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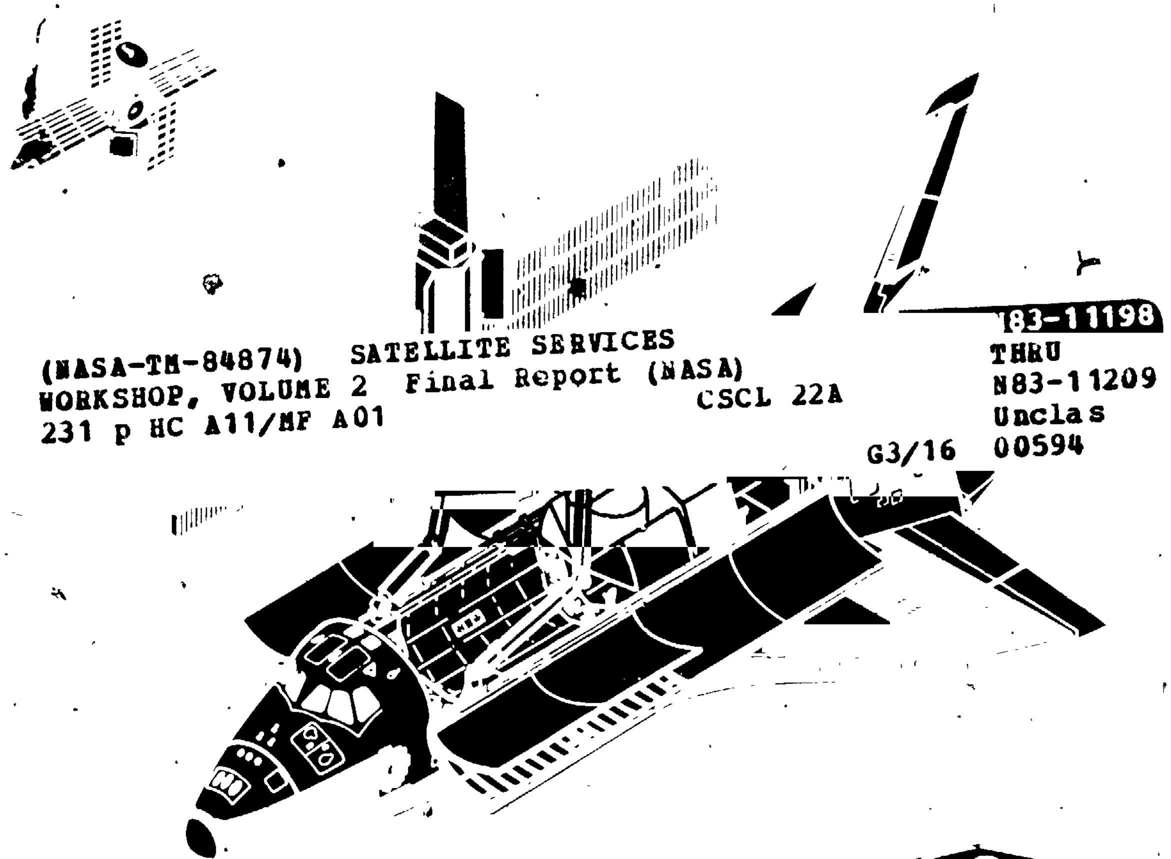

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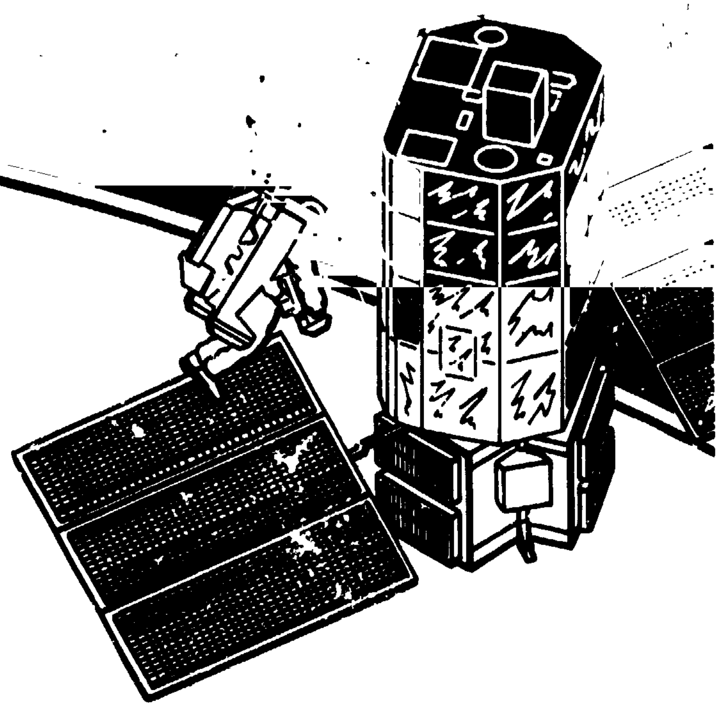

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This document consists of the presentations submitted at the Satellite Services Workshop. Its purpose is to provide a forum for the exchange of information and the identification of key issues associated with on-orbit servicing of satellites. Responsibility for content and technical accuracy lies with each respective author. Prior to formal publication elsewhere, the data presented herein may not be used without the author's permission.

National Aeronautics and Space Administration
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ADVANCED SERVICE EQUIPMENT

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For

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NASA JOHNSON SPACE CENTER
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ADVANCED SERVICE EQUIPMENT

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1. INTRODUCTION

Orbiter-based service equipment needs/usage have been identified by considering a broad spectrum of on-orbit operational scenarios associated with three primary mission events:

- Initial launch
- Revisits
- Earth return.

The scenarios reflected the types of satellite classes shown in Fig. 1, including nominal and alternate modes of operation, contingency situations (as Remote Manipulator System (RMS) inoperative), and Orbiter close proximity operations.

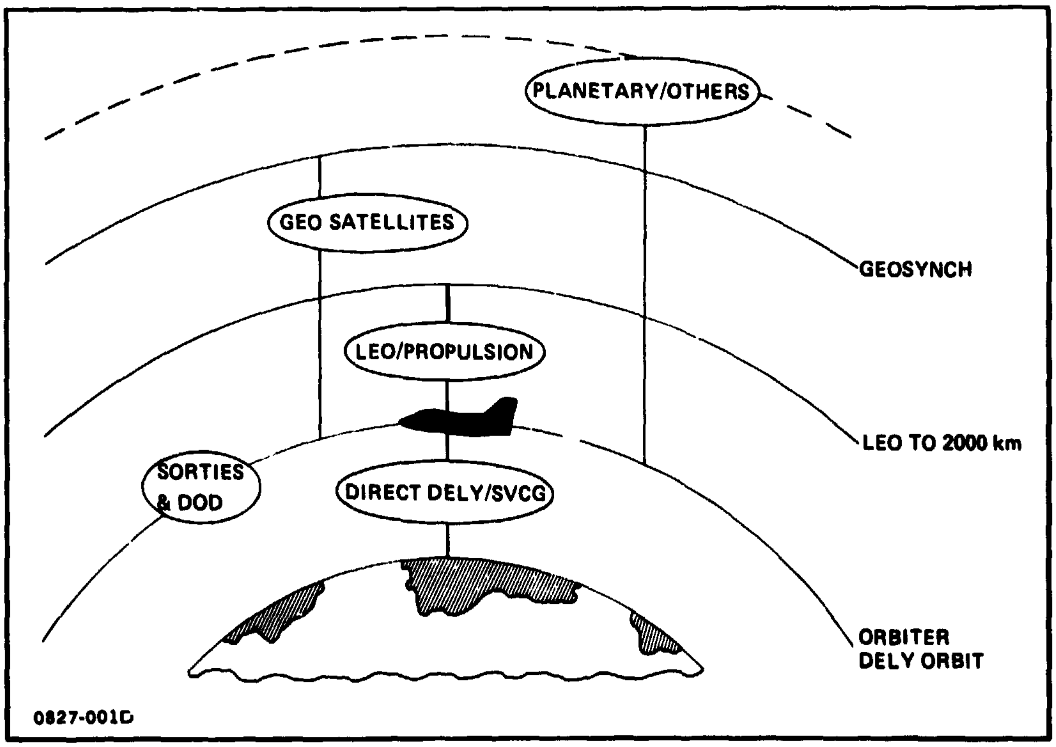

Fig. 1 Satellite Classes

2. ON-ORBIT OPERATIONS

In developing the on-orbit operations servicing scenarios, the following fundamental goals or objectives were sought:

- Attempt to standardize on-orbit service operations
- Maximize the use of existing equipment or those under development
- Enhance the utilization of the Space Transportation System to the satellite user community
  - Minimize service equipment user charges and cost of on-orbit operations
  - Maximize mission success prospects.

A simplified on-orbit sequence of events for a revisit mission is shown in Fig. 2. This sequence of events represents scenarios where the RMS is used to retrieve satellites and place them on a Handling and Positioning Aid (HPA) for on-orbit servicing. In this scenario, the HPA becomes the "standard location" at which servicing and check-out of the satellite is performed, and from which the spacecraft is redeployed from the Orbiter. Retrieval is accomplished by the Orbiter after inspection of the satellite. The service equipment needs associated with a particular event are highlighted in Fig. 2. The initial events call for:

- Maneuverable Television (MTV)
- Remote Manipulator System (RMS) and associated Aft Flight Deck Controls/Displays (AFD C&D)
- Aft Flight Deck Controls and Displays (AFD C&D) for close proximity flight control of the MTV.

Subsequent operations identify:

- HPA
- Work Platform for the HPA
- Open Cherry Picker (OCP) and RMS
- AFD C&D for satellite checkout/servicing support
- Equipment stowage/fluid transfer system for servicing support.

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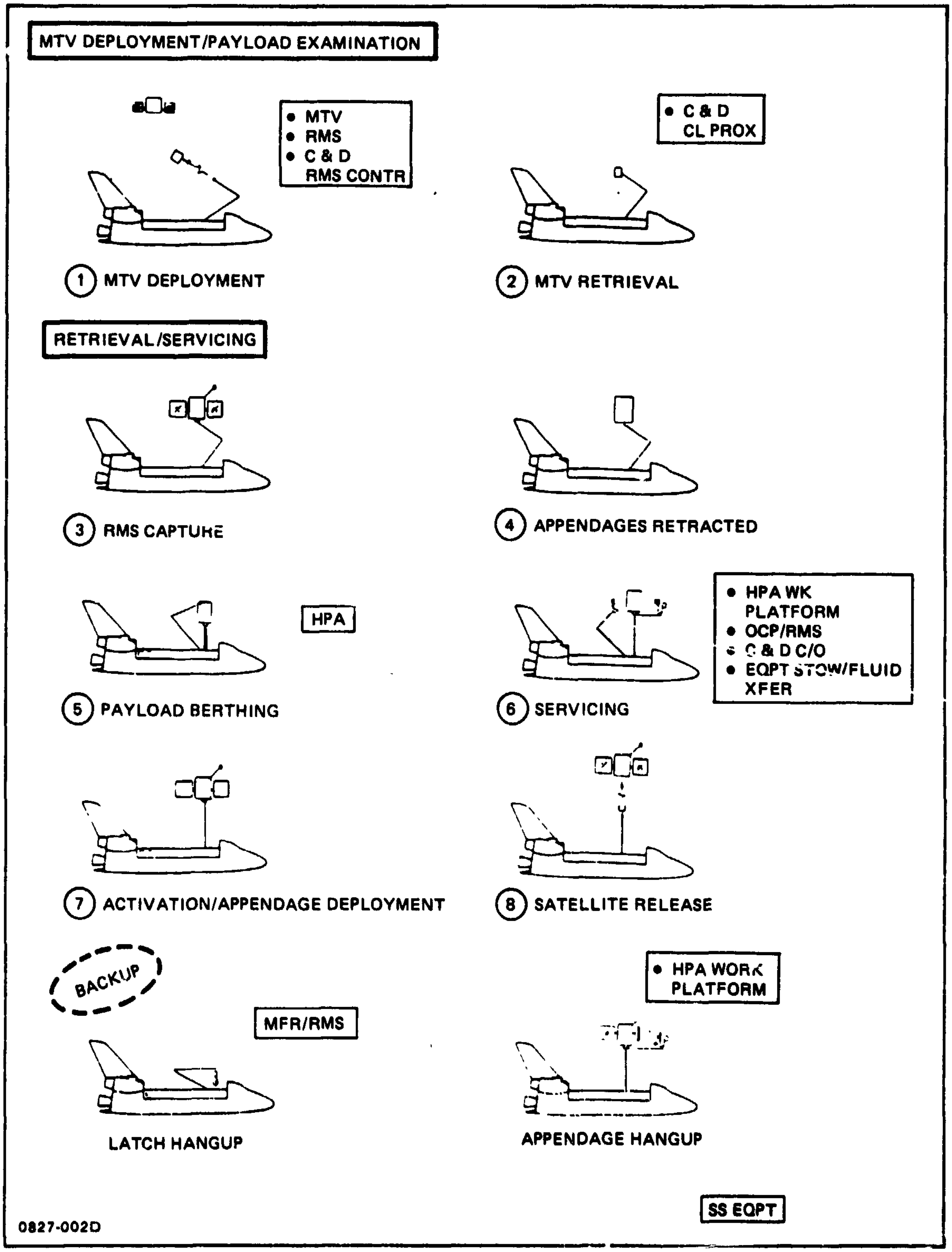

Fig. 2 Representative Revisit Scenario

The backup situations identify the following equipment needs:

- Manipulator Foot Restraint (MFR) to cover latch hangups
- HPA work platform (also identified above) to assist a potential satellite appendage hangup.

The service equipment identified in the broad spectrum of operational scenarios and satellite classes that have been analyzed can be conveniently grouped within the following satellite service operations:

- Payload Deployment
- Close Proximity Retrieval
- On-Orbit Servicing
- Backup/Contingency
- Delivery/Retrieval of High Energy Payloads (LEO/Propulsion Class)
- Earth Return
- Optional Services.

1.1 PAYLOAD DEPLOYMENT EQUIPMENT

Satellite service equipment associated with payload deployment operations includes:

- Retention Structures
- RMS*
- Tilt Table
- Payload Installation/Deployment Aid (PIDA)*
- HPA*
- Spin Table
- (AFD C&D).

Equipment noted with an asterisk (*) is described and illustrated herein.

1.1.1 Remote Manipulator System

The RMS can be used to deploy payloads from the Orbiter payload bay. Of particular note to the satellite user are the standard RMS elements: the snare end effector and its compatible grapple fixture (see Fig. 3). These elements have been

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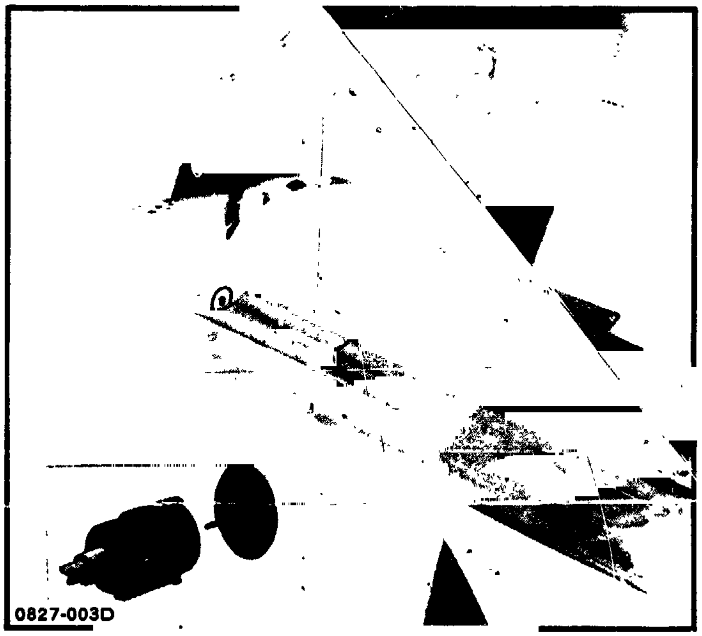

Fig. 3 Deployment Retrieval - Remote Manipulator System

designed to release a satellite with essentially no differential velocity during deployment. Nominal capabilities of the RMS are:

- Payload Handling Capability: 65,000 lb
- Positioning Accuracy: 2 in. ± 1 deg within reach envelope
- Payload Release: ±5 deg attitude < 0.015 deg/sec relative.

The RMS is also used to retrieve satellites, when they are within the reach distance of the RMS arm, to enable on-orbit servicing or earth return.

1.1.2 Payload Installation and Deployment Aid

The Payload Installation and Deployment Aid (PIDA) is a mechanism which enables deployment (and reinstallation) from the payload bay of very large size/mass payloads (e.g., 15 ft diameter and 65,000 lb). The device, which has been under development at the NASA Johnson Space Center, provides automatic deployment and stowing of satellites having minimum clearance envelopes with the Orbiter payload bay. Figure 4 shows the PIDA having lifted a large satellite out of the payload bay and transferred it to the RMS, to enable its subsequent checkout and deployment from the Orbiter.

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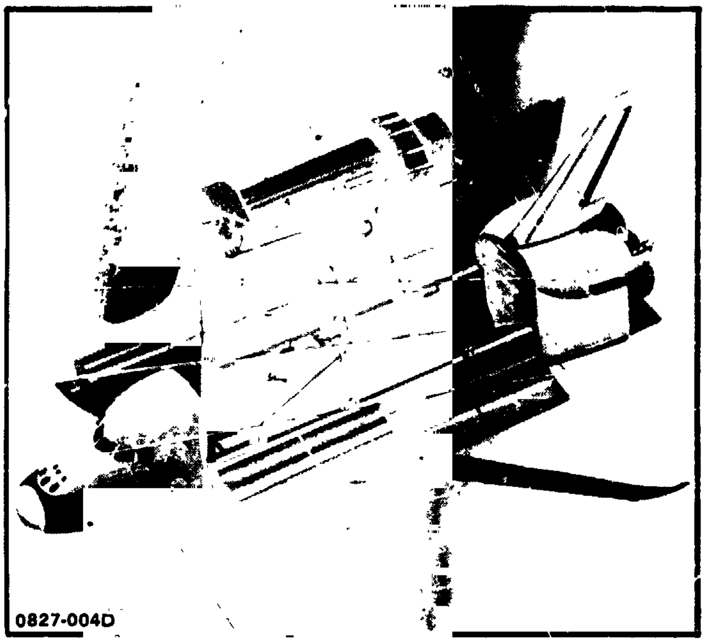

Fig. 4 Deployment - Payload Installation & Deployment Aid

1.1.3 Handling and Positioning Aid

The HPA will support satellites outside the confines of the payload bay and, with its "over-the-side" feature, could enable full deployment of satellite appendages (if desired) prior to release from the Orbiter (Fig. 5.). For initial launch missions, the HPA contains a standardized berthing and umbilical interface for checkout prior to deployment, has provisions for transferring attitude/state vector information to the satellite from the Orbiter navigation system, and provides the means to impart a separation velocity between the satellite and Orbiter during deployment.

Figure 6 is an adaptation of a spin table to the HPA. The spin table would be equipped with a stored energy device to impart a separation Δv for deployment. The HPA platform can also be rotated to direct the separation Δv in a desired direction. Orbiter attitude requirements are thus relieved in meeting separation requirements.

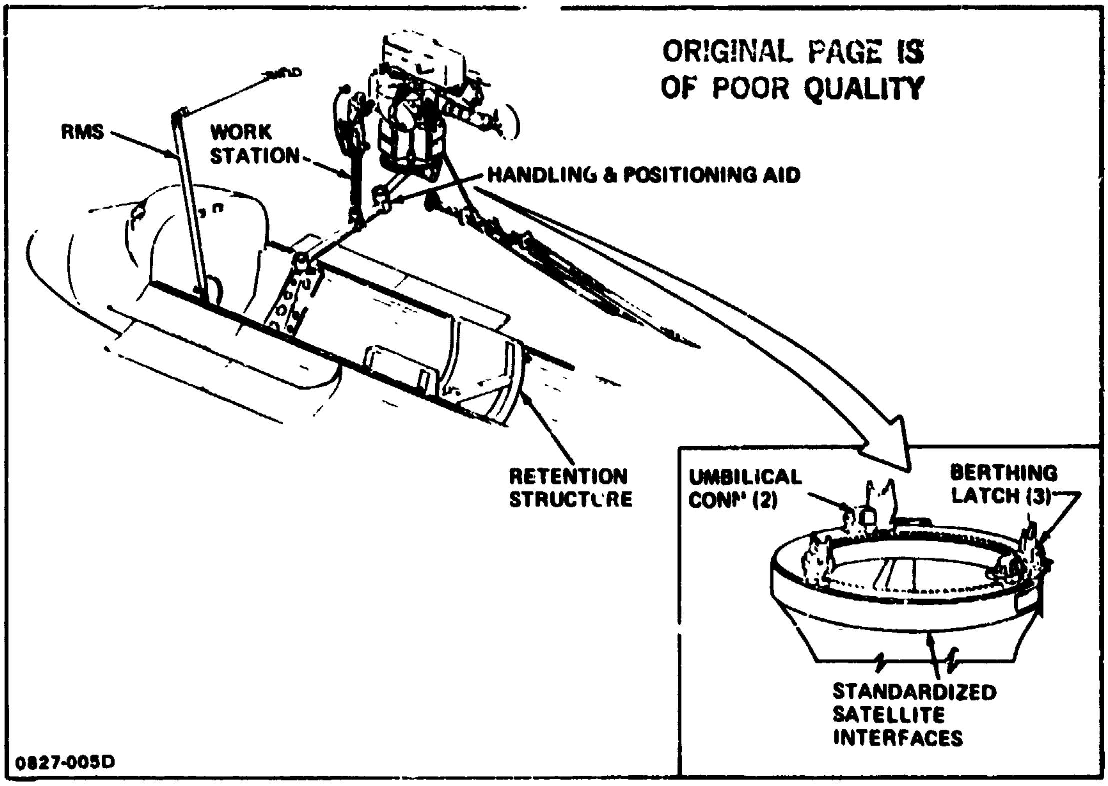

Fig. 5 Handling & Positioning Aid

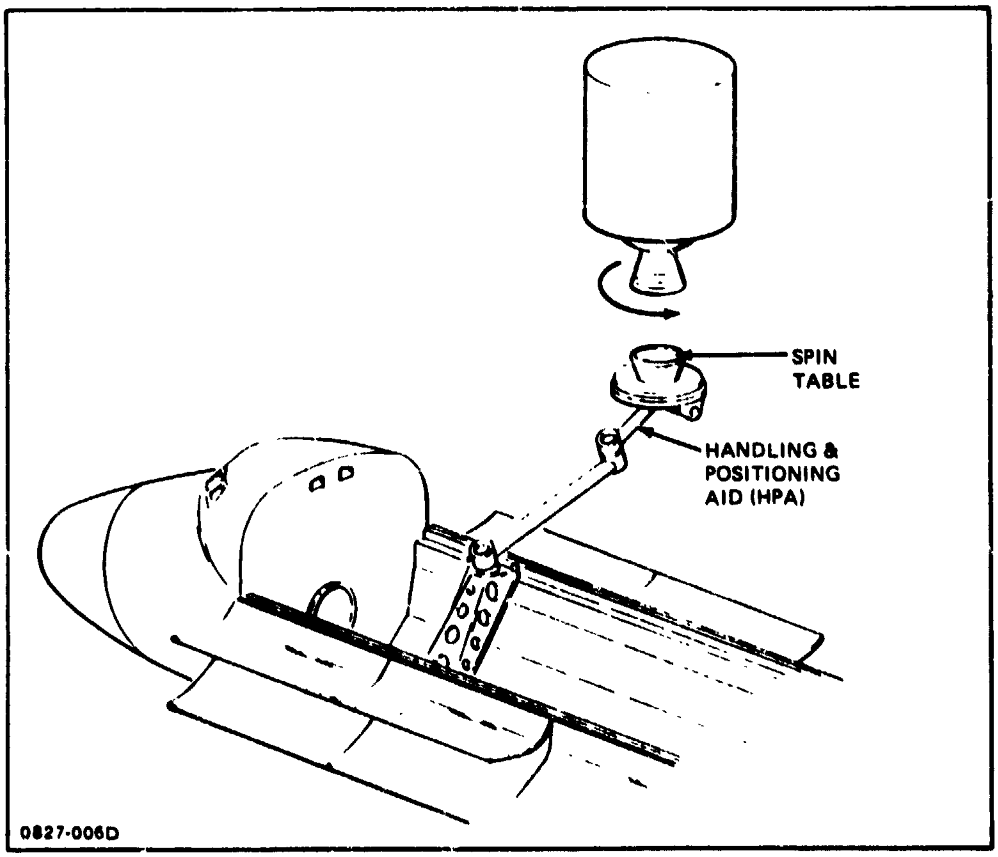

Fig. 6 Spin Table - HPA Adaptation

1.2 CLOSE PROXIMITY RETRIEVAL EQUIPMENT

Satellite service equipment associated with close proximity retrieval operations includes:

- RMS
- MTV*
- Proximity Operations Modules (POM)*
  - MTV Adaptation
  - Manned Maneuvering Unit/Work Station (MMU/WS) Adaptation
- AFD C&D.

Equipment noted with an asterisk (*) is described and illustrated herein.

1.2.1 Maneuverable Television

An MTV, is shown in Fig. 7 being deployed from the Orbiter by the RMS. The MTV is a free-flying spacecraft, remotely flown by the Orbiter crew from the Aft Flight Deck (AFD), with video and telementry transmission back to the Orbiter.

The MTV has a range of about three miles and is used to remotely examine all satellites prior to Orbiter retrieval. It can also be deployed to view and record propulsion stage firings of satellites destined for higher energy LEO altitudes or geostationary orbit. Following its examination mission, the MTV is flown back to the Orbiter and retrieved by the RMS.

Also shown in Fig. 7, in retracted position, is the HPA which is deployed over-the-side to provide a fixed platform for spacecraft servicing aboard the Orbiter.

The MTV is shown in Fig. 8 examining a spacecraft prior to retrieval for servicing. This free-flying spacecraft is remotely flown by the Orbiter crew from the AFD.

