (CCTV) flight monitor and enable the operator to demonstrate enhanced RMS dexterity in dynamically constrained operation.

The flight demonstration system is based upon research and development of sensors and manipulators at JPL and extensive ground testing on the Manipulator Development Facility at JSC (References 1,2,3). The design of the flight demonstration system has introduced unique requirements and, coupled with project time constraints, has necessitated trade-off decisions in favor of shuttle safety and integration compatibility. The flight system is currently in the final design stage. The demonstration flight is projected to be mid 1987. This paper describes the demonstration system and discusses several design issues that result from integration and compatibility requirements of the Shuttle Orbiter.

RMS/FTS SYSTEM DESCRIPTION

The RMS/FTS System can be installed in any of the Orbiter vehicles. When in operation, the RMS/FTS will provide visual (graphic or numeric) real time quantitative force-torque information to the RMS operator. The displayed information may be the measured forces and torques at the mechanical arm (outer wrist joint) - end effector interface or computed forces and torques at specified points on the paylod periphery. Mode selection, scale amplification and on-off operation will be under the control of the RMS operator.

The system will be composed of new elements or subsystems which are interfaced to existing Orbiter subsystems. The new elements include:

a) Force Torque Sensor (FTS)
b) Data Collection Electronics (DCE)
c) Sensor Display Subsystem (SDS)
d) Cabling and a Control Panel.

The existing Orbiter subsystems include:

a) RMS mechanical arm
b) Mechanical arm flight cable
c) A CCTV flight monitor
d) Power and air conditioning
e) Flight data recorder

Figure 1 portrays the general location of the system elements. The

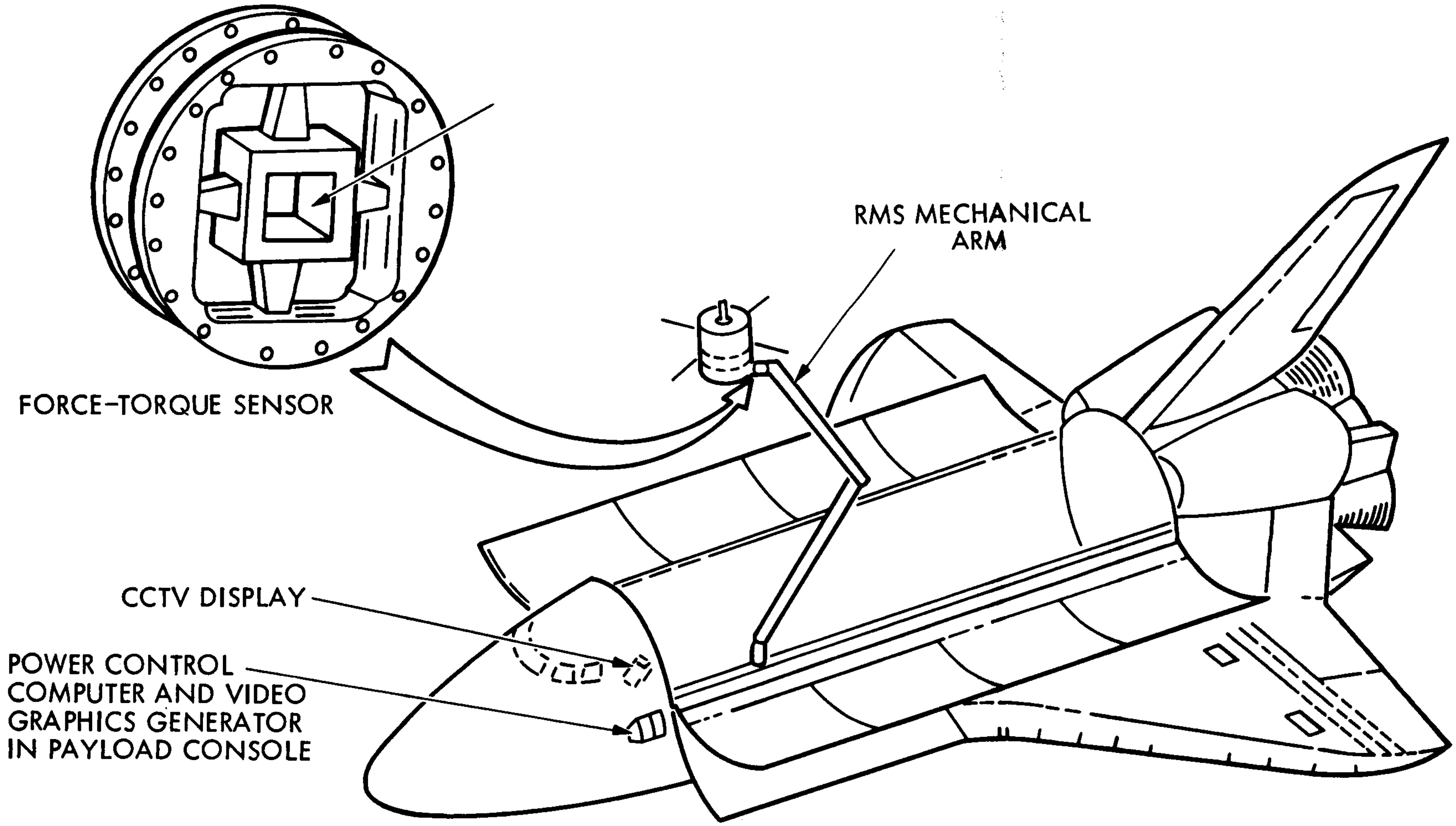


Figure 1. Artist Concept of RMS/FTS System Installation In Shuttle Orbiter

Force Torque Sensor (FTS) will be mounted on the RMS Mechanical Arm in series with the arm and end effector. The Data Collection Electronics (DCE) will be packaged in the center cavity of the FTS. The Sensor Display System and a control console will be located on the aft flight deck in a payload dedicated console. The CCTV Flight Monitor is located at the payload station. As required, special cabling will be provided to connect the elements and to supply Orbiter 28 vdc power.

Figure 2 presents a system block diagram. The Force Torque Sensor and Data Collection Electronics on the Mechanical Arm in the Cargo Bay will receive power and transmit data through payload dedicated wires in the Mechanical Arm Flight Cable. The Sensor Display Subsystem will contain a system computer, video graphics generator, a keyboard and control switches. All data will be written to the video display and recorded on a video tape recorder.

Force Torque Sensor

The FTS is cylindrical in external shape with flanges to permit bolting into the RMS at the mechanical arm (roll portion of the wrist) - end effector interface. Installation into the RMS will be accomplished by separating the standard end effector from the arm, bolting the FTS to the arm and then bolting the end effector to the FTS. The FTS, therefore, becomes an integral part of the RMS structure, and the total length of the RMS will be increased by the axial length of the sensor.

The sensing element within the FTS will be a four spoked configuration - see artist concept, Figure 3. The "x" coordinate axis coincides with the end effector/arm roll axis, the "y" axis with pitch and the "z" axis with yaw. Silicon-based semiconductor strain gages will be bonded to the four deflection bars which form the spoke elements. To minimize thermal drift problems, gages on opposite sides of each deflection bar will be wired into a full bridge circuit. Each bridge will provide a single reading which reflects the differences in strain levels on opposite sides of the bars. A total of eight bridge circuits will be employed to produce eight signal outputs.

The strain gages are defined as part of the force-torque sensor. The interface of the FTS with the DCE will include mechanical attachment, strain gage connections and a thermal conduction path to conduct heat from the electronics package to the outer FTS envelope.

The FTS frame will be machined from one piece of aluminum (7075-T7351) and the deflection bars will be sized for high stiffness. Positive stops (not necessarily integral with the frame) are provided to limit the displacement and loads of the deflection bars. The sensor diameter is 25.4 cm (10 in) and length is 5.1 cm (2 in).

Data Collection Electronics

The data collection electronics (DCE) will process strain gage and instrumentation signals, and supply regulated power to the strain gages, instrumentation and all internal electronics. The DCE will be securely

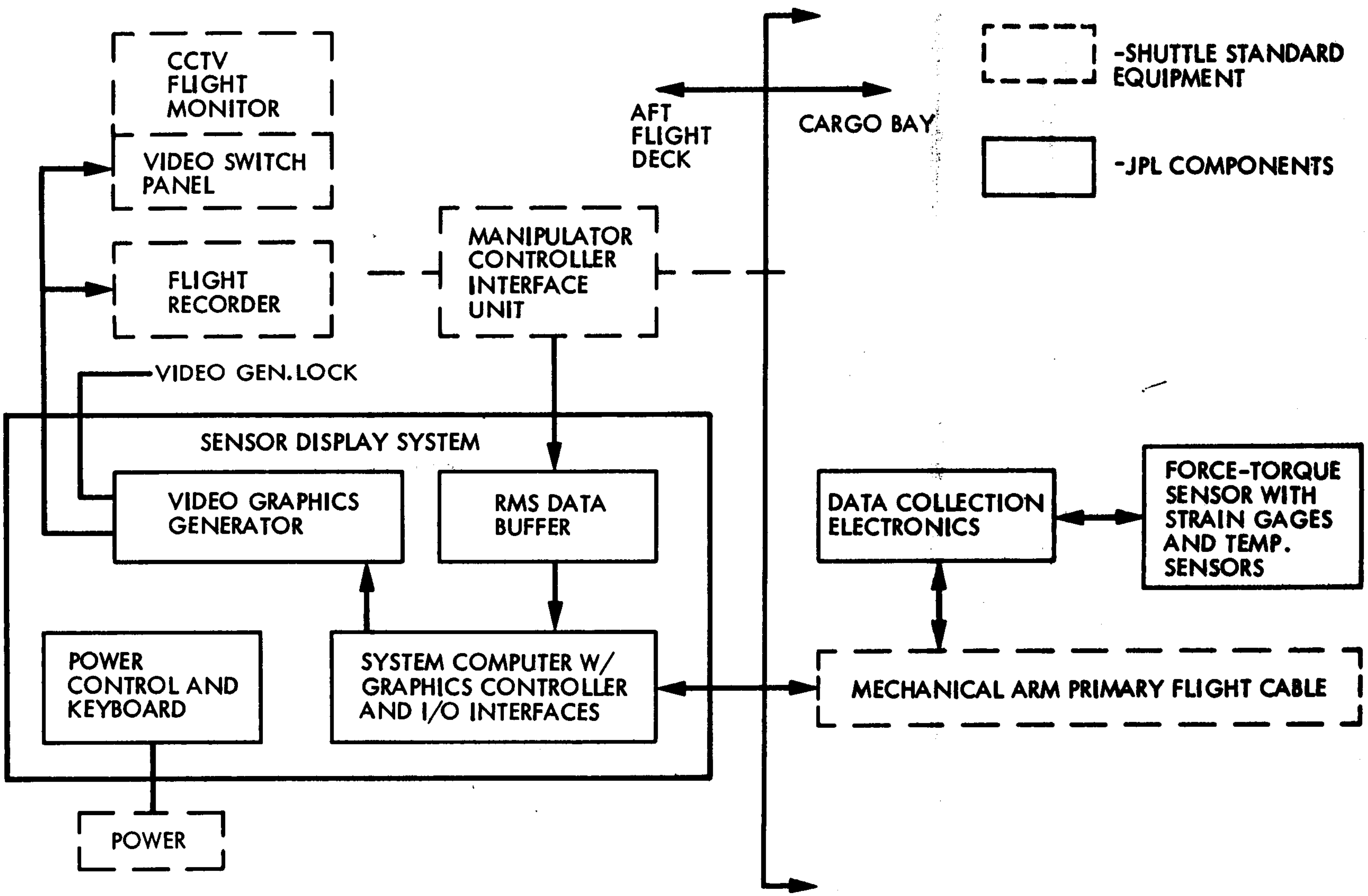


Figure 2. RMS/FTS Functional Block Diagram

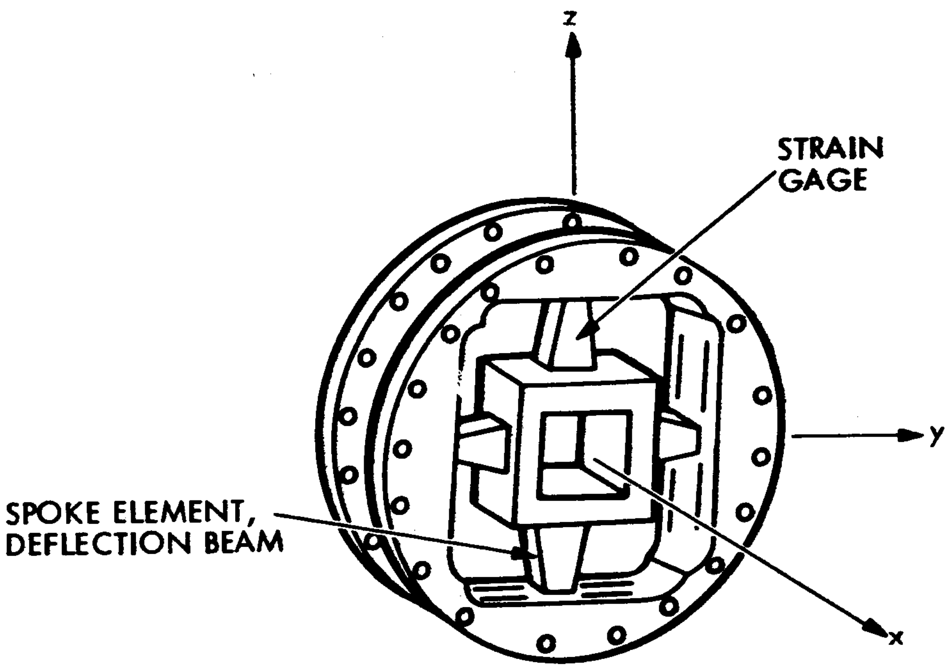


Figure 3. Artist Concept of Force-Torque Sensor Frame

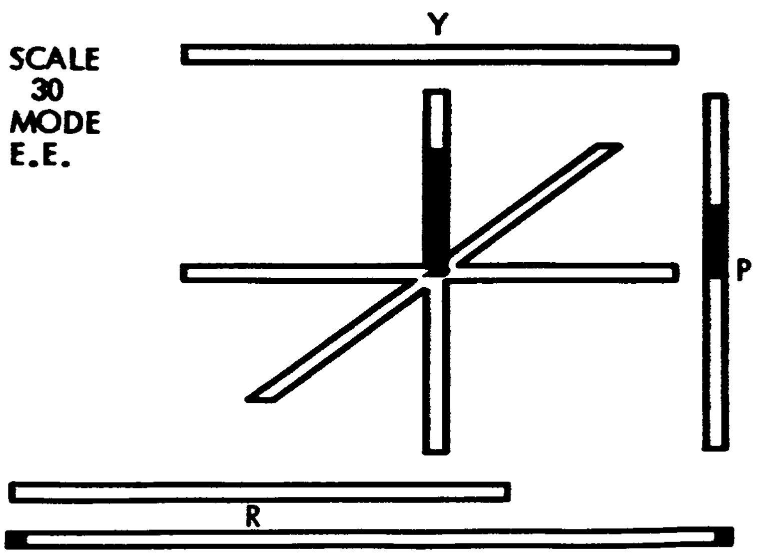


Figure 4. CCTV Display Format

mounted within the center open volume of the force-torque sensor. The eight strain gage bridge circuits on the FTS will be directly wired to the DCE. The strain gage signals will be amplified, converted to digital signals and serialized into a data stream together with engineering temperatures and voltages.

Sensor Display Subsystem

The Sensor Display System (SDS) performs several functions:

1) Provides keyboard or switch panel for operator control of system and function selection,
2) Decodes serialized data from the DCE,
3) Computes forces and torques from the strain gage signals and resolve the output into three orthogonal forces and three orthogonal torques,
4) Generates a video signal for output to an Orbiter closed-circuit TV flight monitor,
5) Accepts and distributes Shuttle power to system components,
6) Outputs force-torque and engineering data to flight tape recorder.

The SDS has the following operator selectable basic capabilities:

1) Scale change - small, medium, large
2) Rotation and translation of point of resolution (i.e., ability to compute forces and torques at discrete payload coordinate points.)
3) Graphic or numeric data display
4) Graphics downsizing for split screen displays.

The Sensor Display System is composed of several functional elements or components. These are a system computer, a video graphics generator, input/output and interface electronics and a control panel and keyboard for system control by the RMS operator.

The RMS operator may select either of the two aft flight deck CCTV monitors for viewing the RMS/FTS outputs. The SDS outputs both text and graphics. Text output reflecting operator keyboard inputs, system status and numerical force/torque levels are under the control of the operator. The format for graphical forces and torques is illustrated in Figure 4.

Forces are portrayed on a two dimensional projection of a three dimensional cartesian coordinate system, see center bars in Figure 4. Lateral forces will be displayed on the horizontal axis (positive to the right), vertical forces on the vertical axis (positive up) and axial forces on the oblique axis (positive down and to the left). Force magnitude is indicated by the length of the filled-in bar on the primary axes. In the diagram, a positive "z" axis force is shown.

Torques are displayed around the edge of the screen as bar charts. Yaw, pitch, and roll axes are labelled with Y, P, and R respectively. Force axes are unlabelled. Alphanumerics in the upper left hand corner of the screen indicate the current display scale (maximum displayable force), and display mode (orbiter or end effector).

System Computer - A developed, space hardened system computer is being procured from Southwest Research Institute. This computer has the designation of SC-1D and will have the following features and capabilities:

a) 8086 Central Processing Unit (CPU), 8087 Floating Point Co-Processor, 8089 Input/Output processor, 128k bytes Random Access Memory (RAM), 64k bytes Eraseable Programmable Read Only Memory (EPROM) Space, 4 parallel I/O Lines, two 16 bit Direct Memory Access (DMA) ports, one R232 serial port, two programmable timers, power supply, chassis suitable for operation in either the aft flight deck or in the cargo bay.

b) An RS422 serial port for sensor electronics interface in addition to the standard RS232 serial port.

c) Provision for connecting a Transterm 1 keyboard to the RS 232 serial port.

Video Graphics Generator (VGG) - The video graphics generator will be fabricated by Southwest Research Institute to a design developed by Parallax, Inc., Model 600 series. The video signal generated will be compatible with the Space Transportation System Closed Circuit TV System including the ability to genlock. Specific features include:

a) Average pixel writing rate at least 10 million pixels per second including set-up time, for all operations.

b) The capability to accept commands to display and erase alphanumerics, vectors, arcs, rectangles, bar graphs, parallelograms, polygons, and to perform area fill operations.

c) Double buffered display memory for smooth screen updating.

d) Zoom out capability. (This is necessary for performing split-screen displays on the shuttle flight monitors. The image must be shrunk to one-half its normal size and remain in the center of the screen in order for the A7 control panel MUX channels in the aft flight deck area to be effective. The area around the shrunken image need not be accessible by the graphics driver software.)

e) Status signals to the software driver for the VGG which include information on retrace timing - both horizontal and vertical. (This allows display memory buffers to be swapped while they are not being displayed.)

f) Compatibility with STS CCTV system. The VGG will appear to the shuttle videosystem as though it is a CCTV camera, except that camera pan/tilt position information, etc., will not be generated. This compatibility also means that the VGG will be able to genlock (i.e., synchronize) with the master synch generator used to synchronize the CCTV cameras.

DESIGN ISSUES RELATED TO SHUTTLE INTEGRATION

Shuttle safety and non-interference of normal operations are the two dominant integration issues. Material selection to meet off-gassing, flammability and stress corrosion are firm requirements. Because the FTS sensor element will be inserted as a structural element into the RMS mechanical arm, a fail-safe structure design is required. Display of forces and torques is not critical to the normal operation of the RMS so single point failures in the electronics or software can be tolerated in this demonstration system.

The initial concept of the sensor element was a cylinder of 25 cm (10 in) diameter and 12.7 cm (5 in) length. Tolerance stackups of the mechanical arms and Orbiter cargo bay proved that a 12.7 cm growth in mechanical arm length could not be allowed. Therefore, the FTS sensor was constrained to a length of 5.1 cm (2 in). The initial longer length housed the DCE electronics very comfortably. With the two inch requirement, the electronic package had to be reduced to fit into the center cavity or relocated on the exterior sensor diameter. With some effort, the interior mounting seemed do-able and less vulnerable to accidents.

By far, the hardest design choice involved selecting the sensor mechanical range. From the point of view of the demonstration experiment, a narrow range, ± 22.7 kg (50 lb.) force and ± 76 kg-m (550 ft.lb.) moment, with low end sensitivity and accuracy, ± 0.45 kg (1 lb) force and ± 0.14 kg-m (1 ft.lb.) moment, is desired. Safety and operational considerations, however, push the design toward a high load capability and high stiffness. The later effect, obviously, erodes small force/torque measurement capability. To amplify on this issue, the discussion will now focus on launch and mechanical arm operation.

Shuttle Launch

During launch, the mechanical arm is supported at the elbow, wrist and wrist roll joint. The most outboard support is the wrist roll joint which leaves the end effector cantilevered from the flange attachment to the arm. When the FTS sensor is put into the arm it will mate to the wrist roll joint flange and further extend the end effector. The sensor, acting like a stiff spring, then supports the end effector. The launch vibration environment will produce movement of the end effector and flexing of the deflection bars in the FTS. Although the launch environment is not well defined, the launch loads on the sensor have been estimated to be 25 g's. This translates to a moment load on the sensor of 243 kg-m (1760 ft.lb.) about the "y" and "z" axes.

One possible design option is to provide mechanical stops in the sensor so that the deflection beams carry the full load up to ± 76 kg-m, then share the load with an alternate path (mechanical stops) above ± 76 kg-m. This reduces the design range of the deflection beams but introduces a hard non-linearity when vibratory inputs cause stop-to-stop motion. The jarring input to the end effector may be unacceptable.

Mechanical Arm Operation

The sensor adds compliance to the arm and therefore reduces arm stiffness. Reduced stiffness is a concern in the controllability of the arm when handling payloads. Absolute lower stiffness levels are not known but guideline values have been established. A stiff sensor design meets the guideline constraints, a very sensitive instrument does not.

Design Compromise

The RMS/FTS design is proceeding with the conservative option of a stiff sensor. This satisfies Orbiter safety and operational constraints and minimizes cost. To achieve the sensitivity of $\pm$ 0.45 kg of force, reliance is being placed on the extremely high sensitivity of the semiconductor strain gages (better than 1 part in a million). The system will be sensitive to temperature changes as well as electronic drifts. Operational procedures will be involved to make frequent zero load bias adjustments to negate temperature induced and electronic drifts. This approach can only be applied in a space environment when the mechanical arm is inoperative.

REFERENCES

(1) Bejczy, A.K. and Dotson, R.S., A Force-Torque Sensing and Display System for Large Robot Arms, Proceedings of the IEEE Southeastcon '82, Destin, FL, April 4-7, 1982

(2) Bejczy, A.K., Dotson, R.S., Brown, J.W., and Lewis, J.L., Manual Control of Manipulator Forces and Torques Using Graphic Display, Proceedings of the 1982 IEEE International Conference on Cybernetics and Society, Seattle, WA, October 28-30, 1982

(3) Corker, K., Bejczy, A.K., and Rappaport, B., Force/Torque Display for Space Teleoperation Control Experiments and Evaluation, Proceedings of the Annual Manual Conference, Columbus, OH, June 17-19, 1985

ORBITAL FLUID RESUPPLY

TANKER DEVELOPMENT

SATELLITE SERVICES WORKSHOP II

NOVEMBER 7, 1985

JOHN W. GRIFFIN

National Aeronautics and Space Administration

S85-33112

Lyndon B. Johnson Space Center
Houston, Texas 77058

ENGINEERING DIRECTORATE
DESIGN•DEVELOPMENT•TEST

ORBITAL FLUID RESUPPLY - PROGRAMMATIC OBJECTIVES	PROPULSION AND POWER DIVISION	
	JOHN GRIFFIN	

- 0 PROVIDE ECONOMIC ORBITAL FLUID RESUPPLY
  - 0 NASA/DOD/COMMERCIAL VEHICLES
- 0 DEVELOP GENERIC TANKERS
  - 0 ADDRESS TOTAL ORBITAL RESERVICING REQUIREMENTS
  - 0 MAXIMIZE COMMONALITY AND FLEXIBILITY
  - 0 MINIMIZE NUMBER RESUPPLY VEHICLES REQUIRED IN NASA INVENTORY
- 0 SUPPORT SATELLITE REFURBISHMENT/MAINTENANCE SIMULTANEOUS WITH FLUID RESUPPLY

ORBITAL FLUID RESUPPLY - DESIGN OBJECTIVES

JOHN GRIFFIN

- O DESIGN FLEXIBILITY/COMMONALITY
  - O SUPPLY FLUIDS FROM ORBITER TO ORBITING VEHICLE(S)
  - O REMOVE/RECEIVE FLUIDS FROM ORBITING VEHICLE TO ORBITER
  - O OPTIMIZE SIZING - PERMIT GROWTH BY INTERCONNECTING TANKERS
  - O SYSTEM MODIFIABLE
    - O EVA OR AUTOMATIC OPERATION
    - O ORBITAL ATTACHMENT TO SPACE STATION, PLATFORMS, ETC AS FLUID CHANGEOUT MODULE
    - O ORBITAL ATTACHMENT TO OMV/OTV FOR REMOTE RESERVICING
- O LOW OPERATIONAL COST/SHORT TURNAROUND SCHEDULE
  - O MINIMIZE ORBITER/USER INTERFACES - PARTICULARLY COMPUTER INTERFACES
  - O MINIMIZE GROUND CHECKOUT FACILITIES
  - O SOFTWARE VERIFICATION INDEPENDENT OF FLIGHT HARDWARE AND MAJOR FACILITIES
  - O FLY AS "FILLER" PAYLOAD - RELOCATABLE IN PLB ON-ORBIT
- O MINIMIZE DEVELOPMENT COST/RISK
  - O MAXIMIZE USE OF EXISTING CERTIFIED HARDWARE

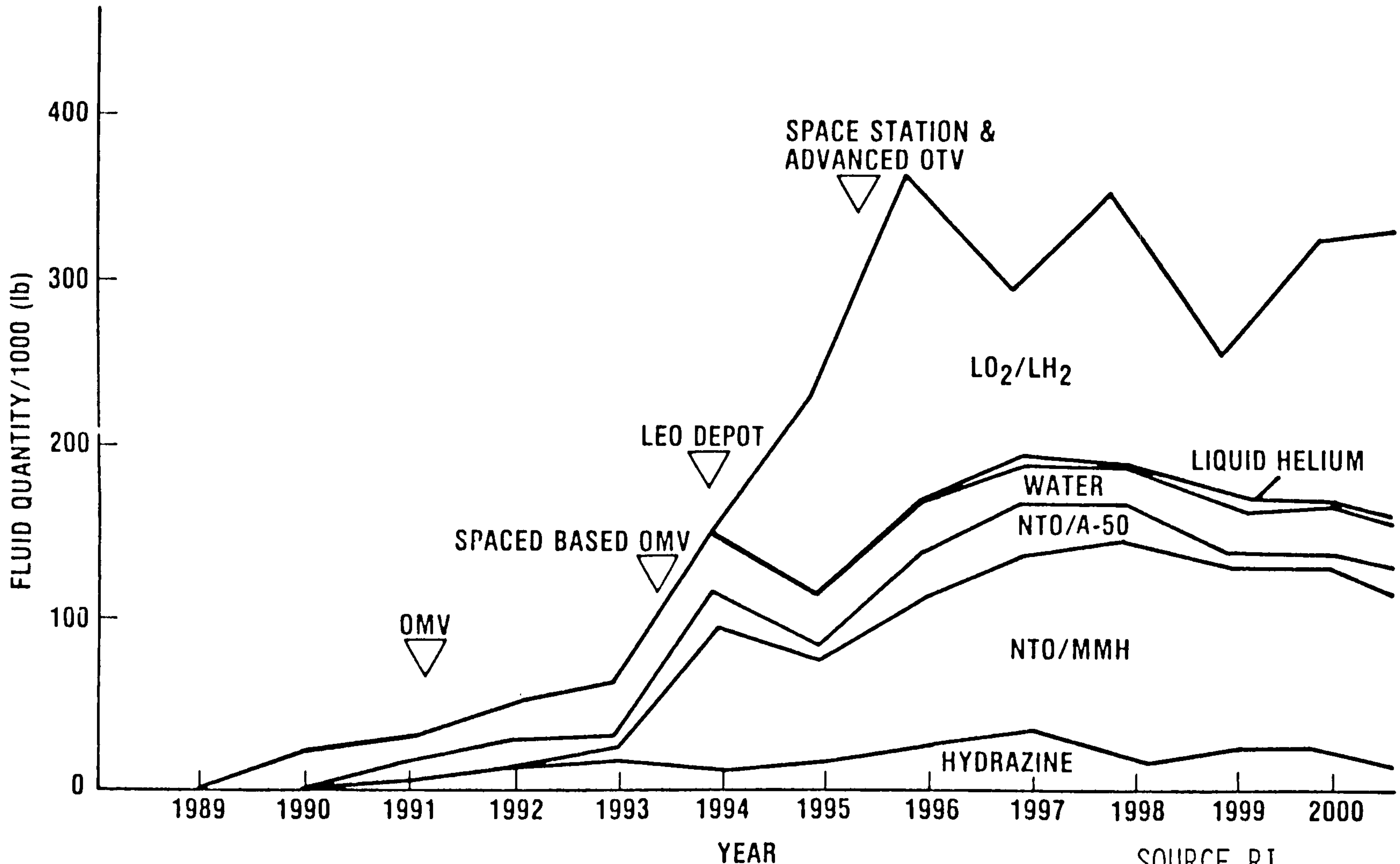
ORBITAL RESERVICING JUSTIFICATION	PROPULSION AND POWER DIVISION
	JOHN GRIFFIN

- O REQUIRED FOR LARGE SYSTEMS BUILT IN ORBIT - NOT RETURNABLE/NOT EXPENDABLE
  - O ESSENTIAL FOR FUTURE DEVELOPMENT OF SPACE
- O CONSERVE COST
  - O FLUIDS LIFE LIMITER FOR MANY SATELLITES - EXTEND LIFE
  - O LOWER EXPENSE THAN RETURN AND RELAUNCH
  - O COMBINED WITH ON-ORBIT MAINTENANCE/REPAIR/UPGRADES
- O REDUCES RISK TO ORBITING SYSTEM - MAJOR RISKS ARE:
  - O LAUNCH ENVIRONMENT
  - O LANDING ENVIRONMENT
  - O GROUND OPERATIONS/ENVIRONMENT EXPOSURE
- O INTRODUCES PAYLOAD DESIGN OPTIONS
  - O LOWER FLIGHT STRUCTURE WEIGHT OR INCREASE PAYLOAD WEIGHT - LAUNCH DRY
  - O REDUCED ENVELOPES/WEIGHTS - LOWER CONSUMABLE REQUIREMENTS