Fig. 7 Maneuverable Television Deployment

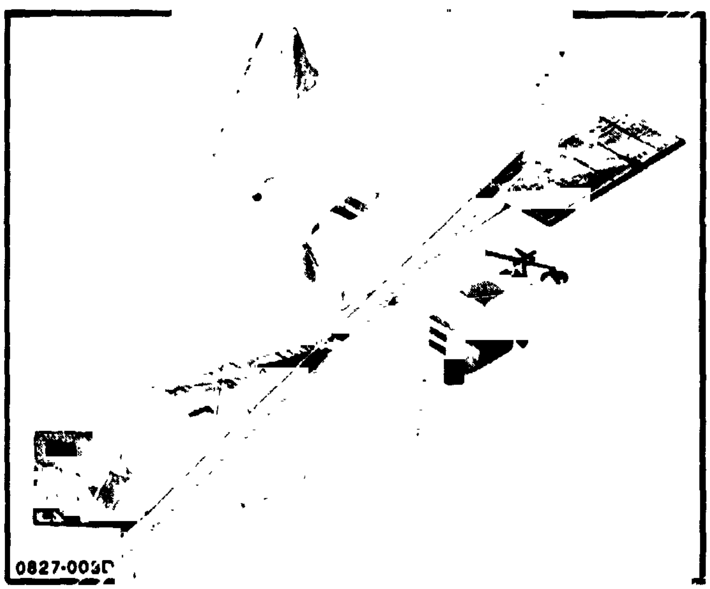

Fig. 8 Satellite Examination via MTV

1.2.2 Unmanned Proximity Operations Module-Satellite Capture/Retrieval

The Orbiter can readily rendezvous with a satellite to within 1000 ft separation distance. However, concerns by some satellite users regarding Orbiter thruster plume impingement or contamination during terminal closure maneuvers could preclude direct Orbiter rendezvous/retrieval of a spacecraft. Retrieval of satellites within a 1000 ft range can be accomplished by an adaptation of the MTV called the Unmanned Proximity Operations Module (POM).

Controlled by the Orbiter crew, the POM would be dispatched to capture the satellite and return it to within the reach distance of the RMS. It would be flown via TV (essentially using MTV equipment) and capture its target via the satellite's RMS-compatible grapple fixture. The POM utilizes a non-contaminating, cold gas propulsion system which provides three axes of control during free flight and satellite towing operations. Figure 9 shows the unmanned POM, equipped with an extendable mast and RMS end-effector, as it is about to capture a satellite.

Figure 9 also shows an unmanned POM towing a satellite to the Orbiter. The POM would stabilize/position the satellite within reach distance of the RMS arm and then detach itself from the satellite's grapple fitting to allow the RMS to capture the satellite. Following capture, the RMS would place the satellite on a Tilt Table or HPA to enable on-orbit servicing.

1.2.3 Manned Proximity Operations Module - Satellite Capture/Retrieval

Retrieval of satellites within a 1000 ft separation distance of the Orbiter can also be accomplished by a manned Proximity Operations Module (POM). The manned POM is an adaptation of a Work Station (WS) that can be used in conjunction with an MMU to retrieve moderate sized satellites of the Multimission Modular Spacecraft class (Fig. 10).

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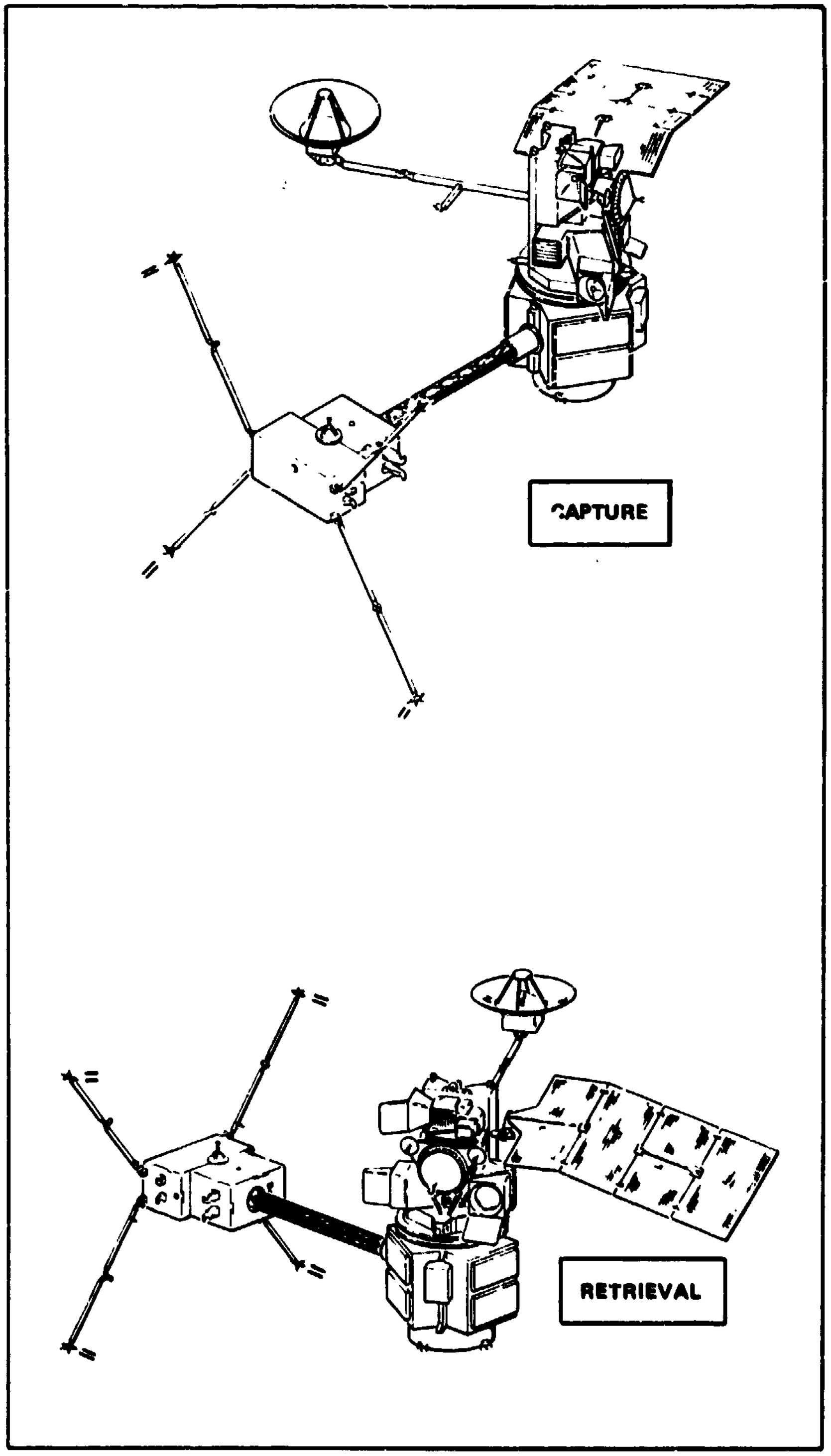

Fig. 9 Unmanned Proximity Operations Module

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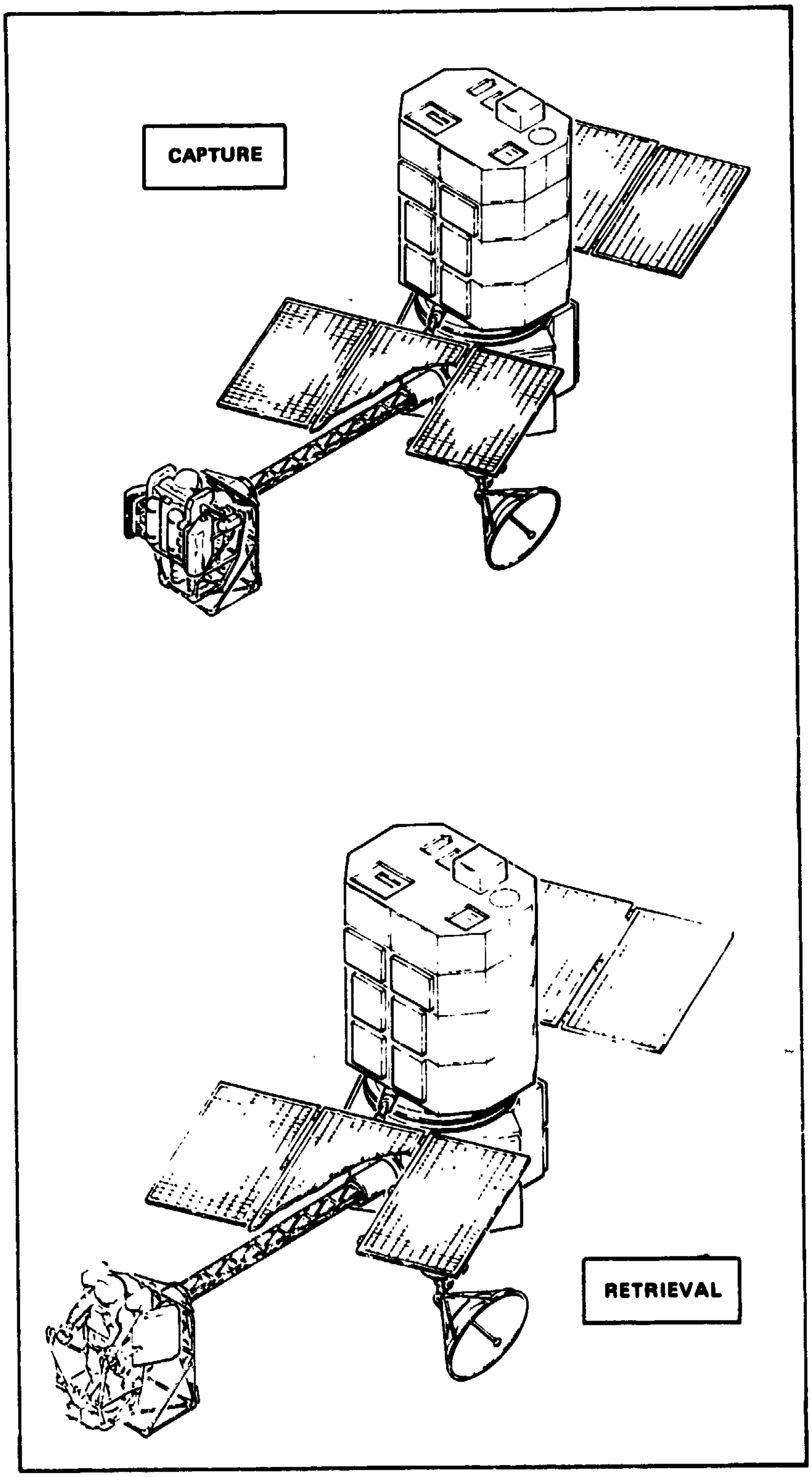

Fig. 10 Manned Proximity Operations Module

The WS is equipped with an extendable mast and an RMS end effector which are mounted to a support structure. This enables the astronaut to fly with the snare end effector in a forward position during satellite engagement and in an aft position during satellite towing operations. An astronaut would fly the manned POM to the satellite, capture it via the satellite's RMS-compatible grapple fixture, and tow the satellite to within reach distance of the RMS. Figure 10 shows the manned POM "flying-in" the end effector to engage the satellite's grapple fixture. As most of the major hardware elements for this concept exist or are in late stages of development, the manned POM is a conceivable choice for near-term satellite retrieval missions.

Figure 10 also illustrates the manned POM towing a spacecraft to the Orbiter. Using the flight control capabilities of the MMU, the astronaut would stabilize/position the satellite within reach distance of the Orbiter RMS arm. The POM would then detach itself from the satellite's grapple fixture to allow the RMS to capture the satellite. Following capture, the RMS would place the satellite on a Tilt Table or HPA for on-orbit servicing.

1.3 ON-ORBIT SERVICING EQUIPMENT

Satellite service equipment associated with on-orbit servicing operations includes:

- Open Cherry Picker/Remote Manipulator System (OCP/RMS)*
- Tilt Table/OCP Work Platform*
- HPA*
- Equipment Storage
- Fluid Transfer*
- Non-Contaminating Attitude Control System (ACS)*
- AFD C&D.

Equipment noted with an asterisk (*) is described and illustrated herein.

1.3.1 Open Cherry Picker Servicing

The OCP is a movable work station controlled by an astronaut on the tip of the RMS arm. Servicing capabilities include lighting, tool storage, a payload handling and transport device, and a stabilizer to rigidly position the astronaut at the work site.

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Figure 11 depicts an astronaut replacing an equipment module on a representative Multimission Modular Spacecraft. The OCP, with its movable foot restraint, reduces the physical effort associated with performing EVA and, with its control station, allows the astronaut to fly himself into the most convenient position to perform service functions within the Orbiter Payload bay.

Fig. 11 Open Cherry Picker Servicing

1.3.2 Open Cherry Picker - FSS Work Platform Servicing

An OCP can be adapted to the FSS Cradle A' Tilt Table as shown in Fig. 12, to provide a convenient work platform for spacecraft servicing. The work platform can be positioned at varying distances from the satellite and, with the 360° rotational feature of the Tilt Table, provide total access to all satellite locations.

1.3.3 Handling and Positioning Aid

The HPA will support satellites outside the confines of the payload bay and, with its "over-the-side" feature, enables full deployment of satellite appendages during servicing, and prior to release from the Orbiter.

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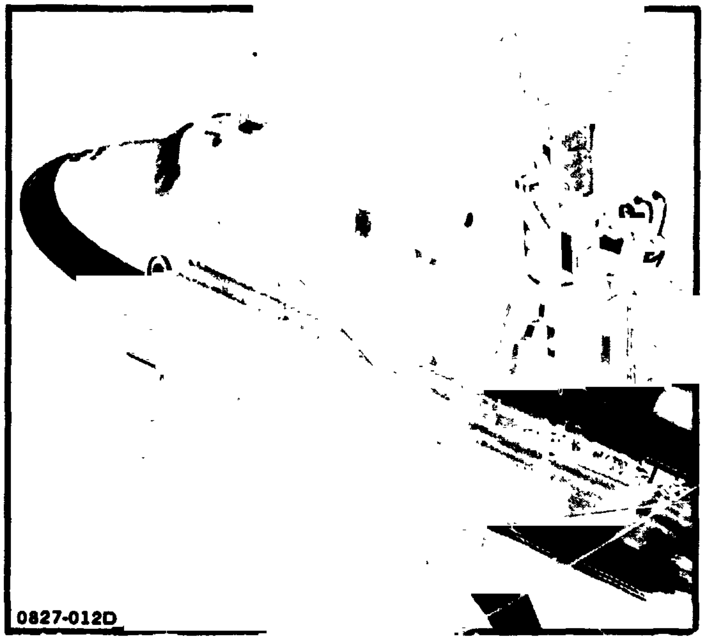

Fig. 12 Open Cherry Picker - FSS Work Platform Servicing

On-orbit servicing is accommodated by rotating turn-table provisions in the HPA and via a movable work platform incorporating an OCP. The work platform has translational and vertical motion capability which, with the HPA turn-table features, enables total access to all satellite locations. The standardized berthing and umbilical interface also contains a fluid coupling interface to transfer propellants during servicing missions.

Figure 13 depicts a two-astronaut servicing capability. One astronaut is shown servicing a segment of the satellite via the OCP mounted to the end of the RMS arm. The second astronaut utilizes the OCP work platform on the HPA.

Although not shown in the illustration, the OCP with its stabilizer feature could attach itself to the satellite, release from the RMS, and enable the RMS to transport equipment from the Orbiter payload bay to the respective work stations.

Fig. 13 Servicing via Handling & Positioning Aid

1.3.4 Fluid Transfer

Provisions are needed for storage and transfer of propellants for satellites and the Versatile Service Stage (VSS). Fluids replenishment could involve both direct tankage/fluid replacement and transfer of propellants via a special fluid transfer system. Either approach, however, is dependent on more detailed definitions of satellite(s) and propulsion stage requirements than are presently available.

Figure 14 depicts a fluid transfer module in the payload bay replenishing propellant for a VSS through an interface connection in the HPA.

1.3.5 Non-Contaminating Attitude Control System

Orbiter servicing of contamination-sensitive satellites can be accomp... by providing a non-contaminating ACS package in the payload b... he package would provide precision, long-term attitude control wit... ne use of the Orbiter's primary or vernier reaction control system... ...ernatively, and if acceptable, the Orbiter could be placed into a ... drift mode.

...gure 15 shows a non-contaminating ACS concept consisting of Skylab-type CMGs located in the payload bay, with cold gas thrusters/N_2 propellant mounted on extensible arms to serve as momentum unloading devices.

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Fig. 14 Fluid Transfer

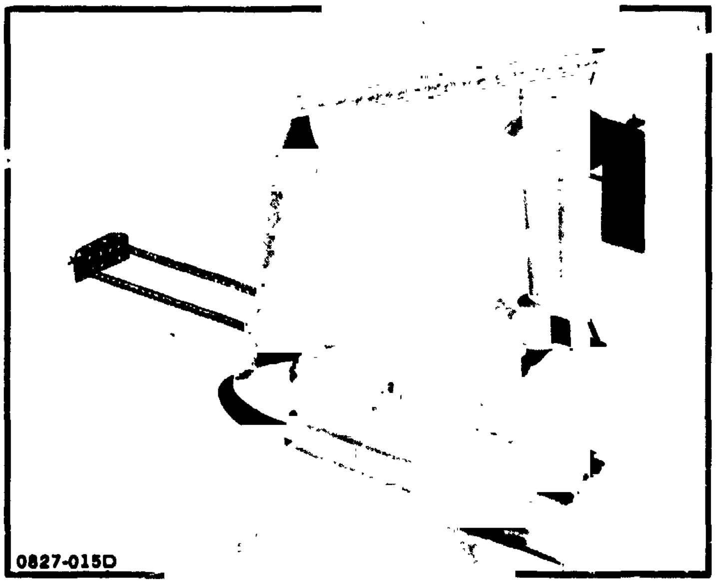

Fig. 15 Non-Contaminating ACS - Servicing of Contamination Sensitive Satellites

1.4 BACKUP/CONTINGENCY EQUIPMENT

Satellite service equipment associated with backup/contingency operation includes:

- Manipulator Foot Restraint/Remote Manipulator System (MFR/RMS)*
- Manned Maneuvering Unit/Work Station Unit (MMU/WS) adaptations*
  - End Effector for Satellite Deployment
  - Stabilizer for Mechanical Hangup Situations
  - Payload Handling for On-Orbit Servicing Support.

Equipment noted with an asterisk (*) is described and illustrated herein.

1.4.1 Manipulator Foot Restraint - Backup for Mechanical Hangups

The Manipulator Foot Restraint (MFR) is mounted on the end of the RMS arm and used to support contingency operations in the payload bay which require EVA. The MFR/RMS serves as a backup for potential hangup of retention latches, mechanical hangup situations associated with satellite appendage deployment, and EVA support of sortie missions.

Figure 16 shows an astronaut being deployed on the MFR to manually release a retention latch and to support sortie missions in the payload bay. In addition to providing the astronaut with a foot restraint which reduces physical effort required to perform EVA tasks, the MFR includes a tool bin to carry supporting tools that may be needed for backup operations.

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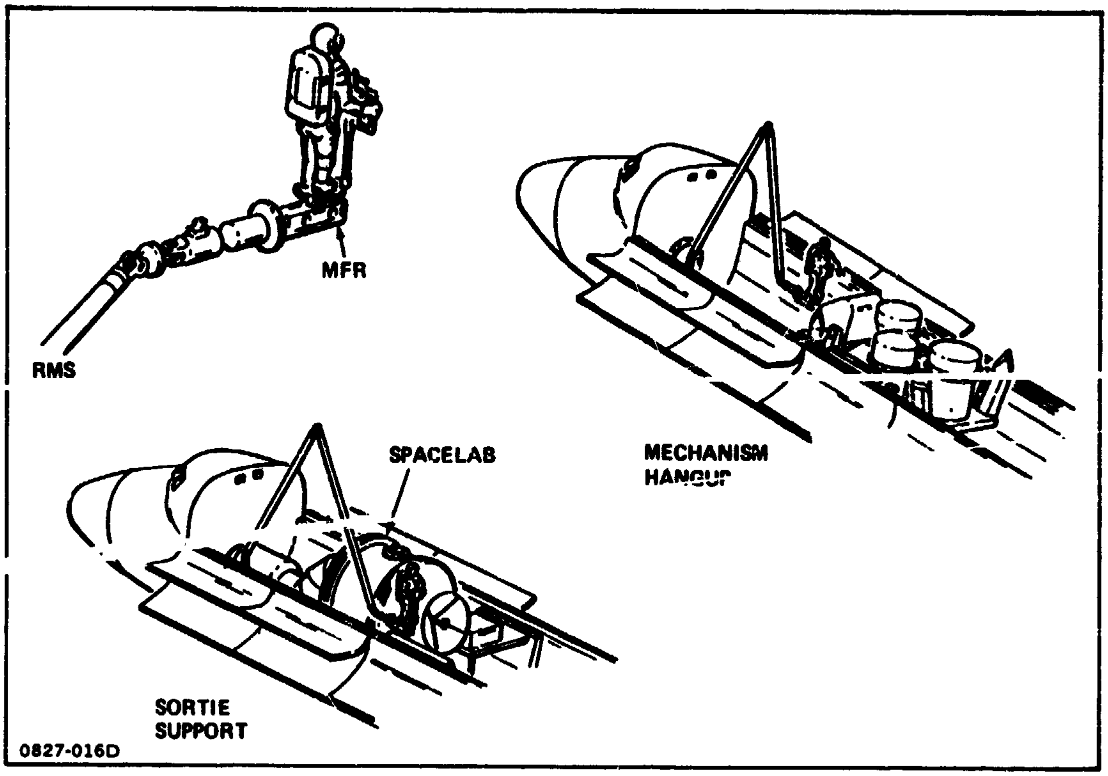

Fig. 16 Manipulator Foot Restraint (MFR)

1.4.2 Manned Maneuvering Unit/Work Station

Three variations of Work Station adaptations that have been identified are shown in Fig. 17. The work station adaptations feature:

- "Kit" variations that are applicable in all situations when the RMS is inoperative
- Enables lift-out/deploy of payloads for RMS inoperative modes
- Provides a portable work station (in and about the payload bay) for planned and contingency servicing operations
- Enables transport of equipment/components to work sites.

An adaptation of a Work Station (WS), used in conjunction with an MMU, would serve as a backup for satellite deployment if the RMS is inoperative or malfunctioning. The WS is adapted with an extensible mast and an RMS snare end effector that is compatible with the satellite's grapple fixture used for deployment. Figure 17 shows an astronaut in the MMU/WS about to lift a satellite out of the payload bay. The astronaut can orient the satellite for deployment and, with the MMU's propulsion system, impart a separation velocity of about 1 ft/sec to the satellite. Once again, if the RMS is inoperative or malfunctioning on a satellite deployment mission, an adaptation of the WS would also serve as a backup for hangups of spacecraft retention latches. The WS is adapted with a stabilizer to position the astronaut rigidly to a work site. Figure 17 shows an astronuat in the MMU/WS with the stabilizer attached to hand rails along the payload bay. The astronaut is preparing to manually release a payload retention latch. The same adaptation of the WS (with stabilizer) could also serve as a backup for hangups of spacecraft appendages that might occur during deployment of satellites by the RMS. For example, with the stabilizer attached to a "hard point" on a satellite, the astronaut could manually release a solar array mechanism.

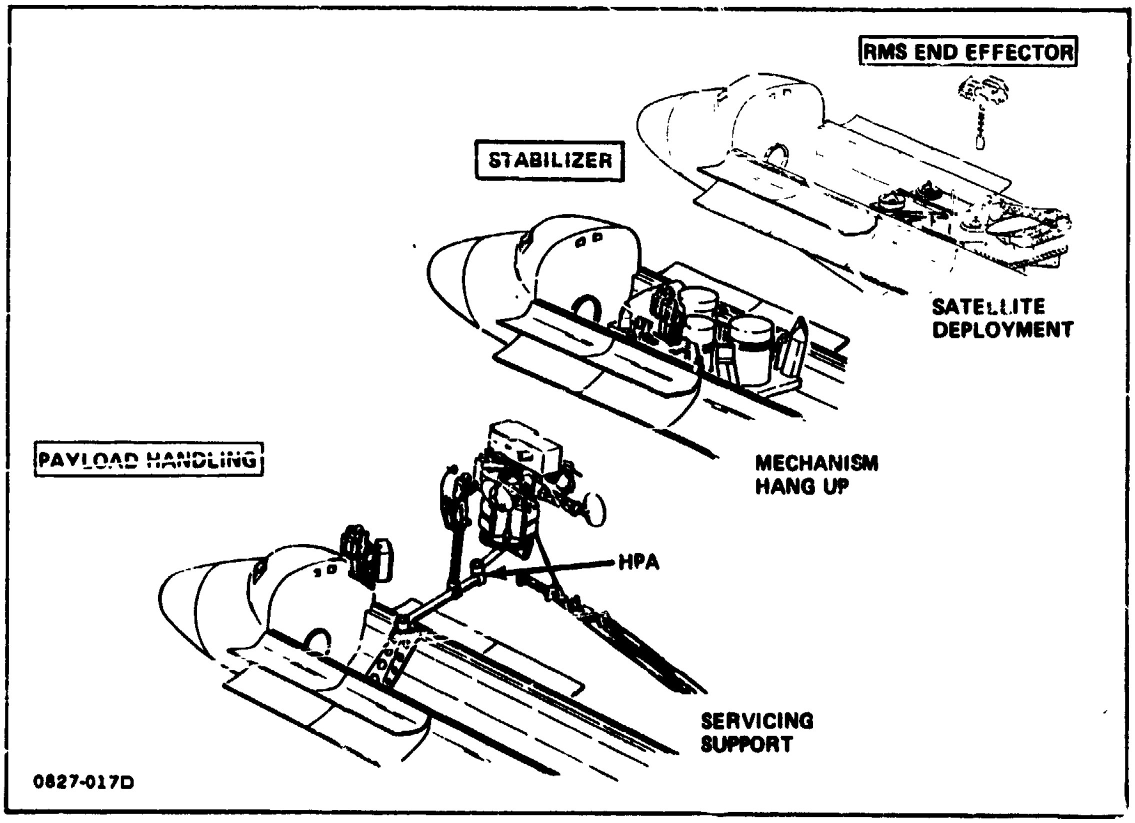

Fig. 17 Manned Maneuvering Unit/Work Station (MMU/WS) Adaptations

With appropriate adaptations of the WS, a revisit service mission could also be performed with the RMS inoperative. A WS adapted with an RMS snare end effector could retrieve payloads within the local vicinity of the Orbiter and position the payloads on a Tilt Table, or on an HPA for on-orbit servicing. Furthermore, a payload handling adaptation of the WS could transport replacement equipment/modules from the payload bay to the work platform at the service site. Figure 17 shows an astronaut using the MMU/WS to transport an equipment module to a second astronaut who is servicing a satellite mounted on the HPA.

The three adaptations of the WS (RMS snare end effector, payload handling, and stabilizer) are implemented in terms of "kits" that are adapatable to a single "core WS" carried on the service mission as illustrated in Fig. 18.

1.5 DELIVERY/RETRIEVAL OF HIGH ENERGY PAYLOADS

Satellite service equipment associated with delivery and retrieval of LEO/Propulsion class payloads includes:

- Versatile Service Stage (VSS)*
  - Delivery, Rendezvous, Docking, and Retrieval Capability
  - End Effector Kit for Non-Cooperative Satellite Stabilization
- AFD C&D.

Equipment noted with an asterisk (*) is described and illustrated herein.

1.5.1 Versatile Service Stage - Satellite Placement and Retrieval

A Versatile Service Stage (VSS) is used to transport and retrieve satellites from higher energy LEO orbits that are not directly accessible by the Orbiter. It is equipped with a high performance propulsion system for performing large Δv maneuvers, and a clean-firing, cold gas propulsion system for close-in satellite retrieval and Orbiter close proximity operations. The VSS contains a television system for satellite examination and to support remote control of the VSS-to-satellite docking/capture operation. Following capture, the VSS/satellite returns to the Orbiter and achieves rendezvous within about 1000 ft. Close proximity flight control of the VSS/satellite is remotely controlled by the Orbiter Crew who "fly" the VSS/satellite to within reach distance of the RMS arm (see Fig. 19).

Servicing of the satellite takes place on the Orbiter. Following servicing, the VSS/satellite is deployed from the Orbiter. The VSS then delivers the satellite to its operational orbit and again returns to the Orbiter.

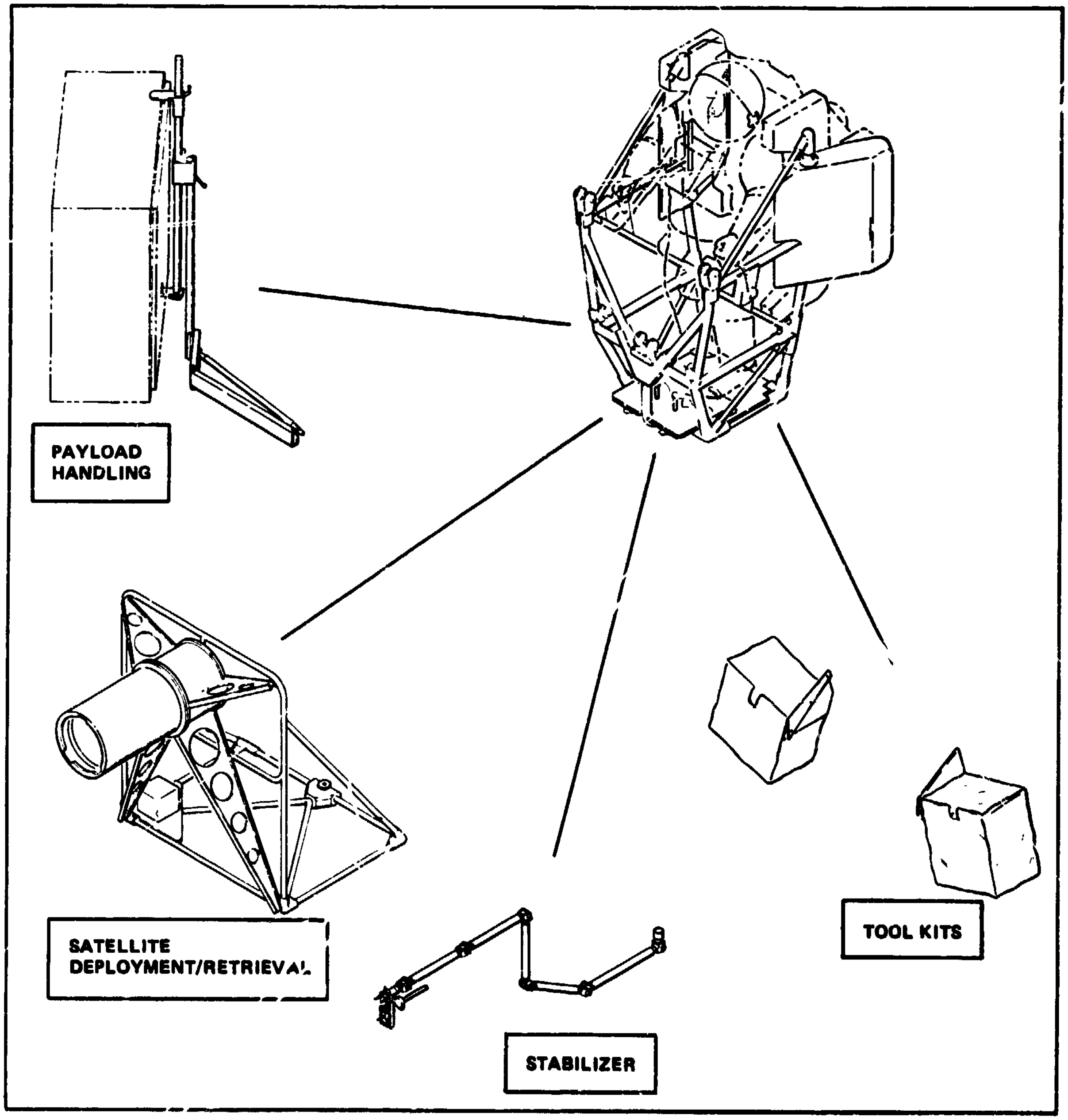

Fig. 18 Work Station Adaptations

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Fig. 19 Versatile Service Stage - Satellite Placement & Retrieval

Fig. 20 Versatile Service Stage - Stabilizing Uncooperative Satellite

1.5.2 Versatile Service Stage - Stabilizing Uncooperative Satellite

The VSS is shown in Fig. 20 "snaring" a satellite that is known to be oscillating at rates higher than acceptable for direct docking by the VSS. A special front-end "kit," provided on the VSS, consists of an extensible mast and RMS snare end effector. The VSS would synchronize its motion with the satellite, extend the end effector to capture the satellite's RMS-compatible grapple fitting, and stabilize it for docking. The operation is remotely controlled via a TV link to the Orbiter (or ground).

1.6 EARTH RETURN EQUIPMENT

Satellite service equipment associated with earth return operations includes:

- Special Retention Structures
- Equipment Storage
- VSS*
  - Debris Capture Kit
    - Debris Retrieval/Return to Orbiter
    - Debris Deorbit
- AFD C&D.

Equipment noted with an asterisk (*) is described and illustrated herein.

1.6.1 Versatile Service Stage - Capture of Satellite Debris

Figure 21 shows the VSS adapted with a special front-end "kit" to capture space debris for deorbit or return to the Orbiter. The front-end "kit" consists of dexterous manipulator arms mounted to a rotating platform.

Capture operations are remotely controlled via a TV link to the Orbiter (or ground). After rendezvous with the debris element, the VSS TV monitors its tumbling motion and is maneuvered to a position where the plane of the VSS rotating platform parallels the tumbling motion. The platform is then spunup to synchronize with the debris tumbling rate. Manipulators engage the satellite and gradually de-spin it via a clutch mechanism in the rotating platform. The debris satellite is then "cinched-up" against bumper stops and held for propulsion maneuvering. The VSS could return to the Orbiter or perform a propulsion maneuver to place the debris element in a desired reentry trajectory, then release the debris to deorbit while the VSS returns to the Orbiter.

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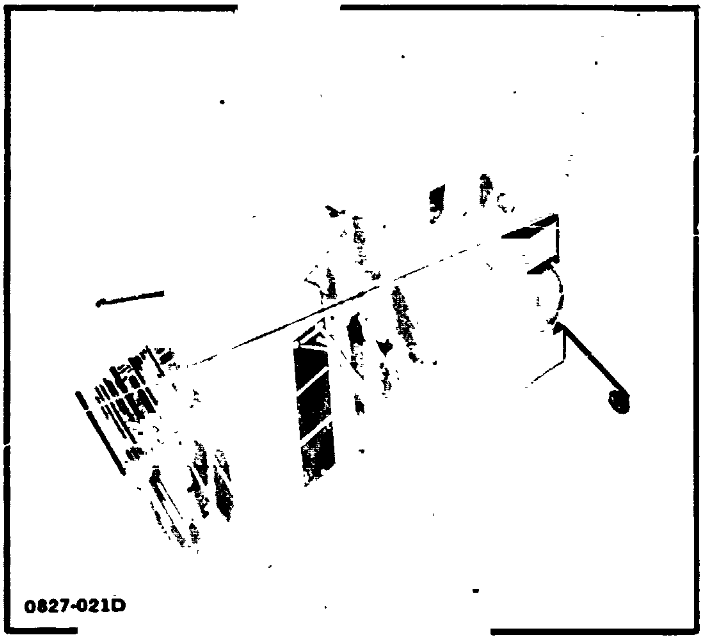

Fig. 21 Versatile Service Stage - Capture of Satellite Debris

Fig. 22 Versatile Service Stage - Close Proximity Flight Control

1.6.2 Versatile Service Stage - Close Proximity Flight Control

Close proximity operations of a free flying VSS are controlled by the crew from the Aft Flight Deck of the Orbiter (see Fig. 22). The VSS would rendezvous within about 1000 ft of the Orbiter and be flown by the crew to within reach distance of the RMS arm. The VSS and its payload would be captured by the RMS and positioned on a support structure (such as the Handling and Positioning Aid) to enable servicing or preparations for earth return.

1.7 OPTIONAL SERVICE EQUIPMENT

Satellite service equipment associated with optional on-orbit service operations, and which can be provided at the discretion of the satellite user, includes:

- Sun shield*
- Orbital Storage*
- Attitude Transfer Package
- Lighting Enhancement.

Equipment noted with an asterisk (*) is described and illustrated herein.

1.7.1 Sun Shield

The sun shield would provide sun-impingement protection to a satellite with the payload bay doors open. The shield would be retracted during launch and have the payload bay doors closed on-orbit. As the payload bay doors open, the shield closes automatically to envelope the payload as illustrated in Fig. 23.

As presently conceived, the large area surface of the sun shield would be composed of thin-film insulation and could be modularly adaptable to accommodate varying length satellite payloads. The deploy-on-orbit approach minimizes the unit's weight by eliminating the need for the shield to accommodate structural/vibration loadings during launch.

1.7.2 Orbital Storage

Orbital storage provides the option of leaving the spacecraft on-orbit for subsequent revisit/repair if a malfunction (detected prior to deployment) categorizes the satellite as non-operational. Orbital storage eliminates the need to carry backup spares (incurring added user charges) or to return a satellite to earth for repair and subsequent relaunch (additional user charges).

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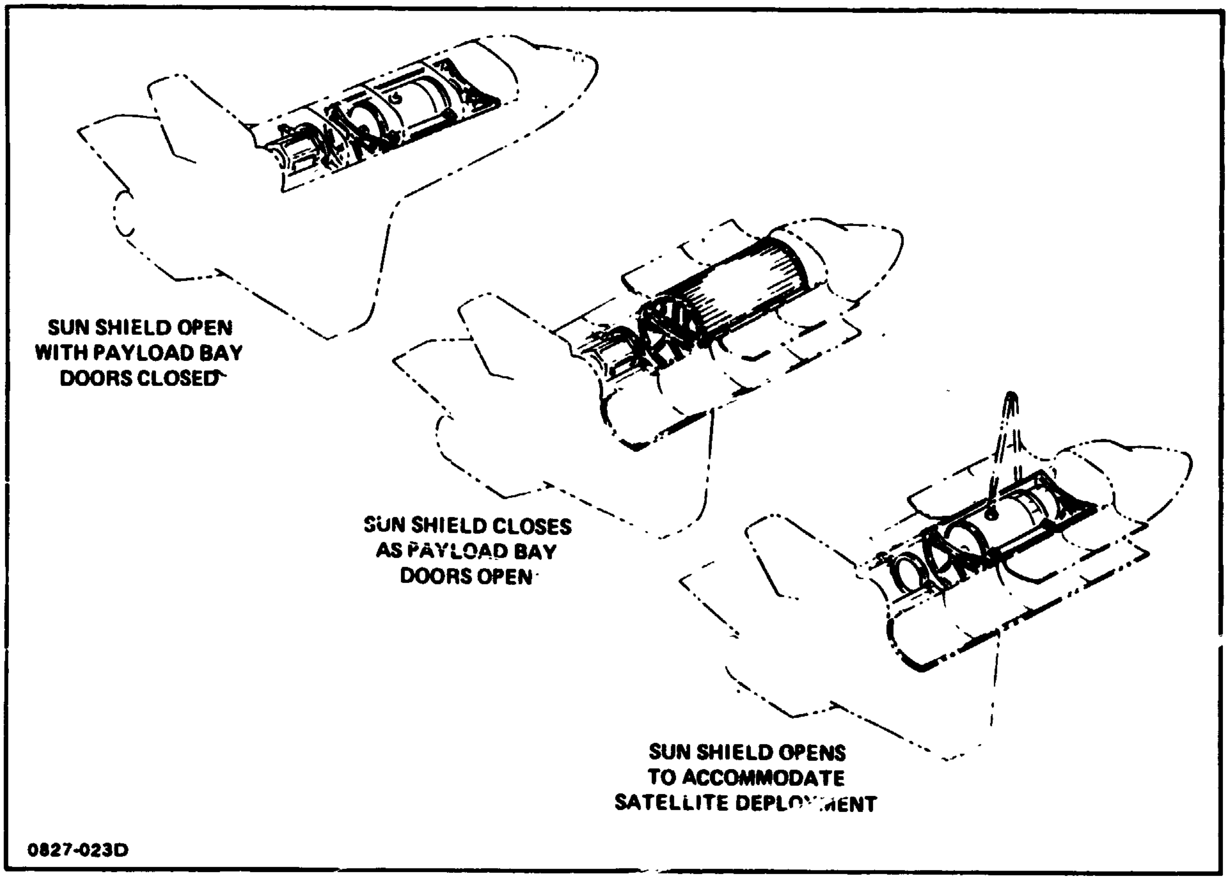

Fig. 23 Sun Shield

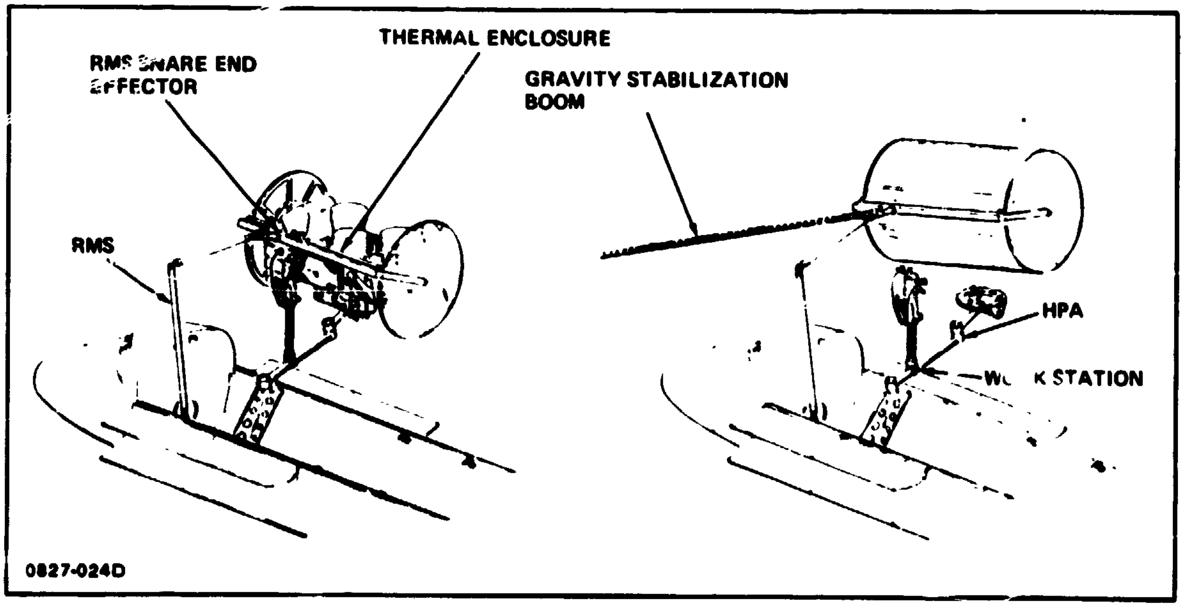

Fig. 24 Orbital Storage

Mounted on the orbital storage enclosure is an RMS-compatible grapple fixture to enable transport from the payload bay to a satellite mounted on the HPA. Within the enclosure structure is an RMS snare-end effector which captures the satellite's grapple fixture and provides the enclosure's hardpoint attachment to the satellite. Figure 24 shows the orbital storage enclosure being placed over the satellite by the RMS.

The figure also shows the satellite released from the HPA and raised above the HPA platform to allow closing of the storage enclosure. From this position, a gravity stabilization boom is activated to provide sufficient on-orbit stability to enable subsequent retrieval for repair/refurbishment of the satellite. With the boom deployed, the satellite is deployed by the RMS in its orbital storage mode.

The thermal enclosure concept employs thin-film insulation with activation of the end bulkheads and enclosure shell involving an inflation technique. The enclosure can also be modularly adaptable to accommodate varying length satellite payloads.

1.8 SUMMARY

A "stable" of generic satellite service equipment has been identified which enables:

- Exploitation of the Space Transportation System's uniqueness vs. expendable launch vehicles, in terms of:
  - Manned presence on-orbit
  - Retrieval/on-orbit maintenance
- Standardization of on-orbit operations/equipment usage to:
  - Minimize user charges/cost of operations
  - Maximize mission success prospects.

This generic service equipment should be developed as soon as possible to enable satellite users to effectively plan for its use. Early flight demonstration of this service equipment and its operation is recommended to provide proof-of-capability to the satellite user community.

References:

1.0 - Satellite Services System Analysis Study, NASA Contr. NAS 9-16120, Grumman Aerospace Corp. August 1981

SATELLITE RETRIEVAL AND SERVICING OPERATIONS WITH PROXIMITY OPERATIONS MODULES

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SATELLITE RETRIEVAL AND SERVICING OPERATIONS WITH PROXIMITY OPERATIONS MODULES

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ABSTRACT

This paper discusses the concept of manned and unmanned Proximity Modules (POM) to assist the Orbiter in retrieval, servicing, and emergency operations of Orbiting payloads. An unmanned POM, capable of examining or capturing and returning to the Orbiter large satellites which are station-keeping at distances up to one kilometer from the Orbiter, is presented and its design features defined.

Also presented is the concept of a manned POM which is capable of capturing and maneuvering smaller payloads in or about the Orbiter payload bay. The manned POM also serves as a free flying work station used to support satellite servicing and provide a back-up to Orbiter RMS inoperative situations.

INTRODUCTION

Satellite Services from the Orbiter payload bay offer the Satellite User Community newly derived benefits which can reduce the cost of operating in space and enhance overall mission success. This capability can be provided by the development of key generic equipment items which will extend the capabilities of the Space Transportation system and support the user needs for satellite deployment and retrieval, on-orbit support, and earth return.

Two such equipment items which will have wide use in nominal and contingency situations are the manned and unmanned Proximity Operations Modules (POM). The POMs are free flying vehicles which utilize a cold gas propulsion system for attitude and translational maneuvering. The POMs can operate either manned or unmanned (manned remote) at distances up to one-half mile from the Orbiter. A cold gas propulsion system is used to minimize the potential contamination to payloads or to the Orbiter resulting from propellant effluents produced by hydrazine and other bi-propellant propulsion systems.

Figure 1 identifies mission applications for an unmanned POM. Initial applications include satellite examination and the examination/retrieval of cooperative satellites. Because of the high useage rate projected for these applications, they are considered key baseline design drivers.

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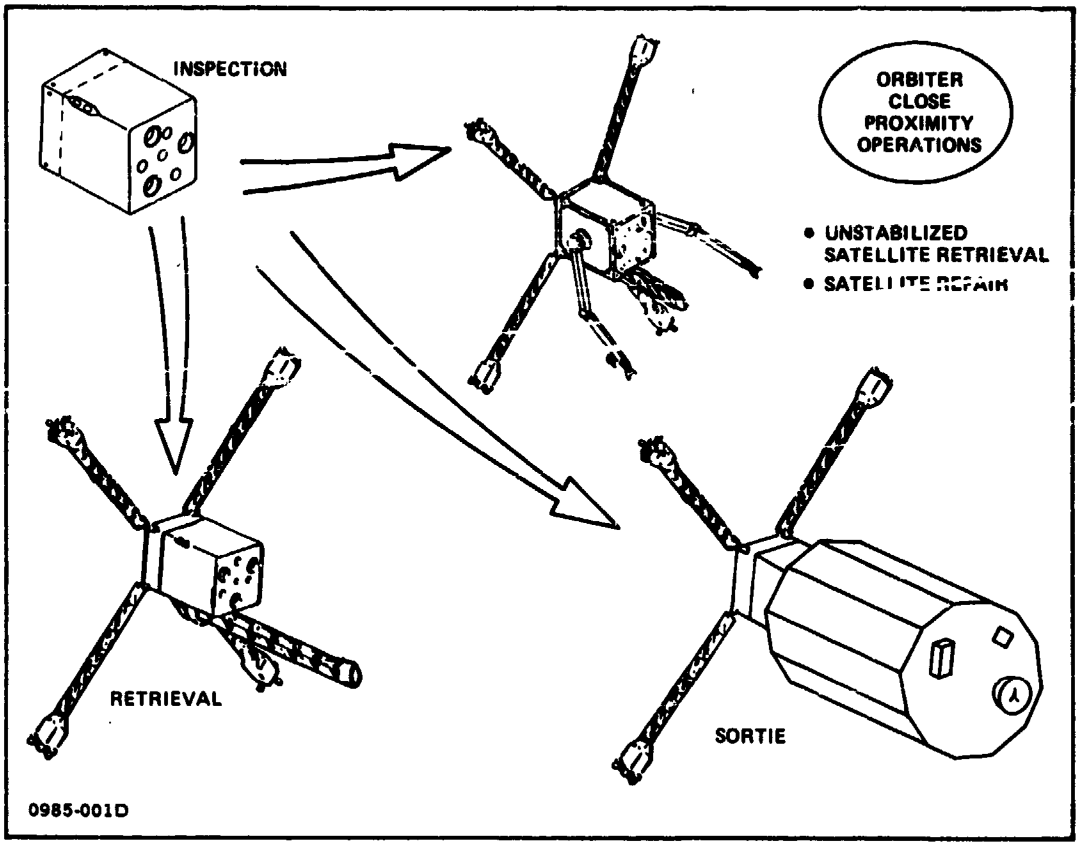

Fig. 1 Unmanned POM System Applications

Another application for the POM is the retrieval of uncooperative satellites or space debris. In this application, the POM synchronizes itself with satellite rotational rates, captures the satellite with manipulator arms, and brings the rotating satellite to very low attitude rates. This mission is considered in the development of add-on kits. This configuration could also be used as a test bed for the development of satellite servicing remote from the Orbiter.

Still another application for the POM is to fly sortie within close vicinity to the Orbiter. Because these requirements are softly defined, this mission application will be considered, but not used as a major configuration driver. However, the capability of the POM to accommodate these missions will be related.

A manned POM, which uses the Manned Maneuvering Unit (MM) as a propulsion system, offers a number of applications for nominal and contingency operations. They include:

- A portable work station for insitu emergency repair such as satellite appendage or payload release mechanism hang-up
- A back-up to RMS for satellite deployment

- A vehicle which can be used to stabilize/retrieve a satellite which has been deployed with higher than nominal attitude rates or which experiences subsystem failure soon after deployment

- On-Orbit crew inspection of the Orbiter

- A free flyer to transfer replacement modules and assist in satellite servicing.

The purpose of this paper is to present concept definitions for both a manned and unmanned POM defined to meet the identified mission applications.

UNMANNED POM DESIGN REQUIREMENTS

To establish system design requirements related to each of the reference mission applications, design reference scenarios were developed to define the sequence of events and operational activities involved for each of the missions. The mission events were then evaluated to define functional requirements from which subsystem requirements could be identified. The results of these analysis indicated that many of the requirements were identical for all applications; the most severe were related to the satellite capture and retrieval scenario. The following paragraphs summarize some of the major design requirements established from these analyses.

Figure 2 identifies the potential satellite retrieval opportunity market for the decade commencing in 1983. These requirements reflect the needs of satellite programs which are currently approved or in planning (exclusive of DoD and Sortie Missions) as defined within the Satellite and Services User Model (S/SUM). The model was developed within the Satellite Services System Analysis Study.

SATELLITE MASS (KG)	NUMBER OF RETRIEVAL EVENTS	EARLIEST TIME FRAME
SATELLITES DIRECTLY REACHABLE BY ORBITER		
UP TO 1000 KG	12	1986
1000 - 2500 KG	9	1983
2500 - 5000 KG	18	1985
5000 - 11,000 KG	15	1987
> 11,000 KG	19	1987
SATELLITES WITH LEO PROPULSION		
UP TO 1000 KG	8	1986
1000 - 2500 KG	38	1986
2500 - 5000 KG	33	1984

*APPROVED OR PLANNED SATELLITE PROGRAMS ONLY

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Fig. 2 Satellite* Retrieval Requirements (1983-1993)

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Satellite retrieval needs are shown grouped in terms of satellite mass and earliest need date because of their importance to POM sizing characteristics and system development schedules. As noted, for satellites which are directly reachable by the Orbiter, satellite retrieval needs in terms of mass are distributed in groups up to 2500 kg, 2500 to 5000 kg, and 5000 kg to 11,000 kg. Retrieval of satellites with a mass greater than 11,000 kg shown in Fig. 2 represent retrieval of Science Application Space Platforms (SASP) and 25 kw Power Modules which are not representative of conventional type satellites. Consequently, the study did not address the needs of these missions.

Satellites which are placed into their final orbit with the aid of propulsion stages are also shown because many may be designed to transfer themselves to lower orbits which can be reached by the Orbiter.

Figure 3 shows the Flight Control Concept selected and baselined for maneuvering and controlling the POM to and from a satellite station-keeping within close proximity to the Orbiter. The concept involves a crewman stationed in the Orbiter Aft Flight Deck equipped with hand controllers and displays flying the POM to a satellite by remote attitude and translation commands. The procedure requires the crewman to maintain the POM in inertial attitude hold during approach to the satellite and controlling inertial LOS rates with Y- and Z axis

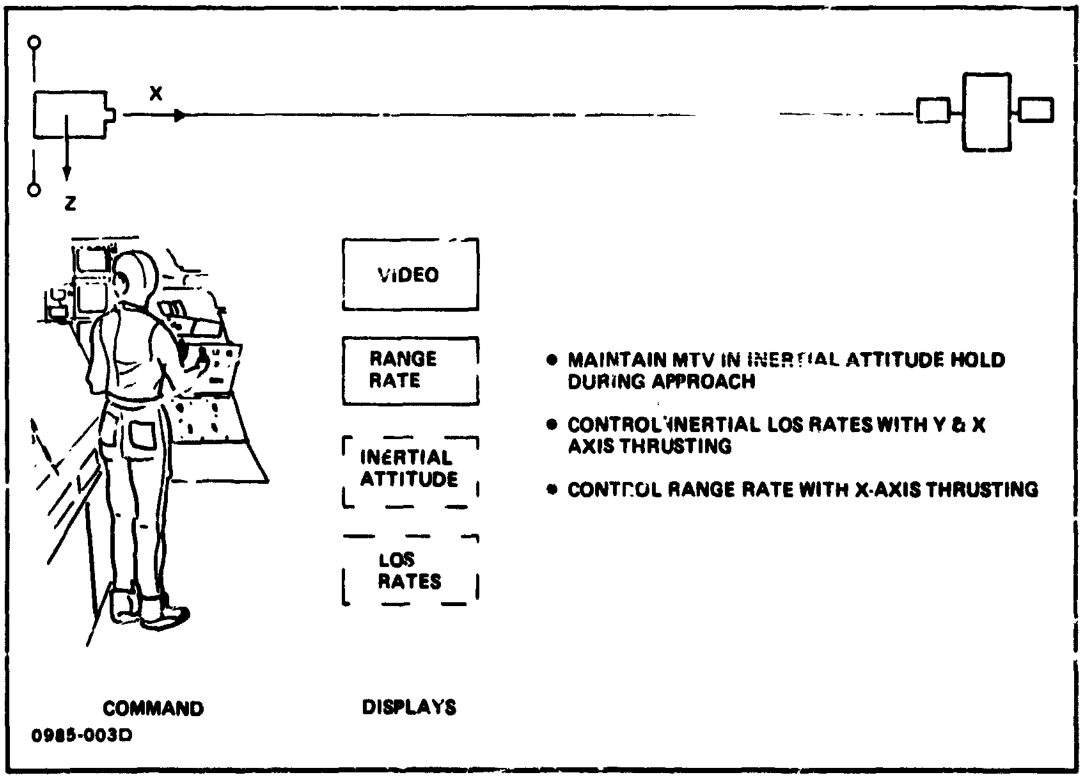

Fig. 3 Flight Control Concept

translation. Range rate is controlled with X-axis translation. This technique was developed during the Apollo program for use in affecting manual terminal rendezvous and docking of the LM to the CSM. Man-in-the-loop simulations have shown that near minimum Δ V penalties are incurred when controlling inertial LOS rates to within low rates (±0.2 mrad/sec). An additional factor which impacts the selection of guidance system components needed to implement this technique is the non-stringent inertial sensor requirements. The gyro drift effects occurring within the approximately 15 minutes of flight time have little impact on inertial attitude reference accuracy.