PRELIMINARY ORBITAL RESUPPLY
FLUID REQUIREMENTS

PROPULSION AND POWER DIVISION

JOHN GRIFFIN



SOURCE RI
NAS-8-35618

REPRESENTATIVE FLUID TRANSFER ENGAGEMENTS

PROPULSION AND POWER DIVISION

JOHN GRIFFIN

YEAR

MISSION CATEGORY	90	91	92	93	94	95	96	97	98	99	00	01	TOTAL
MAINTENANCE (SATELLITES)	2	5	9	10	10	10	6	7	3	5	5	5	77
SPACE-BASED OMV				1	4	2	6	1	4	2	2	1	23
SPACE-BASED OTV						4	5	4	5	4	6	5	33
SPACE STATION			4+	4+	4+	4+	4+	4+	4+	4+	4+	4+	40
DOD		3	2	2	3	2	5	7	3	4	4	4	39
LEO TOTAL	2	8	15	17	21	22	26	23	19	19	21	19	212
GEO					1	3	5	6	6	8	10	5	44
GRAND TOTAL	2	8	15	17	22	25	31	29	25	27	31	24	256
CUMULATIVE TOTAL	2	10	25	42	64	89	120	149	174	201	232	256	

SOURCE RI
NAS-8-35618

MAJOR TANKER DESIGN AND SCHEDULE DRIVERS

JOHN GRIFFIN

- 0 MAJOR DESIGN DRIVERS
  - 0 FLUID TYPE
    - 0 EARTH STORABLE LIQUIDS
      - 0 MONOPROPELLANTS
      - 0 BIPROPELLANTS/HIGH PRESSURE GASES
      - 0 OTHERS - DESIGN SATISFIED BY ABOVE
    - 0 HIGH PRESSURE GASES
    - 0 CRYOGENICS
      - 0 SUPERCRITICAL
      - 0 SUBCRITICAL
    - 0 SUPER FLUIDS - HELIUM II
  - 0 SIZE - MASS OF FLUID REQUIRED
- 0 SCHEDULE DRIVERS - SHORT TERM MARKET
  - 0 SATELLITES - PREDOMINATELY EARTH STORABLE - HYDRAZINE
    - 0 GRO-2000 LBS (1990), LEASECRAFT-3000 LBS (1991), UARS-1200 LBS (1991), PROTEUS-1100 LBS (1992), DOD-5000 LB RANGE MAX
  - 0 IOC SPACE STATION (1992) - PRELIMINARY 90 DAY FLUID REQUIREMENTS > 100 LBS
    - 0 EARTH STORABLES: N_2H_4-2320 LBS, WATER-1893 LBS, P.F. WATER-2391 LBS
    - 0 CRYOGENICS: LN_2-2735 LBS, LO_2-631 LBS, ARGON-573 LBS
    - 0 GAS: AIR-143 LBS

TANKER SELECTION FOR INITIAL DEVELOPMENT	PROPULSION AND POWER DIVISION
	JOHN GRIFFIN

- O MONOPROPELLANT (N_2H_4) WITH HIGH PRESSURE GAS RESUPPLY
  - O WIDELY USED - MOST IMMEDIATE SATELLITE APPLICATION
  - O REFERENCE SPACE STATION PROPELLANT
    - O BASIC DESIGN APPLICABLE TO WATER RESUPPLY
  - O SIZING AT 5000 POUND CAPACITY
    - O SATISFIES IMMEDIATE AND GROWTH MARKET
    - O CONSISTENT WITH AVAILABLE HARDWARE
  - O LOGICAL FIRST STEP IN THE ORBITAL TANKER EVOLUTION
  - O COMBINE WITH PARALLEL BIPROPELLANT STUDY - ASSURE COMMONALITY

JSC PROGRAM APPROACH FOR ORBITAL PROPELLANT TANKER DEVELOPMENT	PROPULSION AND POWER DIVISION
	JOHN GRIFFIN

- O OSCRS (ORBITAL SPACECRAFT CONSUMABLES RESUPPLY SYSTEM)
  - O OPERATIONAL MONOPROPELLANT TANKER WITH MAXIMUM COMMONALITY TO A BIPROPELLANT DESIGN
  - O FEASIBILITY ASSESSMENT
    - O STARTED WITH ORS FLIGHT DEMONSTRATION
    - O ESTABLISHED INTERFACES WITH MAJOR USERS/SUPPLIERS
    - O DEDICATED IN-HOUSE STUDIES COMPLETED REFERENCE DESIGN APRIL 1985
  - O PRELIMINARY DESIGN CONTRACT
    - O EIGHT MONTH STUDY TO BE COMPLETED IN MID 1986
      - O THREE PARALLEL CONTRACTS $1.7M TOTAL
    - O DRAFT RFP RELEASED FOR BIDDER/GOVERNMENT REVIEW 3/14/85
    - O RFP RELEASED 6/7/85, PROPOSALS RECEIVED 7/25/85
      - O FIVE BIDDERS
      - O CONTRACT AWARDS EARLY NOVEMBER 1985
    - O USER REQUIREMENTS DEFINITION - ALL FLUIDS
    - O MONOPROPELLANT PRELIMINARY DESIGN/COST ASSESSMENT
    - O BIPROPELLANT CONCEPTUAL DESIGN/COST ASSESSMENT

- OSCRS (CONTINUED)
  - FABRICATION/CERTIFICATION
    - SCHEDULED START 1988 - OPERATIONAL IN 1991
    - AUTHORITY TO PROCEED WITH PLANNING FROM JESS MOORE - MAY 1985
    - NONADVOCATE COST REVIEW COMPLETED JUNE 1985
    - NEW START REQUEST FOR 1988 BUDGET
  - FIRST CUSTOMER GRO (GAMMA RAY OBSERVATORY)
    - INTERCENTER AGREEMENT - JSC/GSFC - MARCH 1985
  - MAJOR SUPPORTING CONTRACTS
    - STANDARDIZED SPACECRAFT REFUELING COUPLING NAS9-17333
      - FLIGHT CERTIFIED MONOPROPELLANT EVA COUPLING DELIVERED MARCH 1986
      - DESIGN APPLICABLE TO BIPROPELLANTS
      - COUPLING AUTOMATION STUDY
- STANDARDIZED AUTOMATIC PROPELLANT RESUPPLY INTERFACE DEVELOPMENT
  - INTEGRATED INTO OSCRS DEVELOPMENT OR SEPARATE CONTRACT
  - PLANNED TO START BY 1988

JSC PROGRAM APPROACH FOR ORBITAL PROPELLANT TANKER DEVELOPMENT	PROPULSION AND POWER DIVISION
	JOHN GRIFFIN

- O BIPROPELLANT TANKER DEVELOPMENT
  - O PRELIMINARY CAPABILITY: 18,000 LBS N_2O_4/MMH
  - O DESIGN ADAPTABLE FOR OMS/RCS PAYLOAD BAY KIT
  - O PRELIMINARY DESIGN ASSESSMENT - COMBINED WITH OSCRS STUDY CONTRACTS
    - O COMPLETE MID 1986
  - O JSC PRELIMINARY PLANNING
    - O INITIATE TECHNOLOGY/HARDWARE/EXPERIMENTS AS DICTATED BY OSCRS STUDY
    - O PRELIMINARY DESIGN STUDY CONTRACT(S) 1987 TIME FRAME
    - O PHASE C/D - START FY 1989 - IOC 1992

- O CRYOGENIC TANKERS
  - O PRELIMINARY PLANNING UNDERWAY AT JSC
    - O FEASIBILITY/SIZING STUDY CONTRACT LATE 1986 - EARLY 1987
  - O IOC REQUIREMENT VARIATIONS FROM 1992 TO 1996

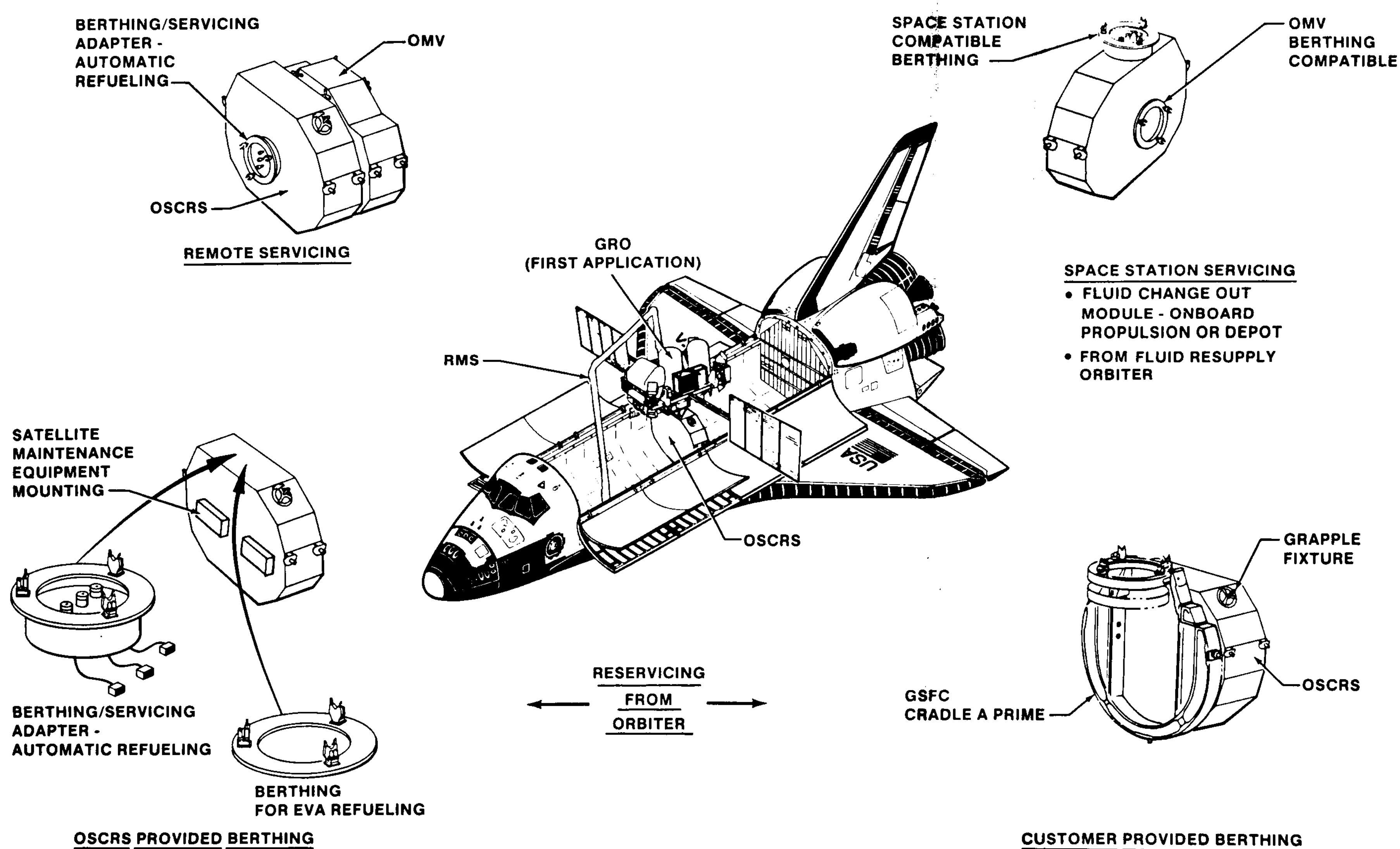
- O SUPER FLUID TANKERS
  - O DEVELOPMENT FOLLOW CRYOGENIC TANKERS
  - O IOC PROJECTED 1996 - 1999: TECHNOLOGY AND FUNDING LIMITED

OSCRS OVERVIEW

OSCRS STUDY CONTRACT-MAJOR BASELINE REQUIREMENTS

- O MEET GENERAL DESIGN OBJECTIVES PREVIOUSLY DEFINED
- O MEET NHB 1700.7A - PAYLOADS SAFETY CRITERIA
- O PRELIMINARY FLUID QUANTITIES (OPTIMIZED DURING STUDY)
  - O MONOPROPELLANT - 5000 LBS N_2H_4 - HIGHER WITH INTERCONNECT
    - 500 LBS GN_2
  - O BI PROPELLANT - 18,000 LBS N_2O_4/MMH
  - O FLUID RESERVICING CAPABILITY INDEPENDENT OF SATELLITE PMD
- O REDUNDANCY - FAIL OP/FAIL SAFE
- O CONTROL - VIA COMMANDS TO SELF CONTAINED COMPUTER
  - O ORBITER BASED - OPERATION INDEPENDENT OF GPC
    - O AFT FLT DECK "PLUG IN" TERMINAL WITH GRAPHIC DATA DISPLAY
  - O SPACE BASED - ADDRESSED THRU USER VEHICLE
- O NO VENTING OF RAW PROPELLANTS
- O EMERGENCY SEPARATION CAPABILITY
  - O NON EVA SEPARATION - ALL ELECTRICAL/FLUID/MECHANICAL INTERFACES

OSCRS OPERATIONAL CAPABILITY



- 0 TASK 1 - REQUIREMENTS DEFINITION (COMPLETE 2 MONTHS FROM ATP)
  - 0 USER REQUIREMENTS DEFINITION TO CY2010 - ALL FLUIDS
    - 0 USER BASE
      - 0 SPACECRAFT THAT ORBITER CAN REACH
      - 0 SPACECRAFT THAT CAN RENEDEZVOUS WITH ORBITER-SELF PROPELLED, OMV, ETC
      - 0 OMV, SPACE STATION, ORBITER OMS/RCS PAYLOAD BAY KIT
      - 0 ETR (EASTERN TEST RANGE) AND WTR (WESTERN TEST RANGE)
  - 0 ORBITER/GROUND FACILITIES/CREW INTERFACES
- 0 TASK 2 - MONOPROPELLANT SYSTEM PRELIMINARY DESIGN (COMPLETE 5 MONTHS FROM ATP)
  - 0 SYSTEM/HARDWARE/SOFTWARE/OPERATIONAL TRADE STUDES
  - 0 PRELIMINARY DESIGN: HARDWARE, SOFTWARE, GSE
    - 0 SPECIFICATION/PROGRAM PLAN/SCHEDULES/COST EST.
- 0 TASK 3 - BIPROPELLANT SYSTEM CONCEPTUAL DESIGN (COMPLETE 5 MONTHS FROM ATP)
  - 0 SYSTEM/HARDWARE/SOFTWARE/OPERATIONAL TRADE STUDIES
  - 0 CONCEPTUAL DESIGN
  - 0 ASSESS/IMPLEMENT COMMONALITY - MONOPROPELLANT TO BIPROPELLANT
- 0 TASK 4 - MONOPROPELLANT SYSTEM PRELIMINARY DESIGN DEVELOPMENT (COMPLETE 8 MONTHS FROM ATP)
  - 0 EXTENSION OF TASK ANALYSIS/DESIGN EFFORT
- 0 TASK 5 - BIPROPELLANT SYSTEM PRELIMINARY DESIGN (COMPLETE 8 MONTHS FROM ATP)
  - 0 EXTENSION OF TASK 3 ANALYSIS EFFORT

SUMMARY/CONCLUSIONS	PROPULSION AND POWER DIVISION	
	JOHN GRIFFIN	

- O ORBITAL RESUPPLY OF FLUIDS IS ESSENTIAL TO DEVELOPMENT OF SPACE

- O NASA COMMITTED TO ORBITAL FLUID RESUPPLY
  - O MEET ITS INTERNAL NEEDS
  - O AS A CUSTOMER SERVICE

- O MULTIFUNCTION TANKERS - KEY TO MINIMIZING COST

- O OSCRS - FIRST GENERATION OPERATIONAL TANKER
  - O MONOPROPELLANT AND HIGH PRESSURE GAS RESUPPLY
  - O PHASE "B" UNDER CONTRACT IN NOVEMBER 1985
  - O IOC PROJECTED IN 1991

- O SUBSEQUENT TANKER DEVELOPMENTS IN PLANNING
  - O BIPROPELLANTS
  - O CRYOGENICS
  - O SUPER FLUID - HEII

SATELLITE SERVICING SAFETY CONSIDERATIONS

by

Gregg J. Baumer

Johnson Space Center
Safety Division

The presence of man in space has expanded the opportunities for successful commercial, scientific, and national defense space ventures . With the Space Shuttle we are commuting into space in a shirt-sleeve environment on a routine basis. We are conducting experiments in the areas of materials processing, biomedical research, drug manufacturing, crystal growth, earth resources, atmospheric research, astronomy, and many others. We are deploying, retrieving, servicing, and repairing satellites. In the near future, we will be building a permanent orbiting space station. In order, however, to obtain the benefits of an astronaut in space, we must take certain measures to assure his safety. We cannot just build a destruct system and push a button when things go wrong.

There will always be safety hazards present in manned spaceflight operations, but accidents can be controlled by eliminating or reducing the risks involved. In past NASA programs, safety and mission success were achieved by direct NASA involvement in the design and development of all flight hardware with total control over risk management. In our present era, NASA has relinquished its control over payload design, and has given the responsibility for payload mission success to the payload organization where it rightfully belongs. Likewise, the responsibility for payload safety has also been delegated to the payload organization. NASA, however, has retained an overview responsibility to assure that all identified hazards have been controlled in accordance with NASA safety policy and requirements.

NHB 1700.7, Revision A, "Safety Policy and Requirements for Payloads Using the Space Transportation System (STS)," dated December 9, 1980, establishes the safety requirements for all STS payloads and their ground support equipment. The requirements are intended to protect flight and ground personnel, the STS, other payloads, and the general public from payload-related hazards. The document contains technical and system safety requirements that are general in nature. The intent of those requirements is to minimize NASA involvement in payload design process, while assuring safety.

Payload organizations have been given freedom to select any design or operational procedure that meets the general safety requirements of NHB 1700.7A and controls identified payload hazards. Safety review panels have been established at JSC and KSC to review the payload for adequate safety implementation. The NASA document that defines the safety review process by which the payload organizations interface with the JSC and KSC safety panels is JSC 13830, Revision A, "Implementation Procedures for STS Payloads System Safety Requirements," dated May 16, 1983.

The end result of the safety review process is the safety certification of the payload at the flight readiness review for its assigned mission. This certification shall be supported by a safety assessment report prepared by the payload organization and concurred in by NASA. To assist the payload organization in the interpretation of the safety requirements and to provide timely safety inputs during the development of the payload the flight and ground safety panels normally conduct four safety reviews. These reviews are phased to correspond to the baselining of a payload's conceptual, preliminary, and final designs, and to the delivery of the as-built hardware.

The depth and number of formal safety reviews may be tailored. Reviews may be combined, depending upon design maturity, design complexity, and STS experience of the payload organization. Where hazard potential is low or the payload design is simple with a minimum of STS interfaces, the formal safety review meeting may be cancelled in lieu of the receipt of a complete safety data package. Where the payload is a standard element (i.e., the payload is a reflight payload or one payload of a series that has already been reviewed), the review process may even be simplified to a recertification of the baseline safety assessment without any formal reviews or comprehensive data packages.

The question is often asked by payload organizations, "If my design is this or that, what are the applicable safety requirements?" The answer quite simply is NHB 1700.7A. The same response would be given if the question was, "What requirements must a payload meet if it is to be retrieved or serviced on orbit?"

The technical safety requirements of NHB 1700.7A can be grouped into two broad categories--those that relate to failure tolerance and those that relate to designing for minimum risk. Failure tolerance is the basic safety requirement that shall be used to control payload hazards. Of special importance is the application of failure tolerance to hazardous functions. Hazardous functions are operational payload events (i.e., motor firings, appendage deployments, stage separations, RF transmission, etc.) where inadvertent operations or loss may result in a hazard. Monitoring and safing apply to control of hazardous functions. Requirements to "design-for-minimum-risk" are used to control payload hazards that relate to structures, materials compatibility, flammability, toxic offgassing, pyrotechnic devices, radioactive materials, and flammable atmospheres.

Failure tolerance requirements shall be used to control all hazards, except in those cases where a "design-for-minimum risk" safety requirement applies. Payload critical hazards shall be controlled such that no single failure or operator error can result in: damage to STS equipment; a nondisabling personnel injury; or the use of unscheduled safing procedures that affect operations of the Orbiter or another payload. Likewise, catastrophic hazards shall be controlled such that no combination of two failures or operator errors can result in a disabling or fatal personnel injury; or the loss of the Orbiter, ground facilities, or STS equipment. Where hazardous functions are involved, inhibit and monitoring requirements will apply. Critical hazardous functions require two inhibits, whereas catastrophic hazardous functions require three inhibits. Monitoring of inhibits is normally required for only catastrophic hazardous functions.

In order to achieve a safe payload, it is the inherent responsibility of the payload organization to conduct detailed component and system level safety analyses. Events or conditions which can result in a risk or accident involving the STS or personnel, should be reported as hazards. If they affect only the success of a payload mission, they are not hazards to the STS.

The purpose of the STS safety review process is to review the payload organization's safety assessment and implementation of the NASA safety policy and requirements. As stated earlier, the safety requirements of NHB 1700.7A are general in nature. This was done deliberately to give the payload organizations maximum control over their design. Because of this generality, questions often arise that seek the safety panel's interpretation of those requirements as they apply to the control of identified hazards of specific designs.

Assistance is available to STS customers. A safety engineer from the Payload Safety Branch at JSC, code NS2, is assigned to each STS payload. This engineer will serve as the single point of contact for safety. He is available for pre-review consultation, issue/question discussion, and to coordinate technical assistance on specific issues. In addition, document JSC 18798, "Interpretations of STS Payload Safety Requirements," is available. This document is a published collection of safety interpretations relative to specific payload designs. These interpretations may be applied to all payloads that utilize similar designs.

Satellite retrieval, servicing, and repair are not new to the STS. The JSC and KSC safety panels have conducted safety reviews for a number of such satellite missions: SPAS 01 (Shuttle Pallet Satellite), LDEF (Long Duration Exposure Facility), ORS (Orbital Refueling System), SMRM (Solar Maximum Repair Mission), HS-376 Spacecraft Retrieval, Leasat (Leased Satellite) Salvage Mission, HST (Hubble Space Telescope), GRO (Gamma Ray Observatory), and Landsat. All of these satellites and experiments were reviewed for compliance with NASA safety policy and requirements as defined in NHB 1700.7A. Most of these satellites share generic type hazards that would apply to most deployable STS payloads. These include:

a. Failure of primary or secondary structure that may result in debris colliding with and damaging the STS. Causal factors include: inadequate strength capability to handle STS loads; propagation of undetected flaws; physical damage during manufacturing, transportation or handling; stress corrosion, etc.

b. Inadvertent firing of solid propellant rocket motors or liquid propellant engines that may result in heat damage, contamination, or collision with the Orbiter. Causal factors include crew error, software failures, inhibit failures, inadvertent commanding, etc.

c. Inadvertent deployment or separation of payload appendages, stages, or other hardware that may collide with the STS, or prevent safe return of the STS. Causal factors are similar to "b" and also include structural failures, etc.

d. Inadvertent turn on of payload transmitter antenna systems that may cause the Orbiter to be exposed to hazardous levels of RF radiation. Causal factors are similar to "b" and "c".

e. Payload fires that may damage the STS. Causal factors include excessive use of flammable materials that create hazardous flame propagation paths to the Orbiter and failure to isolate or control potential ignition sources.

f. Structural failure (overpressure) of fluid systems that may cause damage to the Orbiter from flying debris,contamination, and fire. Causal factors include regulator failures, overheating, propellant decomposition, freezing, incompatible materials, inadequate ultimate factors of safety, etc.

g. Toxic contaminants in the Orbiter cabin that may injure the flight crew. Causal factors include use of materials that offgas excessive organic contaminants, or the leakage of toxic fluids or vapors from payload hardware carried in the crew cabin.

h. Battery failures that may propagate damage to the STS. Causal factors include inadvertent charging, cell reversal, overdischarge, short circuits, electrolyte leakage, etc.

i. A payload or its airborne support equipment that may violate the Orbiter payload bay door envelope and prevent closure of the doors. Causal factors include mechanical failures of drive mechanisms, electrical failures, structural failures, etc.

j. Release of hazardous fluids that may cause fire, contamination, and other damage to the Orbiter. Causal factors include leakage from components, seal failures, overpressurization of fluid system, loss of thermal control, etc.

The above list is far from complete and is only meant to be a sampling of the hazards encountered. There are many other generic hazards that relate to radiation, pyrotechnics, electrical/electronic systems, cryogenics, sealed containers, hazard detection and safing, and command systems.

The hazards that are unique to satellite servicing and repair activities are as varied as the designs of the payloads and the servicing or repair activity to be accomplished. Some such hazards might include:

a. Damage to the EMU (extravehicular mobility unit) that may cause injury to crewmen. Causal factors include payload sharp edges or protrusions, snag and pinch points, hot surfaces, inadequate work envelope, task difficulty, inproper tools, lack of or inadequate restraints or mobility aids, etc.

b. Structural failure of payload safety critical hardware may be caused by crew induced loads.

c. In-flight servicing of electrical hardware may cause electrical sneak circuits that activate hazardous payload functions.

d. Material degradation due to on-orbit environmental conditions and age life may reduce the reliability of existing hazard controls, and result in occurrence of a hazardous event.

e. Unplanned termination of servicing or repair activities may prevent safe satellite deployment and/or loss of Orbiter entry capability. Causal factors include any in-flight emergency that would require termination of extravehicular activity combined with an unsafe payload configuration for deployment or return to Earth. Payload berthing hardware and repair scenarios should be designed and planned to eliminate or severely limit exposure to this type of hazard.

f. Propellant servicing is an inherently hazardous procedure. Leakage, overpressure, overtemperature, adiabatic compression detonation, contamination of EVA crewmen are among the hazards to be controlled.

g. Satellite servicing and repair must address safety of in-flight operations with a satellite that has experienced electrical and mechanical failures. Preflight assessment of the anomally must be made to assure safety critical and hazardous systems are deactivated or configured to prevent a hazard to the STS.

h. Payload radioactive materials may injure EVA crewmen during repair activity. Location, quantity, and strength of all payload radioactive materials must be evaluated prior to any repair activity. Especially important would be assessment of shielding during repair activity that is conducted on internal parts of the satellite.