Using this flight control concept, analytic simulations were performed to establish a Δ V budget for satellite retrieval at a range of 1 km. Results of these studies indicated a Δ V budget of 54 ft/sec was required for the POM to approach the satellite to a stand-off range of 100 ft, maneuver about the satellite for close-in examination, and finally close-in and capture the satellite by its grapple fitting. These analyses also showed a Δ V budget of 39 ft/sec was required for the POM to return a satellite within the RMS reach of the Orbiter. Both Δ V budgets included a 25% contingency allowance.

For these Δ V requirements, the propellant required for a POM to effect satellite retrieval as a function of satellite mass (to be towed) and POM dry weight is shown in Fig. 4. The propellant quantities are based on the use of

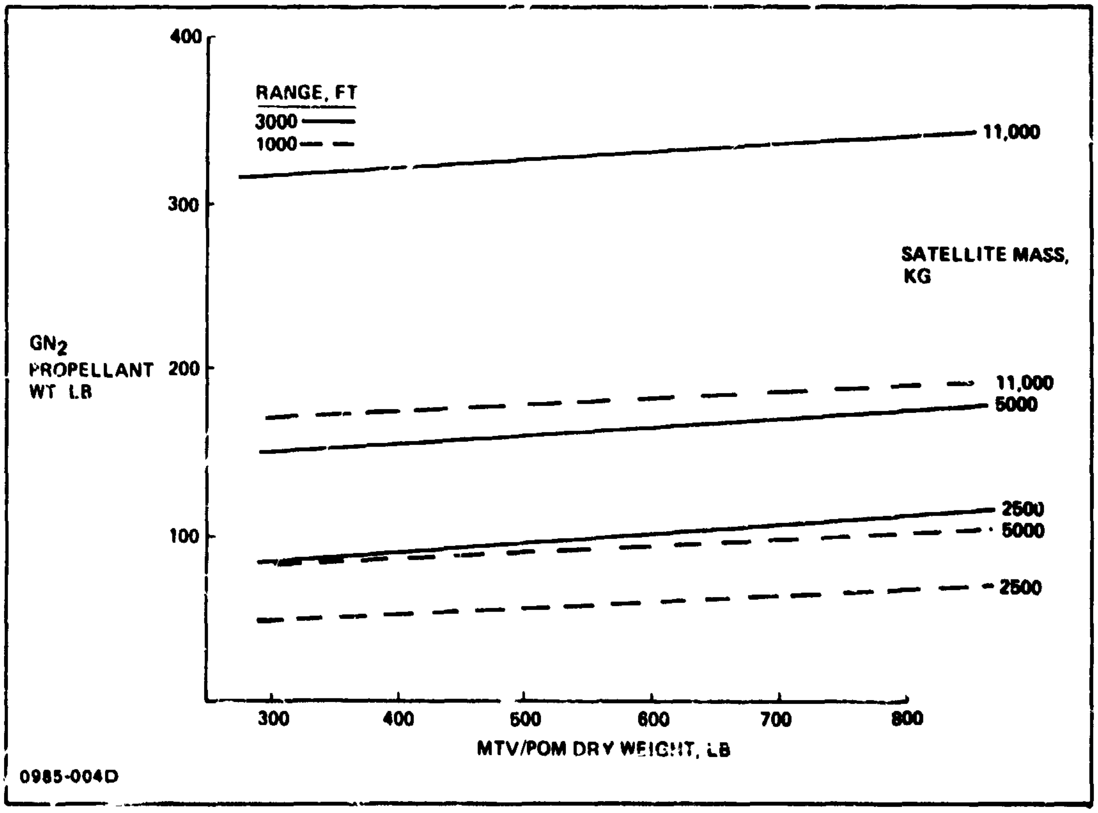

Fig. 4 POM Propellant Weight for Satellite Retrieval

gaseous nitrogen (GN_2) with an I_{sp} of 60 sec. This assumption is somewhat conservative in that it represents the low side of GN_2 performance within the expected temperature control.

Results show, for example, that a propellant quantity of 150-160 lb of propellant is required to retrieve a satellite 5000 kg (LDEF) in mass. Similarly, a satellite of 2500 kg station keeping at range of 1 km could be retrieved with only 75 lb of usable propellant.

Another subsystem which can significantly impact the POM configuration is the Attitude and Translation Control System. Figure 5 identifies the variation in vehicle characteristics which must be accommodated. Inertia variations, for example, can differ by a factor of 100 to 1 when the POM is in free flight, by itself, as compared with towing a satellite of more than 5000 kg. Similarly, mass variations can vary over a range of 20 to 1 when towing satellites up to 11,000 kg as compared to POM inflight alone. A wide spread in combined cg locations is also a factor to be accommodated.

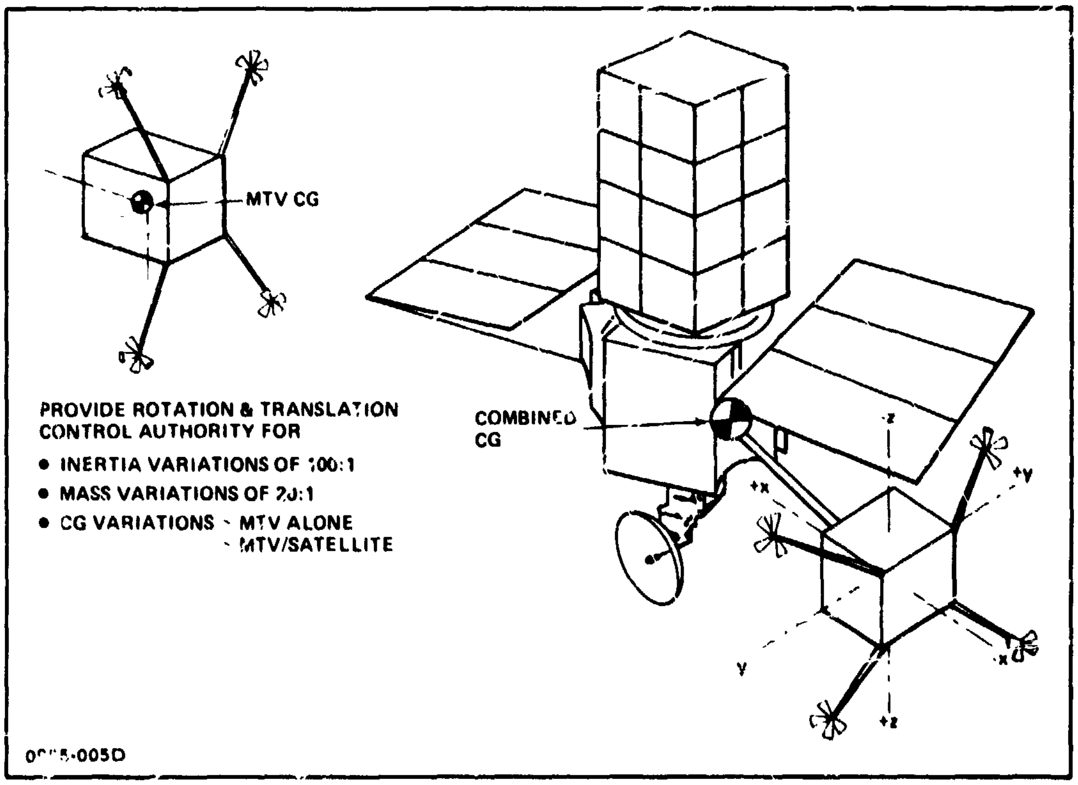

Fig. 5 Attitude & Translation Control-Subsystem Requirements

The following criteria were used to define a system configuration.

- Provide a time of flight of not more than 15 minutes for returning a satellite up to 11,000 kg in mass to the orbiter

- Consume propellant within Δ V Budget allocation

- Provide adequate rotational maneuvering response during all mission phases including satellite berthing, satellite maneuvering, and satellite hand-off to orbiter. Maneuvers must be executed using pure balanced couples about all axis

- Provide adequate translational maneuvering response to recover from trajectory dispersions during all phases. Translation maneuvers shall not induce rotational disturbances in presence of wide cg variations.

Other subsystems considered included communications, TV systems, and Power Supply and Distribution.

The communications system requirements were based on the use of a dedicated system for transmitting real time commands from the orbiter to the POM and transmitting telemetry data, video, POM status data, and command verifications from the POM to the orbiter. Television is to be provided via analog wideband FM modulation. Data transmission will use omni directional antenna coverage on the POM and hemispheric (Orbiter Cooperative) from the Orbiter at a range up to one nautical mile.

Design requirements for the POM TV system were based on its usage as a navigation aid during the trajectory approach and for use in satellite examination.

Thus, the TV system is required to acquire targets at low light levels at a range up to 1 km and include a variable field of view with zoom, iris, and focus control. It is also required to provide high quality black and white imagery at ranges from 25 ft to 3000 ft.

Power requirements were determined by evaluating energy usage over the mission timelines for each of the system applications. Results showed that the mission driver was retrieval of uncooperative satellites, requiring approximately 1200 watt-hrs of energy and requiring a capacity of approximately [illegible] amp-hrs. Peak current was estimated at 50 amps.

UNMANNED POM SYSTEM CONFIGURATIONS

The overall approach used to develop the POM configuration was to define a baseline core module which can be integrated with add-on kits to support the complement of mission applications. The baseline core module will include common subsystems sized to incorporate growth in system capability where mission requirements dictate. Add-on kits would include the mission unique capabilities which could be brought on-line in a time frame consistent with their mission needs.

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Figure 6 illustrates the overall design and general equipment arrangement selected for the core module. Where possible, major subsystems were organized into component groupings for ease in maintenance and refurbishment. For example, one side wall contained the power supply and distribution system with room provided for a second 40 amp-hr battery. The top panel was used to mount the communications system components. The second side wall was used to support the G&C components. The bottom surface was used to mount the passive berthing docking mechanisms. The front face was used to mount the TV system. Additional room was provided for the extendable mast and stow canister (kit 3). These components are to be interfaced with the TV system in accordance with specified dimensions.

Overall size of the core module measures 44 in. W x 29 in. H x 35 in. L. System weight, including subsystems, is estimated at 407 lb including a 15% contingency.

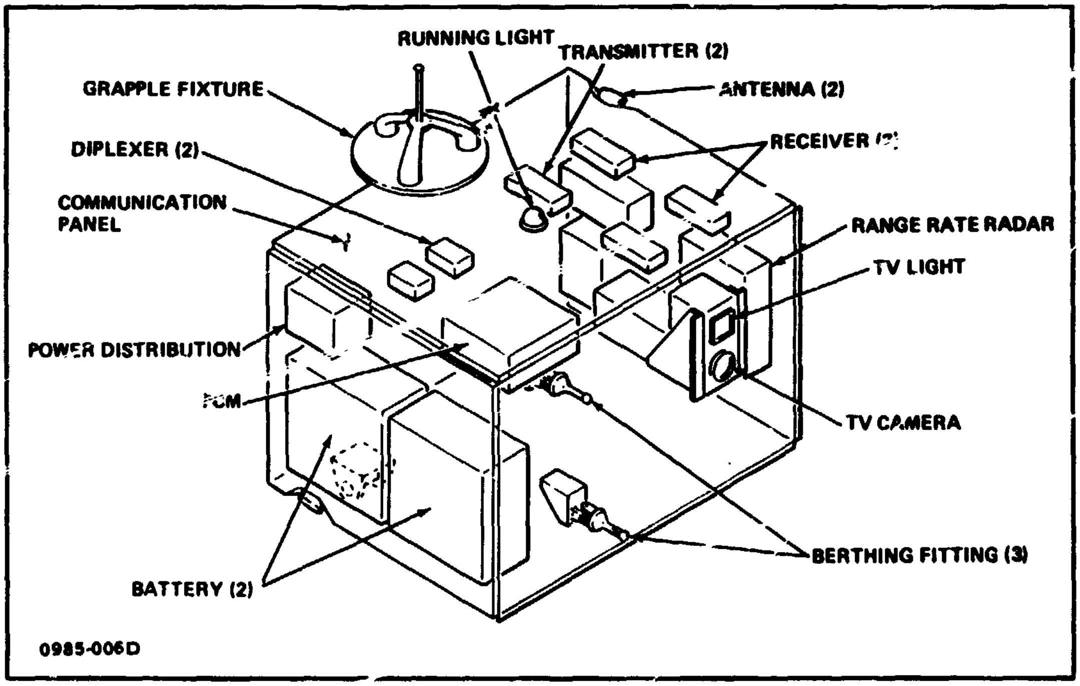

Fig. 6 Unmanned POM Core Module

Figure 7 shows a composite of add-on kits which can be integrated with the core module to perform the complement of mission applications.

Kit 1 consists of a small propulsion kit for use in satellite examination and smaller payload retrieval. The kit contains two 20 in. spherical tanks, which hold approximately 86 lb of useable propellant, and the control system thrusters. The thrusters are arranged in quads mounted to each corner. The

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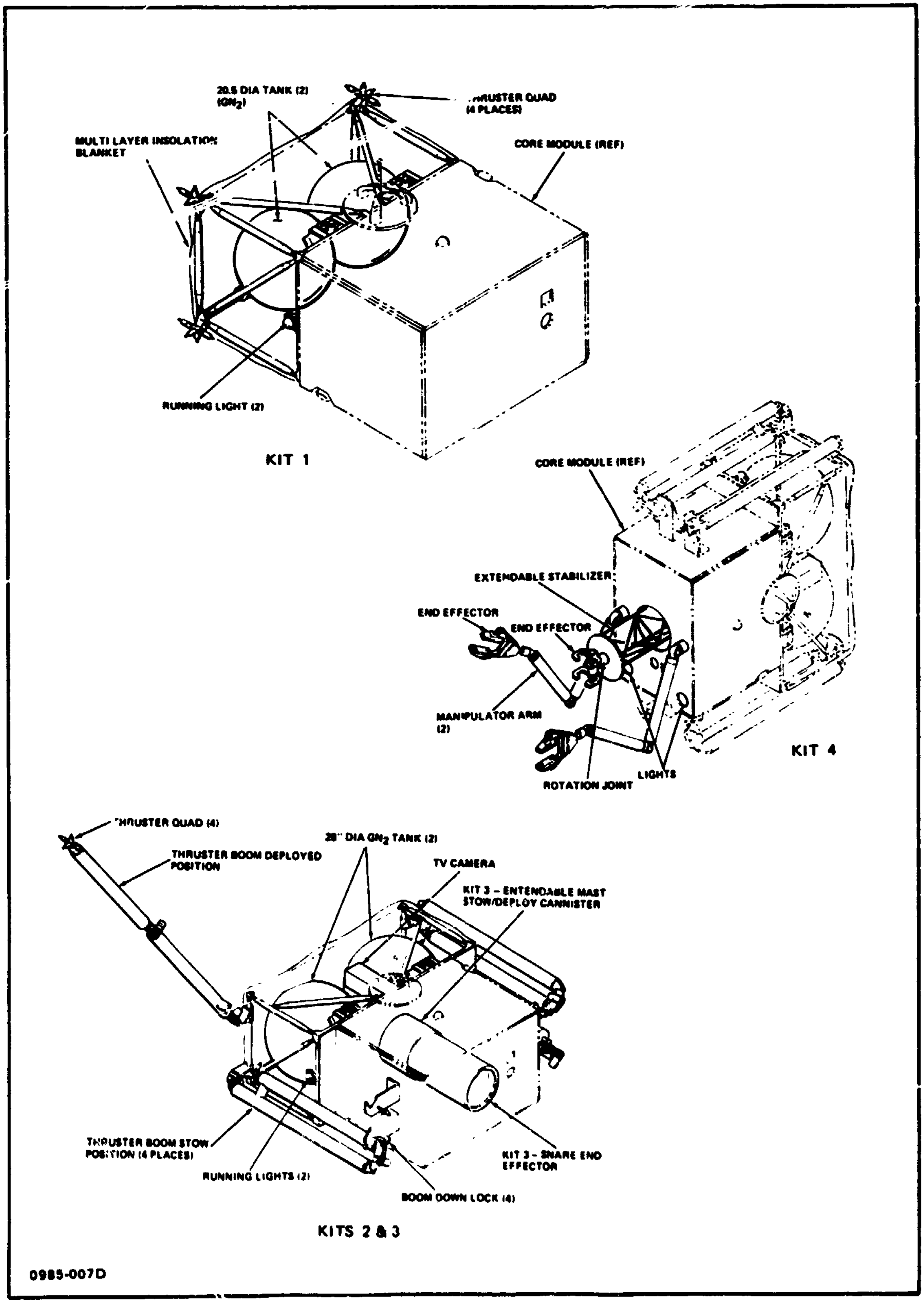

Fig. 7 POM Add-On Kits

width and height of the kit are sized identical to the core module so as to appear as one continuous module. The overall vehicle size with core module attached is 44 in. W x 29 in. H x 58 in. L and weighs approximately 675 lb (including contingency). This configuration is capable of performing two satellite missions or retrieve a satellite up to 2500 kg at a range of up to 1 km. It could also provide a Δ V of 100 ft/sec for maneuvering a 500 kg payload about the orbiter for sortie experiment flights.

Kit 2 contains two 28 in. diameter spherical tanks which provide about 175 lb of useable GN_2 propellant. Also contained within the kit is an aft facing TV camera which is used for navigating the POM to the Orbiter while satellite towing. Four 101 in. deployable thruster booms, each carrying a thruster quad mounted to the end, provide the attitude and translation control authority required for satellite towing. The boom length was sized to minimize thrust loss due to impingement for satellites as larage as 15 ft in diameter.

Also incorporated is kit 3, the snare end effector, which is attached to an extendible boom and used for satellite grappling. Overall size of this configuration measures 76 in. W x 29 in. H x 66 in. L and is estimated at 1110 lb. This configuration is used to retrieve satellites up to 5000 kg at a range of up to 1 km. It could also be used for larger payload sortie missions providing a Δ V of 100 ft/sec for 1000 kg experiment packages. Not shown is kit 2A which incorporates four 28 in. diameter tanks and which can be used to retrieve payloads up to 11000 kg at 1 km.

Also shown in Fig. 7 is kit 4 manipulator system integrated with the core module and propulsion kit 2. The manipulator arms, each 5 ft in length, are mounted to the front face of the core module to provide full access to the work site. The arms store along the core module side panels within the thruster boom stowage envelope. An extendible stabilizer is used to secure the POM to the work site. The kit also includes a tilt and pan television system with two additional lights mounted to the front face.

Figures 8 and 9 show artist's renderings of the unmanned POM (kit 2 & 3 configuration) during satellite capture and towing operations. Figure 8 depicts the POM about to capture the LDEF satellite by its grapple fitting. Figure 9 shows the POM towing the LDEF to the Orbiter.

Figure 10 shows the unmanned POM as it is stowed in the orbiter payload bay at its flight support station. It is deployed and berthed using the RMS and can be readily carried with a near full complement of payloads.

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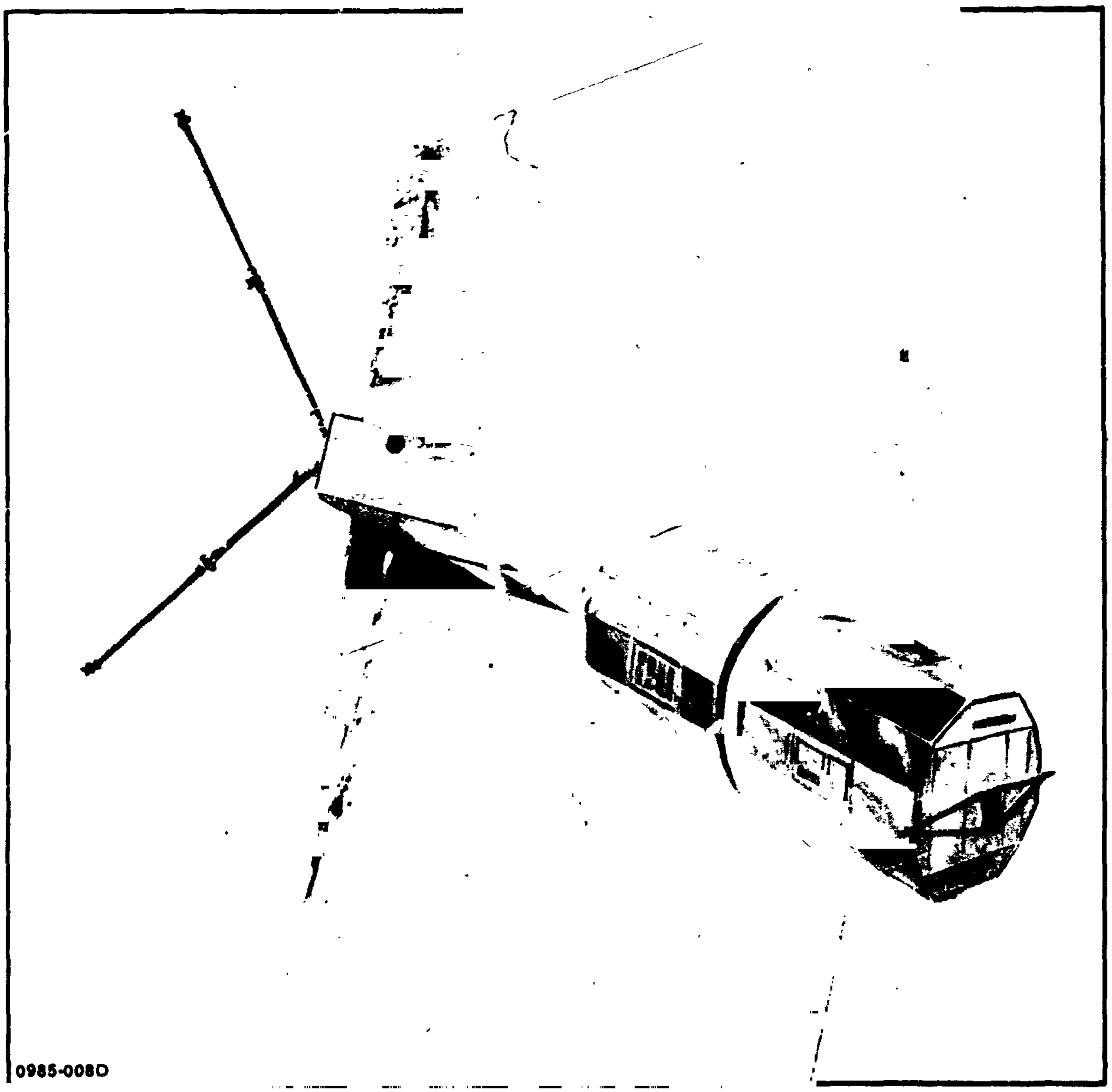

Fig. 8 MTV Proximity Operations Module – Satellite Captive

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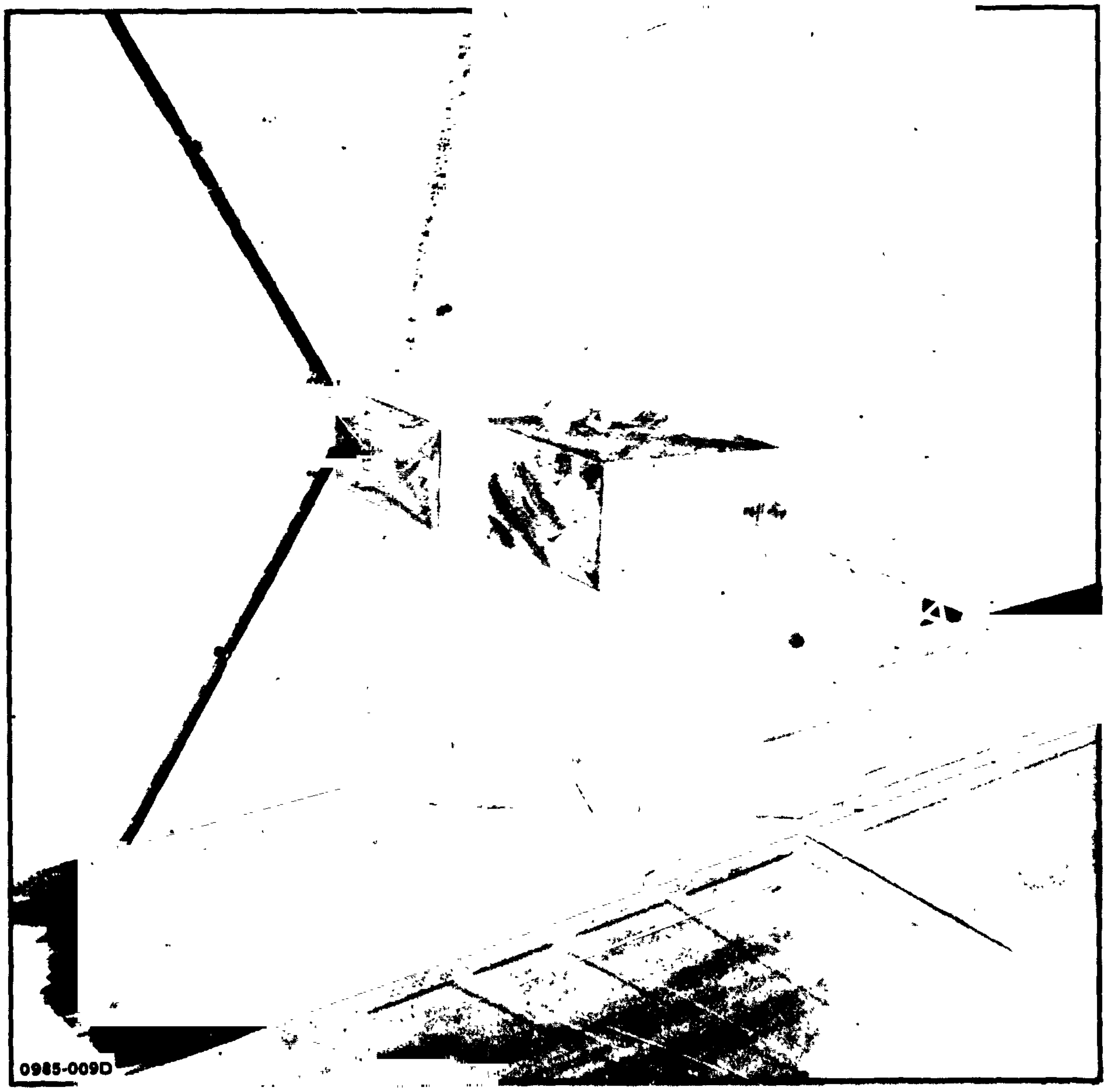

Fig. 9 MTV/Proximity Operations Module – Satellite Retrieval

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Fig. 10 MTV/Proximity Operations Module – Stow Configuration

MANNED POM - REQUIREMENTS/CONFIGURATION

The manned POM concept, which utilizes the MMU for maneuvering, is a free flying work station capable of performing several mission operations. As such, the major requirements are to provide a crew man with a foot restraint system which will attach to various types of surfaces and, after attachment, will provide access to a large work zone. The system should also be capable of capturing and maneuvering satellites and transporting small modules in or about the orbiter payload bay.

The design approach used to meet these requirements was to develop a baseline core vehicle work station which would accept various attachment mechanisms suited to the specific application. Figure 11 depicts the baseline work station configuration which was developed to perform the identified mission applications. It consists of an aluminum frame structure with attachment mechanisms for holding the MMU to the work station. These attaching devices are identical to those used to hold the MMU to the payload bay Flight Support Station.

A control handle on either side of the frame actuates the pivot mechanisms on each supporting assembly. The astronaut is held to the module by these attachment points.

The front face of the frame has been designed with a minimum of frontal structure to provide maximum access to the work site. The foot restraint is a fixed platform which the astronaut stands on while working.

Also shown in Fig. 11 is the stabilizer add-on kit which attaches to the bottom centerline of the foot restraint platform. The stabilizer consists of four connected links which can pivot ±90 degrees and which attach to the underside of the work station. The astronaut flies up to the satellite and attaches the end effector to an appendage. He can then position himself to the work zone for optimum reach and visibility. Actuation of controls on his right-hand panel will rigidize the links in the stabilizer, holding the work station in a fixed position. He can then step forward of his MMU onto the work platform.

An artist's rendering showing a crewman repairing hang-up of a satellite appendage is depiected in Fig. 12. The crewman is shown released from the MMU and held by the foot restraint to provide him with unobstructed access to the work zone available within his immediate frontal projection.

Figure 13 shows application of the MMU/Work Station for use in Orbiter examination. The astronaut is shown removing samples of tank insulation which can be returned to earth for analysis. In this application, an attachment kit consisting of two barbed petons and a bumper pad are incorporated with the work station. The petons penetrate and grab into the external tank insulation material and hold the work station firmly attached to the tank. A bumper pad which attaches to the lower part of the core module provides a third point to distribute astronaut applied loads thru the core vehicle to the external tank. Two containers are included to store sample returns and carry hand tools which may be required at the work site. The total kit weight is estimated at 25 lb.

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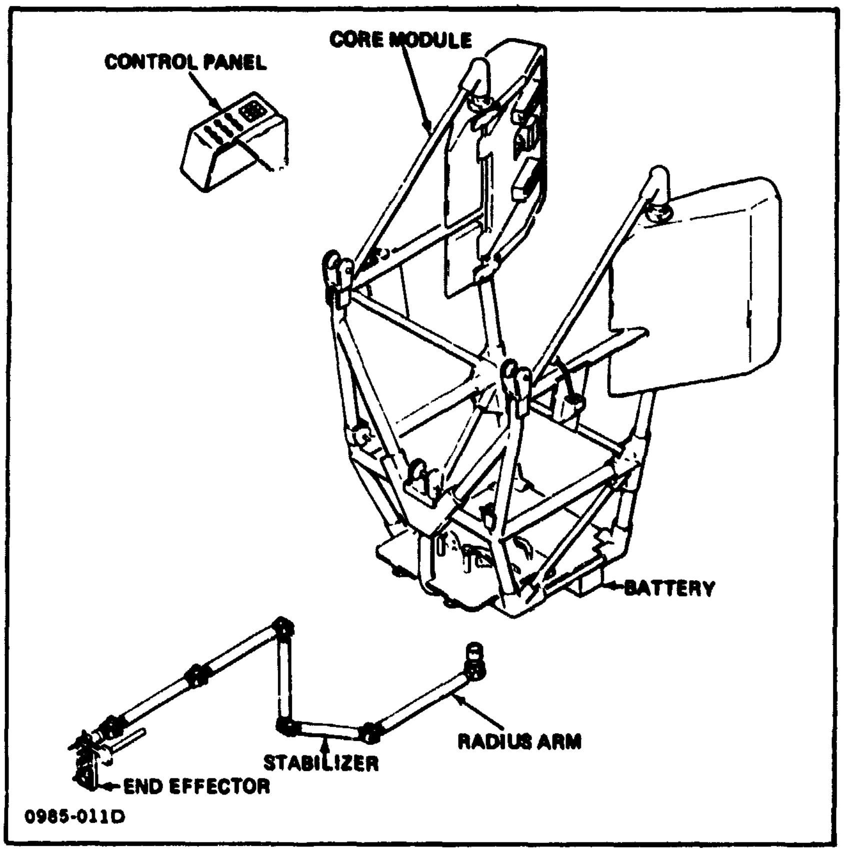

Fig. 11 Work Stations-Stabilizer

Retrieval of satellites at distances up to 1000 ft from the orbiter can also be accomplished by a manned POM. For this use, the work station is equipped with an extendable mast and an RMS end-effector that is mounted to a support structure.

The astronaut flies with the snare end-effector in a forward position during satellite engagement and in an aft position during satellite towing operations. An astronaut would fly the manned POM to the satellite, capture it via the RMS-compatible grapple fixture, and tow the satellite to within the reach distance of the RMS.

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Fig. 12 MMU/Work Station – Backup for Satellite Appendage Hangups

Figure 14 shows the manned POM "flying-in" the end effector to engage the satellite's grapple fixture. As most of the major hardware elements for this concept exist or are in late stages of development, the manned POM could be a more readily available approach for near-term satellite retrieval missions.

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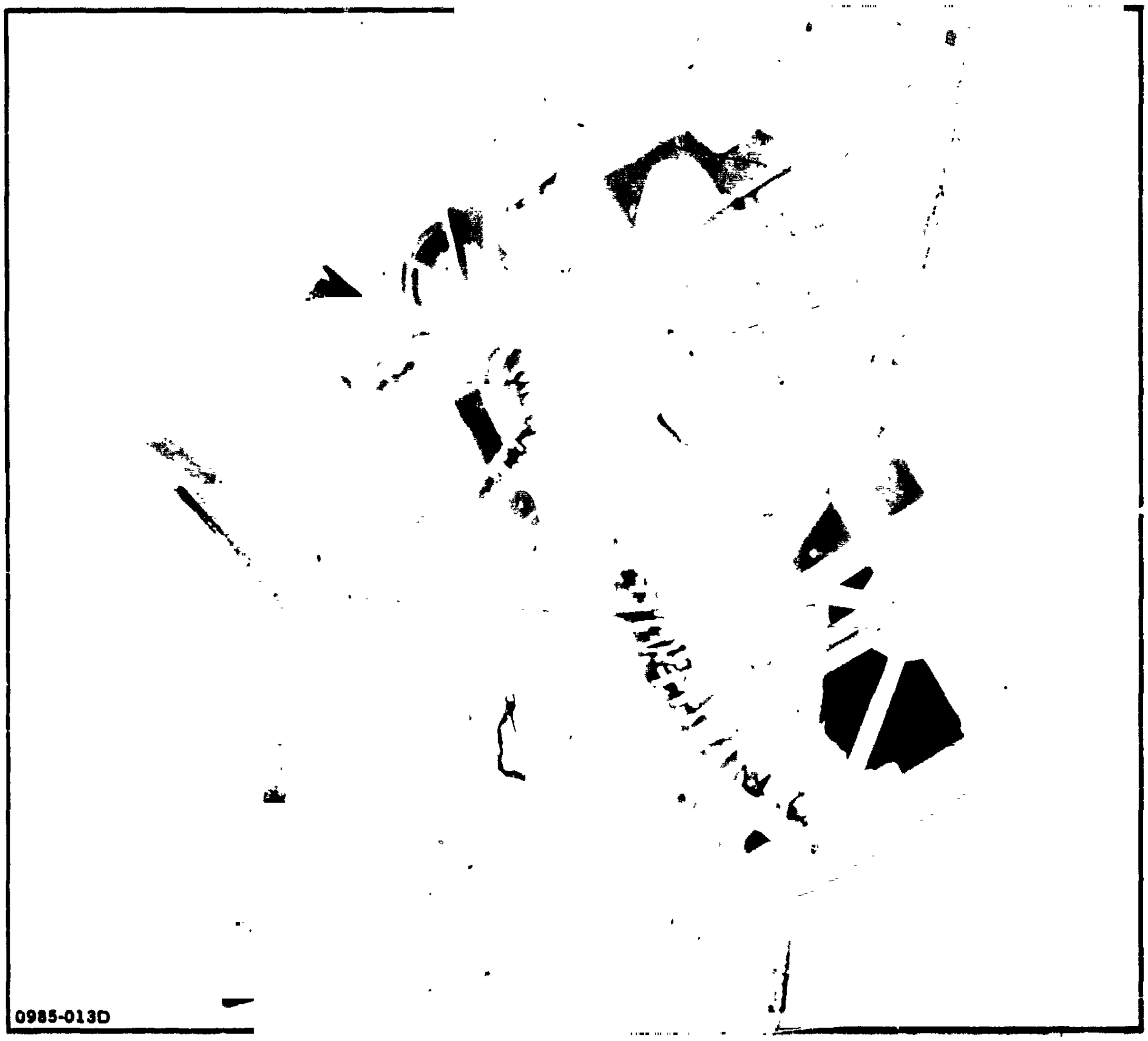

Fig. 13 MMU/Work Station – External Tank Inspection

Figure 15 shows a manned POM towing a spacecraft to the orbiter. Via the flight control capabilities of the MMU, the astronaut would stabilize/position the satellite within the reach distance of the Orbiter's RMS arm. The POM would then detach itself from the satellite's grapple fixture to allow the RMS to capture the satellite. Following capture, the RMS would place the satellite on a Tilt Table or Handling and Positioning Aid to enable on-orbit servicing.

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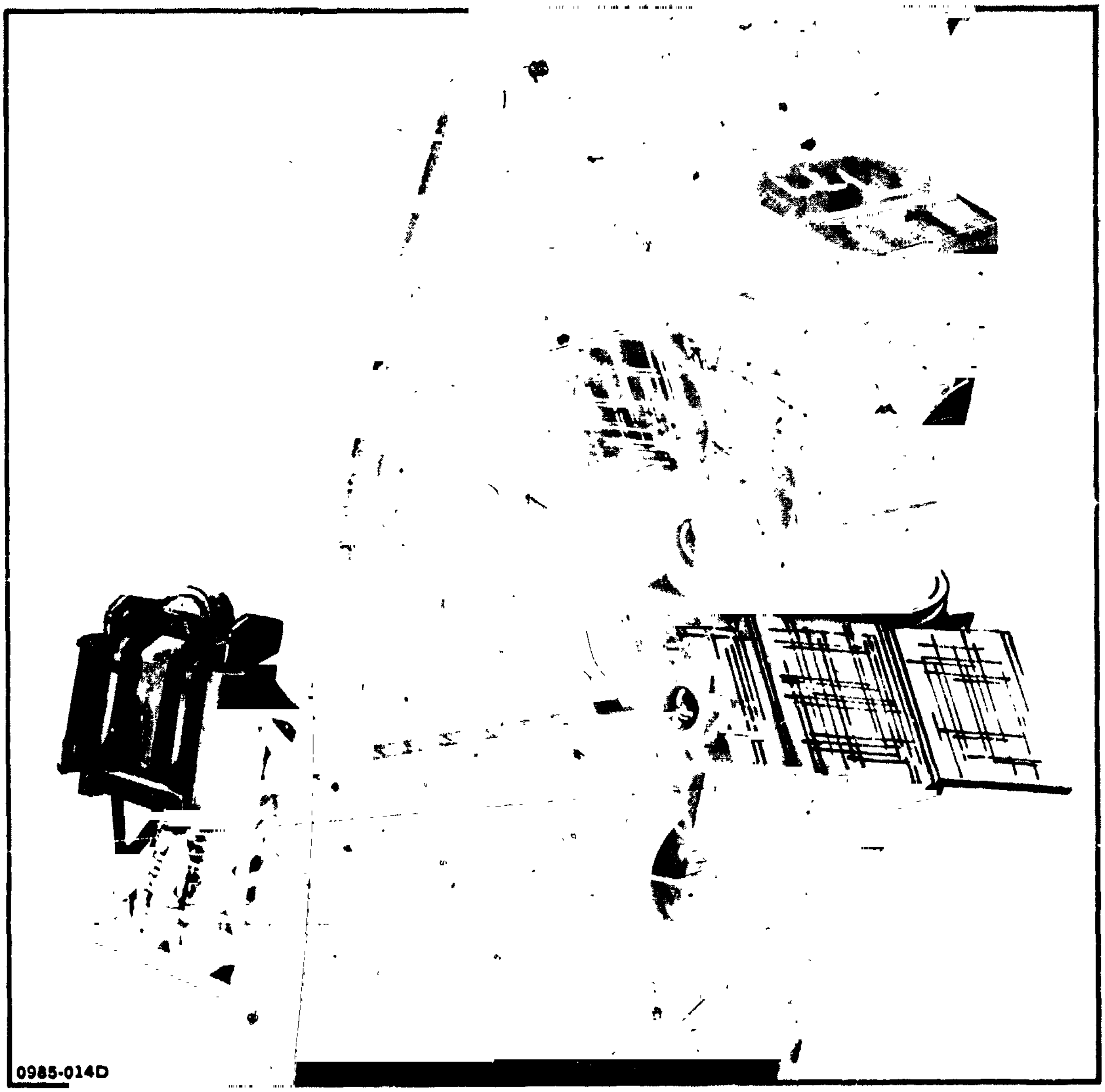

Fig. 14 Manned proximity Operations Module – Satellite Capture

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Fig. 15 Manned proximity Operations Module – Satellite/Retrieval

CONCLUSIONS AND RECOMMENDATIONS

The development of manned and unmanned POMs can significantly enhance the orbiter's capabilities and efficiency to operate in space (Fig. 16). Both systems can be developed within the current technology base using off the shelf equipments and brought on line with minimum development.

- MTV/PROXIMITY OPERATICNS MODULE OFFERS THE FOLLOWING BENEFITS
  - NON-CONTAMINATING RETRIEVAL
  - MINIMUM ORBITER RCS USEAGE FOR CLOSE PROXIMITY OPERATIONS
  - EXPANDS SATELLITE CAPTURE POTENTIAL BEYOND RMS CAPABILITIES
  - SORTIE EXPERIMENT MISSION APPLICATIONS
- MMU/WORK STATION OFFERS THE FOLLOWING CAPABILITIES FOR NOMINAL AND CONTINGENCY OPERATIONS
  - WORK STATION ADAPTABLE TO VARIOUS SITES
  - TRANSPORTER OF SMALL PAYLOAD MODULES TO WORK SITES
  - PROXIMITY OPERATIONS MODULE FOR SATELLITE CAPTURE AND RETRIEVAL ..... SATELLITE STABILIZATION
  - RMS BACKUP FOR SATELLITE DEPLOYMENT
- DEVELOPMENT OF MANNED AND UNMANNED POM'S ARE BOTH TECHNICALLY AND ECONOMICALLY ATTRACTIVE
  - CAN BE DEVELOPED WITH IN CURRENT TECHNOLOGY BASE USING OFF-THE-SHELF EQUIPMENTS
  - MINIMIZES DEVELOPMENT COSTS
  - OFFERS AFFORDABLE GROWTH

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Fig. 16 Conclusions

An unmanned POM can be used to examine/retrieve satellites at a range up to 1 km from the orbiter with minimum contamination to satellite users and the orbiter. It could be readily integrated into the orbiter payload bay with little impact to cargo manifesting, and significantly conserve RCS propellant for use in scientific and on-orbit experiment operations. It also provides a re-usable vehicle from which an assortment of sortie experiment flights can be flown.

The manned POM also provides a number of mission functions which can improve both nominal and contingency operations. It can serve as a free flying work station capable of attaching to various locations in or about the payload bay. It can be used to maneuver replacement modules during on-orbit satellite servicing and perform back-up to the RMS in the event of RMS system malfunctions.

Because of the wide utility offered by both unmanned and manned POMs, it is recommended that early system development for both equipments be undertaken.

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SPACECRAFT AUTOMATIC UMBILICAL SYSTEM

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June 22, 1982

SPACECRAFT AUTOMATIC UMBILICAL SYSTEM

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Space operations in the late 1980s and early 1990s will require semi-automatic docking and berthing mechanisms with supporting automatic umbilicals. Long life satellites and payloads, such as the 25 kW Power System, will require special services to be supplied across the interface umbilical to accommodate the various multiple, replaceable payloads. Required operational features of the umbilical are expected to be a remotely controlled mate/demate, a manual override as backup, numerous cycle connect/disconnect durability, and a continuous five-year operational life with on-orbit maintenance and repair capability utilizing current Orbiter Replacement Unit (ORU) technology.

In a specific application to the 25 kW Power System, the umbilical services will include transfer of nominal 25 kW of electrical power, heat rejection through a fluid transfer loop, various commands and signals for effecting control, data transfer, and communication services to any mated single or multiple payload which also enables extension of mission time.

The complex mission requirements resulted in selecting a total of 14 electrical connectors that included eight power connectors, four signal connectors, and 2 coax connectors. To meet fluid transfer requirements, two one-half inch diameter line connectors were selected to handle fluid pressures up to 3000 psi. These requirements led to a large platen (12 inches by 22 inches) with self-aligning devices, a free-floating, self-aligning ram system, manual and emergency release systems, and ORU capabilities. The orbital repair, replacement, and maintenance consideration strongly governed the design and led to a system configuration consisting of two major ORU assemblies.

To demonstrate the design capability, a prototype umbilical was built and a series of preliminary operational tests were performed and documented.

SPACECRAFT AUTOMATIC UMBILICAL SYSTEM

INTRODUCTION

Space operations in the late 1980s and early 1990s will require semiautomatic docking and berthing mechanisms with supporting automatic umbilicals. Long-life satellite and Space Shuttle payloads, such as the 25 kW Power System, will require special services to be supplied across the interface to accommodate the various multiple replaceable subsatellites. To provide the necessary capabilities, NASA MSFC issued a set of requirements, and then awarded a study to Lockheed Missiles & Space Company, Inc. (LMSC), to first conceptually define an appropriate berthing technique and then to complete a prototype design of an automatic umbilical system suitable for use with payloads carried by the Space Shuttle.

After considering a variety of docking concepts, a four-element berthing system was selected, as is shown in Fig. 1. This concept consists of a set of four remote manipulator system (RMS)-type end-effector capture/tie-down devices on the power system and corresponding RMS-type grapple fittings on the payload. In operation, the RMS maneuvers the payload to a position where the four grapple fittings can be snared within the end-effectors and then secured to the power system. The concept takes advantage of mechanisms and operating techniques developed for attaching the RMS to a payload.

The umbilical installation is shown in Figs. 2 and 3. Although it is attached to the docking interface structure, its operation is independent and it must comply with the following set of primary requirements specified by Marshall Space Flight Center (MSFC):

- The umbilical shall contain:
  - 2 12.7-mm self-sealing quick-disconnects for use with 3000-psi (gaseous nitrogen) fluid lines
  - 8 Four-pin electrical connectors, shell size 40
  - 4 Sixty-pin electrical connectors, shell size 36
  - 2 Sixty-pin electrical connectors, shell size 24
  - 2 Coaxial connectors, shell size 20
- Angular misalignment: ±3°, carrier-to-plate
- Maximum travel from retracted lock position to carrier plate: 6 inches
- Umbilical: floating, nonload carrying
- Mating: a separate function, but may be simultaneous with docking
- Mating and demating: by remote command, backed by extravehicular activity (EVA) and/or pyrotechnics

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- Operating life: 100-cycle
- Repairable or orbital replaceable units (ORUs)

In the discussions which follow, the general characteristics of the umbilical system are described, followed by detailed descriptions of each individual mechanism.

UMBILICAL GENERAL CONCEPT

The required set of connectors is mounted on rigid platens in the configuration shown in Fig. 4. Two platens, facing each other, constitute a remotely controlled mating/demating interface. One platen is fixed rigidly to the spacecraft frame while the other is movable upon command. One side has its connectors in floating mounts to allow for manufacturing tolerances, differential thermal expansion, and misalignments. The overall installation includes the ORU provisions, which requires two more pair of secondary platens, and is shown in Fig. 5. The complete system thus consists of six platens. Two platens at the two ORU interfaces are passive: one mounted on the power system and the other on the payload. Their counterparts are, respectively, on the two on-orbit replaceable assemblies: the movable platen assembly and the slave platen assembly. These two sets of hand-connected platens provide the ORU interconnect capabilities of the umbilical system. These enable removal of either assembly one at a time or both assemblies as a unit, with the umbilical platen connected as shown in Fig. 5. A schematic of the system is given in Fig. 6 to clarify the geometry and identify the primary component nomenclature. The umbilical system thus consists of two assemblies incorporating two sets of secondary platens, a slave platen and a movable platen.

DETAILS OF MECHANISMS

Eighteen different mechanisms, as shown in Table I, are necessary to perform all the functions required for the operation of this automatic umbilical. Compliance with the ORU requirements dictates that all powered mechanisms must also be manually operable within the working capabilities of an astronaut in EVA. Additionally, the operation of the ram system, which connects the movable platen to the slave platen, must not apply a reaction/separation force to the docking system. An external ram system, as shown in Fig. 7(a) would transfer the insertion load via a long and elastic path through the docking system. Such an arrangement is obviously unsatisfactory inasmuch as it would be conducive to poor control of the umbilical operation and could lead to inadvertent separation of the connectors in case of spacecraft disturbances. A short rigid load path, independent of structure and docking interface, is shown in Fig. 7(b). It is the solution that has been adopted. Its mode of operation (shown in Fig. 8) requires driving the specially threaded end of the Saginaw ram screw into the securing nut with sufficient guidance so that cross-threading will not occur. Rotation of the Saginaw nut then completes the umbilical interconnect operation. Reversal of the sequence separates the umbilical and secures the movable platen in the stowed position with a suitable preload.

The ram mechanism is shown in Fig. 9. This device is based on the use of a Saginaw screw and ball bearing nut, each of which is driven separately by worm screw and gear arrangements. Two such units are mounted under the movable platen. One unit is motorized and drives the other by means of two standard chains

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and sprockets. Operating the screw drive moves the screw with respect to the platen; operating the nut drive moves the platen along the screw. The two units are connected by chain guards designed to resist the chain loads and protect the umbilical cabling. These are shown in Fig. 10.

The movable platen is mounted in a frame as shown in Fig. 11. It is guided by rollers at each end and rests on four adjustable posts when stowed. When disconnected from the tie-down nuts, it is free to rise within the limits of the cabling and has ±6.4-mm clearance fore and aft, and laterally. These clearances also provide significant rotational freedom about the three axes so that a realistic range of docking position tolerances can be accommodated.

To prevent possible connector misalignments, one of each connector-socket pair is elastically mounted with 3 degrees of freedom: two axial and one rotational as shown in Fig. 12. Centering is provided by springs or by strips of elastomers. The flange rests on a teflon pad to minimize friction forces. Typical installation of a size 40 connector is shown in Fig. 13. It should be noted that the two shells are designed to self-center before the pins contact their sockets. A guiding keyway (not shown) ensures proper indexing by forcing the socket-side shell to rotate against its centering springs. Figure 14 shows the centering precision that can be expected at various stages of umbilical connection. The precision drops from ±3 mm at docking to ±0.4 mm at shell insertion; and then to normal pin clearance at pin insertion. This design approach should ensure safe and repeatable pin insertion in the specified 100-cycle operations.

MISCELLANEOUS MECHANISMS

A number of secondary mechanisms are required to: (1) guide the movable platen, (2) secure the movable platen frame assembly and the slave assembly, (3) provide positioning guidance for ORU removal or insertion of these units, and (4) ensure positive locking of all removable components employing EVA astronaut capabilities aided by a standard 11.11-mm (7/16-in.) hexagonal socket driver.

Figure 15 shows the typical configuration of the movable platen guide system. It consists of aluminum channels used as tracks and sets of two delrin rollers having diameters 6.4 mm smaller than the track width. This system provides also a ±3.2-mm lateral freedom.

Figure 16 shows the general arrangement of the movable platen tie-down system. The system consists of a pair of angle tracks with centering ramps so that the EVA astronaut needs only to set the unit within the track and push it in place over the ram pin. The ram mechanism shown in Fig. 17 consists of a special centering pin and clamp with a high mechanical advantage linkage driven by a worm gear mechanism. Thus, with little effort, an EVA astronaut can connect the secondary platens; or conversely, safely disconnect them. A simple lock device is added at the corners of the frame for rigidity and to relieve possible loads on the secondary ram system. It is also designed for simple manual operation.

Figure 18 shows the tie-down arrangement of the slave platen. The platen ram system is identical to that of Fig. 17. The frame lock-pin is considered a necessity in this case to avoid having this unit cantilevered on the ram pin. Without the side lock-pin, it would be difficult to maintain the close platen alignment required for correct threading of the Saginaw screw ends of the main ram system.

Protective doors are used to guard against debris and to prevent accidental damage to the connectors when the umbilical is in the stowed position. These doors are required both on the movable and on the slave platen sides. The doors, mounted on telescopic tracks as shown in Fig. 19, are driven by electric motors. Only the movable-platen door is powered. The slave door has a catch by means of which it is pushed open by the other door. A spring device ensures its closure. This approach is taken because the payload may be passive with no access to its control system until the umbilical is connected.

Figure 20 shows the installation of the power drive units on the movable platen ram system. They are attached to the side of the chain guard and drive the worm gears through reduction drives. A typical power drive unit is shown in Fig. 21. It consists of two electric motors (for redundancy) driving a reducing gear train of specified ratio. The motors are mounted in spring-loaded sockets, with their pinions meshing the main gear. In case of failure, a motor can be partially withdrawn (out of mesh) or removed and replaced by a new one. Current would be supplied to these motors through the sockets so that an EVA astronaut can plug and unplug them when servicing the umbilical. The total umbilical system requires three to four power units (six to eight motors).[1]

ANCILLARY DESIGN FEATURES

The electric power cables specified for the large connectors are multistrand size "O." Their outer diameter is close to 12 mm, which necessitates special preforming to enable assembly of the platen cabling configuration as shown in Fig. 22. However, a more practical approach may be obtained using the configuration of Fig. 23, which will most likely be used for the flex fluid hoses. A total of 32 No. O cables is to be installed, plus 4 bundles of 61 small cables and 2 large coaxial cables.