As stated earlier, the list of unique hazards are as varied as the designs of the satellites. It should be re-emphasized that it is the responsibility of the payload organization to conduct a comprehensive safety assessment of its payload. The purpose of the STS payload safety review panel is not to conduct that safety assessment, but rather to assist the payload organization in interpretation of the safety requirements, and to assess the implementation of those requirements. The requirements of NHB 1700.7A allow great latitude to the payload community to develop unique designs to control hazards. Through careful and close coordination between the NSTS and the payload community, accidents related to satellite servicing and repair can be controlled or eliminated.

LASER DOCKING SENSOR (LDS)

NASA-JSC
HARRY O. ERWIN
NOVEMBER 1985

LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
	HARRY O. ERWIN	11/85

LDS OUTLINE

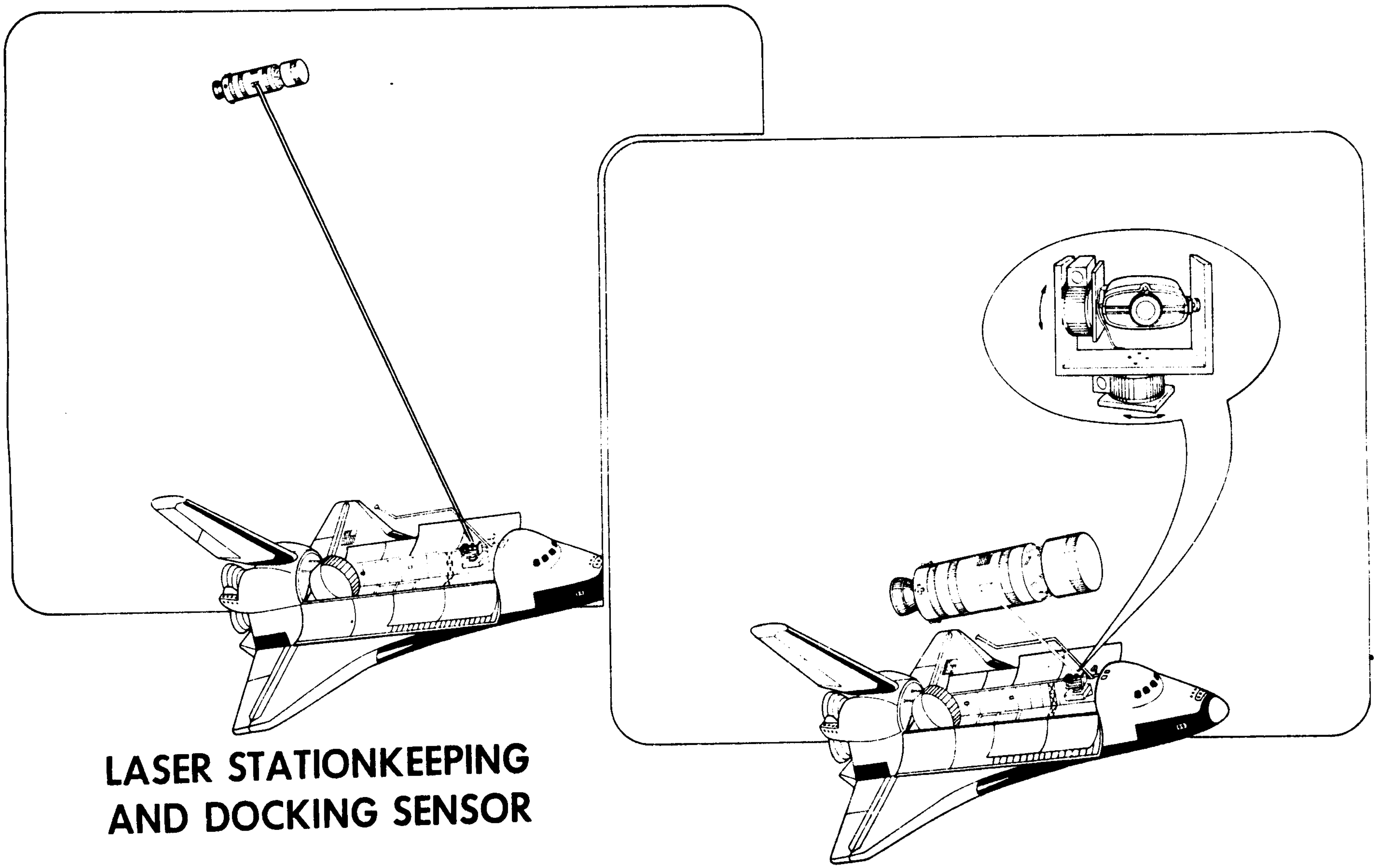
- OVERVIEW/DESCRIPTION
- REQUIREMENTS
- BENEFITS
- CURRENT RESEARCH
  - SINGLE REFLECTOR APPROACH
  - DEAD RECKONING OPTOELECTRONIC INTELLIGENT DOCKING (DROID)
  - RENDEZVOUS AND DOCKING TRACKER (RDT) SYSTEM
  - IN HOUSE EFFORT
- CONCLUSIONS

LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
	HARRY O. ERWIN	11/85

LDS OVERVIEW/DESCRIPTION

- PASSIVE DOCKING AIDS (REFLECTORS) ON TARGET
- TRANSMITTER (LASER BEAM) AND RECEIVER ON INTERCEPTOR VEHICLE
- DETERMINES RELATIVE POSITION AND ATTITUDE
- PERFORMS STATIONKEEPING AND DOCKING
- AUGMENTS/REPLACES VISUAL TRACKING
- EVENTUALLY WILL TOTALLY AUTOMATE RENDEZVOUS AND DOCKING

LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
	HARRY O. ERWIN	11/85



LASER STATIONKEEPING AND DOCKING SENSOR

LDS REQUIREMENTS

- STANDARD CONFIGURATION FOR PAYLOAD MOUNTED TRACKING AIDS NEEDED SOON
- OPERATE WITHIN RANGE OF ABOUT 1 KM
- AUTOMATIC SOFT DOCKING
- ACCURATELY MEASURE RELATIVE POSITION, ATTITUDE AND RATES
- LOW CONTACT FORCES FOR SENSITIVE EXPERIMENTS
- ACCURACY AT LONGER DISTANCES TO MINIMIZE PLUME IMPINGEMENT PROBLEMS

LDS BENEFITS

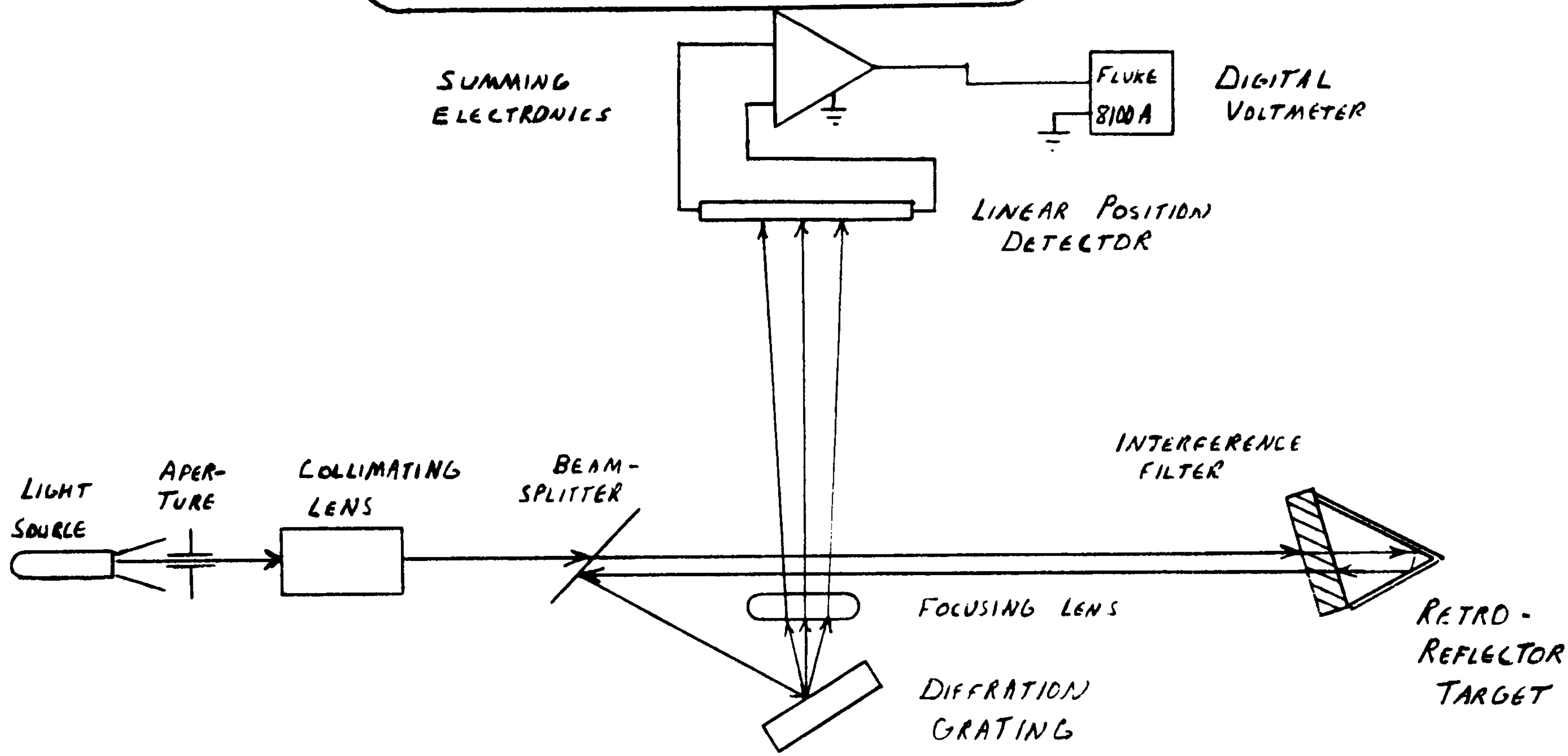
- LIGHT WEIGHT AND SMALL SIZE (USE ON SMALLER VEHICLES)
- GREATER ACCURACY THAN CONVENTIONAL SYSTEMS
- MINIMIZES CONTACT FORCES AND MOMENTS
- CONSERVES FUEL (USES ORBITAL MECHANICS VS BREAKING)
- RELIEVES CREW FOR OTHER DUTIES
- MINIMIZES PLUME IMPINGEMENT AND ENVIRONMENT CONTAMINATION
- QUICKER DOCKING
- STANDARDIZES DOCKING
- LOW POWER (NO POWER FROM TARGET)
- SPINOFFS TO ROBOTICS, ETC.
- MINIMIZES PERTURBATION OF TARGET VEHICLE
- IMPROVES SAFETY - REDUCES HUMAN ERROR
- FREES KU-BAND FOR DATA TRANSMISSION

SINGLE REFLECTOR APPROACH

- MCDONNELL DOUGLAS ASTRONAUTICS COMPANY
- 3 - AXIS REMOTE ATTITUDE SENSING
  -- SINGLE AXIS LAB DEMONSTRATION USES COLLIMATED WHITE LIGHT
- SINGLE RETROREFLECTOR WITH INTERFERENCE FILTER
- REFLECTED LIGHT WAVELENGTH DETERMINED BY ANGLE OF INCIDENCE TO INTERFERENCE FILTER
- REFLECTED LIGHT SPREAD OUT BY DIFFRACTION GRATING, AND STRIKES LINEAR POSITION DETECTOR
- ANGULAR RESOLUTION 0.5 DEGREE
- LAB DEMONSTRATION HAD 2 DEGREE RESOLUTION

LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
	HARRY O. ERWIN	11/85

ATTITUDE SENSOR TEST SETUP



DROID SYSTEM

- DEAD RECKONING OPTOELECTRONIC INTELLIGENT DOCKING
- ENERGY OPTICS, INC
- TOTALLY AUTONOMOUS (MULTIPLE MICROPROCESSORS)
- PROVIDES TARGET INFORMATION DIRECTLY TO HOST COMPUTER
- USES EXISTING TECHNOLOGIES
- CONCENTRATES ON SHORT RANGES (<1000M)
- ANTICIPATES USING GPS (GLOBAL POSITIONING SYSTEM)
- OPERATES ON SINGLE LASER WAVELENGTH
- ALL-SOLID STATE (EMITTERS AND DETECTORS)
- NO MOVING PARTS (MIRRORS, BEAM STEERERS, FOCUSING, ETC)
- DOCKS PASSIVE TARGETS (RETROREFLECTORS AND DOCKING IMAGE PLATE)
  - -- CAN INTERFACE WITH OTHER ACTIVE SYSTEMS SUCH AS ANOTHER DROID SYSTEM
- USES MULTIPLE SENSORS-DECREASES SENSITIVITY TO AMBIENT LIGHT

DROID CONTINUED

- 3 SENSOR CONFIGURATIONS
  - -- PULSED LIDAR (1000 M TO 30 M)
  - -- CW LIDAR (30 M TO 3 M)
  - -- CCD LIDAR (3 M TO 0.2 M)
  - -- ONLY ONE IN CONTROL AT ANY TIME
  - -- AUTOMATICALLY RECONFIGURES AS RANGE CLOSES

LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
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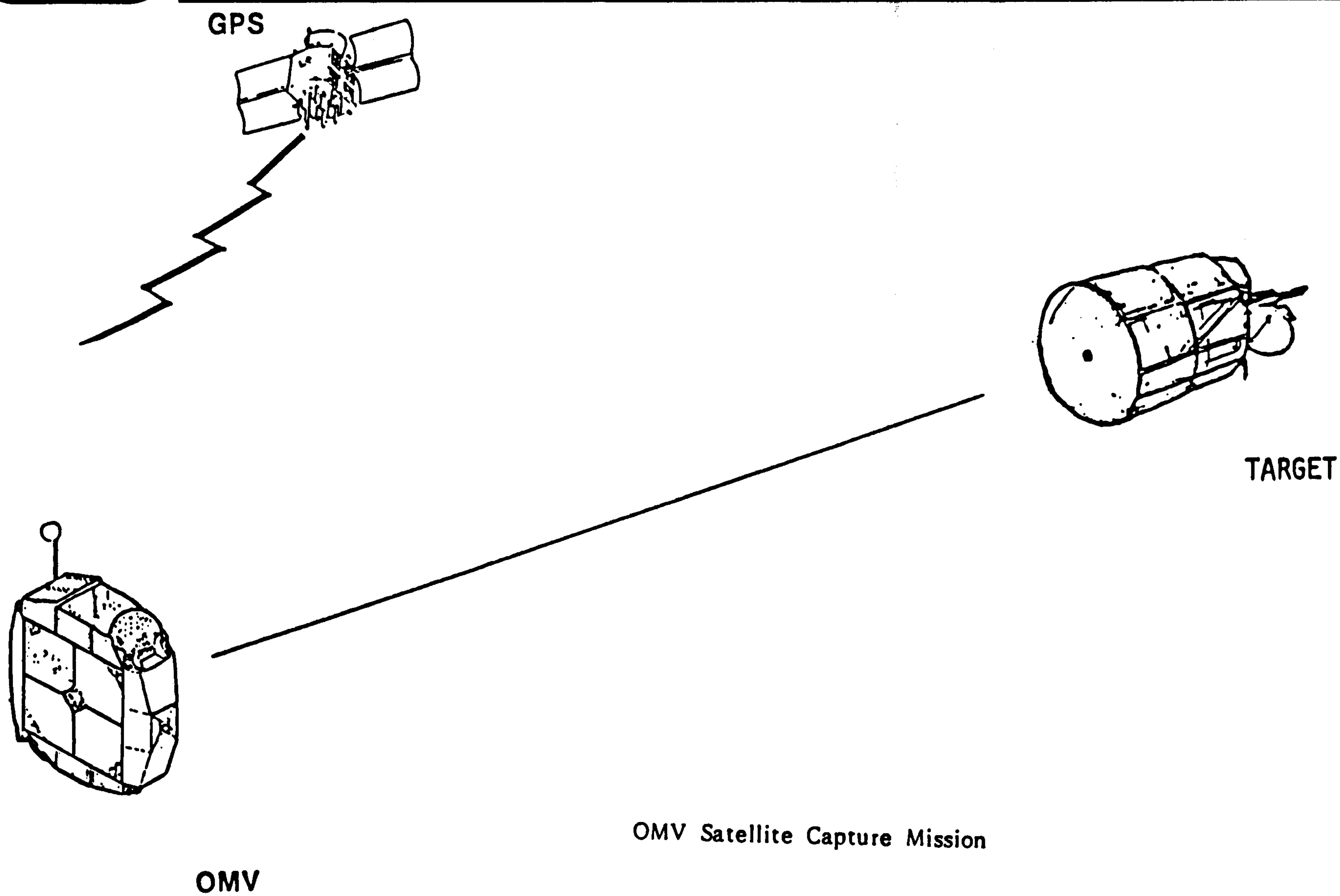
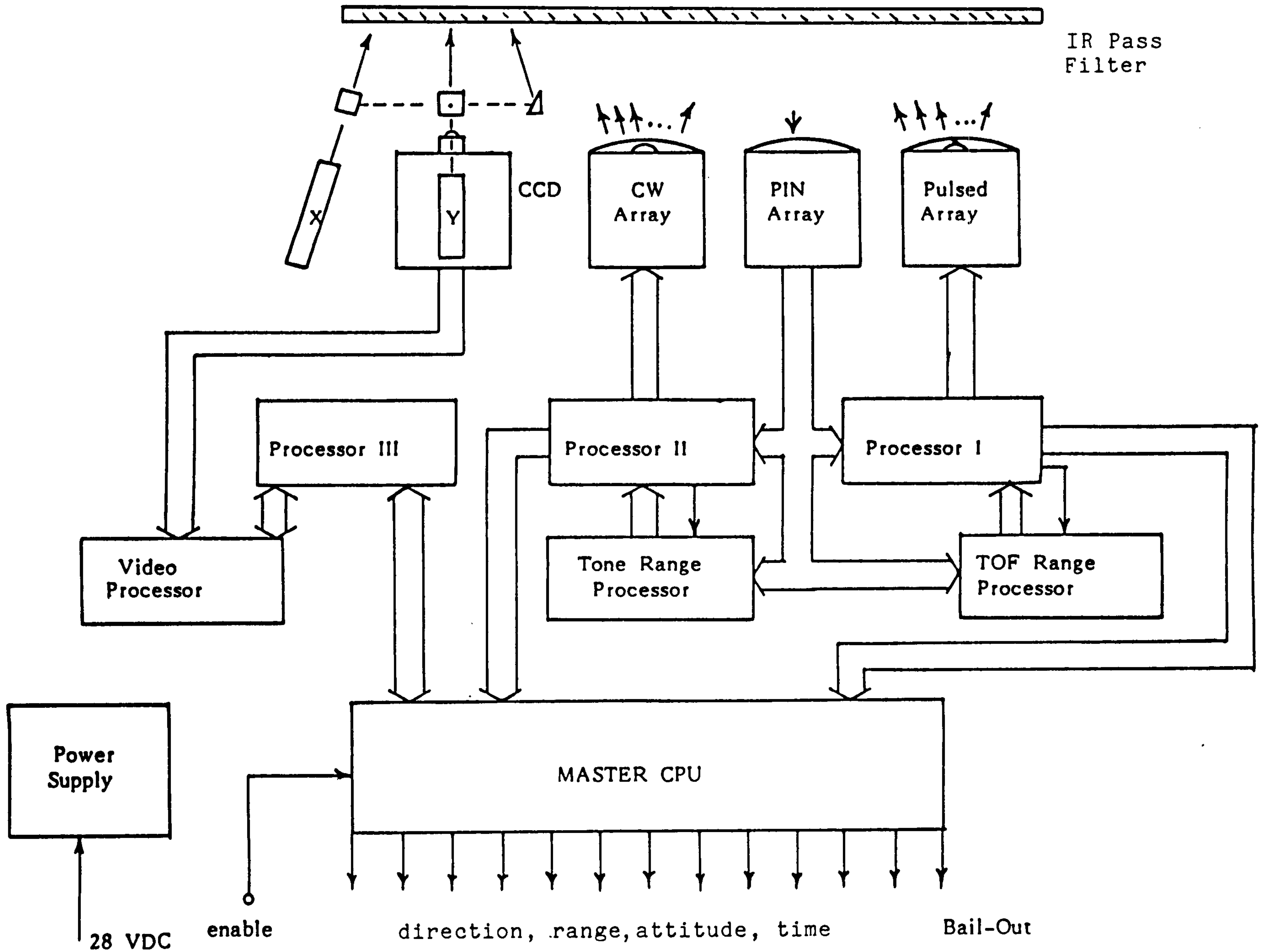


OMV Satellite Capture Mission



DROID System Block Diagram

LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
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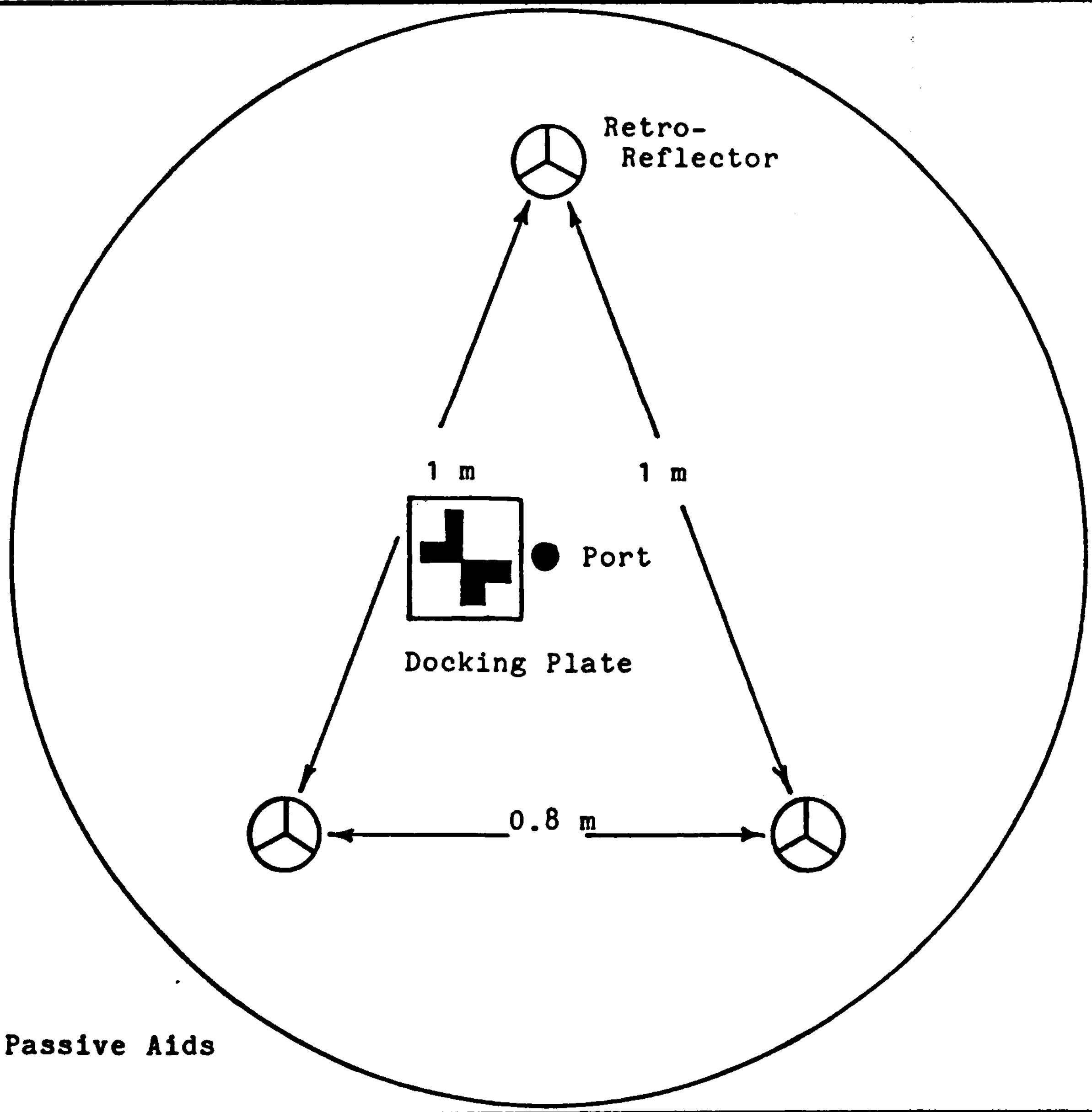

Target Passive Aids

LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
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RENDEZVUUS AND DOCKING TRACKER (RDT) SYSTEM

- BALL AEROSPACE SYSTEMS DIV.
- RANGE AND ATTITUDE DETERMINATION
- PASSIVE REFLECTORS ON TARGET VEHICLE
- SEQUENTIAL ILLUMINATOR SYSTEM ON CHASE VEHICLE
- STEREOGRAPHIC TRACKING OF MULTIPLE TARGETS
- USES CID (CHARGED INJECTION DEVICE) SENSOR
- 3 TRACKING MODES
  - -- SEQUENTIAL LASER PULSES (1000 M TO 100 M)
  - -- SIMULTANEOUS LASER OPERATION (100 M TO 10 M)
  - -- 4TH SOURCE ILLUMINATES TIGHTER SET OF TARGETS (<10M)
- NEWTON'S METHOD USED TO INTERACTIVELY FIND POSITION
- USES PRECISION STAR TRACKER AT HEART
- USES SPIRAL SEARCH FOR ACQUISITION
- LED'S USED FOR SHORT RANGE
- LASER DIODES USED FOR LONG RANGE