In case of emergency, such as pin seizure, main power failure, or jettison of payload to protect the crew, it may be necessary to separate the umbilical platen forcibly and/or quickly. This can be achieved by using protechnic separation nuts for releasing the Saginaw rods and separation actuators as shown in Fig. 24. The pyrotechnic bellows motors have the advantage of providing a large force in a compact volume, and they do not release any contaminants. Because they are mounted on the slave platen, they will be accessible for replacement by an EVA astronaut.

The electrical system consists of power supply to the electric motors and wiring for a set of position sensors (microswitches) to provide for a status readout for each function. A typical setup suitable for ground testing is shown in Fig. 25. It requires four command switches: doors, screws, platen, and emergency release.

[1]Normally three power units. A fourth may be needed to provide proper door tracking.

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RESULTS AND CONCLUSIONS

An umbilical system design has been completed that incorporates all the features specified for a power system-to-payload interconnect capability. A proof-of-concept prototype of the umbilical system has been built to determine experimentally the suitability of the threading characteristics of the ram mechanism and to verify freedom from cross-threading. It will also be used to measure connector insertion loads, first in an ambient laboratory environment and later in a space-simulated environment (thermal-vacuum chamber). Laboratory ambient testing is being performed by the engineering staff and students at San Jose State University.

The following highlight key conclusions derived from the study:

1. Berthing systems that utilize the RMS can be simplified by using RMS targets, closed-circuit TV cameras, tie into the RMS control system, and grapple-fixture and end-effector-like capture and secure mechanisms.

2. To effect a remotely controlled umbilical interconnect in proximity with a manned spacecraft and to provide for EVA backup and ORU maintenance capabilities, 18 different mechanisms are found to be necessary.

3. The weight impact of providing for ORU capability in a large multiple connector umbilical system was found to be in the order of +60 percent.

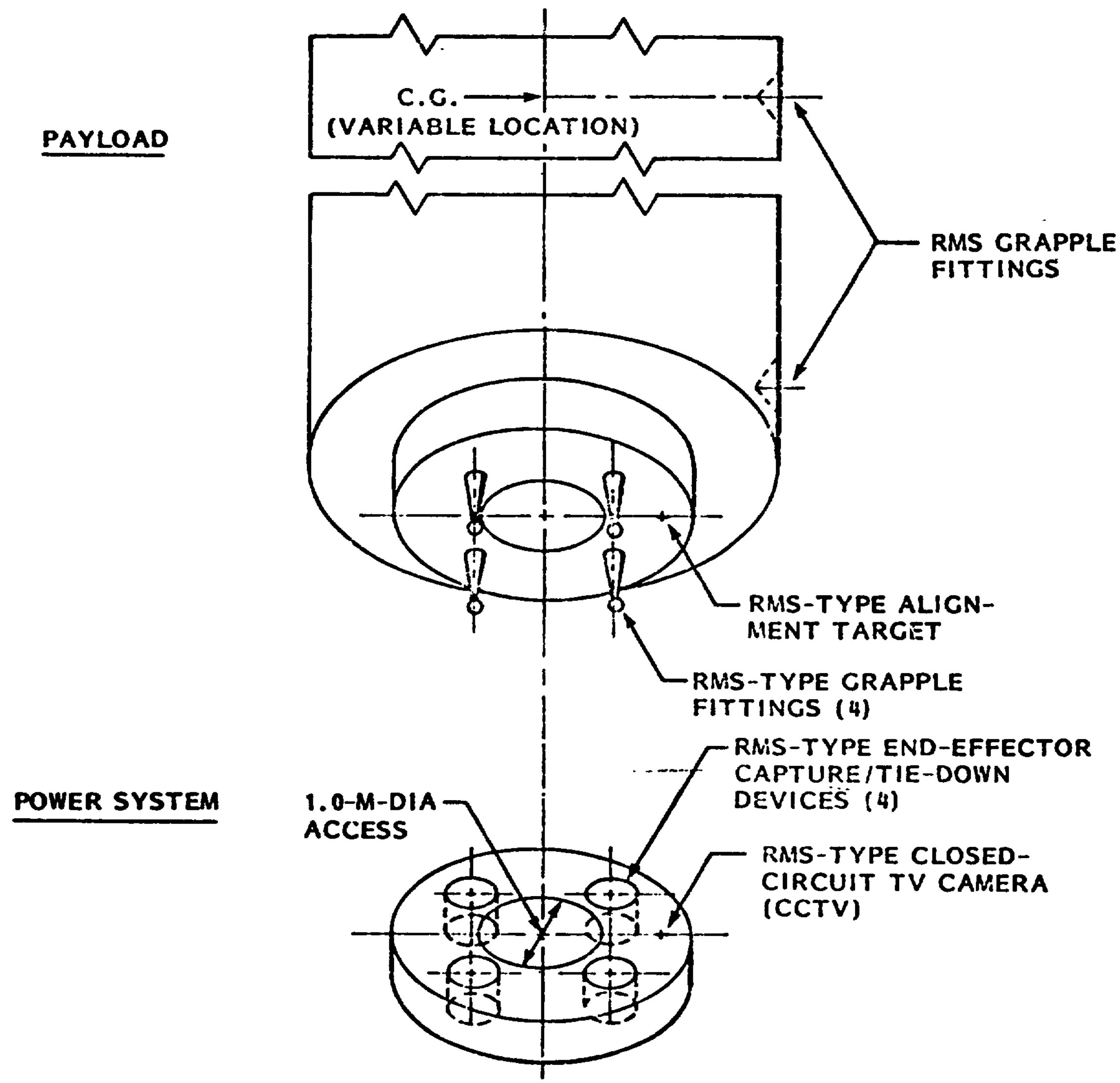

Figure 1. Berthing System Concept

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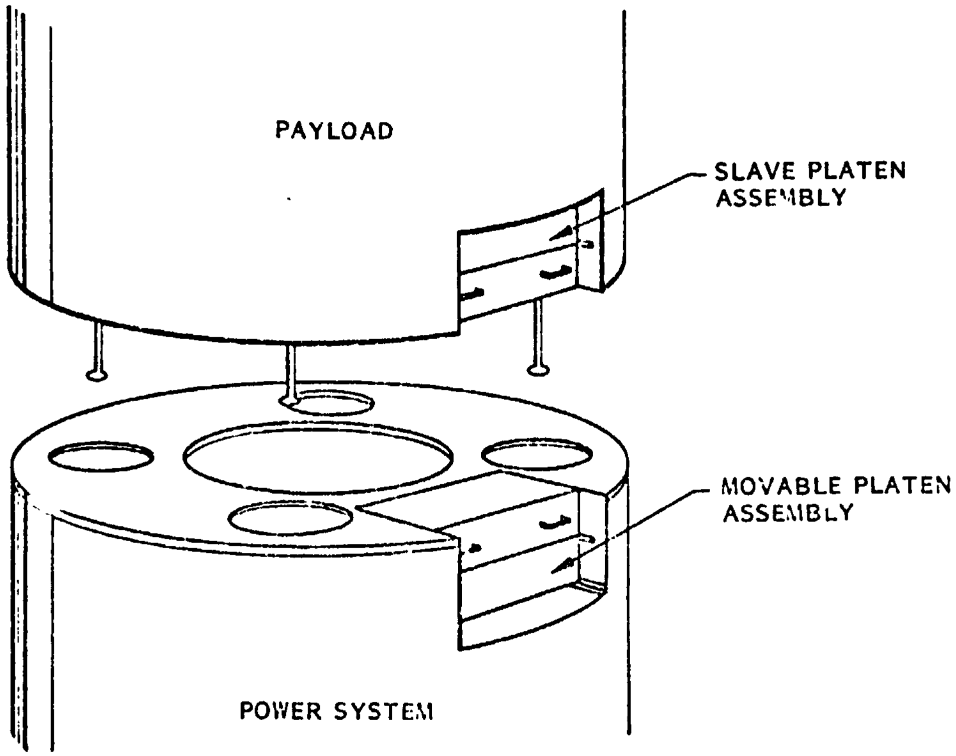

Figure 2. Berthing and Umbilical Installation

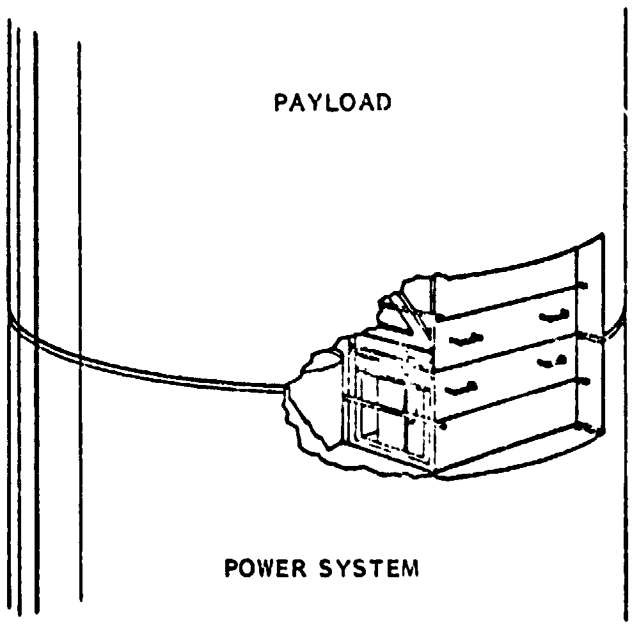

Figure 3. Umbilical Prior to Connection

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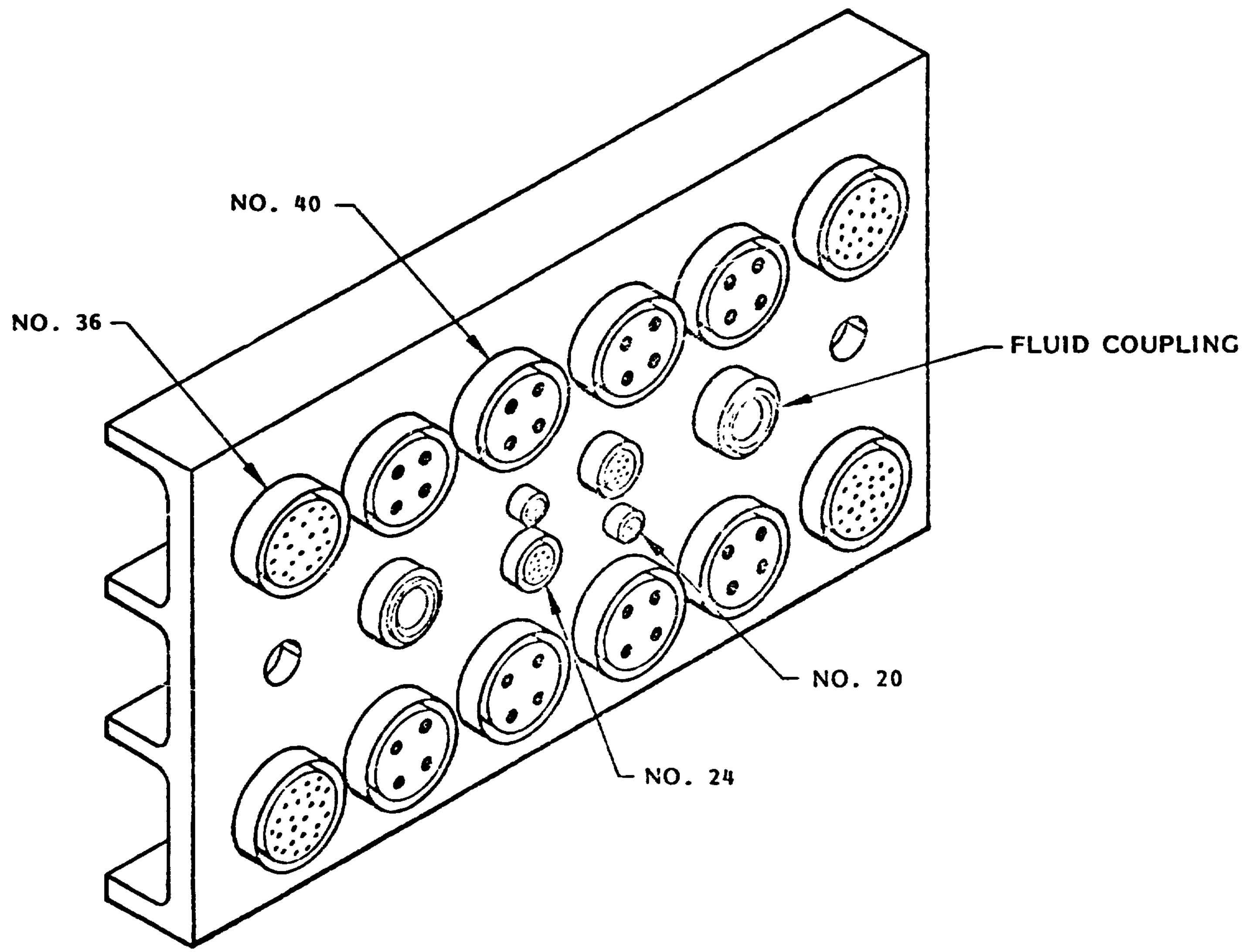

Figure 4. Typical Platen Connector Arrangement

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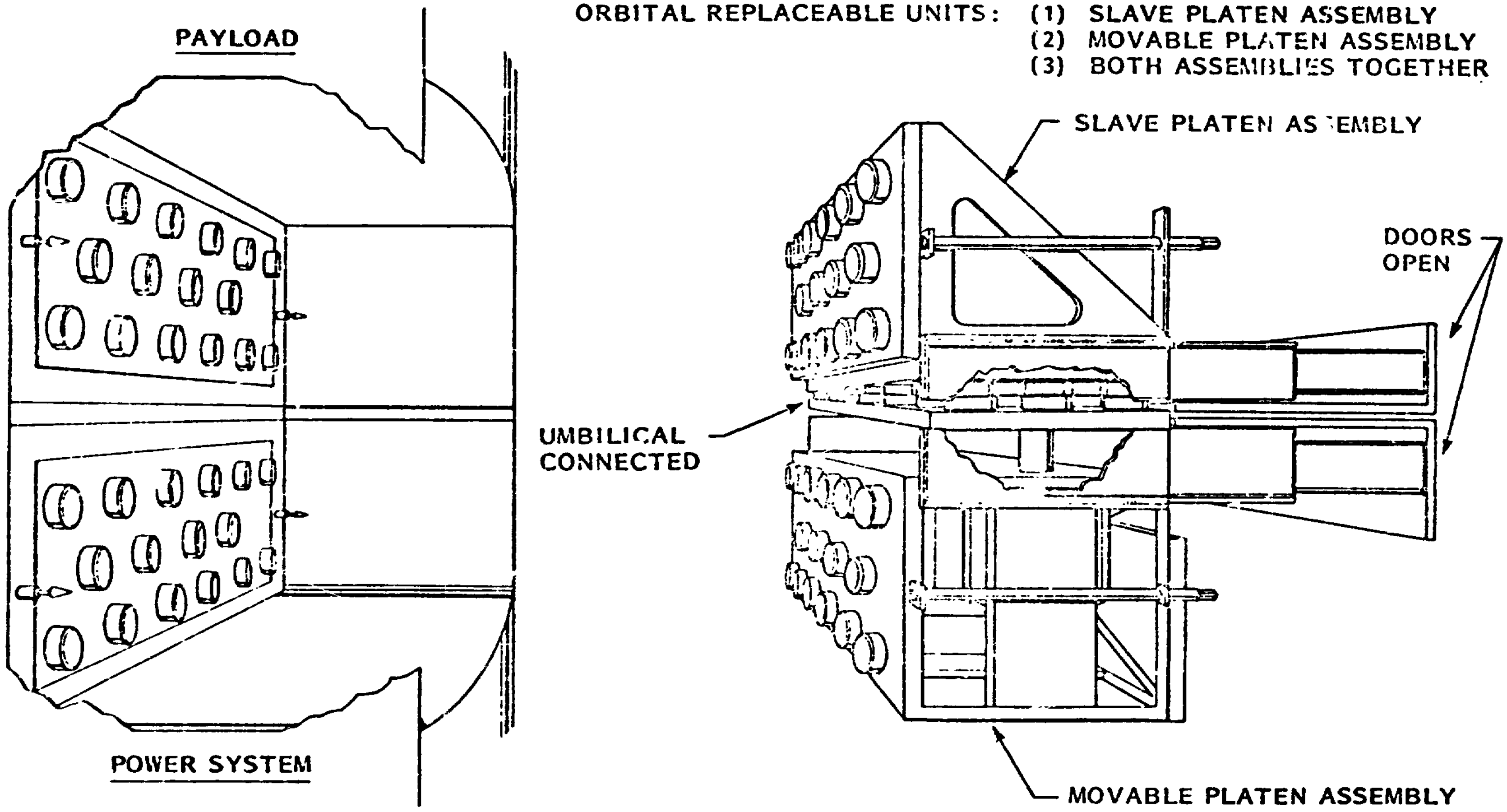

Figure 5. Orbital Replacement Unit

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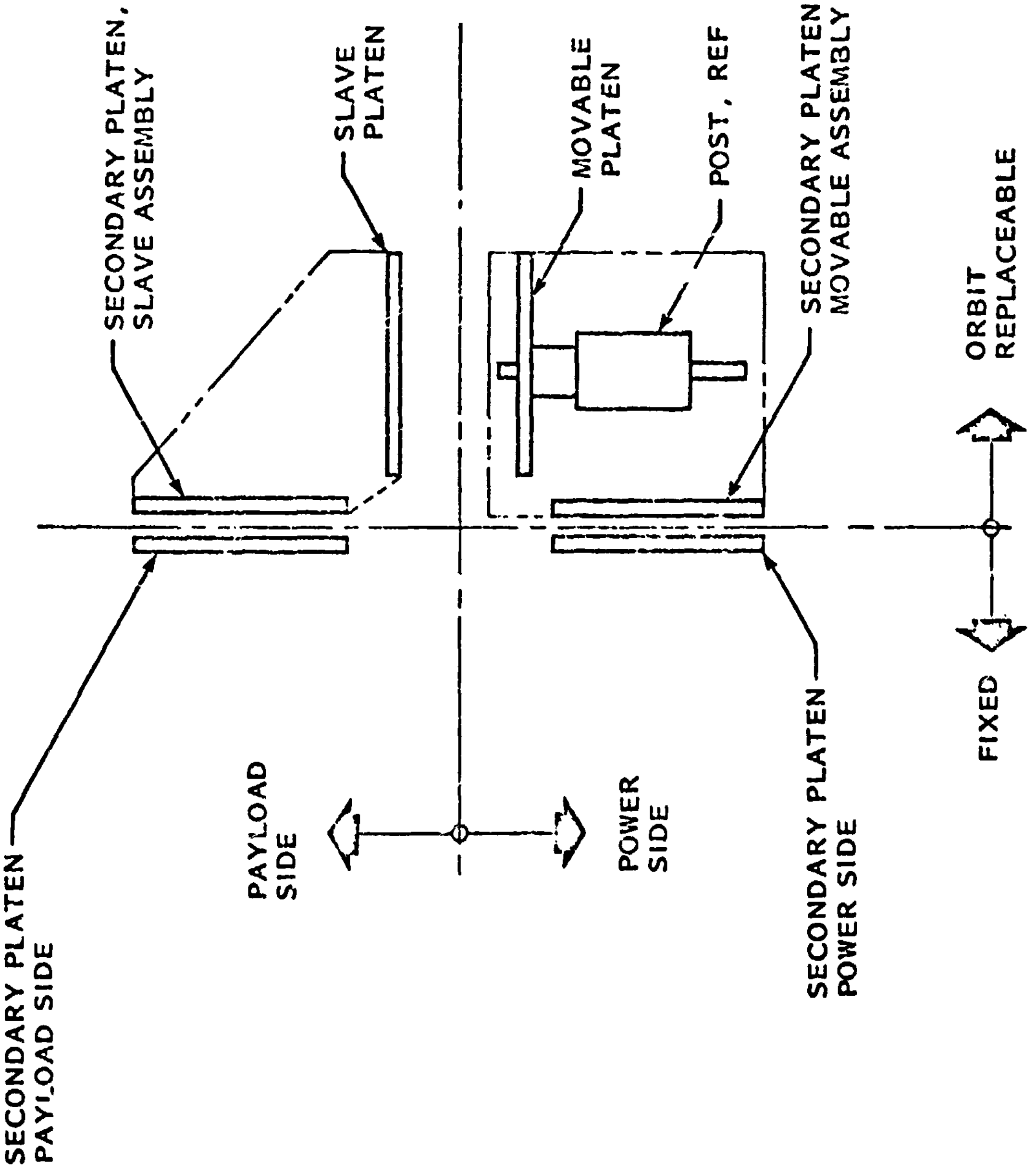

Figure 6. Schematic and Nomenclature

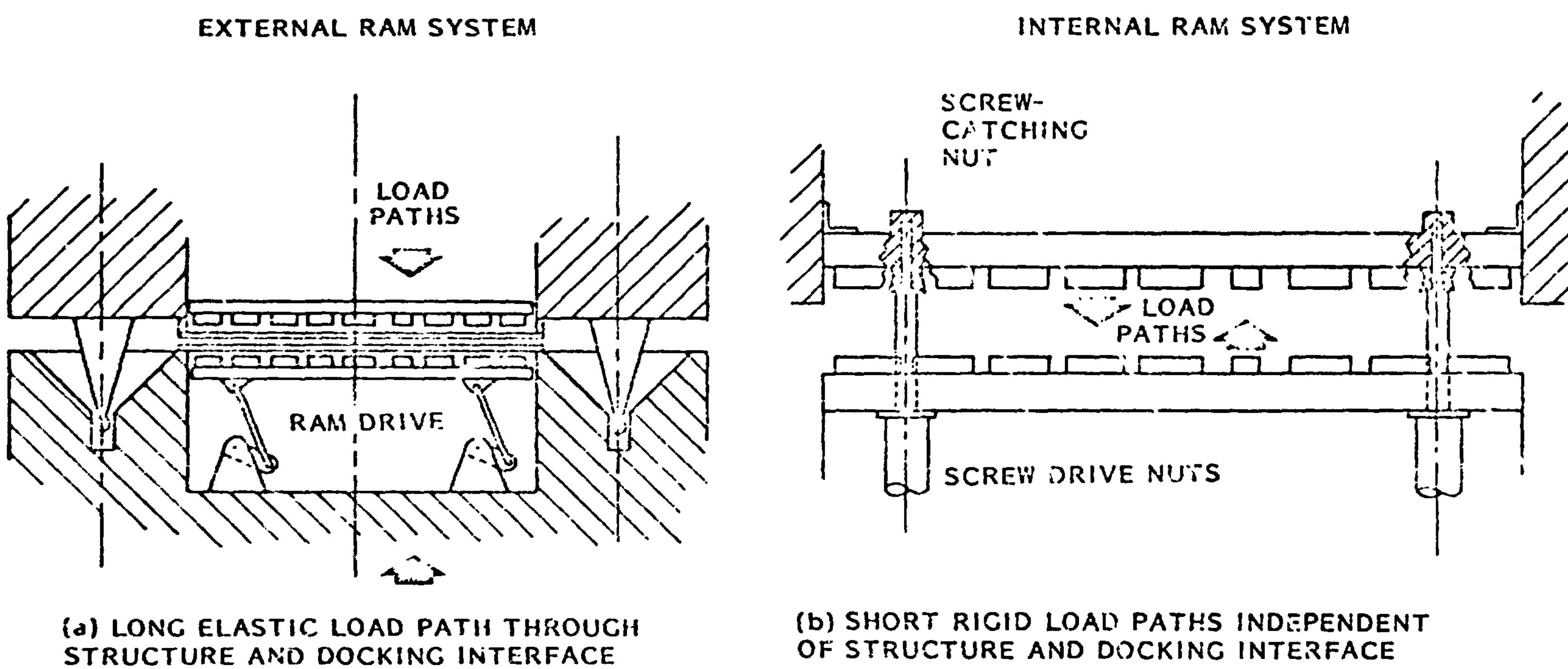

Figure 7. Umbilical Drive System

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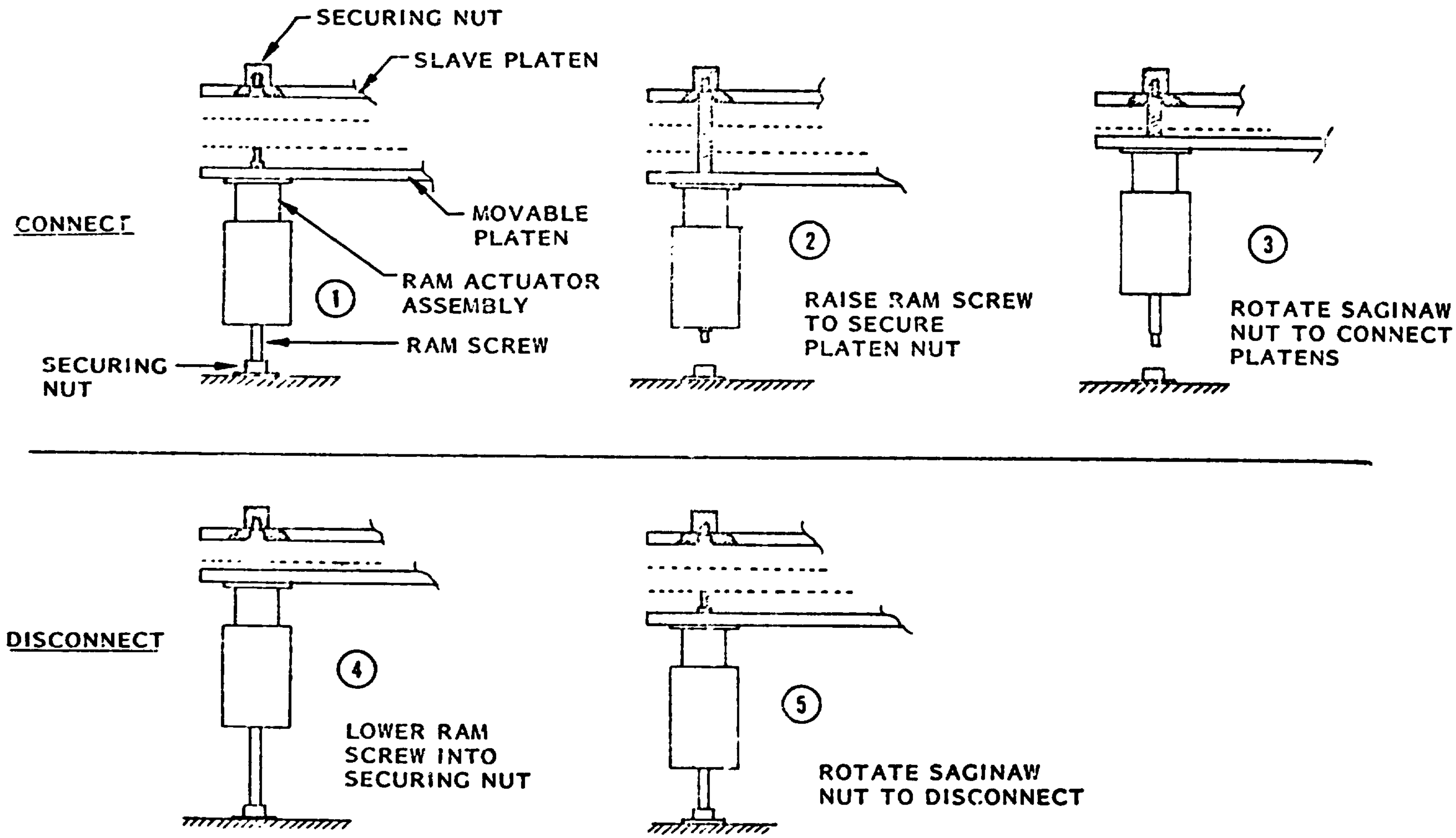