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LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
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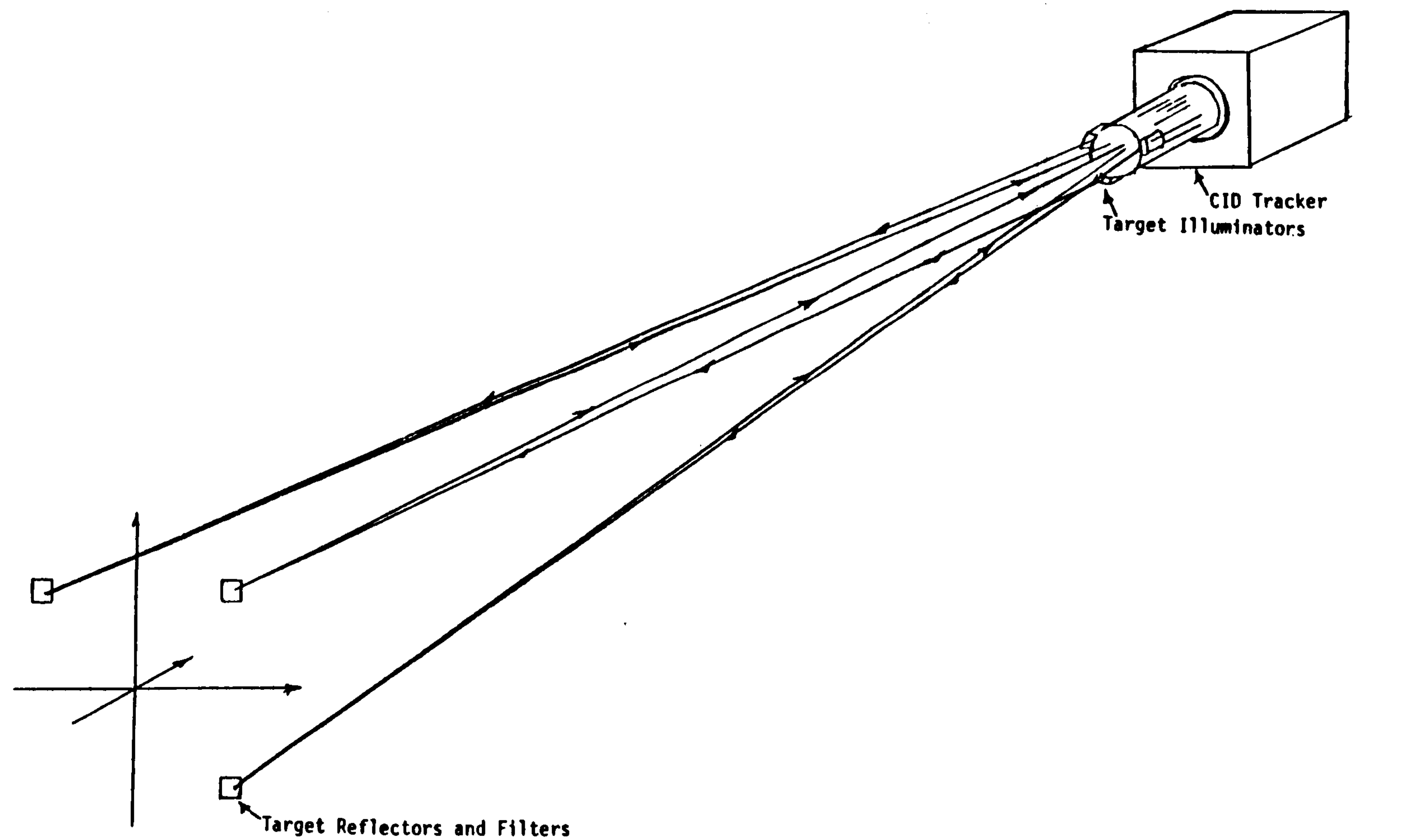
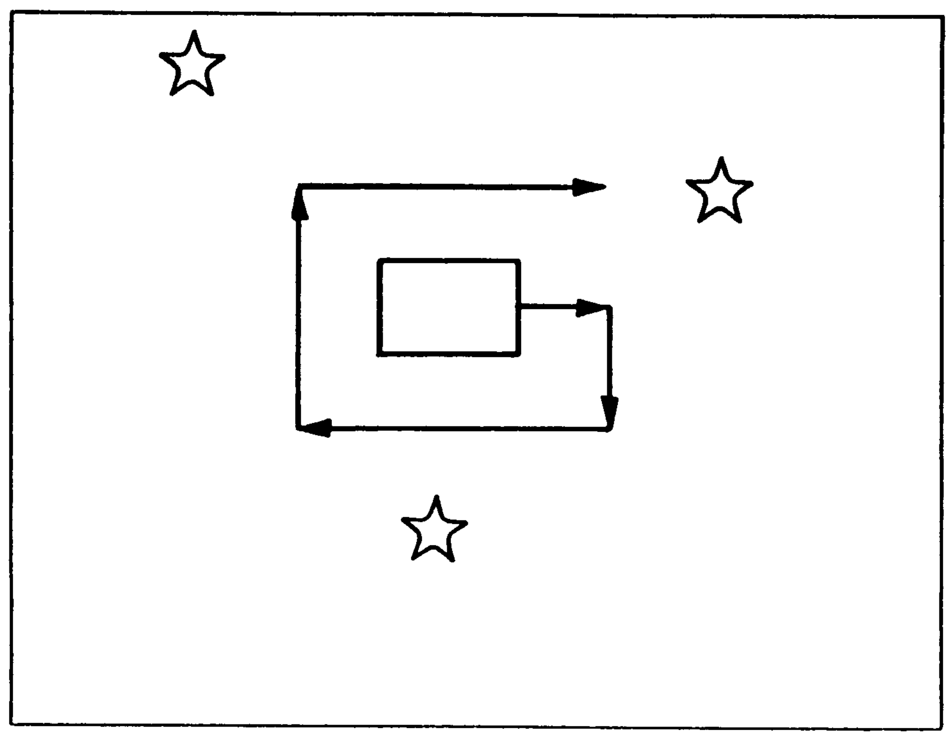


RENDEZVOUS AND DOCKING TRACKER (RDT) SYSTEM

- 16 × 16 PIXEL PARTITION
- SPIRAL SEARCH



LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
	HARRY O. ERWIN	11/85

IN-HOUSE EFFORT

- NASA/JSC
- IR LASER DIODE TRANSMITTER
- GALVANOMETER BEAM STEERERS (150 - 300 HZ RATE)
- PMT (PHOTOMULTIPLIER TUBE) IMAGE DISSECTOR, CCD ARRAY, OR AVALANCHE PHOTODIODE RECEIVER
- JOINT NASA/CONTRACTOR EFFORT TO PRODUCE WORKING BREADBOARD MODEL
- 3 TONE RANGER
  - -- 3750 HZ TONE RANGES TO 40 KM (10 M RESOLUTION)
  - -- 375 KHZ TONE RANGES TO 400 M (10 CM RESOLUTION)
  - -- 15 MHZ TONE RANGES TO 10 M (2 MM RESOLUTION)
- 3 RETROREFLECTORS
- POSSIBLE FLIGHT EXPERIMENT

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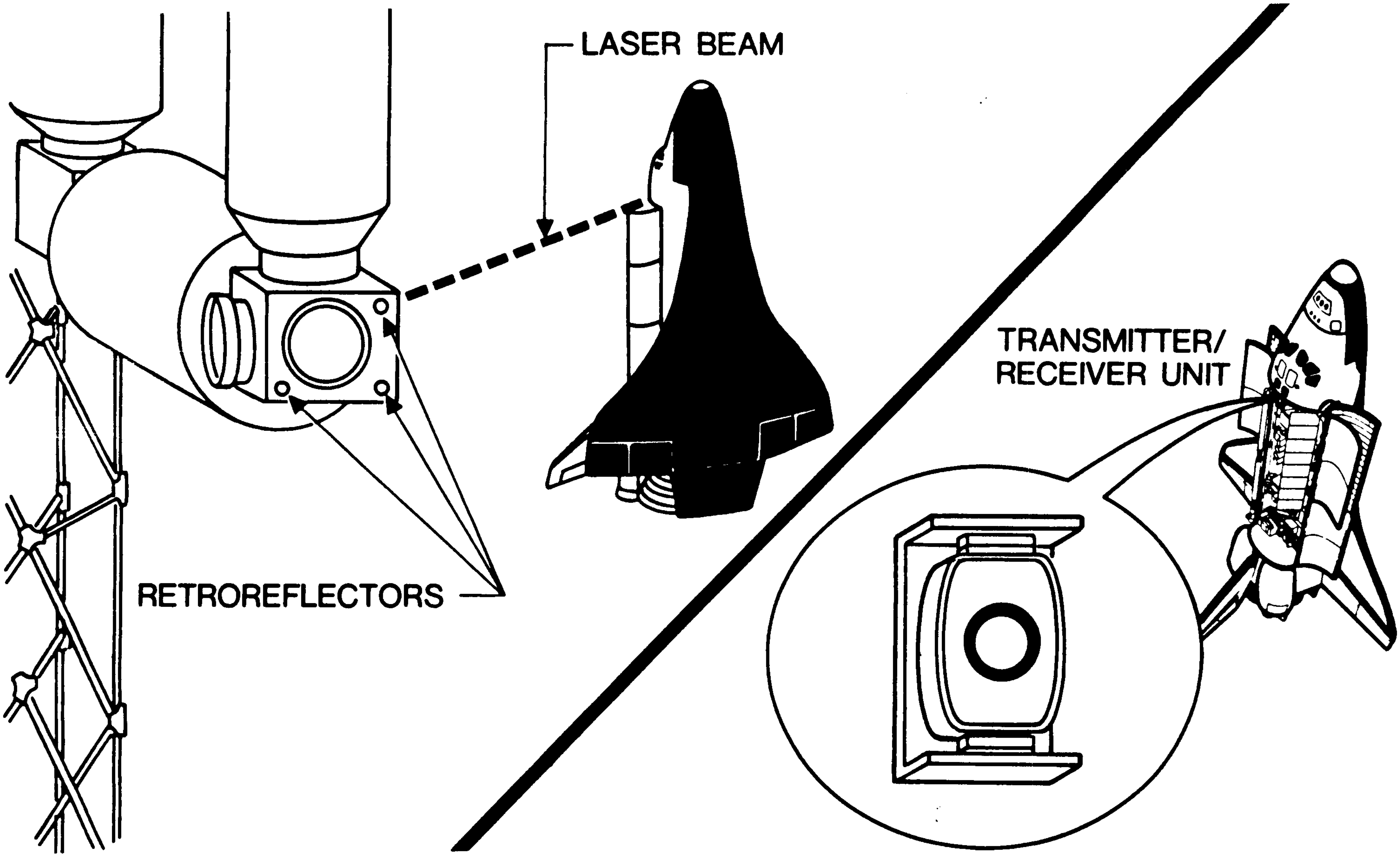
LASER DOCKING SENSOR	TCD/TRACKING TECHNIQUES BRANCH	
	HARRY O. ERWIN	11/85



CONCLUSIONS

- PLAN TO BREADBOARD AND EVALUATE EACH SYSTEM
- WINNING EVALUATION TO BE FLIGHT TESTED ON ORBITER
- COMBINE BEST FEATURES OF EACH SYSTEM TO PRODUCE HYBRID

NEED FOR FLIGHT EXPERIMENT

- o DEMONSTRATE AND EVALUATE A CAPABILITY TO TRACK A PASSIVE ORBITAL TARGET SPACECRAFT WITH SUFFICIENT ACCURACY TO ENABLE SOFT DOCKING WITH MINIMAL THRUSTING NEAR THE TARGET VEHICLE.
- o TEST AND EVALUATE LDS WITH REALISTIC ENVIRONMENT AND SCENARIOS.
  - DYNAMICS AND RANGES
  - LIGHTING (SUN, MOON, EARTH, STARS, GLINT, SHADOWS)
  - VACUUM (NO REFRACTION/SCATTERING BY ATMOSPHERE)
  - PLUME EFFECTS
  - THERMAL EFFECTS
- o OBTAIN REALISTIC PERFORMANCE DATA AND CREW REPORTS NEEDED TO REFINE DESIGN FOR FINAL OPERATIONAL SYSTEM.

SUMMARY RESULTS OF THE
RENDEZVOUS AND PROXIMITY OPERATIONS WORKSHOP
JSC: February 19 - 22, 1985

Dr. Roscoe Lee
TRW Defense Systems Group
Houston, Texas

SATELLITE SERVICES WORKSHOP II

NASA Lyndon B. Johnson Space Center

November 6 - 8, 1985

INTRODUCTION

The Rendezvous and Proximity Operations Workshop, which was held at the NASA Lyndon B. Johnson Space Center on 19 - 22 February 1985, was jointly sponsored by the NASA Office of Space Flight, the NASA Office of Aeronautics and Space Technology, the NASA Goddard Space Flight Center, the NASA Johnson Space Center, the NASA Marshall Space Center, and the Jet Propulsion Laboratory. NASA had initiated activities to understand the current experience base of rendezvous and proximity operations and to establish focused emphasis for future applications. Specifically, the NASA plans were to define technologies, advanced developments, and operations to accommodate users with cost effectivity and to define flight experiments and demonstrations to obtain early operational experience and build confidence in the user/customer community. The Workshop was selected as the mechanism for focused interchange of information, planning, and potential program initiatives among the NASA, DoD, industry, and academic communities.

Rendezvous and proximity operations are synergistic with satellite services. These operations provide the transportation, delivery, retrieval, maneuvering, EVA, and teleoperations functions, which are necessary to support satellite services, both for retrieved systems and for in-situ operations. Indeed, the Workshop was structured to provide perspectives from the users, operators, and builders of systems for rendezvous and proximity operations, to initiate an integrated system definition process.

WORKSHOP OBJECTIVES

- Provide a historical perspective of what has been accomplished to date, with respect to rendezvous and proximity operations.
- Establish an interchange of information concerning ongoing technology and advanced development activities related to rendezvous and proximity operations.
- Assess technology and operational readiness to meet requirements; identify areas of high payoff for future technology and advanced development expenditures.
- Establish a long-range planning framework for the development and demonstration of rendezvous and proximity operation capabilities

needed in the future.

- o Define the minimum capabilites and self-protection that a user must provide to ensure successful Space Transportation System rendezvous and proximity operations.

WORKSHOP STATISTICS

The response to the Workshop was overwhelming, with over 650 applications received for the available 500 spaces. 505 attendees were registered with the following distribution: 137 NASA; 13 DoD; 319 domestic industry; 2 academia; 3 ESA/ESTEC; and 31 international industry.

WORKSHOP AGENDA

Three opening plenary sessions were conducted to provide perspectives from the system operators, users, and builders. Subsequently, eight parallel session were directed toward specific areas of technology, advanced development, or systems engineering. A wrap-up plenary session provided summary reports by the parallel session chairmen, a summary of the Workshop results, and discussion of post-Workshop activities. The papers presented in each session will be listed during the highlights summary of this report.

The presentors at the Workshop represented the following categories: 33 NASA presentations; 40 domestic industry/academia presentations (20 different organizations); and 8 international presentations.

SESSION HIGHLIGHTS

SESSION 1 - Introductory Plenary Session: Co-Chairmen: W. Smith/NASA HQ and K. Cox/JSC

Session Themes:

- o Describe rendezvous and proximity operations and technical and programmatic concerns.

- Identify lesson learned to date
- Describe the operator's perspectives on rendezvous and proximity operations

1. Keynote Address: Space Operations National Infrastructure - W. Smith/NASA HQ
2. Role of Rendezvous and Proximity Operations in Integrated Orbital Operations - K. Cox/NASA JSC
3. Mission Control Center Perspectives - J. Cox/NASA JSC
4. Flight Crew Perspectives - Solar Maximum Mission - D. Walker/NASA JSC
5. Mission Operations Perspectives - K. Young/NASA JSC

In the Keynote Address, Mr. Smith identified the need to develop capabilities to effectively support integrated space operations, incorporating the NSTS, Space Station Program elements, user spacecrafts and space systems, and space support systems.

Dr. Cox identified the major challenges of early integration across multiple programs and projects, the need to enable high productivity in parallel operations of multiple space systems with varying maturities, and a drive to enhance accommodation of users of the space infrastructure. Some of the recommended major program thrusts were to develop rendezvous and proximity operation services keyed to user requirements, develop pragmatic, operationally cost-effective autonomy/automation techniques, and establish a plan for evolving space traffic contol development among elements of the space fleet.

Highlights of the operator perspectives were: that proximity operations will tend to be custom designed for each payload; propellant will always be a premium; large masses can be man-handled with sufficient planning and handholds; and high fidelity simulations are crucial to technique development and crew training. From a flight crew perspective, close integration of man and machine (e.g., crewman/Orbiter, crewman/RMS, and crewman/MMU) has been demonstrated and can facilitate satellite servicing.

Session 2 - User Requirements Plenary Session: Co-Chairmen: J. Purcell/GSFC and J. Steincamp/MSFC

Session Themes:

- Describe actual and potential solutions to first-use application problems and worst-case conditions.
- Emphasize viable division of responsibilities between user and STS.
- Aggregate user requirements to influence development of proper and timely rendezvous and proximity operation capabilities.

1. Rendezvous and Proximity Operations during EURECA Missions - E. Graf/ESA
2. SPARTAN Rendezvous - S. Lambros/GSFC
3. Solar Maximum Repair Mission - K. Grady/GSFC
4. Leasecraft/Materials Processing - Rendezvous, Prox Ops, and Costs - R. O'Brien/Fairchild
5. Space Telescope and AXAF - T. Styczynski/LMSC
6. Advanced X-Ray Astrophysical Facility (AXAF) - Servicing Mission Concepts - J. Steincamp/MSFC
7. Contamination Effects during Rendezvous and Prox Ops - L. Leger/JSC

User systems (e.g., EURECA, SPARTAN, Leasecraft, Space Telescope, and AXAF) are currently negotiating NSTS interfaces, which are resulting in significant trades between the operator and user of the NSTS. There was much concern on the requirements for cooperative equipment on the user system, which are potentially big cost drivers. Also, the users indicated that they see a need for increased reliability in retrieval schedules and that the uncertain and changing manifesting schedule result in large uncertainties in the design of the user flight systems and their flight plans. There is a general desire to minimize the shutdown of user systems during rendezvous and proximity operations, since such interruptions tend to reduce the system reliability.

Session 3 - Planned Systems Capabilities Plenary Session: Co-Chairmen: G. Butler/MDAC and W. Huber/MSFC

Session Theme:

- Review major systems and emerging service capabilities

1. STS-Based Services - A. Louviere/JSC

2. Tethered Satellite System - A. Lorenzoni/PSN/CNR and C. Rupp/MSFC
3. OMV: The Key to Satellite Services - A. Stephenson/TRW
4. OMV Servicing Capability - F. Bergonz/MMA
5. Orbital Maneuvering Vehicle - R. French/LTV
6. Orbit Transfer Vehicle - R. Austin/MSFC
7. Future Space and Ground Network Capability - J. Cooley/GSFC

The STS on-orbit services were graded as follows: delivery is mature; retrieval and satellite servicing is immature and evolving.

The OMV is a vital link in integrated orbital operations. It will enable satellite servicing and retrieval/redeployment from the Shuttle or Space Station. The OMV integrated orbital services systems will include a docking mechanism, storage racks, and manipulator arm. Plans are evolving for ground demonstrations, cargo-bay demonstrations, and free-flight verification of remote servicing.

Session 4 - Space Traffic Control: Co-Chairmen: R. Lee/TRW and P. Kramer/JSC

Session Themes:

- Define what constitutes space traffic control
- Establish perspectives on drivers/needs for space traffic control
- Discuss concepts, strategies, technologies, and equipment for space traffic control.

1. Issues and Constraints driving Space Traffic Control Policies - Overview - R. Lee/TRW
2. Impact of Space Traffic Level on Space Transportation System Fleet Size - D. Morris/LaRC
3. Operational Control Zones - A. DuPont and B. Nader/JSC
4. Proximity Operations - Antenna Pattern Coverage for Space Traffic Control - T. Campbell/LaRC
5. Formation Flying Techniques - D. Henderson/TRW
6. Trajectory Control Rendezvous - F. Clark/LEMSCO
7. Mars Orbit Automated Rendezvous and Docking System - R. Anderson/LinCom
8. Rendezvous and G&N Technology Needs - A. Klumpp/JPL

Space traffic control was defined to be the active operations of rendezvous, formation flying, EVA, tether control, and docking; flight planning of these activities; the tracking, monitoring, and control of active, interacting systems; and the tracking and monitoring of debris and predictions of collisions. There is a need for early development of space traffic control policies across the operator, user, and builder communities and across programs and projects.

No show stoppers were identified, but there is a need to focus on end-to-end integration of systems and operations and on the development of automatic rendezvous and proximity operations GN&C systems.

Session 5A - Mechanisms: Co-chairmen: R. Olsen/GAC and C. Cornelius/MSFC

Session Themes:

- o Review current state-of-the-art and planned future mechanisms, which will support rendezvous and proximity operations.
- o Define requirements for additional mechanism capabilities to support integrated rendezvous and proximity operations.
- o Identify the rendezvous and proximity operations vehicle and system interfaces from mechanisms perspective.

1. Manipulators for Berthing/Docking - R. Schappell/MMC
2. The Payload Deployment/Retrieval Performance of Space Shuttle Remote Manipulator System - P. Nguyen/SPAR
3. Transportation - Space Station Interfaces: Mechanical and Structural - I. MacConochie/LaRC
4. Berthing Mechanisms - G. Burns/MDAC
5. Remote Satellite Servicing - D. Scott/MSFC
6. Satellite Capture Mechanisms and Simulations - N. Shields/Essex Corp.

The heritage and current activities in STS, OMV, and Station mechanisms were reviewed, including berthing/docking and capture mechanisms, RMS performance and capabilities, and satellite servicing, simulation studies, and facilities. It was agreed that there is an urgent need to identify and develop subsystems to satisfy user needs in terms of satellite capture and control mechanisms; docking and berthing mechanisms; servicing systems such as manipulator arms, tools, and end effectors; and payload/module handling and attachment systems. Satellite servicing tasks will be simplified by the development of man-tended automated mechanisms and systems.

Session 5B - Man and/or Machine Operations: Co-Chairmen: G. Hanley/RI and J. Anderson/NASA HQ

Session Themes:

- o Roles and technology needs for automated rendezvous and proximity operations
- o Human roles and technology needs and rendezvous and proximity operations

1. Automated Rendezvous and Docking Systems - J. Tietz/MMC
2. Automated Satellite Servicing On Orbit - H. Meissinger/TRW
3. A Japanese Effort in Space Robotics - K. Machida/Electrotechnical Laboratory
4. Telepresence Systems: Analysis and Neutral Buoyancy Verification - D. Akin/MIT Instrument Laboratory
5. Increased Intelligence and Autonomy in Space Operations - A. Bejczy/JPL

In situ servicing by teleoperations is feasible only if transmission delays are reasonably small (i.e., 0.25 to 1.0 sec). Automation will enhance or enable satellite servicing functions, with evolutionary expansion of the role of artificial intelligence. Early concepts of tailored human/machine combinations are emerging, with provisions for evolution. Such combinations include manned teleoperators, small manned spacecraft, and telerobotics.

Session 6 - Laser and Radio Frequency Systems and Technology: - Co-Chairmen: J. Paul/Hughes, F. Allario/LaRC, and K. Krishen/JSC

Session Themes:

- o Review state-of-the-art and future projections of components and subsystem technology applicable to tracking for rendezvous and proximity operations.
- o Promote exchange of ideas between NASA and industry on needs for tracking and ranging for proximity operations.
- o Identify overall system requirements for laser and RF radar systems for proximity operations.

1. Review of Laser and RF Systems - H. Erwin/JSC
2. Highly Stable Nd:YAG Laser Technology for Range and Range Rate Measurements - R. Byer/Stanford University
3. A Prospectus for Semiconductor and Advanced Solid-State Laser Technology for Tracking - P. Moulton/Schwartz Electro-optics
4. Long-Lifetime Stable CO_2 Lasers for Radar - R. Hess/LaRC
5. Review of Millimeter-Wave Component State-of-the-Art Technology - D. Ball/Hughes
6. A Millimeter Wave Range/Range Rate System for OMV - E. Feagler/Bendix
7. Rendezvous and Proximity Sensor Candidates - B. Kunkel/MBB-ERNO
8. Effects of Tethers on Rendezvous and Prox Ops - J. Carroll/Cal State Institute

"Ball park" requirements sensor systems were established for Space Station proximity operations. Further definition is needed for the selection of sensors on the Space Station and orbiting vehicles/systems. The requirements on lasers were defined to be 100 milliwatt average power; range capability of 0 to 1 km; range accuracy of 1 mm (with retroreflectors on the target vehicle); range rate accuracy of 1% of range rate; and an angular resolution of 0.01 deg. For millimeter-wave radar, the requirements were defined to be a frequency in the range of 100 GHz; range of 3 to 200 feet; range accuracy of 1 mm; and range rate accuracy of 0.1 fps. A cooperative retroreflector target cross-section is necessary.

Session 7 - Displays and Human Factors: Chairmen: J. Hatfield/LaRC

Session Theme:

- Focus on advanced enabling control/display technologies, information requirements and display formats, control/display integration, work-station configuations, and human factor guidelines in the rendezvous and proximity operations tasks.

1. The Geometrical and Symbolic Content of a Perspective Display for Commercial Aviation - Implications for Spatial Proximity Operations Displays - S. Ellis/ARC
2. Use of Perspective Displays for Situational Awareness in Space Station Proximity Operations - M. McGreevey/ARC
3. Advanced R&T Base Control/Display Technology with Potential for Rendezvous and Proximity Operations - R. Parrish/LaRC
4. System Integration of the New High Technology Work Station - J. Hussey/GAC

5. Application of Voice Interactive Systems - C. Moore/VERAC
6. THURIS - The Human Role in Space - S. Hall/MSFC
7. Stereo Video and Display Systems - N. Shields/Essex Corp.

It was concluded that there is a need to accelerate the development of language interfaces to standardized graphics; high-performance, highly-integrated graphics hardware; information management methods/technologies; voice recognition and synthesis; and 3-D stereo display concepts. A D&C test bed and dynamic environment simulator are required to validate D&C designs.

Session 8 - System Integration and Development Support: Co-Chairmen: A. Nathan/GAC and F. Vinz/MSFC

Session Theme:

- o Identify techniques for orderly progression of concepts from system requirements and design to fully developed Space Station capabilities.

1. OMV Utilization for Large Observatory Mission Support - F. Swalley/MSFC
2. Interaction between the Space Transportation System and the Operations and Technology Requirements - D. Eide/LaRC
3. System Integration Methodologies - F. Vinz/MSFC
4. Crew Systems, Engineering Space Telescope Maintenance and Repair; Lessons Learned - F. Sanders/MSFC
5. Systems Integration/Simulation Tests - A. Nathan/GAC
6. Rendezvous and Docking Technology Development for European Missions - W. Fehse/ESTEC-ESA
7. Demonstration Mission for Autonomous Rendezvous with the EURECA Platform - P. Natenbruk/MBB-ERNO

The value of front-end systems integration in the design process was identified for such planned activities as flight system, RMS, and EVA operations with space telescope servicing. Simulations are critical to system integration and support early identification of design and interface problems and allow a focus on candidate common elements.

Session 9 - Controls: Co-Chairmen: R. Iwens/TRW and J. Dahlgren/JPL

Session Themes:

- o Define principal operations requiring control and review technology status.
- o Establish perspectives on drivers/needs for advanced controls and define major unsolved problems.
- o Discuss concepts, strategies, and technologies required for effective control, including cost, performance, and schedule.

1. Overview of Control Technology related to Rendezvous and Proximity Operations - R. Ogleview/RI
2. RCS Architectural Issues for Space Vehicle Rendezvous and Proximity Operations - E. Bergmann/CSDL
3. Retrieval of Tumbling Satellites - A. Ray/MMC
4. Thruster-Assisted Tethers for Proximity Operations - R. Laskin/JPL
5. Controllability Issues for Remote-Piloted Vehicles - R. Dabney/MSFC
6. Simulation of the Last Meters of Approach and Docking - B. Claudinon/MATRA

Enhanced funding and broader application focus were recommended for a proximity operations sensor, integration of proximity operations, and remote manipulators. The role of man in proximity operations will impact the controls designs. For example, design trades are associated with MMU operations versus teleoperations versus complete automation.

Session 10 - Optical Sensors: Co-Chairmen: S. Russak/MMC and R. Breckenridge/LaRC

Session Themes:

- o Review current optical sensor technology used in Shuttle proximity operations and define needs for future Space Station proximity operations.
- o Review ongoing technology developments in solid-state imaging devices and camera systems.
- o Present advanced concepts for optical sensors, including smart and adaptive sensors with emphasis on subsystem information extraction.

The evolution of smart/integrated sensors is being stimulated by private industry. Smart sensors with man-in-the-loop will benefit rendezvous and proximity operations. The capabilities of television, image intensifiers, and new diode laser structures were described. It was recommended that the constraints imposed by optical sensors be eliminated or reduced. For example, the implementation of simple passive target aids and reduction of the sensitivity of the optical sensors to interference from the background or target would be of significant benefit. It is clear that data processing should be integrated into the sensors.

<u>Session 11 - Data Systems:</u> Co-Chairmen: J. Patel/Honeywell and K. Wallgren/NASA HQ

Session Theme:

- o Discuss data systems technology that is supportive of rendezvous and proximity operations.

1. The Effect of Rendezvous and Proximity Operations on the Space Station Data Management System Architecture - W. Clark/TRW
2. The Effect of Rendezvous and Proximity Operations on the Space Station Data Management System Architecture - G. Love/MDAC
3. High Performance Networks for Combined Video, Voice, and Data - N. Murray/LaRC
4. Video Imaging Processing for Space Station Operations - K. Hoyme/Honeywell
5. Signal Processing Elements and Computers for Proximity Operations - H. Benz/LaRC
6. Optical Mass Storage Systems - G. Claffie/RCA

The presentations covered a broad field of data systems technology applicable to the Space Station, including architecture, networks, processors, storage, and software. It was determined that proximity operations is not a data systems driver. However, digital video could be bandwidth taxing (e.g., 100 mbps). It is predicted that network bandwidths of greater than 500 mbps will be available for the Station.

POST-WORKSHOP ACTIVITIES

A need was identified to establish, within NASA, a long-range planning framework for rendezvous and proximity operations. The plan should be cooperative among the principal participants, respond to needs, promote timely technology development, address the significant integration issues, make effective use of development and test facilities, take advantage of flight test opportunities, and reflect and influence new initiatives. Subsequent follow-on meetings were anticipated, including this Satellite Services Workshop. There will be spin-off, special subject workshops and or technical meetings and a future conference/workshop will be held to revisit rendezvous and proximity operations.

SUMMARY

The Workshop was considered to be highly successful with widespread participation of both domestic and international organizations.

Key perspectives from the operators were that: (1) automation/autonomy should be integrated into manned space operations and (2) significant constraints associated with safety and mission success will continue.

Key perspectives from the users were that: (1) servicing capabilities need to be developed or matured, with reliable scheduling of these services; and (2) overhead (hardware and software) to user systems for such services should be minimized.

Key perspectives from the builders were that: (1) integration of engineering design and system operations is required to effectively (cost and productivity) meet user needs and operating constraints; and (2) there are a number of enhancing technologies which must be developed or focused.

A major challenge will be the effective merging of multiple programs, projects, and vehicles of varying maturity into integrated orbital operations. The use of mature systems for inflight verification or demonstration of needed technologies and advanced developments should be promoted; several specific recommendations were presented for Orbiter experiments and demonstrations directed toward integrated orbital operations. The Space Station, itself, could eventually serve as a technology/advanced development test bed.

SATELLITE SERVICING AT SPACE STATION

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SATELLITE SERVICING AT SPACE STATION

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INTRODUCTION

The first on-orbit repair mission on a spacecraft that was designed to be repaired was the Solar Maximum Repair Mission (SMRM) conducted in April of 1984. With the successful completion of the Solar Maximum repair and subsequent spacecraft checkout, the era of the throw-away spacecraft ended. This on-orbit spacecraft repair was a milestone for the concept of on-orbit servicing, and is the cornerstone on which we are building our plans for the 1990's. This paper will describe the facilities available for the servicing and maintenance of the Space Station platforms and attached payloads, and satellites brought to the Space Station. The basic configuration of the service facilities will be addressed, as well as the type of services to be provided by the Space Station. A satellite servicing scenario will be described to provide an understanding of how the Servicing Facility and its capabilities will be utilized.

The Space Station will open up a new era in the ability to service on-orbit. It will give us the opportunity to use space in more rational, economical and imaginative ways than we have up to now. With the orbiting maintenance base called the core station, and its associated Orbital Maneuvering Vehicle (OMV), we will be able to reach, retrieve, and service the satellites, attached payloads, and platforms in an unprecedented manner. By performing periodic servicing, replenishing consumables, changing or enhancing instruments, and updating components and subsystems in orbit, we will lengthen the life and scientific and applications utility of the space satellite missions from a few years to up to 20 years.

SERVICING OBJECTIVES

Servicing has the primary objective of extending the operational life of satellites, platforms, or Space Station attached payloads by repair, refurbishment, resupply, replacement of failed or worn out units, or by changeout of payload instruments. Another important objective is the assembly in orbit of space systems too large or heavy to be launched on a single Shuttle flight. On-orbit assembly will also find application for those systems, such as instruments containing cryogenically cooled detectors, which normally would be constructed on the ground and launched by the Shuttle. However, a substantial improvement in performance can be realized by eliminating the requirement to design the integrated system to withstand Shuttle launch vibration loads. Instrument subsystems would be housed in separate shipping containers, launched, and then assembled at the Space Station. With reduced structural interfaces, the thermal coupling can be significantly improved giving substantially better detector performance.

Servicing of platforms and other satellites that are outside the reach of the Space Station and Space Station-based OMV's and OTV's, such as polar platforms and satellites in low-altitude orbits not coplanar with the Station, will require either the use of the Shuttle Orbiter or servicing in-situ. Although not directly involved with the Space Station Servicing Facility, the servicing techniques and hardware items used in the Shuttle-based and in-situ servicing will, in many instances, be identical to those used in Station-based missions with equipment commonality being an objective for cost effectiveness.

NEAR-TERM SERVICING MISSIONS

Examples of some near-term science and application missions and their projected servicing needs excerpted from the Langley Data Base are as follows:

- o Hubble Space Telescope (HST)
  - Axial and Radial Science Instrument (SI) replacement
  - Solar array replacement
  - Battery replacement
  - Support System Module ORU replacement
- o Space Infrared Telescope Facility (SIRTF)
  - Cryogen replenishment
  - Battery replacement
  - Electronics assembly replacement or repair
- o Gamma Ray Observatory (GRO)
  - Propulsion System refueling
  - Power and Data Management Module (ORU) replacement
- o Advanced X-Ray Astrophysics Facility (AXAF)
  - Science Instrument (SI) replacement
  - Cryogen replenishment
  - Spacecraft ORU replacement
- o Solar Terrestrial Physics Observatory (STO)
  - SI Repair or replacement
  - Gas bottle replacement
  - Subsystem (ORU) replacement or repair

Of these servicing missions, the Hubble Space Telescope is the major NASA on-orbit servicing activity between 1985 and the early Space Station era. This facility, designed to last for two decades, will be launched in 1986. It is designed not only with replaceable critical spacecraft subsystems, but also the five instruments are in-orbit replaceable for observatory upgrading over the life of the mission. The original plans were on-orbit repair and update at approximately two and one-half years, and to return to earth at five years for major refurbishment. Present plans are to repair and refurbish at the Space Station. Any servicing required prior to Space Station activation will be conducted by the STS Orbiter.

SPACE STATION FACILITY FOR SATELLITE SERVICING

The Servicing Facility defined by the NASA Initial Operating Capability (IOC) Reference Configuration consists of 6 principle elements. They are:

- Servicing Bay
- Satellite Storage Bay
- Refueling Bay
- Instrument Storage Area
- Tool/ORU Storage Area
- Work Station in Pressurized Lab Module

Figure 1 is a drawing of the IOC Reference Configuration showing the servicing facility elements and their locations. These elements when taken together and supported by a manipulator system (MRMS), the Manned Maneuvering Unit (MMU), the OMV, standard tools, and EVA support equipment, make up the Space Station servicing capability.

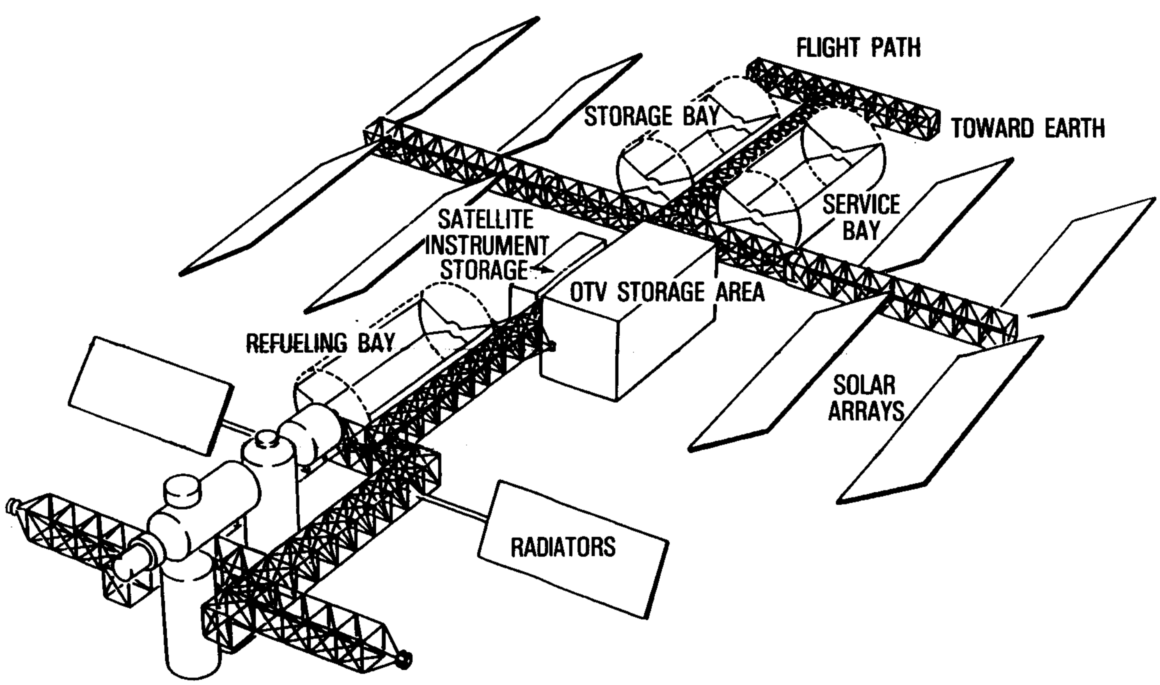

Figure 1 - IOC Reference Configuration of Space Station Servicing Facility

Details including the functional characteristics of each of the elements follow:

Servicing Bay

The Servicing Bay is approximately 9 meters (30 feet) in diameter, and 21 meters (70 feet) in length. The volume enclosed allows for berthing a 4.5 meter (15 foot) diameter by 18 meters (60 foot) long satellite with clearances for movement of EVA crew and placement of workstations. The bay is cylindrical in shape with a segmented retractable cover to allow partial to full opening for access as required. In addition, a separate small port at either end of the bay is envisioned for personnel and small equipment entry. Figure 2 is a diagram of the Servicing Bay with cover partially open and the MRMS handling a payload.

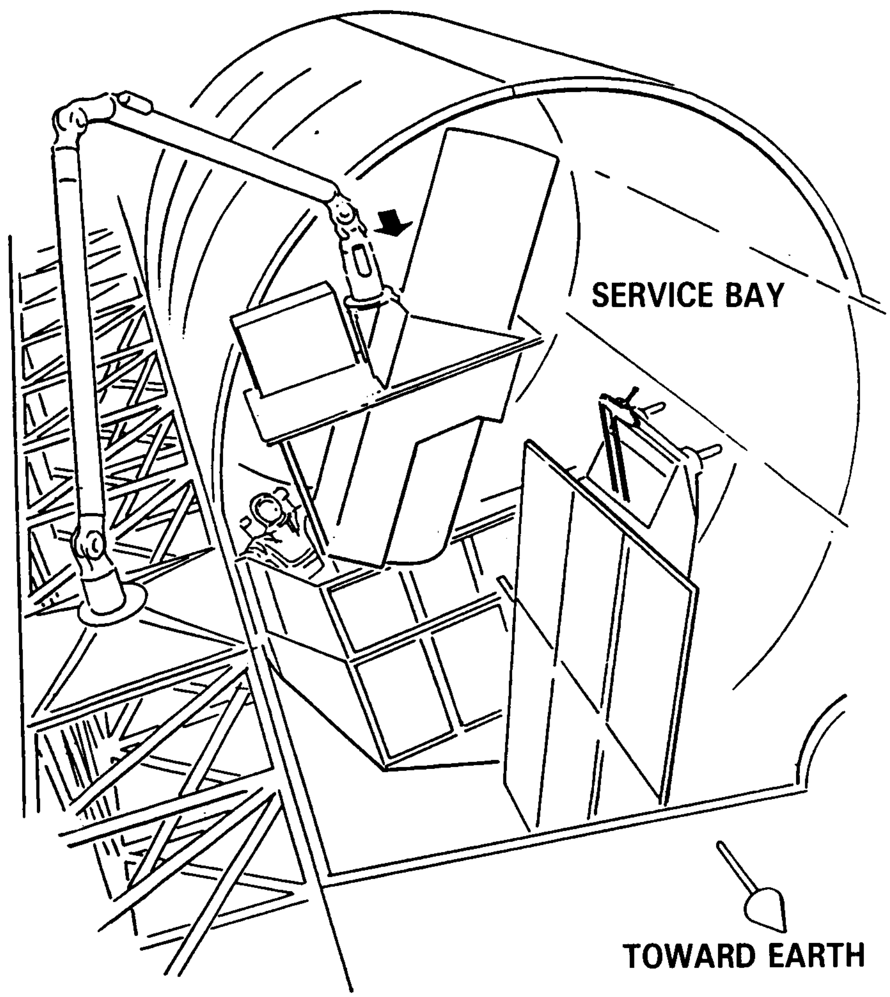

Figure 2 - Servicing Bay

The enclosed bay will provide environmental protection and contamination control. The bay will be outfitted with fixtures to hold satellites and payloads in fixed positions for servicing, and also a satellite positioning system allowing 90° tilt and +/-180° rotation. A thermal control system with interfaces to the Space Station Active Thermal Control System will be utilized to control the internal temperature of the bay, as well as the temperature of the satellite being serviced. Power for the Service Bay is estimated to be 6 KW with 3.5 KW designated for satellite use. In addition to power control and distribution in the bay, the electrical system will include data processing and communication interfaces to the Space Station Information System. For support of servicing operations, the bay will contain the necessary lighting, EVA support equipment, closed circuit television system, and facility monitors such as contamination sensors, and thermal sensors.

Satellite Storage Bay

The Storage Bay is located adjacent to the Servicing Bay, but on the other side of the keel. It is identical in size and shape to the Servicing Bay and utilizes the segmented retractable cover. It is sized for storing large observatories, platforms, or satellites awaiting repair or refurbishment. It will provide a safe haven for long-term storage of satellites, such as HST, while repair procedures, equipment, or replacement units are being prepared on the ground for transfer to the Space Station.

Fixed satellite and instrument retention systems will be available in the Storage Bay. Preliminary assessments indicate that the satellite positioning system used in the Servicing Bay is not required, but could be added for the growth configuration. The power and data systems will be identical in function, but reduced in capability since satellite testing would not be planned for the Storage Bay. A modular approach to the design of the power, data, utilities, and monitoring system will be utilized for commonality and cost effectiveness. Power requirements primarily for the facility, monitors, and satellite thermal control are estimated at 500 watts.

The Storage Bay will have a means to control and monitor contamination. A thermal control system similar to the Servicing Bay will be provided to maintain satellites and instruments within safe storage ranges.

Since the Servicing Bay, Storage Bay, and Refueling Bay are identical in external configuration and access, the operational procedures, including those with the MRMS and OMV, can be standardized which is a significant advantage gained by the commonality of facilities.

Figure 3 shows an instrument replacement using the Servicing Bay and Storage Bay on the main truss. It illustrates how an instrument changeout, utilizing the MRMS, would be accomplished. Here the replacement instrument #2 is removed from the instrument carrier in the Storage Bay and installed onto the Spartan Operational Carrier in the Servicing Bay and the instrument #1 removed and placed back in storage.

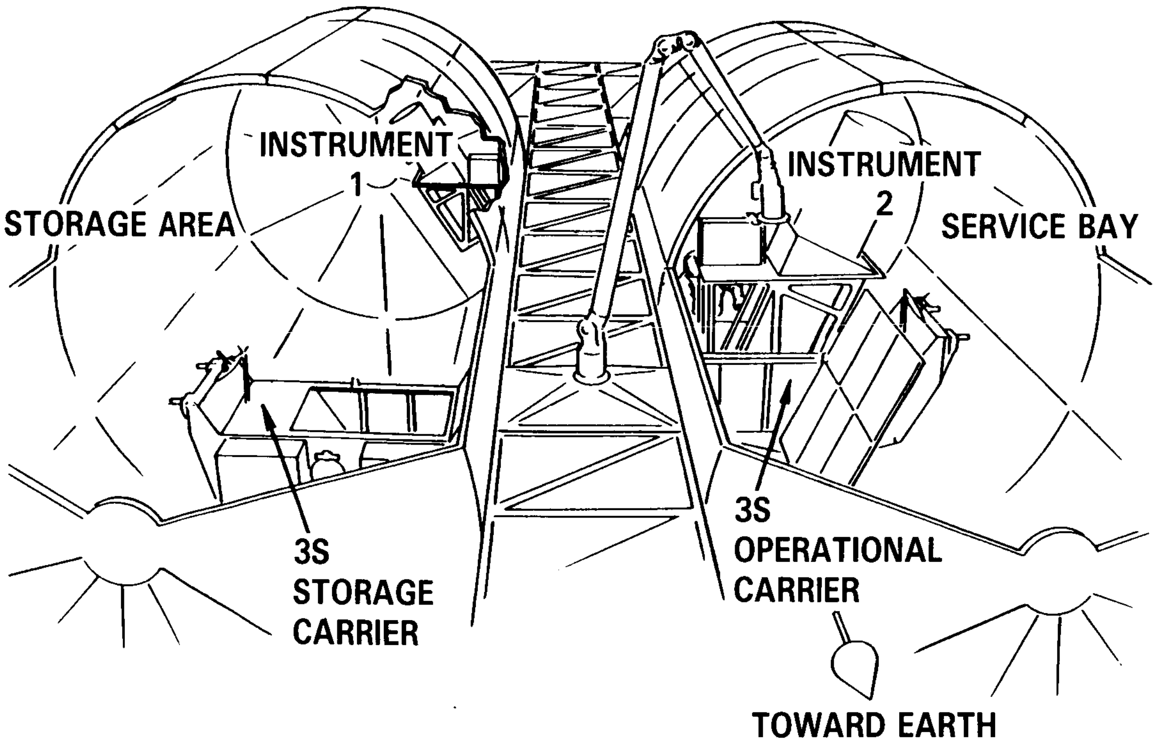

Figure 3 - Satellite Instrument Replacement Illustrating Use of Storage and Service Bays

Refueling Bay

The Refueling Bay is located on the keel near the earth pointing end of the Space Station. The bay will be identical in shape and size to the Servicing Bay, and will use the segmented retractable cover. It is located remote from the Servicing and Storage Bays to prevent any contamination which may occur from refueling from reaching these bays.

Equipment will be provided to refuel satellite propulsion systems and replenish cryogen systems for instruments. Other pressurants and fluids such as liquid nitrogen (LN_2) will be provided in this bay. A satellite retention/positioning system will be available to berth the satellite for refueling.

Power for the Refueling Bay is estimated at 1500 watts. The power and data systems will be identical to the Servicing Bay, but reduced in capability since only satellite monitoring and caution and warning system operations are planned for the Refueling Bay. The modular approach for the electrical system (power, data, etc.) will allow commonality and minimize cost.

A thermal control system also identical to that in the Servicing Bay will be utilized to control the internal temperature of the bay and the temperature of the payload being serviced. A similar contamination monitor and control system will be provided. The utilities and crew EVA support systems will basically be those provided for the Servicing Bay.

Figure 4 shows the Refueling Bay.

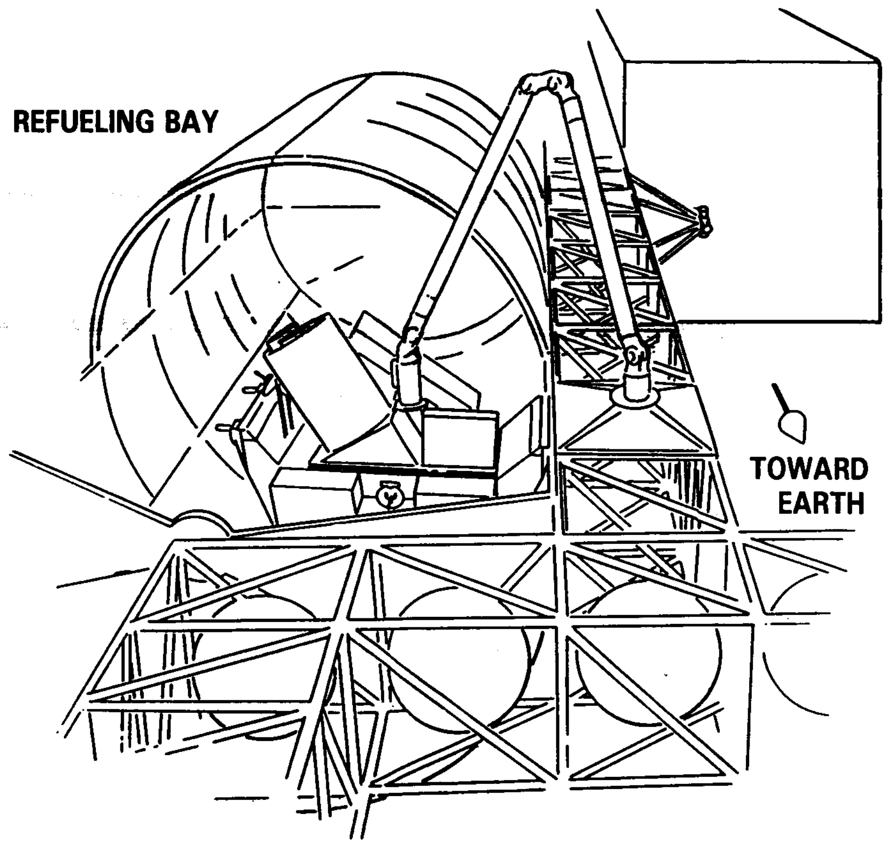

Figure 4 - Refueling Bay

Instrument Storage Area

The Instrument Storage Area is an enclosed volume located just below the solar array boom, and is nominally 3M x 6M x 9M (10 ft. x 20 ft. x 30 ft.). It is used to store replacement instruments and other large modules awaiting installation or return to earth. Instruments stored in this facility will be mounted in an instrument holding/retention fixture or in shipping containers. The area will be thermally controlled and designed to be contamination free. 200 watts of power are estimated for thermal control and facility monitoring. The facility will include the basic utilities of lighting, CCTV, monitors, and EVA support equipment.

Tool/ORU Storage Area

Ten ORU storage lockers and four tool storage lockers, each .9M x 1.5M x 1.5M (3 ft. x 5 ft. x 5 ft.) are located across the center of the solar array mounting truss. These lockers have individual thermal control systems and covers. Power for all 14 boxes is estimated at 150 watts. Holding fixtures for ORU's and tools will be provided. Equipment for facility monitoring and EVA support is also included. A tool inventory system will be used for the tool lockers to insure tool accountability.

Work Station in Pressurized Module

The work station in the Science Lab Module consists of a workbench and a control station. The workbench will be used for repair and replacement of items at the box level, which is two levels lower than the ORU level. The bench will be equipped with tools to support component replacement, printed circuit board changeout, wire soldering/crimping, and continuity testing.

The control station will contain equipment to remotely monitor and control the customer servicing activities. The status of all facilities including temperature, cleanliness level, etc., will be displayed. Viewing of the activities in each of the facilities will be provided via the CCTV System.

Further details on the IOC Reference Servicing Facility are shown in Table 1 - Service Facility Configuration Matrix.

Automation and Robotics

Automation and Robotics (A&R) has the potential to significantly improve the productivity of On-orbit Assembly and Servicing. Its application could greatly reduce the crew EVA requirements. NASA has a mandate from Congress to identify specific Space Station systems which will advance A&R technologies. GSFC is formulating an aggressive A&R program of technology development centered around the Assembly and Servicing area.

The cornerstone of the GSFC plan is the development of a robotic facility that will allow evaluation of robotic scenarios through demonstrations, and tests with full scale items expected to be used on Space Station. As the Servicing Facility design matures, a full scale ground model of the Servicing Bay will be constructed and outfitted with the Robotics System.

A representative sample of the kinds of tasks for Automation and Robotics are as follows:

- Removing and installing payloads and modules;

- Performing precision mechanical assembly as in the build-up of an instrument system;

- Servicing of manufacturing facilities such as for materials processing systems;

- Making electrical and fuel line connections using automatic mechanisms;

- Automated spacecraft testing and facility checkout.

SCENARIO OF SATELLITE SERVICING AT THE SPACE STATION[1]

As stated earlier, a key tool required for supporting the servicing of satellites is the OMV. Figure 5 is a diagram showing the OMV reference configuration. While it is being developed outside the Space Station Program, this orbit and deorbit assist capability is a vital element to the Space Station satellite servicing scenario. This versatile, reusable, remotely controlled free-flying vehicle is capable of performing a wide range of on-orbit services such as transportation to and from the Station and in-situ servicing.

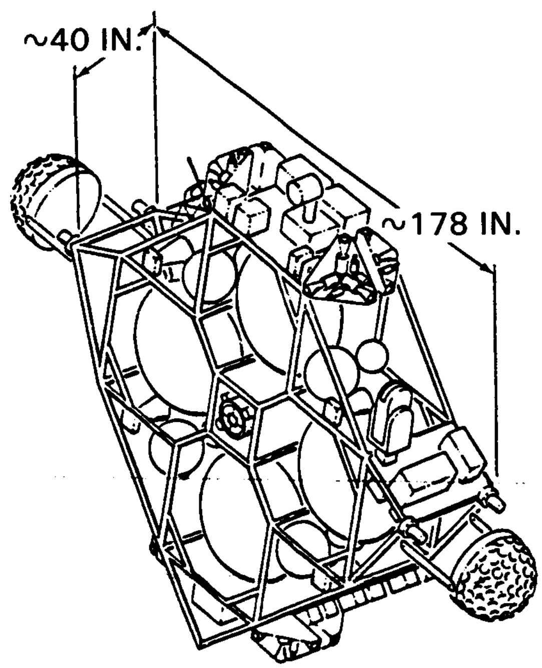

Figure 5 - Orbiting Maneuvering Vehicle (OMV) Reference Configuration

1 This section has been adapted from "On-Orbit Platform Servicing in the 1990's", a paper by Francis J. Logan, presented at AAS/AIAA Astrodynamics Specialist Conference in Vail, CO, August 12-15, 1985.

A scenario for servicing a satellite or platform follows.

Using its propulsion system, the platform or satellite needing service translates to within 20 KM's of the co-orbiting Space Station. The OMV is deployed from the Space Station, and maneuvers a short distance from the Station using its cold gas system. It then translates to the satellite location using its standard propulsion system. Reverting back to its cold gas system, the OMV docks with the powered-down satellite using a grapple fixture positioned to insure that the OMV thrust vector is near the center of gravity of the satellite. The OMV performs the appropriate impulse maneuver to place the OMV/satellite combination to within 15 meters of the Space Station. The MRMS then grapples the OMV/satellite and places it in the Servicing Bay. After the satellite is secured in the satellite servicing fixture, the OMV is released and the MRMS transfers the OMV to a service area for refueling.

The satellite umbilical connector is automatically engaged and functional checkout performed from the satellite ground control center via the transparent Space Station Data System. The satellite is then powered down, and is available for servicing, repair, or reconfiguration. Figure 6 shows the satellite being serviced in the Servicing Bay. A variety of tasks could be accomplished, either by teleoperated manipulators, full robotics, or EVA. A science or applications mission could have its instrument exchanged or upgraded; a cryogenically cooled instrument, such as those on SIRTF or AXAF, could have its dewar refilled; a materials processing mission could have its end product removed and the required raw materials installed in the processor; and any mission experiencing a failure could be repaired.

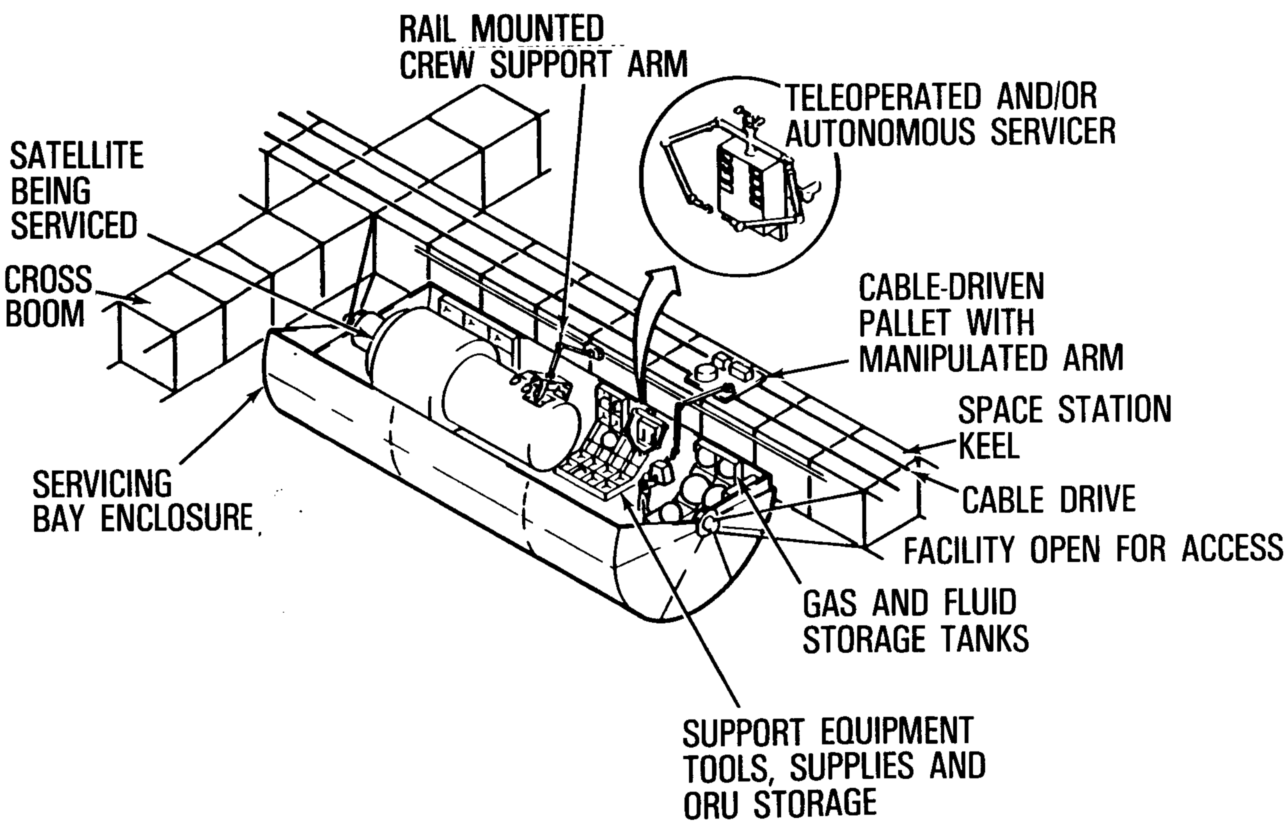

Figure 6 - Satellite Being Serviced in Servicing Bay

Upon completion of the servicing, and a successful functional test, the satellite is judged ready for deployment, which is the reverse of the procedure used for docking. After the OMV has transported the satellite to the proper release location, another satellite checkout is run via the Space Station. The satellite appendages (solar arrays, antennas, etc.) are deployed and verified and the OMV releases the satellite. The OMV then returns to the Space Station and the satellite translates to its operational orbit and continues its mission. Should the satellite not have a propulsion system, as with HST, the OMV would translate to the satellite's orbit and tow it to the Space Station and return it following servicing, to its operational orbit.