Figure 8. Operation of Selected Umbilical Drive System

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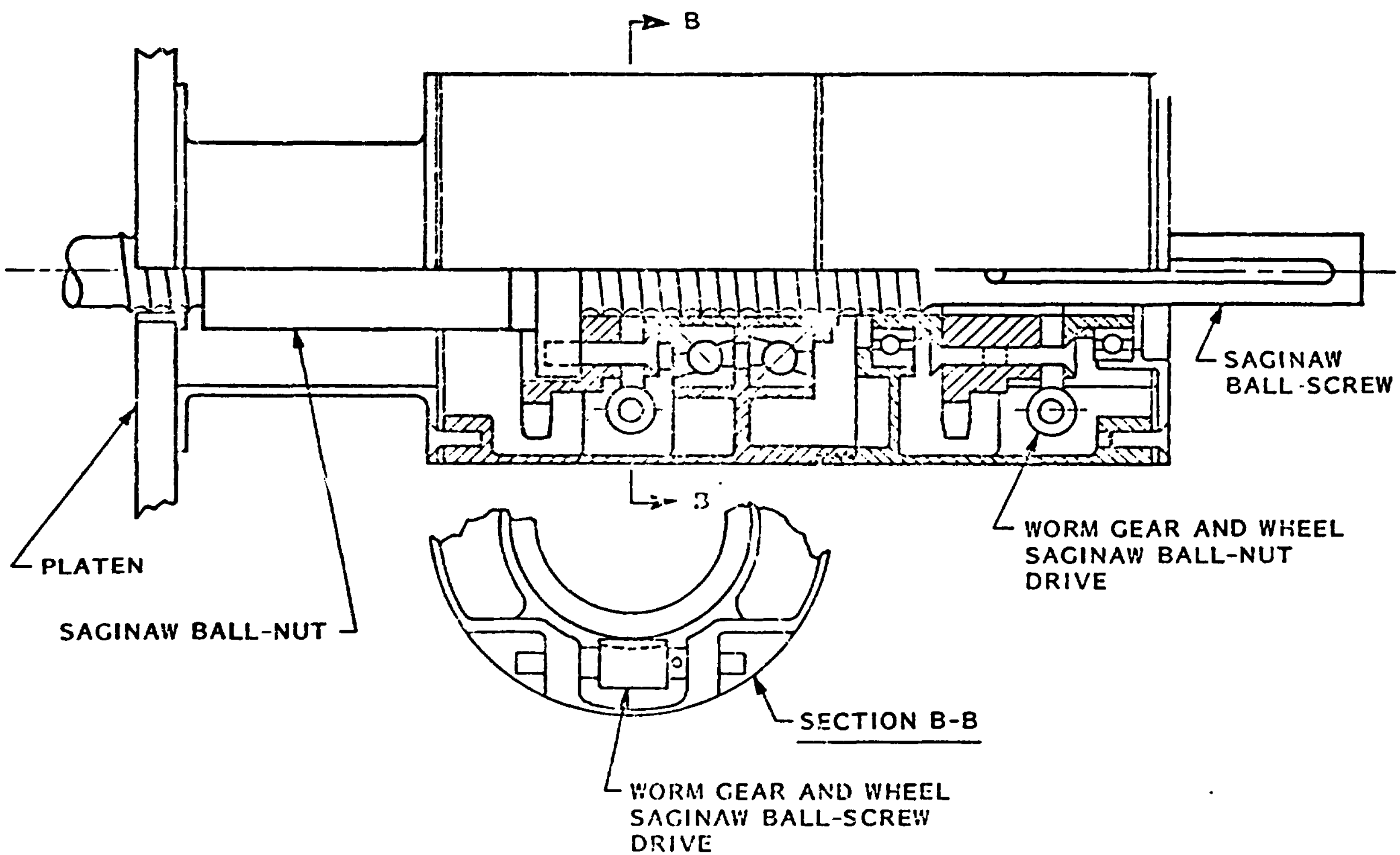


Figure 9. Ram Screw Drive System

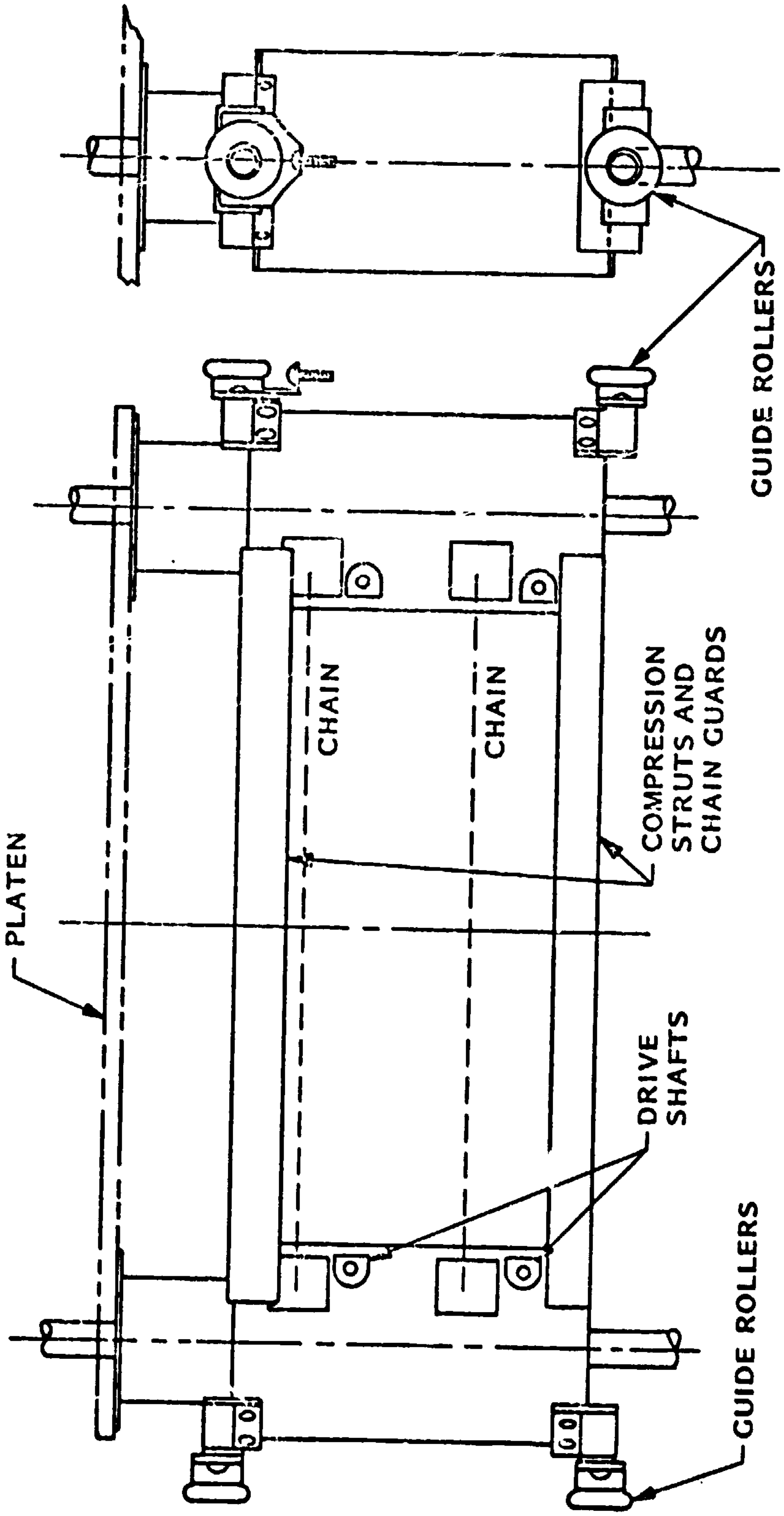
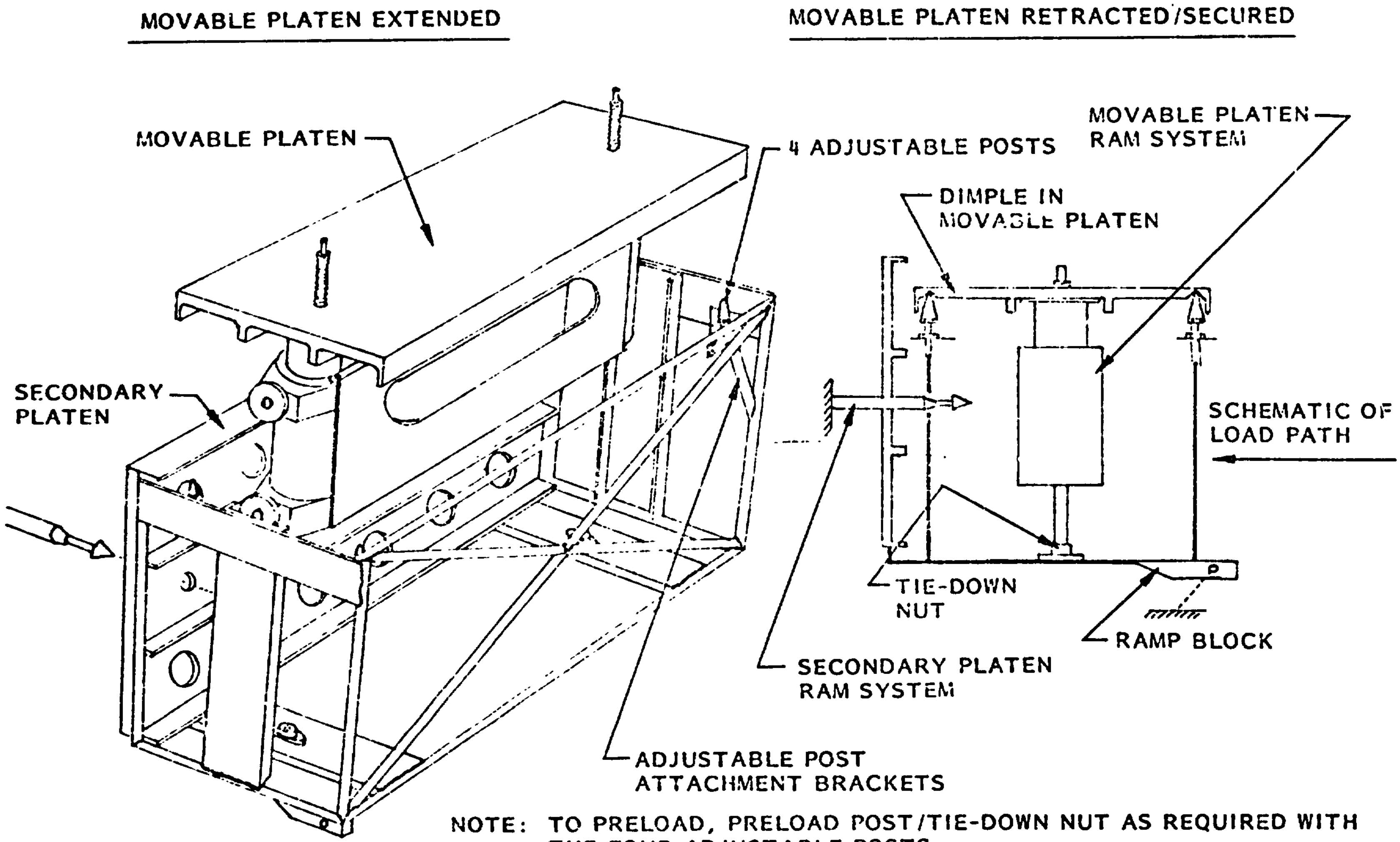


Figure 10. Details of Ram Drive Installation



NOTE: TO PRELOAD, PRELOAD POST/TIE-DOWN NUT AS REQUIRED WITH THE FOUR ADJUSTABLE POSTS

Figure 11. Movable Platen Tie-Down System

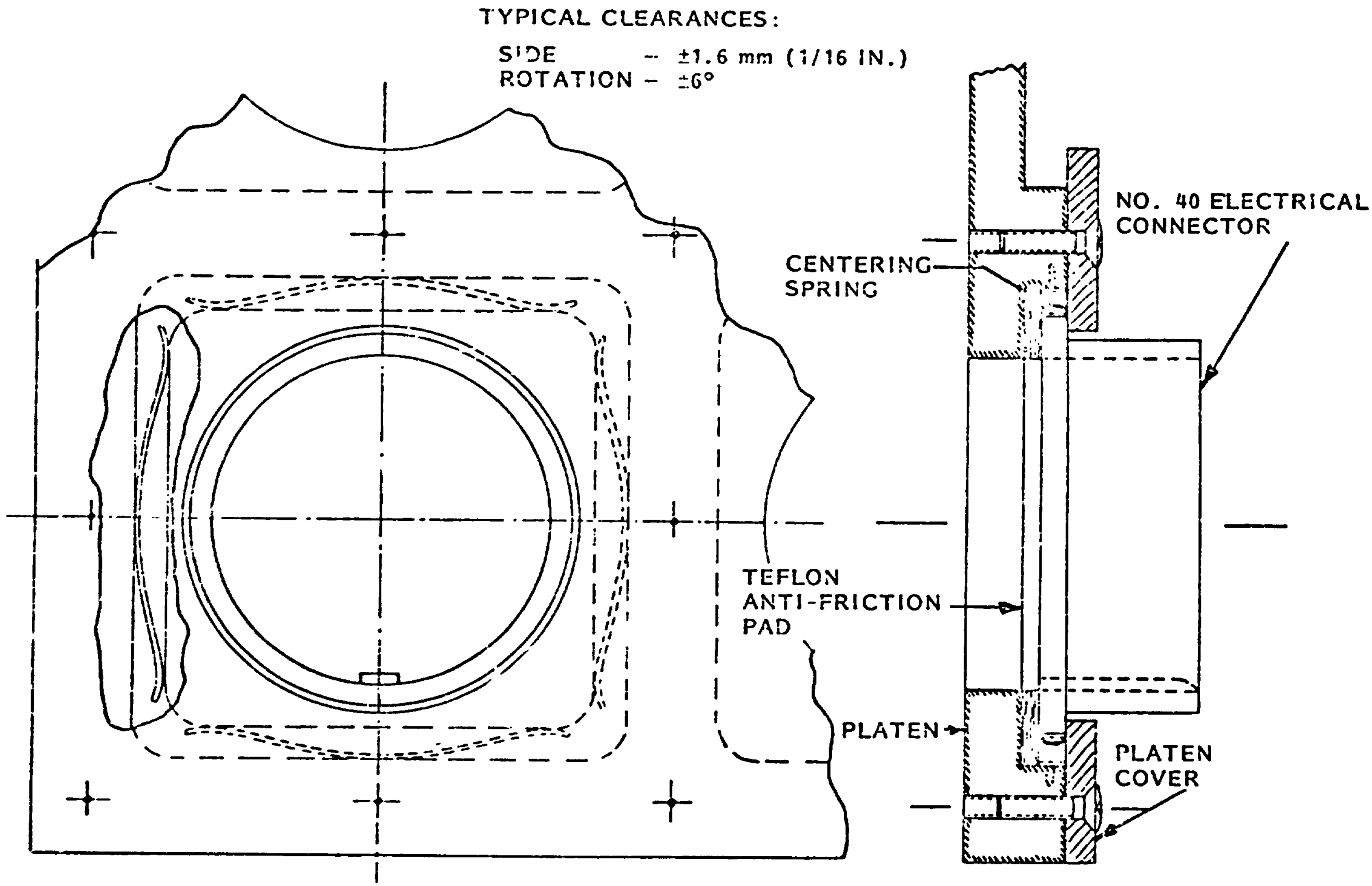


Figure 12. Floating Electric Connector/Platen Installation

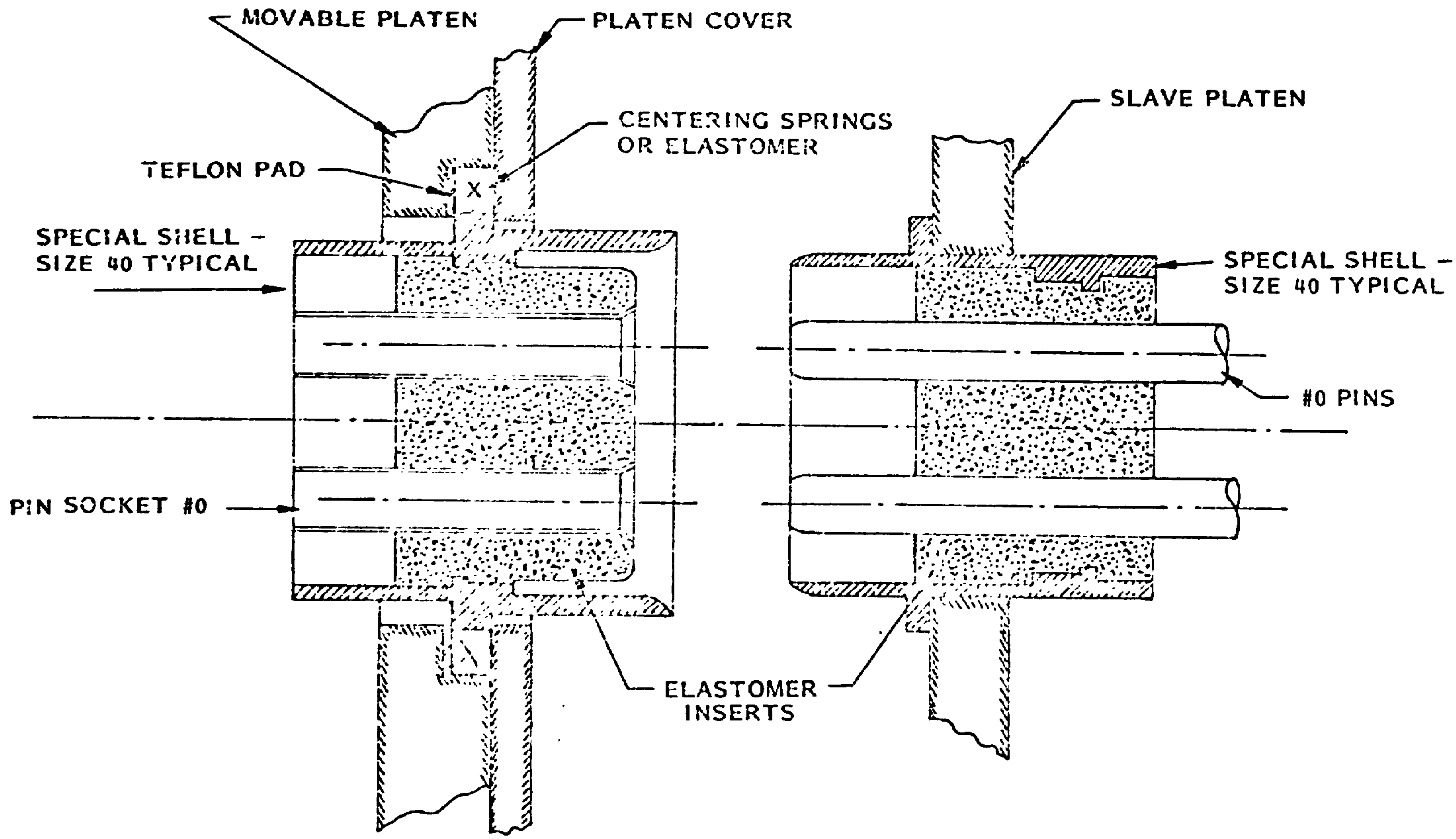


Figure 13. Typical Electric Connector Configuration

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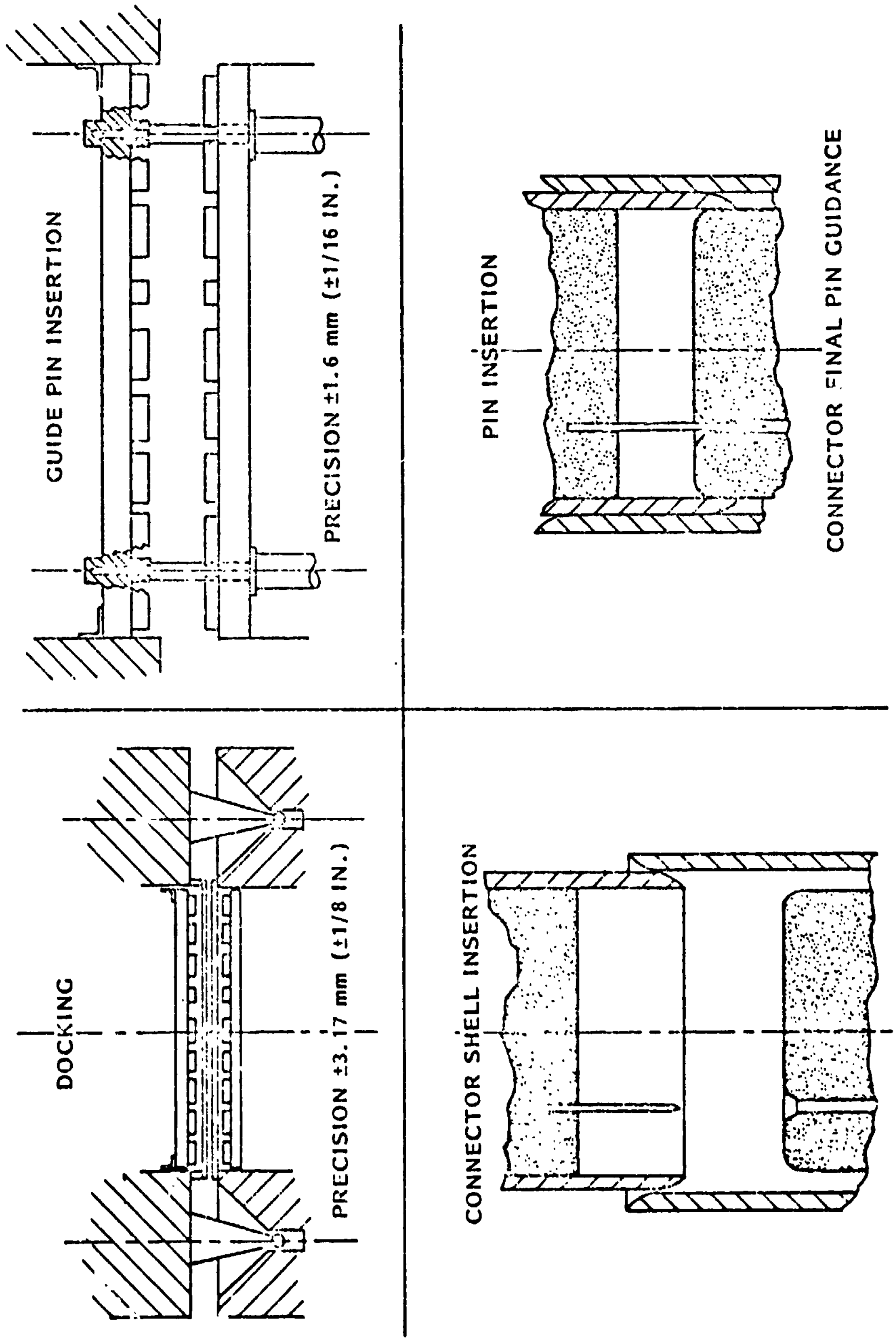