This scenario is a general description of a satellite servicing with Space Station. GSFC's two Phase B contractors are presently in the sixth month of an eleven-month effort where options are being evaluated for the design of the Servicing Facility and servicing procedures. After the definition phase, the contractors will perform a ten-month preliminary design study from which a more detailed servicing scenario will be produced.

USER REQUIREMENTS FOR SERVICING AT SPACE STATION

For users to take full advantage of servicing capabilities at Space Station, certain basic requirements must be satisfied. Satellites must have a standard grapple for capture and a standard berthing interface, such as the Shuttle Orbiter sill and keel trunions, for attachment to the holding and positioning fixtures in the service facility. Standard electrical interfaces with remotely controlled umbilical connectors will need to be implemented. Space Station safety requirements must be met to preclude damage to the Space Station or injury to the EVA crew. Sensitive instruments will need to implement remotely controlled protective devices, such as shutters which can be closed to prevent damage. Satellite thermal systems must be designed to maintain survival temperatures during transfer from orbit to the Space Station Servicing Facility. Large appendages, such as solar arrays, should be retractable to maximize efficiency in handling, translation, accommodation, and servicing. Finally, systems and components must be accessible for servicing.

CONCLUSION

The Space Station will offer a significant capability for on-orbit servicing that will lengthen the life and scientific and applications utility of satellite missions from a few years to an indefinite time. The Space Station Servicing Facility will be designed to provide for replacement of consumables, changing or enhancing of instruments, repair and replacement of systems, storage of replacement units and satellites, and on-orbit assembly. It will also provide the support systems for retrieving and deploying satellites needing servicing. Key to the effective utilization of these facilities is the implementation by users of basic requirements for accommodation and servicing.

TABLE 1

SERVICE FACILITY CONFIGURATION MATRIX

Element	Servicing Bay	Refueling Bay	Satellite Storage Area	Instrument Storage Area	Tool/ORU Storage Area
1. Size					
- Interior Dimen.	9M x 21M (30'x70')	9M x 21M (30'x70)	9M x 21M (30'x70')	6M x 9M x 3M (20'x30'x10')	14@ 1.5M x 1.5M x .9M (5'x5'x3')
- Shape	Enclosed Cylinder w/cover	Enclosed Cylinder w/cover	Enclosed Cylinder w/cover	Enclosed Box w/cover	Enclosed Box w/cover
2. Power Input (Max)	6000 W Peak=10 KW for 10 Min.	1500 W	500 W	200 W	150 W Total
3. Electrical System					
Control Assembly					
- Heater Controllers	X	X	X	X	X
- Data & Com. I/F	X	X	X	X	-
- Power Regulation/ Control	X	X	X	X	X
- Lighting Control	X	X	X	X	-
- Servicing Facility Equipment	X	X	X	X	-
- CCTV Interface	X	X	X	X	-
- Monitors for Facil. Equipment	X	X	X	X	X
- Automatic Self Test System	X	X	X	X	X

SERVICE FACILITY CONFIGURATION MATRIX (CONTD)

Element	Servicing Bay	Refueling Bay	Satellite Storage Area	Instrument Storage Area	Tool/ORU Storage Area
4. Facility Utilities					
- Lighting	X	X	X	X	-
- CCTV (2 Systems)	X	X	X	X	-
- Umbilical Connector System	X	X	X	-	-
5. Access					
- Facility Cover	X	X	X	X	X
- Hatch	X	X	X	-	-
6. Mech. EQ. & Fixtures					
- Satellite Retention Positioning Sys.	X	X	-	-	-
- Fixed Satellite Retention Sys.	X	X	X	-	-
- Adaptors for Special Satellite Retention (i.e., SMM, HST)	X	X	X	-	-
- Inst./ORU Holding Fixture	X	-	X	X	X
- Astronaut Hand Holds & Foot Restraints	X	X	X	X	X
- Portable, Adjustable Foot Support System	X	X	X	-	-

SERVICE FACILITY CONFIGURATION MATRIX (CONTD.)

Element	Servicing Bay	Refueling Bay	Satellite Storage Area	Instrument Storage Area	Tool/ORU Storage Area
7. Accommodations for Use of:					
- MRMS	X	X	X	X	-
- OMV	X	X	X	-	-
8. Thermal Control					
- Insulated Retractable Cover	X (segmented)	X (segmented)	X (segmented)	X	X
- Heaters & Thermal Coatings	X	X	X	X	X
- Fluid Coupling Lines/Cold Plates from SS ATCS	X	X	-	-	-
- Temp. Sensing Devices for Remote Monitoring	X	X	X	X	X
9. Customer Data & Information Links					
- Data Sys. Hardware Interface to SS Data System	X	X	X	X	X
- System for Radiative Trans. of S/C Data to SS Data Sys.	X	-	-	-	-
- System for Voice Communication	X	X	X	X	X

SERVICE FACILITY CONFIGURATION MATRIX (Contd.)

Element	Servicing Bay	Refueling Bay	Satellite Storage Area	Instrument Storage Area	Tool/ORU Storage Area
10. Contamination Control					
- Line of Sight Portable Barrier	X	X	X	-	-
- Contamination Monitor	X	X	X	X	-
- Leak Detector	X	X	X	X	-
11. Safety					
- Emergency Control from Remote Area	X	X	X	X	X

SPACE TRANSPORTATION SYSTEM

PAYLOAD INTEGRATION

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STS PAYLOAD INTEGRATION OBJECTIVES

- ESTABLISH A SYSTEM THAT CAN ACCOMMODATE BOTH COMPLEX AND SIMPLE PAYLOADS
- CREATE AN STS CAPABILITY, WHICH WOULD PROVIDE
  - MINIMUM INTERFACES FOR SIMPLE PAYLOADS
  - SUFFICIENT TECHNICAL DEPTH REQUIRED FOR INTEGRATION OF COMPLEX PAYLOADS
  - COORDINATED STS REACTION TO PAYLOAD NEEDS
  - ACCESS TO WORKING LEVEL PERSONNEL WHERE APPROPRIATE
- ESTABLISH A COORDINATED/SIMPLIFIED PAYLOAD DOCUMENTATION SYSTEM

PAYLOAD INTEGRATION PROCESS

S-84-01668A

- ESTABLISH A TECHNICAL TEAM TO ACCOMPLISH THE INTEGRATION
- DEFINE THE PROGRAMMATIC AND TECHNICAL AGREEMENTS AND BASELINES — THE "PAYLOAD INTEGRATION PLAN" (PIP)
- DEFINE THE INTERFACE SPECIFICATIONS FOR THOSE INTERFACES SPECIFIED IN THE PIP — THE "INTERFACE CONTROL DOCUMENT" (ICD)
- DEFINE THE DATA IN THE PIP ANNEXES NECESSARY TO CONFIGURE STS FLIGHT AND GROUND SYSTEMS TO PROVIDE THE SERVICES REQUESTED IN THE PIP
- ASSIST THE CUSTOMER IN PROPERLY INTERPRETING NASA SAFETY REQUIREMENTS — THE "SAFETY REVIEW PROCESS"
- ASSURE THAT STS IMPLEMENTATION IS CONSISTENT WITH DOCUMENTED REQUIREMENTS — THE "FLIGHT REVIEW PROCESS"

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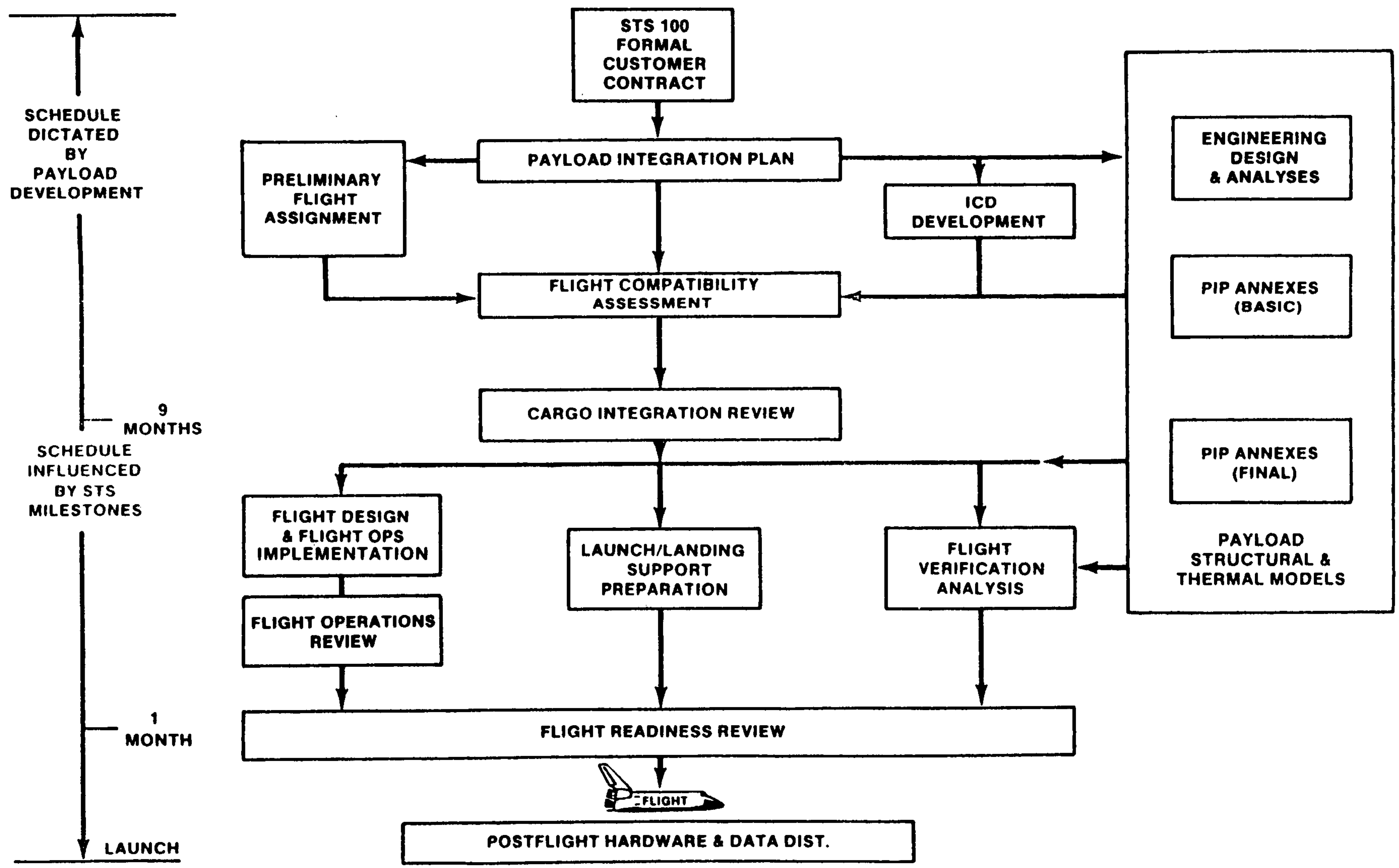
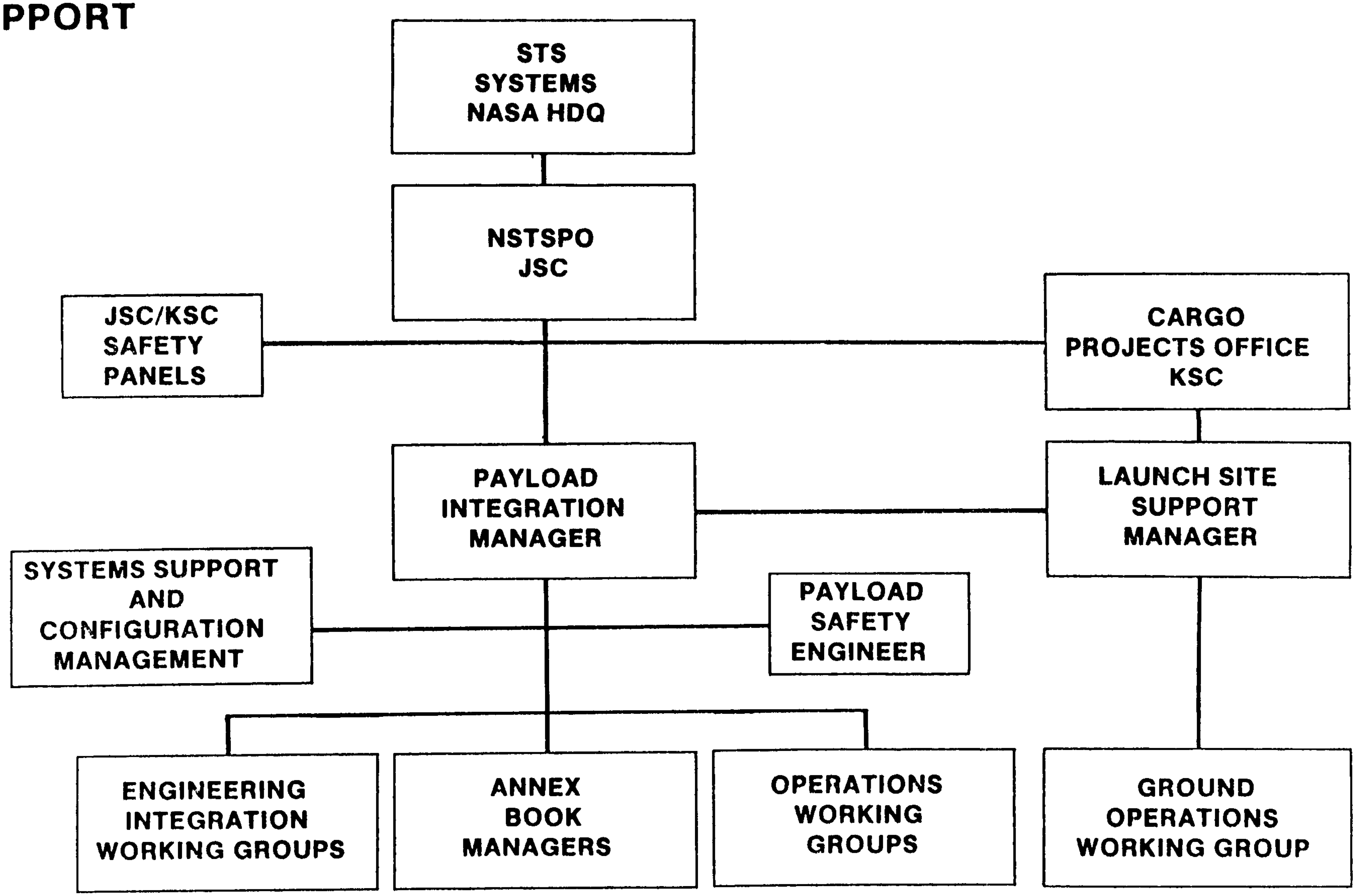
PAYLOAD INTEGRATION PROCESS OVERVIEW

S-84-01018

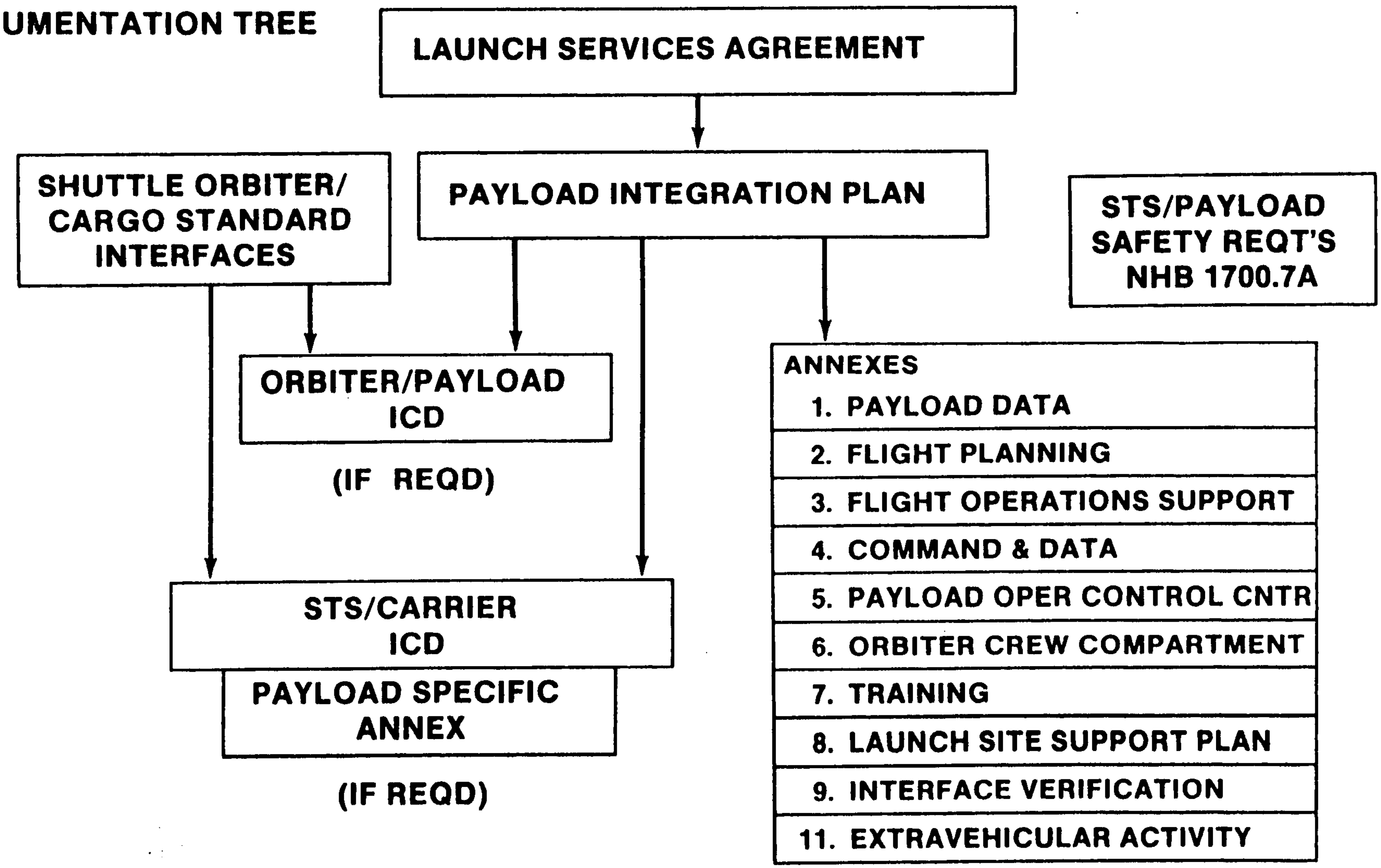


S-84-01080A

PAYLOAD INTEGRATION PROCESS MANAGEMENT/TECHNICAL SUPPORT



CUSTOMER INTEGRATION
DOCUMENTATION TREE



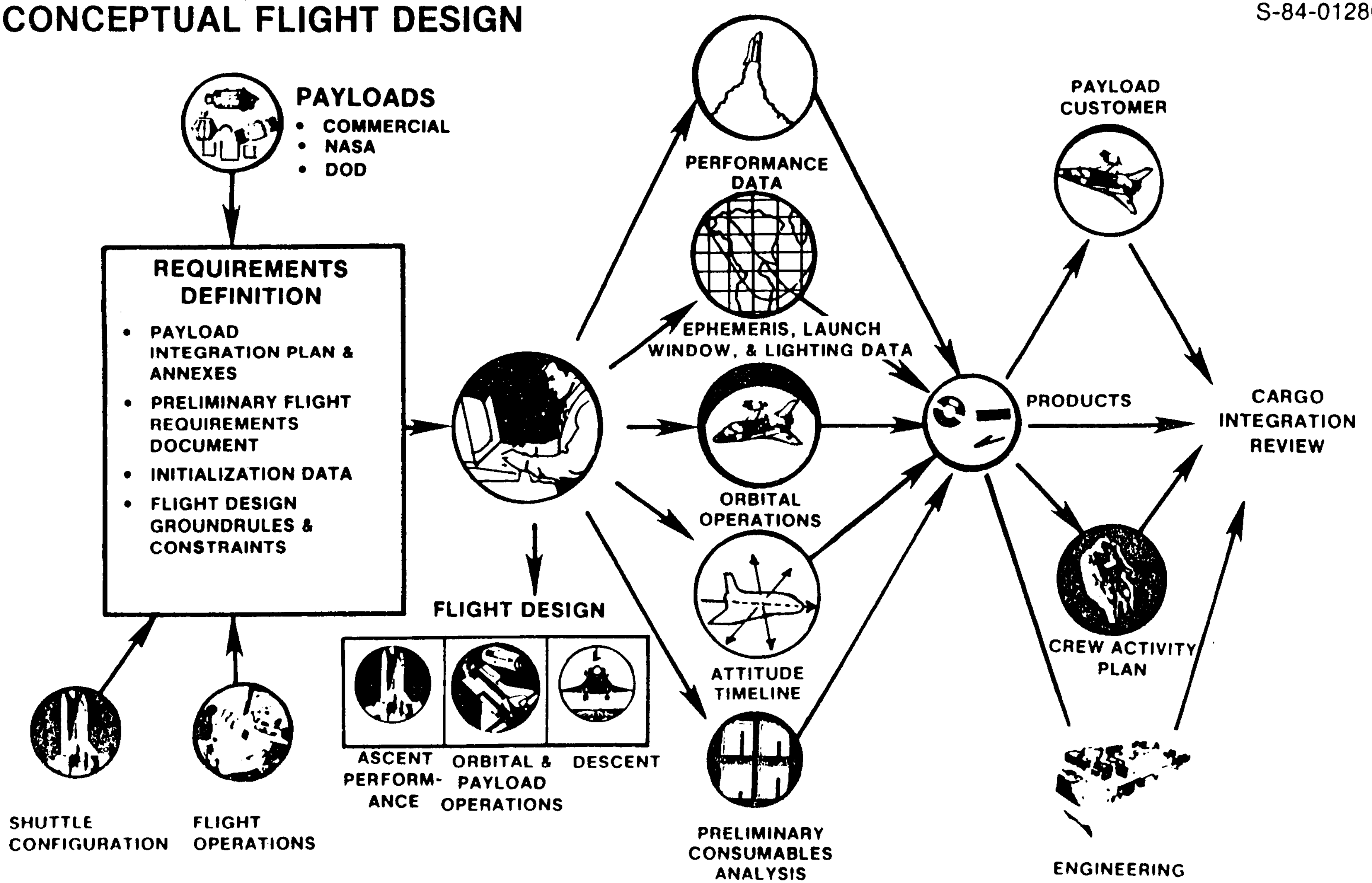
FLIGHT DESIGN PROCESS

S-84-02071

- FLIGHT DESIGN IS THE PREPARATION OF THE TRAJECTORY, ATTITUDE, AND CONSUMABLES PROFILES FOR A SPECIFIC FLIGHT
- INCLUDES TWO MAJOR ELEMENTS — COMPLETED SEQUENTIALLY
  - CONCEPTUAL FLIGHT DESIGN
    — DEVELOPS EARLY DATA BASE FOR CUSTOMER AND NATIONAL SPACE TRANSPORTATION SYSTEM ORGANIZATIONS PLANNING AND ASSESSMENT IN PREPARATION FOR THE CARGO INTEGRATION REVIEW
  - OPERATIONAL FLIGHT DESIGN
    — FINAL PREFLIGHT DESIGN TO SUPPORT RECONFIGURATION OF GROUND BASED AND FLIGHT SYSTEMS

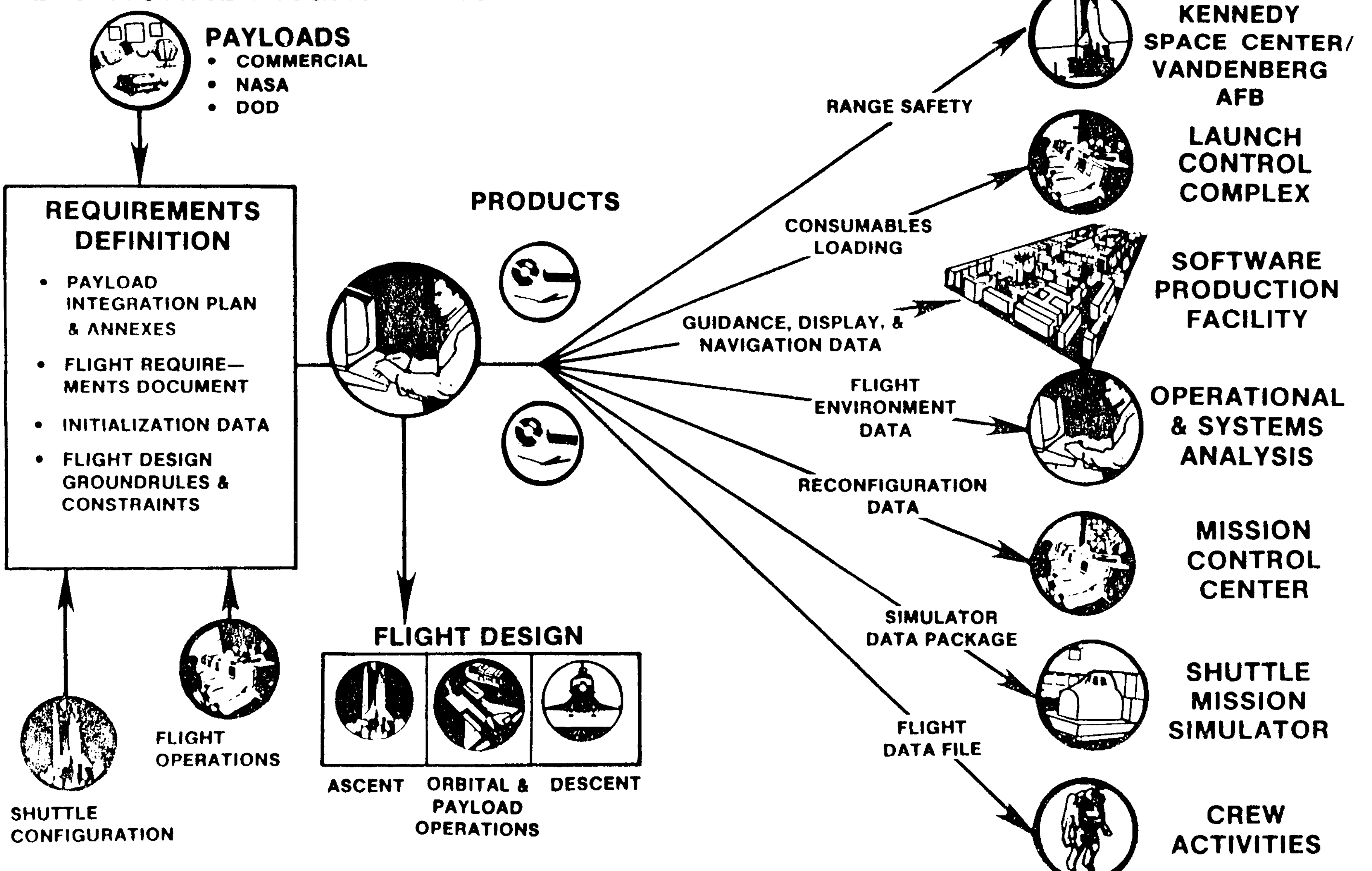
CONCEPTUAL FLIGHT DESIGN

S-84-01286



OPERATIONAL FLIGHT DESIGN

S-84-01282



STANDARD MISSION CRITERIA
KENNEDY SPACE CENTER LAUNCH

S-84-01434A

- **SHARED FLIGHTS**
  - ORBITAL INCLINATION — 28.45° AND 57.0°
  - INITIAL ORBIT ALTITUDE — 160 N. MI. (296 km) CIRCULAR
  - LAUNCH WINDOW (FOR 28.45°)
    - — TWO LAUNCH WINDOWS CENTERED NEAR NOON AND MIDNIGHT GREENWICH MEAN TIME
    - — LAUNCH WINDOW DURATION — 2 HRS FOR EACH WINDOW
  - AT LEAST 80% OF STANDARD LAUNCH WINDOW SHOULD BE USABLE
  - SPACECRAFT SHOULD BE COMPATIBLE WITH DEPLOYMENT ON 2 CONSECUTIVE ORBITAL REVOLUTIONS WITH ANOTHER OPPORTUNITY AT LEAST 12 HRS LATER
- **DEDICATED FLIGHTS**
  - ORBITAL INCLINATION — 28.45° AND 57.0°
  - INITIAL ORBIT ALTITUDE — 160 N. MI. (296 km) CIRCULAR
  - LAUNCH WINDOW DURATION — 1 HR

GEOSYNCHRONOUS TRANSFER ORBIT GEOMETRY FOR STANDARD SHARED FLIGHT LAUNCH WINDOW

S-84-01440

- LAUNCH WINDOW FOR SHARED FLIGHTS COMPATIBLE WITH CLASS OF SPACECRAFT TO BE PLACED IN GEOSYNCHRONOUS ORBIT
- THESE SPACECRAFT REQUIRE THE SUN AND EARTH PARKING ORBIT NODAL CROSSINGS TO BE APPROXIMATELY ALONG TRANSFER ORBIT LINE OF APSIDES
- DUE EAST LAUNCH RESULTS IN LIFT-OFF APPROXIMATELY 90° FROM NODAL CROSSINGS; THEREFORE EARLY MORNING OR LATE EVENING LOCAL TIME LAUNCH IS REQUIRED
- FIRST NODAL CROSSING FOR LAUNCH FROM KENNEDY SPACE CENTER IS NEAR GREENWICH MERIDIAN; THEREFORE LIFT-OFF IS NEAR NOON OR MIDNIGHT GREENWICH MEAN TIME (GMT)

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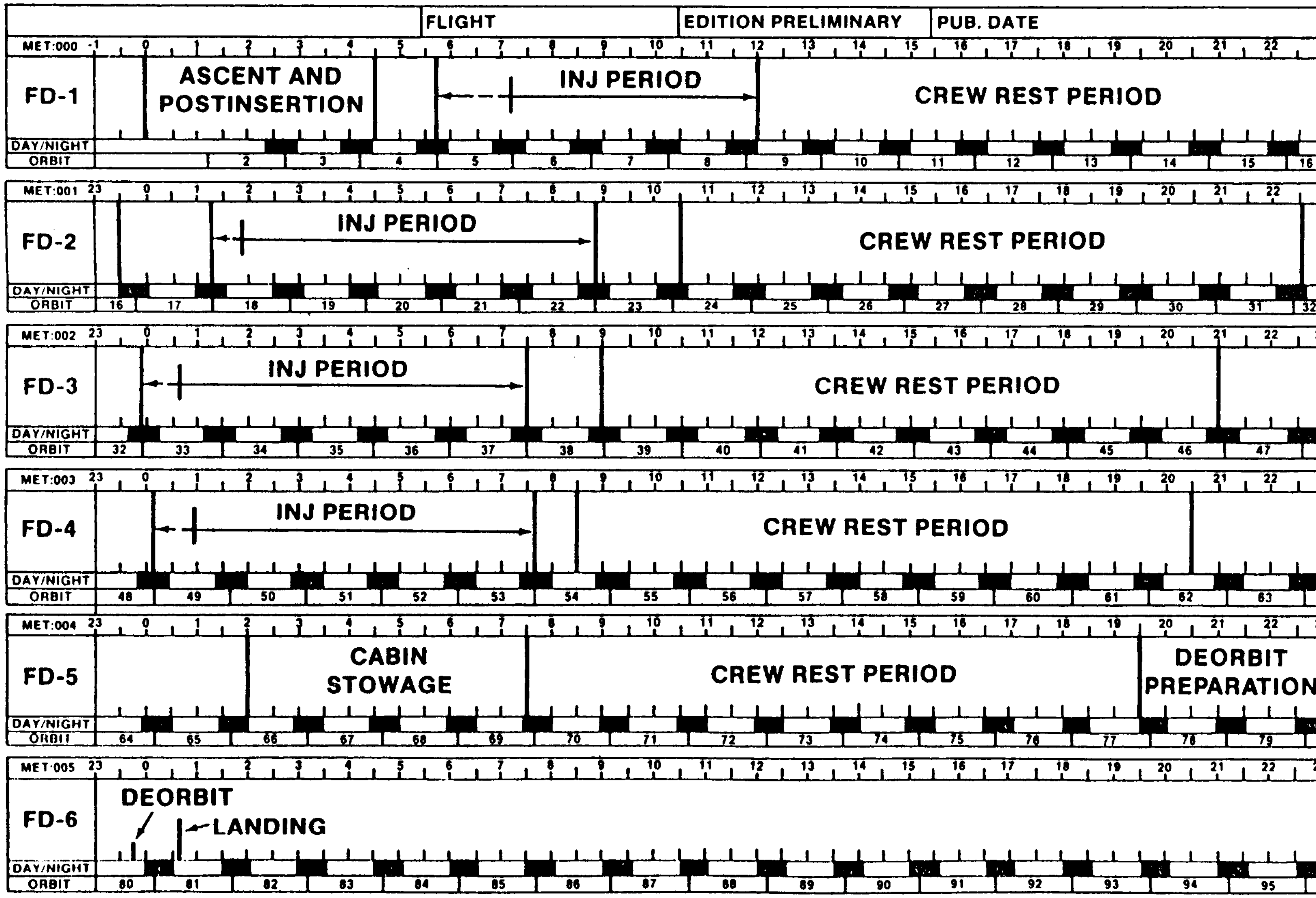
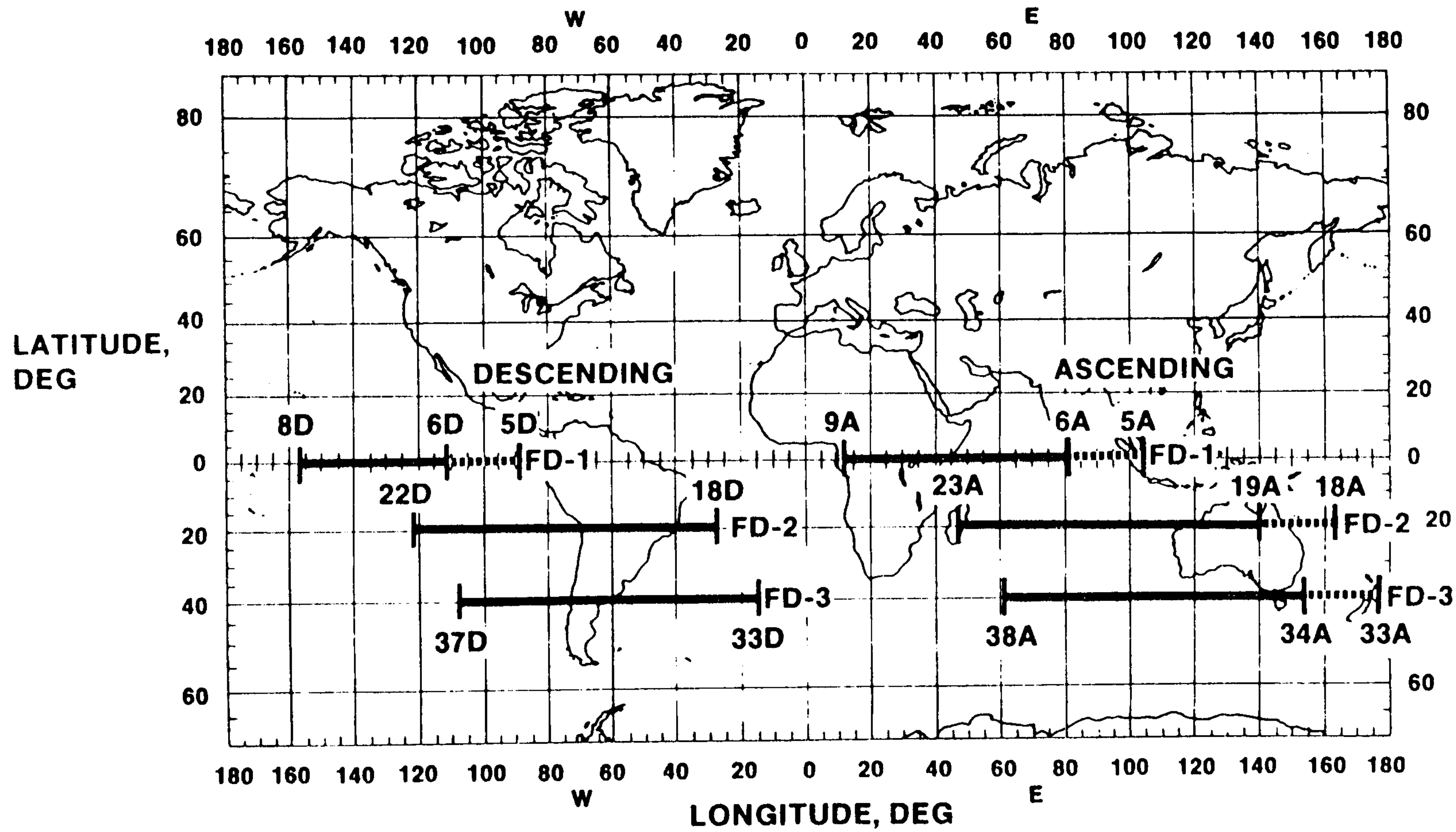
TYPICAL CREW ACTIVITY PLAN

S-84-01288



GEOSYNCHRONOUS TRANSFER ORBIT INJECTION LONGITUDES

S-84-01287



FD - FLIGHT DAY 6A - INJECTION ON SIXTH ASCENDING NODE

MISSION OPERATION PROCESS
MISSION OPERATIONS PRODUCTS

- ONBOARD DISPLAY AND CONTROL
  - NASA USES THE INFORMATION PROVIDED IN THE COMMAND AND DATA ANNEX AND THE CREW COMPARTMENT ANNEX TO CONFIGURE ORBITER COMPUTER DISPLAYS AND PANELS FOR THE CREW TO ACCOMPLISH PAYLOAD OBJECTIVES
- CREW PROCEDURES
  - THE FLIGHT OPERATIONS SUPPORT ANNEX (NO. 3) PROVIDES THE CUSTOMER REQUIREMENTS FOR CREW PROCEDURES. NASA WILL TRANSLATE THESE PROCEDURES INTO THE FLIGHT DATA FILE (FDF) STANDARDS, WHICH ARE USED FOR FLIGHT. THE FINISHED PROCEDURES WILL BE PROVIDED TO THE CUSTOMERS FOR REVIEW
- CREW ACTIVITY PLAN
  - THE FLIGHT PLANNING ANNEX (NO. 2) PROVIDES THE CUSTOMER REQUIREMENTS FOR ALLOCATION OF CREW TIME. THIS INFORMATION IS USED TO DEVELOP THE OVERALL MISSION CREW ACTIVITY PLAN
- MISSION RULES
  - PREMISSION THE NASA CONTROL TEAM IDENTIFIES AS MANY DECISION POINTS AS POSSIBLE. CUSTOMER INPUTS ARE REQUESTED IN ANNEX 3 TO IDENTIFY POTENTIAL FAILURES AND THE APPROPRIATE COURSE OF ACTION FOR EACH FAILURE

MISSION OPERATION PROCESS (CONT)
MISSION OPERATIONS PRODUCTS

S-84-05043

- **PAYLOAD DATA ROUTING REQUIREMENTS**
  - NASA HAS THE CAPABILITY TO ROUTE VARIOUS TYPES OF SHUTTLE AND PAYLOAD DATA TO REMOTE LOCATIONS FOR CUSTOMER CONVENIENCE. ONCE THE DATA TYPES AND ROUTING LOCATIONS ARE IDENTIFIED, THE PROPER INTERFACES WILL BE EXERCISED SEVERAL TIMES PRIOR TO FLIGHT
- **TRAINING AND DEVELOPMENT**
  - IN ORDER TO TRAIN THE NSTS FLIGHT CREW AND MISSION CONTROL TEAM TO PROPERLY ACCOMPLISH THE PAYLOAD OBJECTIVES, NASA MUST DEVELOP SEVERAL TRAINING AIDS. THOSE TRAINING AIDS INCLUDE BRIEFINGS, WORKBOOKS, AND SIMULATOR MODELS. NASA USES THE DATA AVAILABLE IN THE PIP AND ANNEXES TO DEVELOP THESE TOOLS
- **MISSION CONTROL OPERATING PHILOSOPHY**
  - THE PAYLOAD OFFICER IS THE NASA MISSION CONTROL TEAM REPRESENTATIVE FOR ALL CUSTOMER ACTIVITIES. THE FLIGHT OPERATIONS SUPPORT ANNEX SHOULD CONTAIN THE CUSTOMER PLANS FOR INTERFACING WITH THE PAYLOAD OFFICER TO ENSURE THE CUSTOMER'S PARTICIPATION IN MISSION ACTIVITIES

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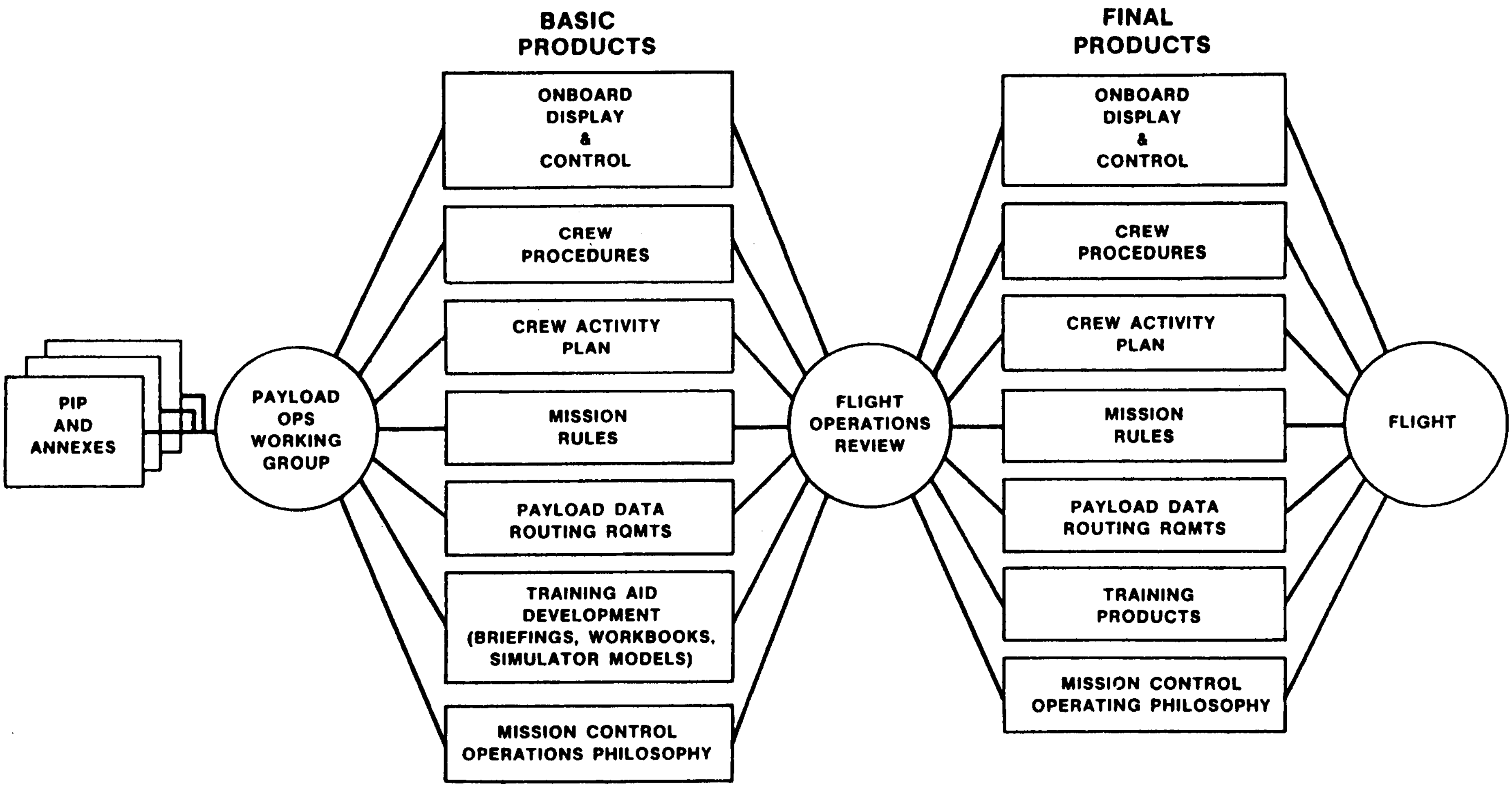
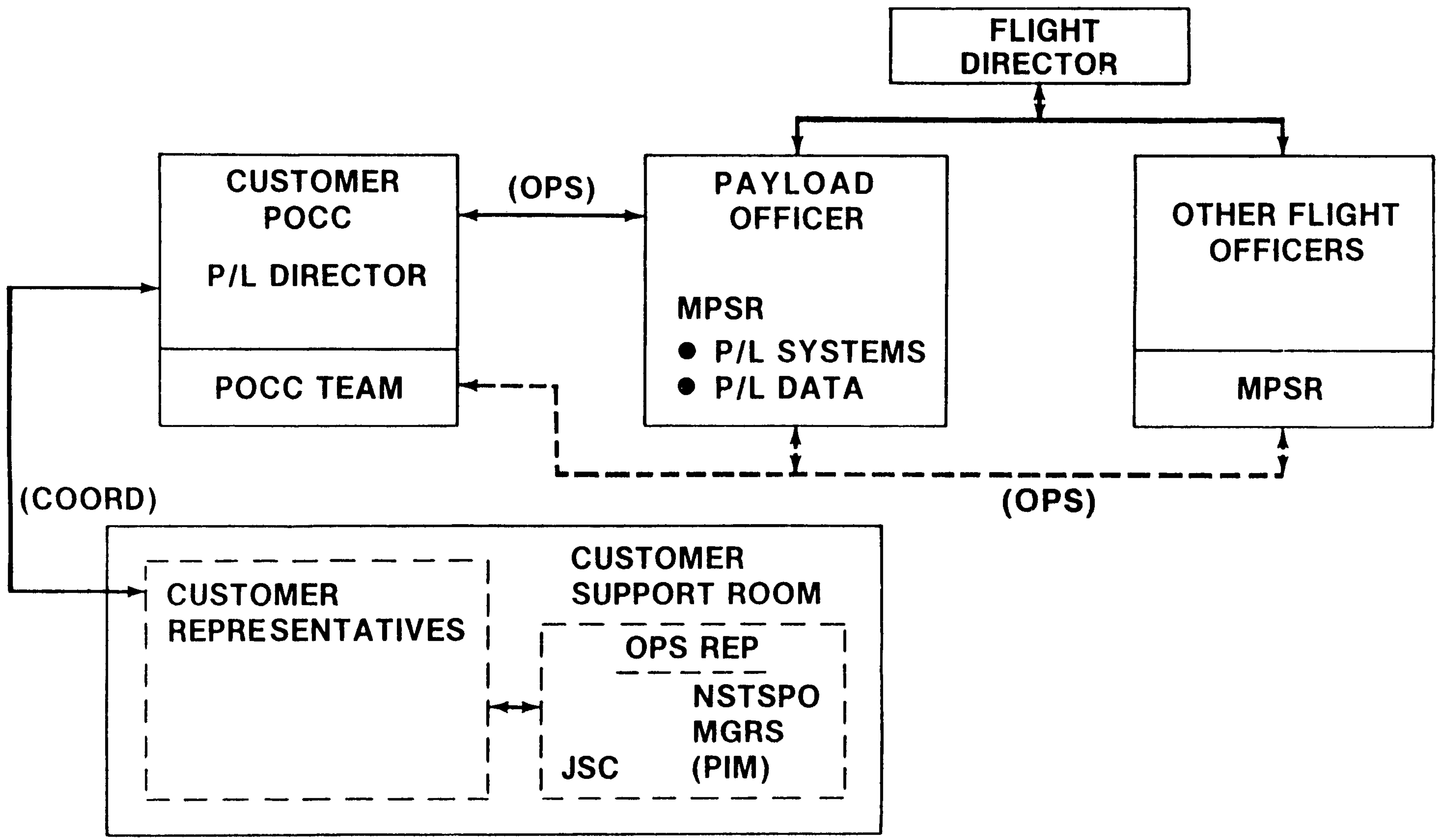
NASA

MISSION OPERATIONS PROCESS

S-84-05034



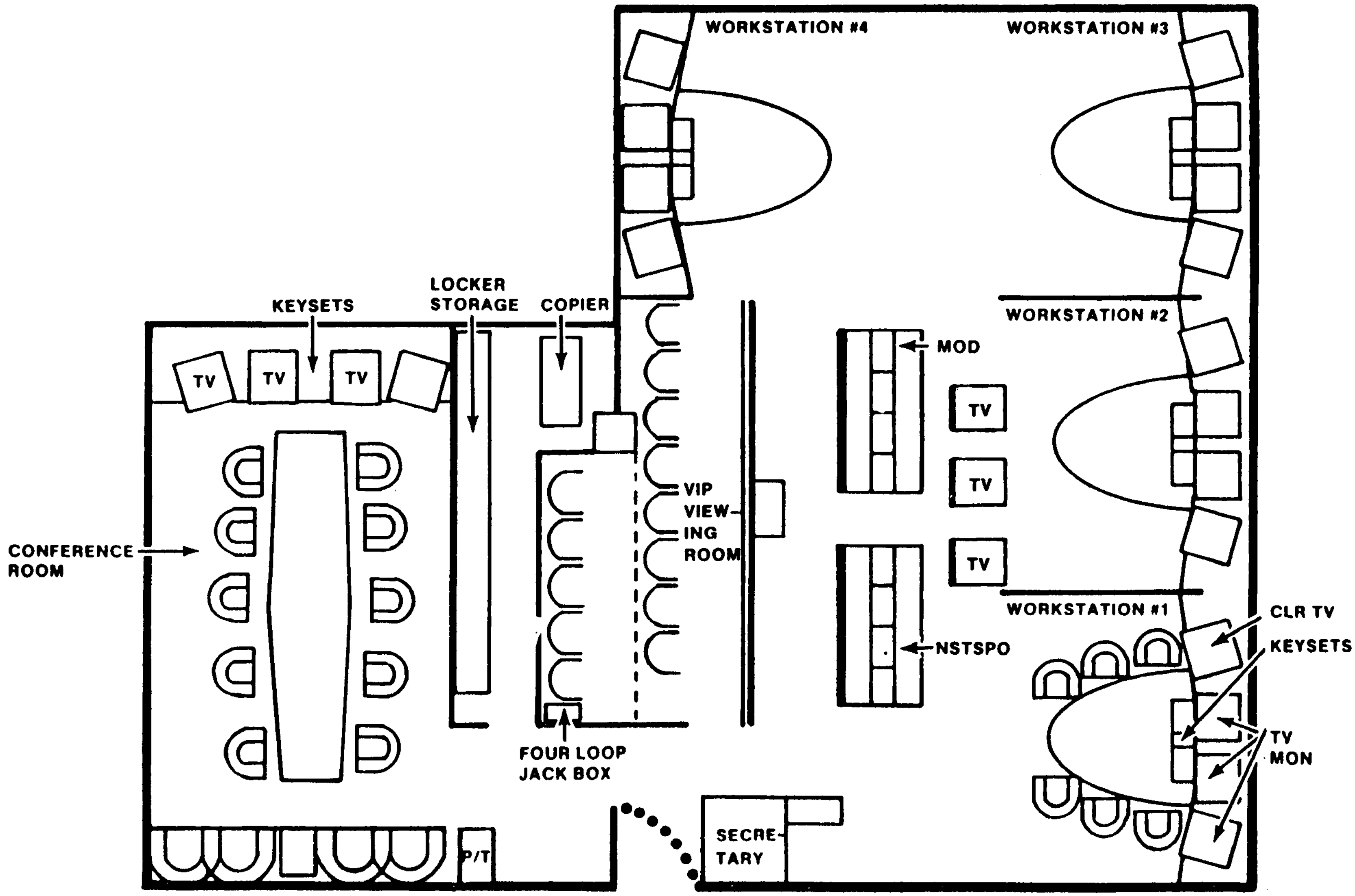
FLIGHT OPERATIONS SUPPORT OVERVIEW



SPACE
SHUTTLE
PAYLOAD
DESIGN AND DEVELOPMENT

S-84-01008A

CUSTOMER SUPPORT ROOM 236



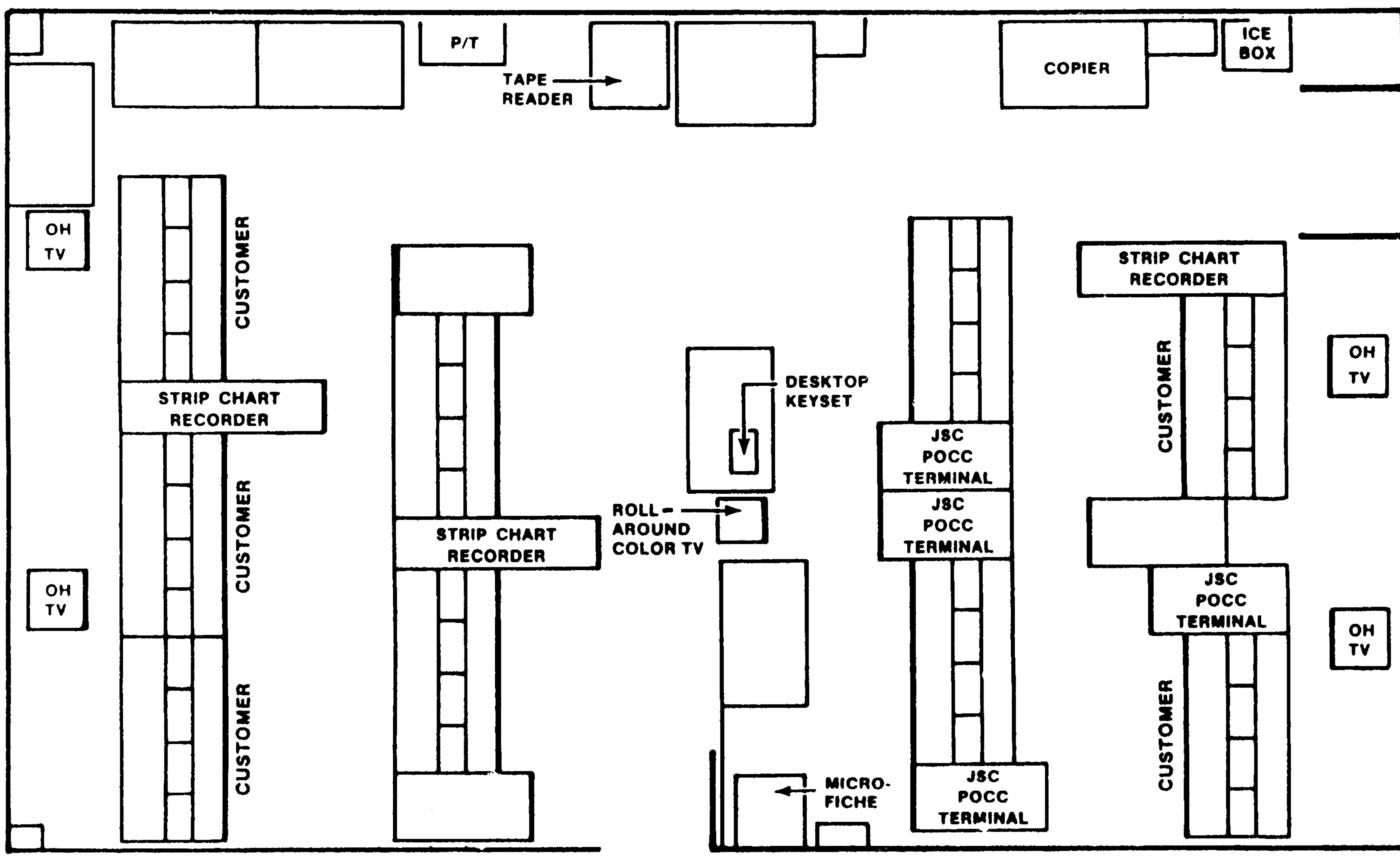
SPACE SHUTTLE PAYLOAD

DESIGN AND DEVELOPMENT

NASA

PAYLOAD MPSR ROOM 217

S-84-01009A



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THE FLIGHT REVIEW PROCESS

- **CARGO INTEGRATION REVIEW**
  - PURPOSE IS TO VERIFY CARGO MANIFEST AND ENSURE THE COMPATIBILITY OF THE STS-PROVIDED HARDWARE AND SOFTWARE WITH INDIVIDUAL PAYLOAD REQUIREMENTS
  - CUSTOMER
    - — SUBMITS DETAILED REQUIREMENTS VIA PIP, PIP ANNEXES, AND ICD AND REVIEWS HARDWARE/SOFTWARE DOCUMENTS TO VERIFY THEIR REQUIREMENTS ARE MET
  - STS
    - — PREPARES PRELIMINARY FLIGHT PLANS, INSTALLATION DRAWINGS, INTEGRATION HARDWARE DRAWINGS, AND SOFTWARE DOCUMENTATION TO DETERMINE THE COMPATIBILITY OF THE CARGO ELEMENTS

THE FLIGHT REVIEW PROCESS (CONT)

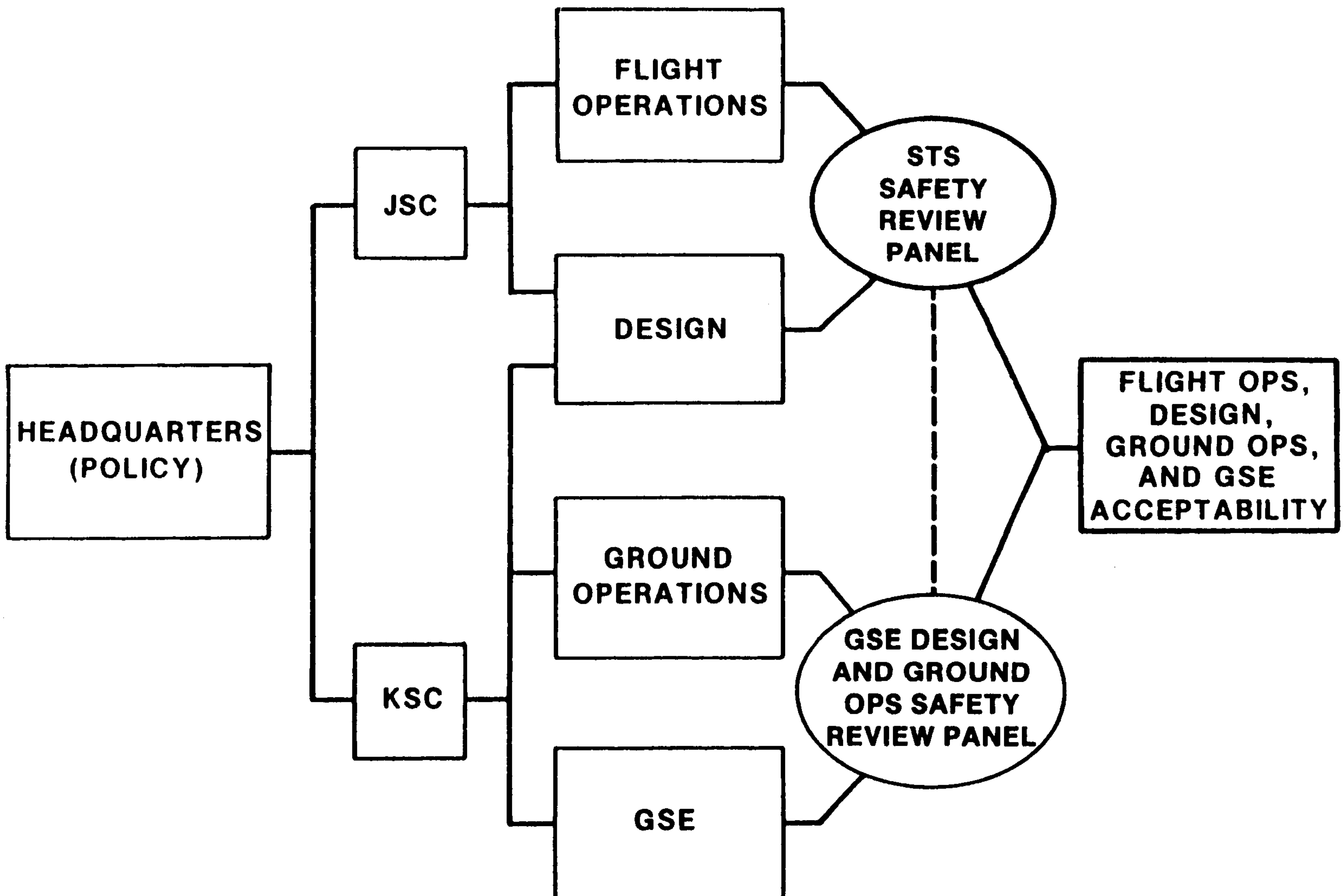
S-84-01672

214

- FLIGHT OPERATIONS REVIEW — REVIEW OF FLIGHT OPERATIONS DOCUMENTATION TO DETERMINE COMPLIANCE WITH CUSTOMER CREW TRAINING AND FLIGHT OPERATIONS REQUIREMENTS
  - CUSTOMER — SUBMITS DETAILED REQUIREMENTS VIA PIP AND PIP ANNEXES AND REVIEWS DOCUMENTATION TO VERIFY REQUIREMENTS ARE MET
  - STS — PREPARES FLIGHT PROCEDURES, MISSION RULES, AND FLIGHT PLAN DOCUMENTATION TO MEET CUSTOMER AND MISSION REQUIREMENTS
- FLIGHT READINESS REVIEW — VERIFIES COMPLETION OF ALL STS/CARGO INTEGRATION ACTIVITIES AND CERTIFIES READINESS OF ALL FLIGHT ELEMENTS
  - CUSTOMER — SUBMITS PAYLOAD CERTIFICATE OF FLIGHT READINESS
  - STS — REVIEWS ALL READINESS STATEMENTS

STS PAYLOAD SAFETY PROCESS

S-84-01021A



STS/PAYLOAD INTEGRATION
THE SAFETY REVIEW PROCESS

DURING FORMAL MEETINGS BETWEEN THE PAYLOAD ORGANIZATION AND THE STS SAFETY REVIEW PANEL, THE COMPATIBILITY OF PAYLOAD DESIGN WITH STS SAFETY REQUIREMENTS IS ASSESSED

- NASA
  - DEFINES REQUIREMENTS (NHB 1700.7A)
  - DEFINES PROCEDURES (JSC 13830)
  - CONDUCTS ASSESSMENT REVIEW
  - APPROVES SAFETY REPORT
- CUSTOMER
  - CONDUCTS SAFETY EVALUATION
  - PREPARES HAZARD REPORTS
  - SUPPORTS REVIEWS WITH SAFETY DATA PACKAGES
  - VERIFIES COMPLIANCE WITH REQUIREMENTS

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STS/PAYLOAD INTEGRATION
SAFETY REVIEW PROCESS (CONT)

- BASIC REVIEW PROCESS CONSISTS OF A PHASE 0, 1, 2, AND 3 REVIEW CORRESPONDING TO BASIC PAYLOAD DEVELOPMENT MILESTONES OF CONCEPTUAL DESIGN, PRELIMINARY DESIGN, FINAL DESIGN, AND HARDWARE DELIVERY
  - PHASE 0 REVIEW IDENTIFIES HAZARDS
  - PHASE 1 REVIEW IDENTIFIES HAZARD CONTROLS
  - PHASE 2 REVIEW VERIFIES DESIGN, IMPLEMENTS CONTROLS
  - PHASE 3 REVIEW VERIFIES HARDWARE AS BUILT, IMPLEMENTS CONTROLS

STS/PAYLOAD INTEGRATION
SAFETY REVIEW PROCESS (CONT)

218

- NUMBER AND DEPTH OF REVIEWS MAY BE TAILORED
  - COMBINED PHASES
    - — DESIGN MATURITY
    - — COMPLEXITY
  - TELECON REVIEW
    - — LOW HAZARD POTENTIAL
    - — SIMPLE DESIGN
  - NO FORMAL REVIEW
    - — SERIES OF REFLIGHT PAYLOADS WITH MINIMUM CHANGES

S-84-01676A

APPLICATION OF LESSONS LEARNED

- EARLY CONTACT WITH PROSPECTIVE CUSTOMERS HAS AN INFLUENCE ON PAYLOAD AND INTERFACE DESIGN FOR MUTUAL BENEFIT
- CONSIDERING COMBINATION OF JSC (FLIGHT) AND KSC (GROUND) SAFETY REVIEWS IN ORDER TO REDUCE CUSTOMER INTERFACE REQUIREMENTS
- MISSION CONTROL ACCOMMODATIONS FOR CUSTOMERS IMPROVED AND MODERNIZED TO PROVIDE ADDITIONAL MISSION OPERATIONS CAPABILITIES

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S-84-01431A

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APPLICATION OF LESSONS LEARNED (CONT)

- TRY TO AVOID IRREVERSIBLE DEPLOY SEQUENCES
  - ANTENNA DEPLOYS
  - UMBILICAL RETRACT
- REQUIRED ORBITER ACTIVITIES SUCH AS WATER DUMPS, FUEL CELL PURGES, AND FLUSH EVAPORATOR OPERATIONS HAVE LITTLE OR NO EFFECT ON PAYLOAD OPERATIONS
- THE PROCEDURES TO INTEGRATE OPERATIONS BETWEEN JSC AND REMOTE POCC'S HAVE BEEN DEVELOPED AND SUCCESSFULLY UTILIZED
- CUSTOMERS USING REMOTE FACILITIES OUTSIDE OF THE CONTINENTAL UNITED STATES SHOULD PROVIDE DUAL COMMUNICATIONS CAPABILITY

APPLICATION OF LESSONS LEARNED (CONT)

- EXPERIENCE WITH PAYLOAD ACCOMMODATIONS AND FLIGHT PRODUCTS HAS RESULTED IN EXPANDED CAPABILITIES
  - DEPLOYMENT
  - SERVICING
  - RETRIEVAL
  - REPAIR
- ENVIRONMENTS ARE WELL DEFINED; THUS CUSTOMER CAN DESIGN TO STATED ENVIRONMENTS WITH CONFIDENCE THAT THE REQUIREMENTS WILL NOT BE CHANGED

221)

Payload Retrieval Package

Statement of Work

The **payload retrieval package** is a required set of optional services provided to a customer requesting transportation service to replace, to retrieve, or to service a previously deployed, orbiting payload. The specific tasks provided in this fixed-price package are as follows:

- Rendezvous/proximity operations
- Plume impingement and contamination analysis
- Ku-band rendezvous radar analysis
- Flight design
- Crew and ground personnel training
- Use of remote manipulator system (RMS)
- Simulator time for mission planning, analysis, and training
- Payload berthing/stowing in the Orbiter

Retrieval of satellite for return to Earth and repair

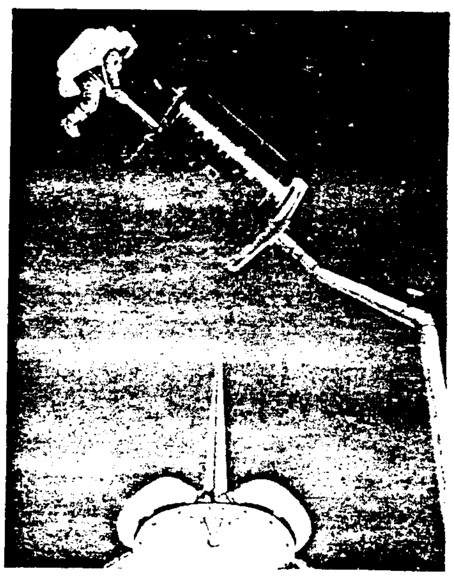


Table 3-1. – Continued

[In fiscal year 1982 U.S. dollars]

Optional service	Department of Defense	Civil U.S. Government	Non-U.S. Government
Payload retrieval package			
Simple	$ 865 000	$ 878 000	$ 1 403 000
Intermediate	1 135 000	1 178 000	1 797 000
Complex	1 568 000	1 625 000	2 330 000

Depending on the individual payload requirements, the retrieval package may be required in one of three degrees of complexity - **simple, intermediate,** and **complex**. The **simple payload retrieval package** generally applies to missions consisting of a routine repeat launch of the Orbiter to the payload orbit to retrieve the payload and a return to the ground, using standard flight design and crew activity procedures. The **intermediate payload retrieval package** generally applies to missions requiring payload servicing associated with the changeout of parts or the repienishment of consumables. The **complex payload retrieval package** would apply to retrieval from a first-of-a-kind rendezvous orbit and/or to more extensive on-orbit payload support such as station-keeping, payload observation, payload checkout, or payload verification.

Terms and Conditions

Payload Integration Managers and Annex Managers at the Johnson Space Center have the final responsibility for determining which, if any, of the fixed-price payload retrieval package categories apply to the specific payload retrieval requirements.

The total price for a customer requiring NSTS services to retrieve a payload on orbit consists of three major components: the payload retrieval package as described in this section of the document, the standard NSTS transportation charge to and from the retrieval orbit, and, if required, the additional optional services described in other sections of this document. A stand-alone payload retrieval pricing policy governing all retrieval services and pricing principles is under review within NASA for publication in the near future.

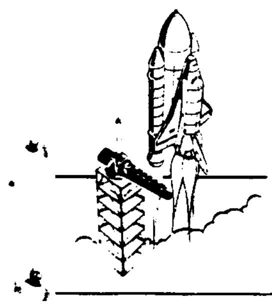
Period of Performance

The tasks associated with the payload retrieval package will normally begin 24 months before launch (L - 24).

ZZZ

Example of activities associated with the categories of payload revisit

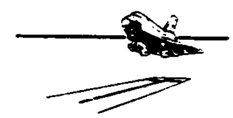
Simple



GRAPPLE PAYLOAD WITH REMOTE MANIPULATOR SYSTEM (RMS)

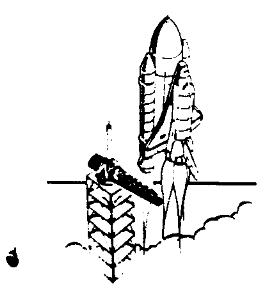
PLACE PAYLOAD INTO PAYLOAD BAY

LAUNCH



LANDING

Intermediate

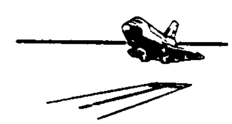


GRAPPLE PAYLOAD WITH RMS

PARTS CHANGEOUT WHILE PAYLOAD IS BERTHED IN PAYLOAD BAY

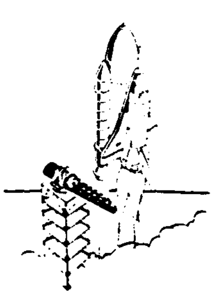
RELEASE PAYLOAD BACK INTO ORBIT

LAUNCH

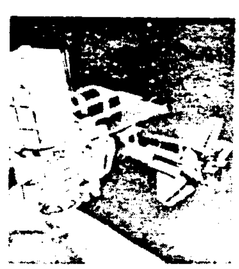


LANDING

Complex



ORBITER STATION-KEEPING WITH PAYLOAD

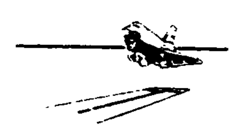


MANNED MANEUVERING UNIT (MMU) GRAPPLE WITH PAYLOAD

PARTS CHANGEOUT IN PAYLOAD BAY

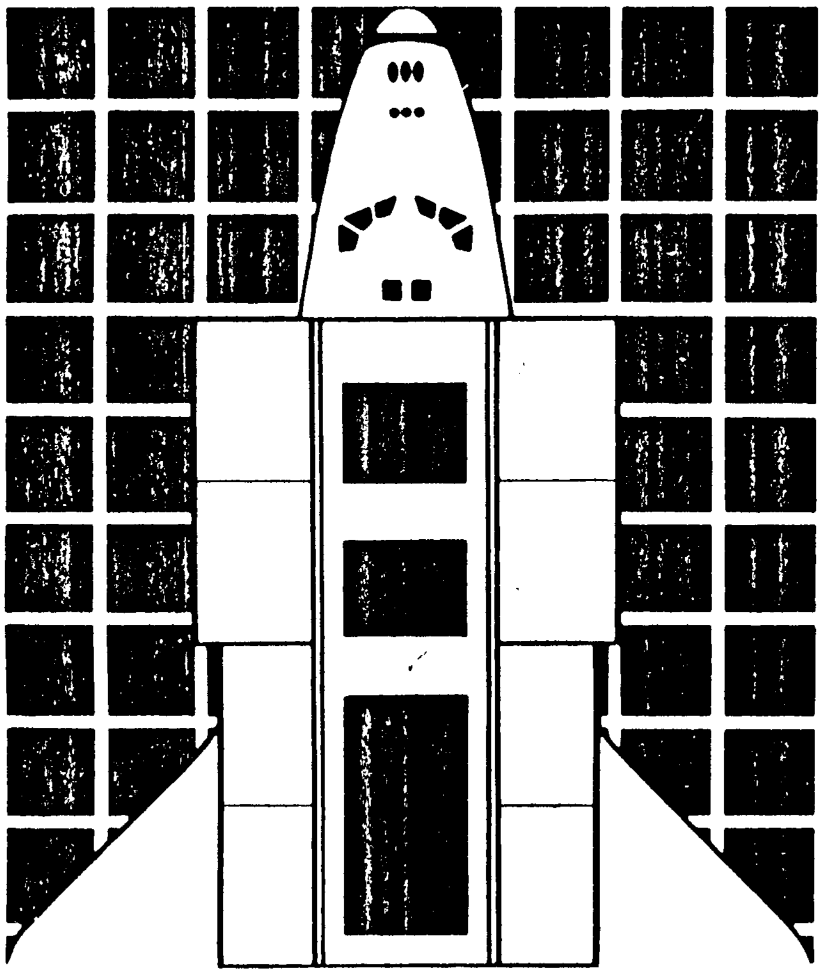
RELEASE PAYLOAD BACK INTO ORBIT

LAUNCH



LANDING

S84-05145



SPACECRAFT STANDARD RETRIEVAL POLICY

OPERATIONAL ORBIT

- DEPLOY BY NSTS AT 160-N. MI. ALTITUDE IN 28.45° INCLINATION ORBIT AND RETRIEVE AT 170-N. MI. ALTITUDE
  - LARGE NUMBER OF CUSTOMERS REQUIRE THE STANDARD 160-N. MI. ALTITUDE ORBIT. A SECOND "STANDARD" WOULD SEPARATE CUSTOMERS INTO UNMIXABLE GROUPS, MAKING MANIFESTING AND FLIGHT DESIGN MORE DIFFICULT
  - INCREASED ORBIT ALTITUDE AND INCLINATION REDUCES NSTS LIFT CAPABILITY — APPROXIMATELY 10 000 LB FOR A 250-N. MI. DEPLOY ALTITUDE
- SPACECRAFT SHOULD PLAN TO BOOST ORBIT TO HIGHER ALTITUDE (TYPICALLY 200 TO 300 N. MI.) AND BACK TO 170 N. MI. FOR RETRIEVAL
  - HIGHER ALTITUDE REQUIRED TO PROVIDE ADEQUATE ORBIT LIFETIME

TIME INTERVAL — DEPLOY TO RETRIEVAL

S-84-05028A

- MINIMUM TIME BETWEEN DEPLOY AND RETRIEVAL FLIGHTS IS 6 MONTHS
  - PROVIDES NECESSARY DECOUPLING BETWEEN THE TWO FLIGHTS. THIS PROVIDES ADEQUATE TIME TO REMANIFEST AND REPLAN THE RETRIEVAL FLIGHT IN THE EVENT OF AN ANOMALOUS OR FAILED DEPLOY MISSION
- ONE RETRIEVAL FLIGHT WILL BE PLANNED BY THE NSTS DURING A 90-DAY INTERVAL, WHICH BEGINS NO EARLIER THAN 6 MONTHS AFTER THE DEPLOY FLIGHT
  - WHEN RETRIEVALS ARE MIXED WITH THE STANDARD LAUNCH WINDOW FOR DEPLOYABLE SPACECRAFT, LAUNCH OPPORTUNITIES OCCUR EVERY 6 WEEKS (3 WEEKS, IF REQUIREMENTS PERMIT). THE 90-DAY INTERVAL PROVIDES A REASONABLE ASSURANCE THAT THE NSTS CAN MANIFEST A MISSION THAT HAS A SUITABLE CARGO AND LAUNCH OPPORTUNITY
- IN THE EVENT A RETRIEVAL IS NOT ACCOMPLISHED AS PLANNED, A RETRIEVAL REFLIGHT OPPORTUNITY WILL BE PROVIDED NO EARLIER THAN 6 MONTHS AFTER THE PLANNED RETRIEVAL
  - CONSISTENT WITH REFLIGHT POLICY FOR ALL NSTS CUSTOMERS
  - THE RETRIEVAL SPACECRAFT SHOULD BE ABLE TO ACCOMMODATE ALL POLICY REQUIREMENTS ON BOTH THE NOMINAL RETRIEVAL AND REFLIGHT OPPORTUNITY
  - IMPLIES REBOOST TO HIGHER ORBIT TO ACHIEVE REQUIRED ORBIT LIFETIME

SPACECRAFT MANEUVER AND TARGETING REQUIREMENTS

S-84-05031

- AT DEPLOYMENT THE NSTS WILL SPECIFY AN INERTIAL ORBIT PLANE ORIENTATION IN WHICH THE SPACECRAFT MUST BE AT RETRIEVAL
  - THIS REQUIREMENT ALLOWS THE NSTS THE OPTION OF PERFORMING MULTIPLE RETRIEVALS ON A SINGLE FLIGHT — AN IMPORTANT ASPECT OF A COST-EFFECTIVE RETRIEVAL SERVICE
  - THIS ORIENTATION WILL BE SELECTED SO THAT IT IS ACHIEVABLE BY FLYING WITHIN ±15 N. MI. OF THE NOMINAL ALTITUDE AND USING NODAL REGRESSION TO CONTROL THE PLANE ORIENTATION. THIS MINIMIZES OPERATIONAL AND PROPELLANT REQUIREMENTS
  - THE SPACECRAFT IS RESPONSIBLE FOR COMPENSATING FOR DEVIATIONS CAUSED BY ATMOSPHERIC DRAG UNCERTAINTIES
- THE SPACECRAFT POSITION WITHIN THE PLANE AT RETRIEVAL WILL BE SPECIFIED BY THE NSTS
  - REQUIRES THE SPACECRAFT TO PERFORM A RENDEZVOUS OF ITS OWN WITH A "GHOST TARGET" PRIOR TO ORBITER ARRIVAL
  - ALLOWS THE NSTS TO COMPLETE RENDEZVOUS WITHIN TIME, MANEUVER CAPABILITY, AND LAUNCH OPPORTUNITY RESTRICTIONS IMPOSED BY TYPICAL SHARED RETRIEVAL FLIGHTS

SPACECRAFT MANEUVER AND TARGETING REQUIREMENTS (CONT)

S-84-05032

- ACCURACY WITH WHICH THE SPACECRAFT MUST PERFORM THIS "GHOST TARGET" RENDEZVOUS IS DEFINED BY A REGION AROUND THE TARGET KNOWN AS THE "CONTROL BOX"
  - NECESSARY TO KEEP NSTS RENDEZVOUS PROPELLANT BUDGETS REASONABLE AND TO ENHANCE THE PROBABILITY OF A SUCCESSFUL RENDEZVOUS
  - THE EXACT SIZE AND SHAPE OF THE CONTROL BOX STILL IS BEING DEVELOPED
- TARGETS FOR THE SPACECRAFT RETRIEVAL WILL BE PROVIDED AS FOLLOWS:
  - INITIAL TARGETS SHORTLY AFTER DEPLOYMENT TO DEFINE PLANE ORIENTATION, DATE, AND TIME OF REVISIT
  - UPDATED TARGETS 30 DAYS PRIOR TO RETRIEVAL FLIGHT LAUNCH TO DEFINE PLANE ORIENTATION, DATE AND TIME OF REVISIT, AND REQUIRED POSITION (PHASE ANGLE) IN THE ORBIT PLANE
  - FINAL TARGETS SHORTLY AFTER RETRIEVAL LAUNCH TO DEFINE THE FINAL PLANE ORIENTATION AND THE CONTROL BOX START TIME

ON-ORBIT TIMELINE

S-84-05030

- A "GO-FOR-DESCENT" WILL BE GIVEN TO THE RETRIEVAL CUSTOMER WITHIN THE FIRST FEW HOURS AFTER A SUCCESSFUL NSTS LAUNCH
  - ENSURES NSTS SUCCESSFULLY ON-ORBIT AND OPERATIONAL BEFORE COMMITTING TO MANEUVER THE SPACECRAFT TO THE RETRIEVAL ORBIT. MAXIMIZES PROBABILITY OF SUCCESSFUL RETRIEVAL BEFORE SPACECRAFT PROPELLANT IS EXPENDED
- THE SPACECRAFT MUST BE CAPABLE OF COMPLETING DESCENT TO THE CONTROL BOX WITHIN 48 HOURS AFTER "GO-FOR-DESCENT" IS GIVEN. THIS INCLUDES TRACKING AND STATE VECTOR DETERMINATION
  - MORE THAN 48 HOURS MAY BE AVAILABLE ON A PARTICULAR FLIGHT, DEPENDING ON CARGO AND TIMELINE
- THE SPACECRAFT MUST BE CAPABLE OF MAINTAINING A SAFE RETRIEVAL-READY MODE FOR AT LEAST 3 HOURS
  - THIS INCLUDES HAVING AN ACCEPTABLE VALUE FOR ORBITER/SPACECRAFT DIFFERENTIAL DRAG
- UP TO 3 HOURS WILL BE DEDICATED TO PAYLOAD BERTHING AFTER RENDEZVOUS AND GRAPPLE
  - THE SUBSEQUENT CREW WORK PERIOD MAY BE PURCHASED AS AN OPTIONAL SERVICE FOR ADDITIONAL TIME

SUMMARY

- NSTS WILL DEPLOY THE SPACECRAFT IN 160-N. MI. ALTITUDE ORBIT AT 28.45° INCLINATION
- SPACECRAFT WILL PROVIDE BOOST TO ACHIEVE OPERATIONAL ORBIT (TYPICALLY 200 TO 300 N. MI.)
- ONE RETRIEVAL OPPORTUNITY WILL BE PROVIDED DURING A 90-DAY INTERVAL BEGINNING NO EARLIER THAN 6 MONTHS AFTER DEPLOYMENT
- RETRIEVAL ORBIT ALTITUDE WILL BE 170 N. MI.
- SPACECRAFT DEBOOST TO RETRIEVAL ORBIT WILL BEGIN AFTER NSTS LAUNCH FOR THE RETRIEVAL FLIGHT AND THE ORBITER SYSTEMS ARE VERIFIED OPERATIONAL FOR RETRIEVAL
- DESCENT TO RETRIEVAL ORBIT WILL BE COMPLETED WITHIN 48 HOURS AFTER "GO-FOR-DESCENT" IS GIVEN
- SPACECRAFT WILL PROVIDE CONTROL OF POSITION IN THE ORBITAL PLANE (PHASE ANGLE) TO FACILITATE RENDEZVOUS AND WILL PROVIDE LIMITED ORBIT PLANE ORIENTATION CONTROL
- A RETRIEVAL REFLIGHT WILL BE PROVIDED, IF NECESSARY, DURING A 90-DAY INTERVAL BEGINNING NO EARLIER THAN 6 MONTHS AFTER THE PLANNED RETRIEVAL

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