Figure 14. Precision of Connector Insertion

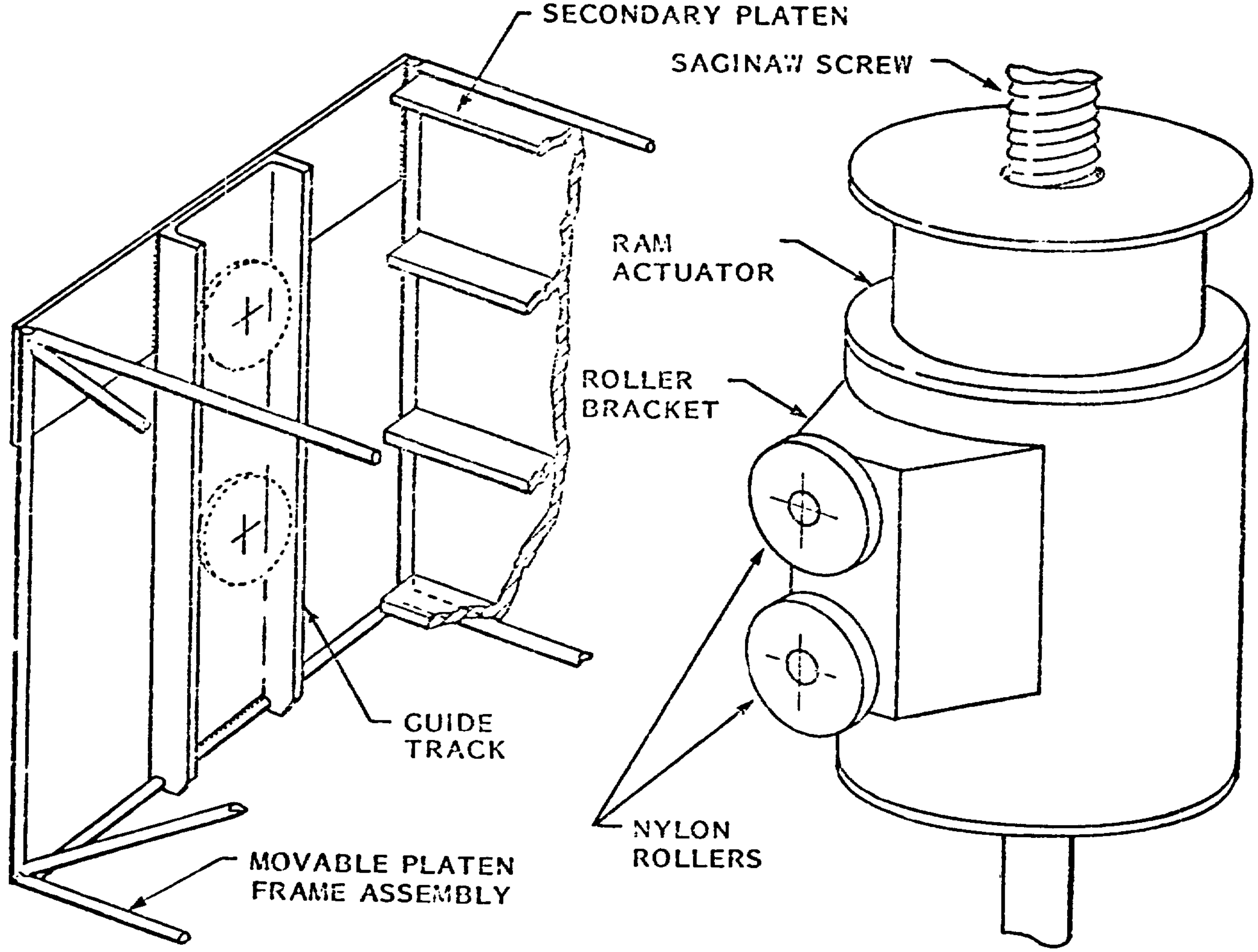

Figure 15. Detail of Guide Track

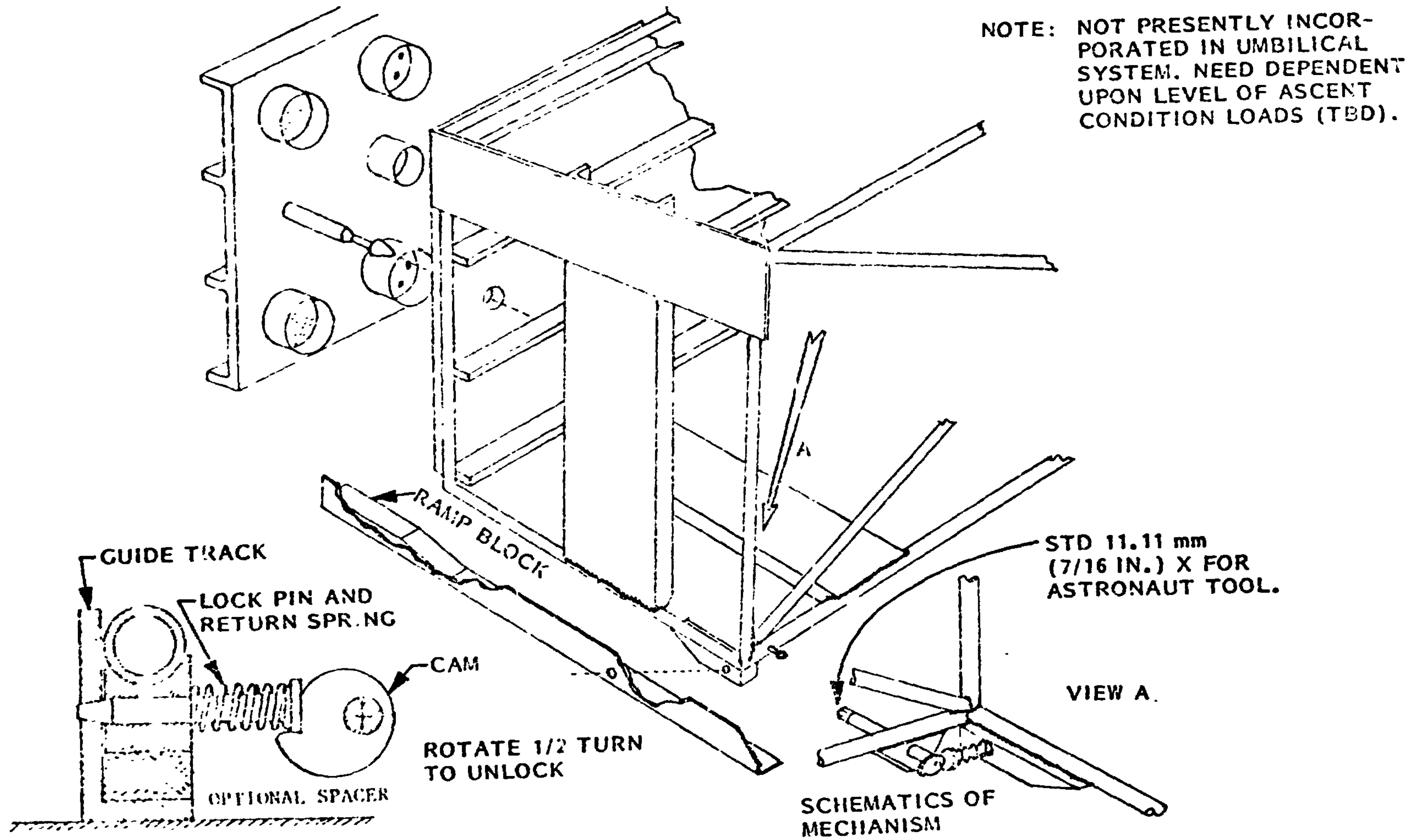

Figure 16. Frame Assembly Tie-Down Mechanism

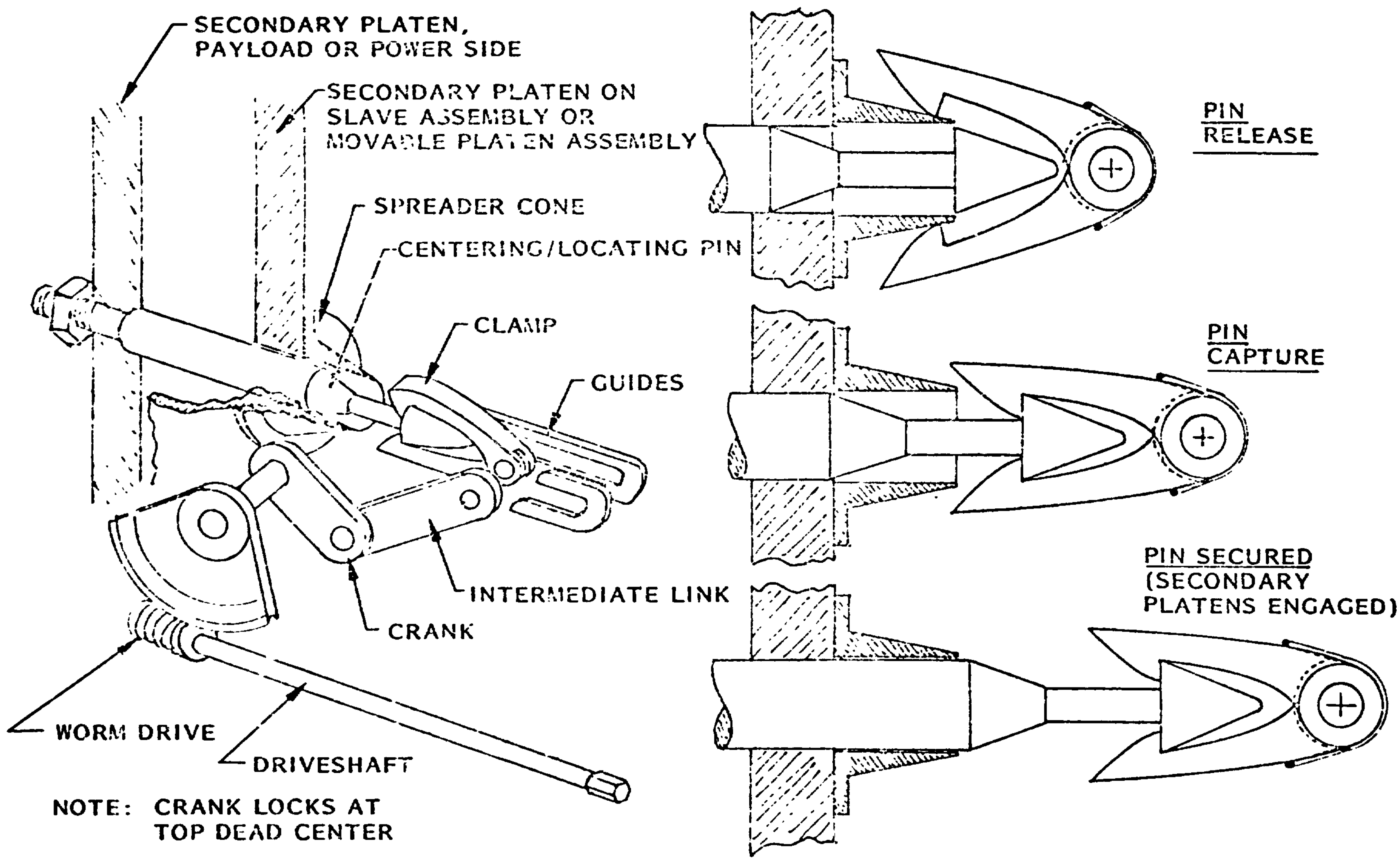

Figure 17. Secondary Platen Tie-Down Mechanism

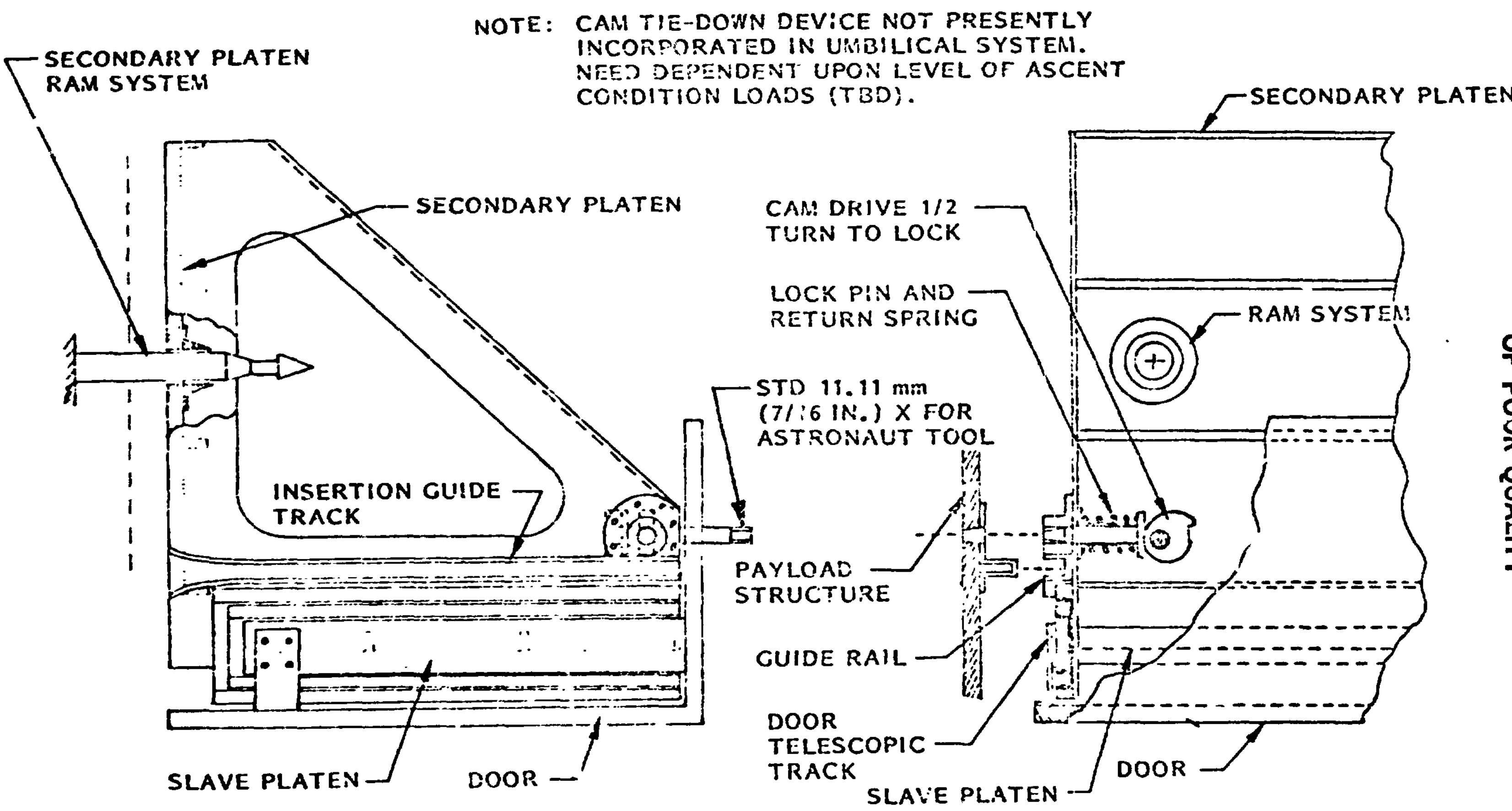

Figure 18. Slave Platen Assembly Tie-Down Mechanism

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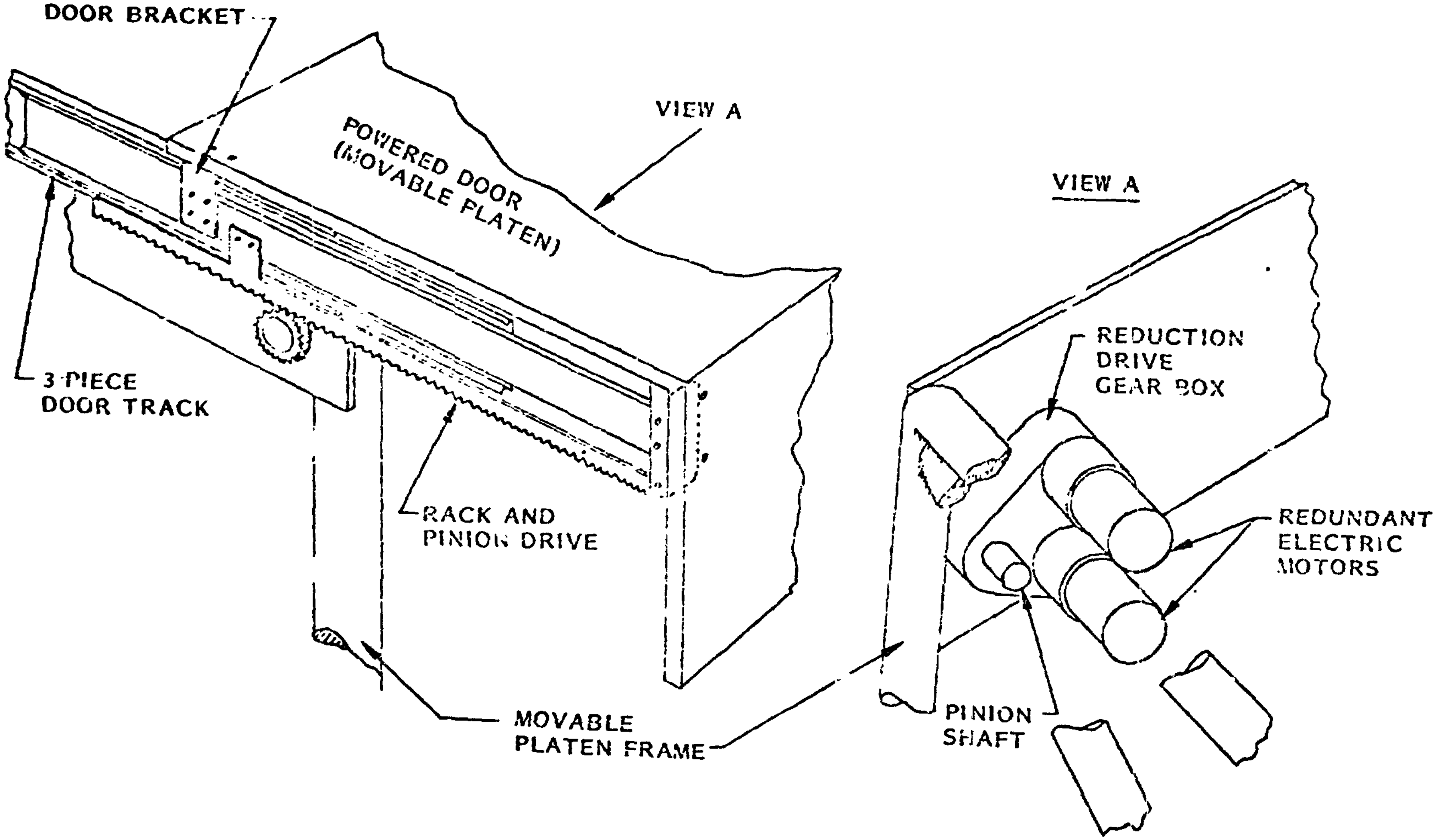

Figure 19. Powered Door Slide and Drive

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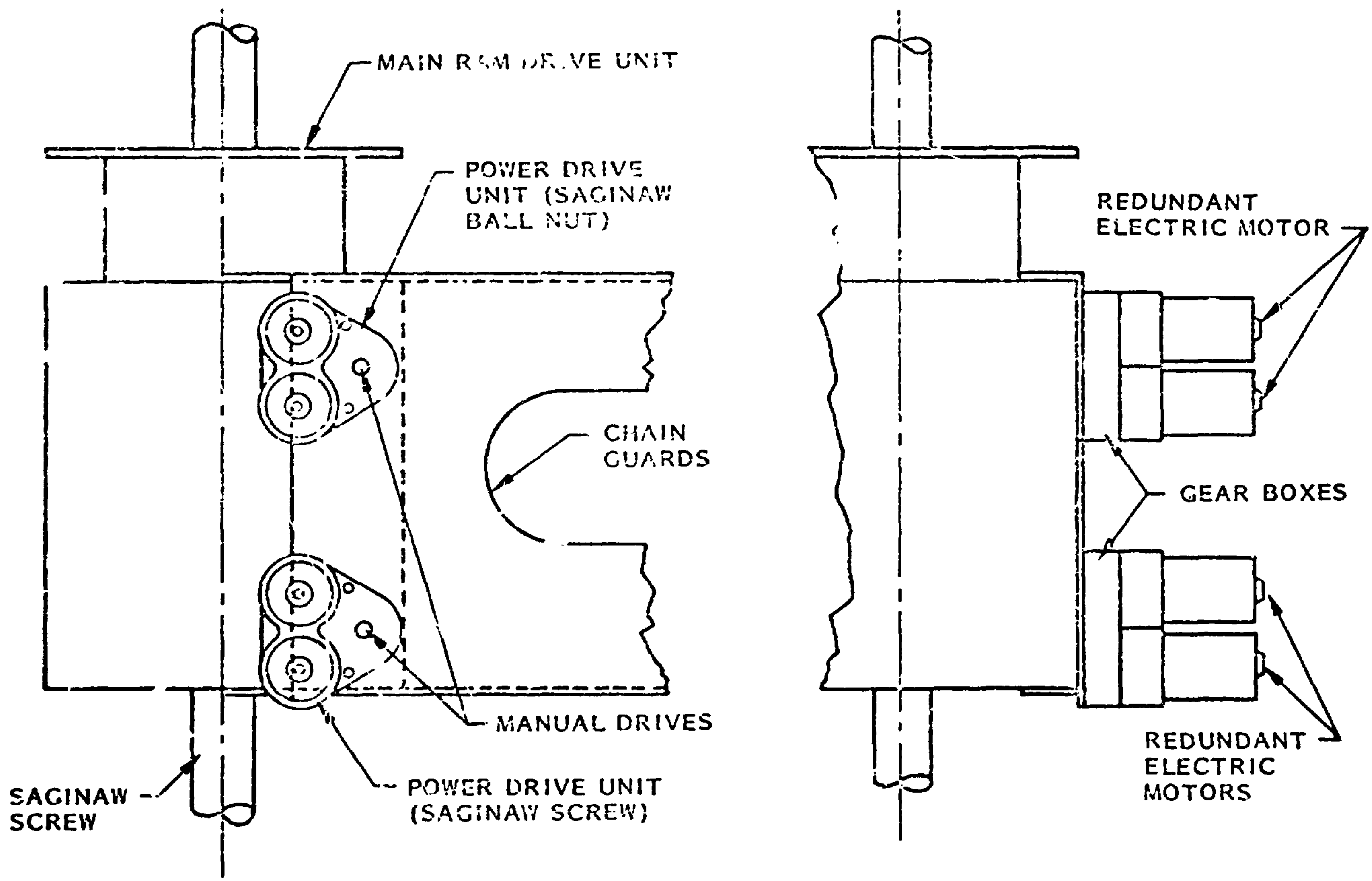

Figure 20. Power Drive Installation (Ram System)

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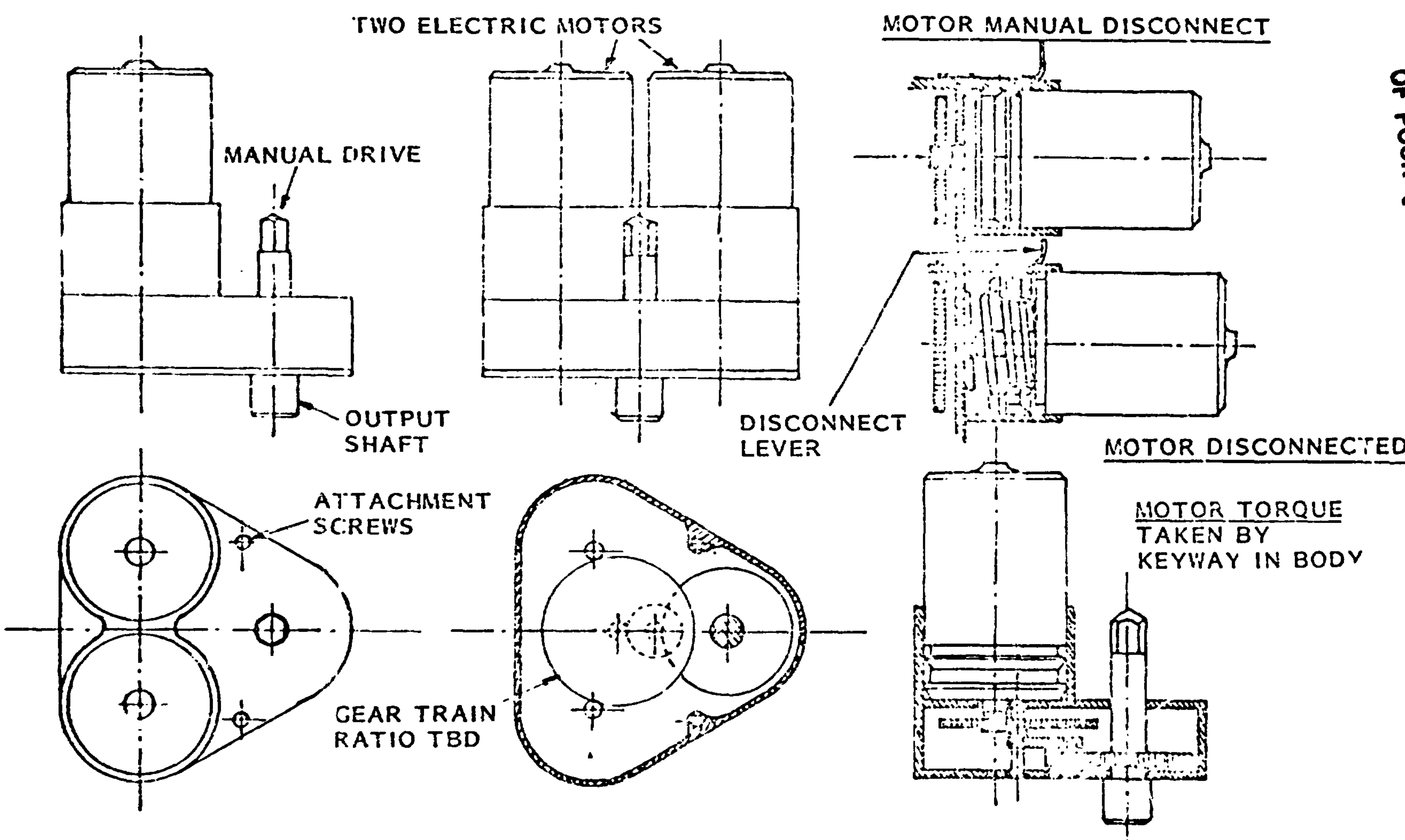

Figure 21. Redundant Power Drive System

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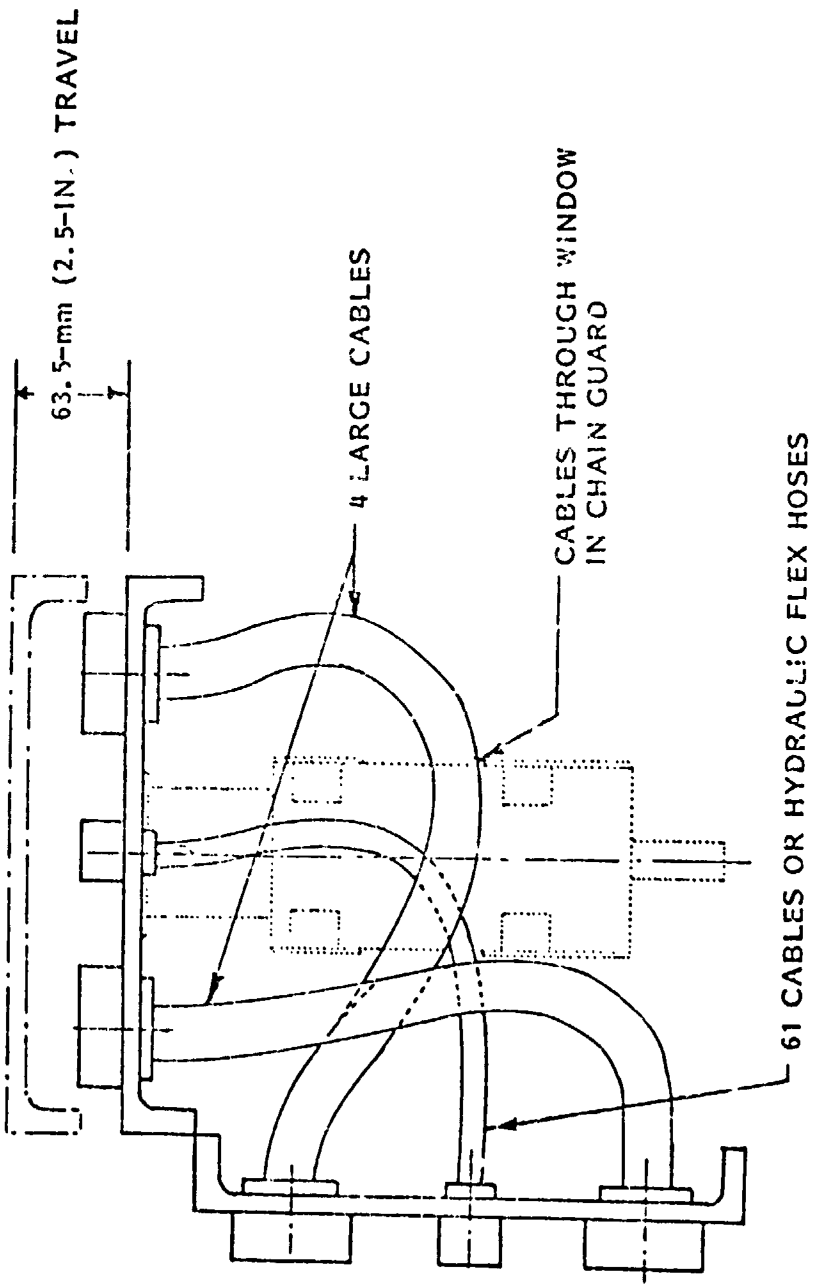

Figure 22. Preliminary Concept of Platen Cabling

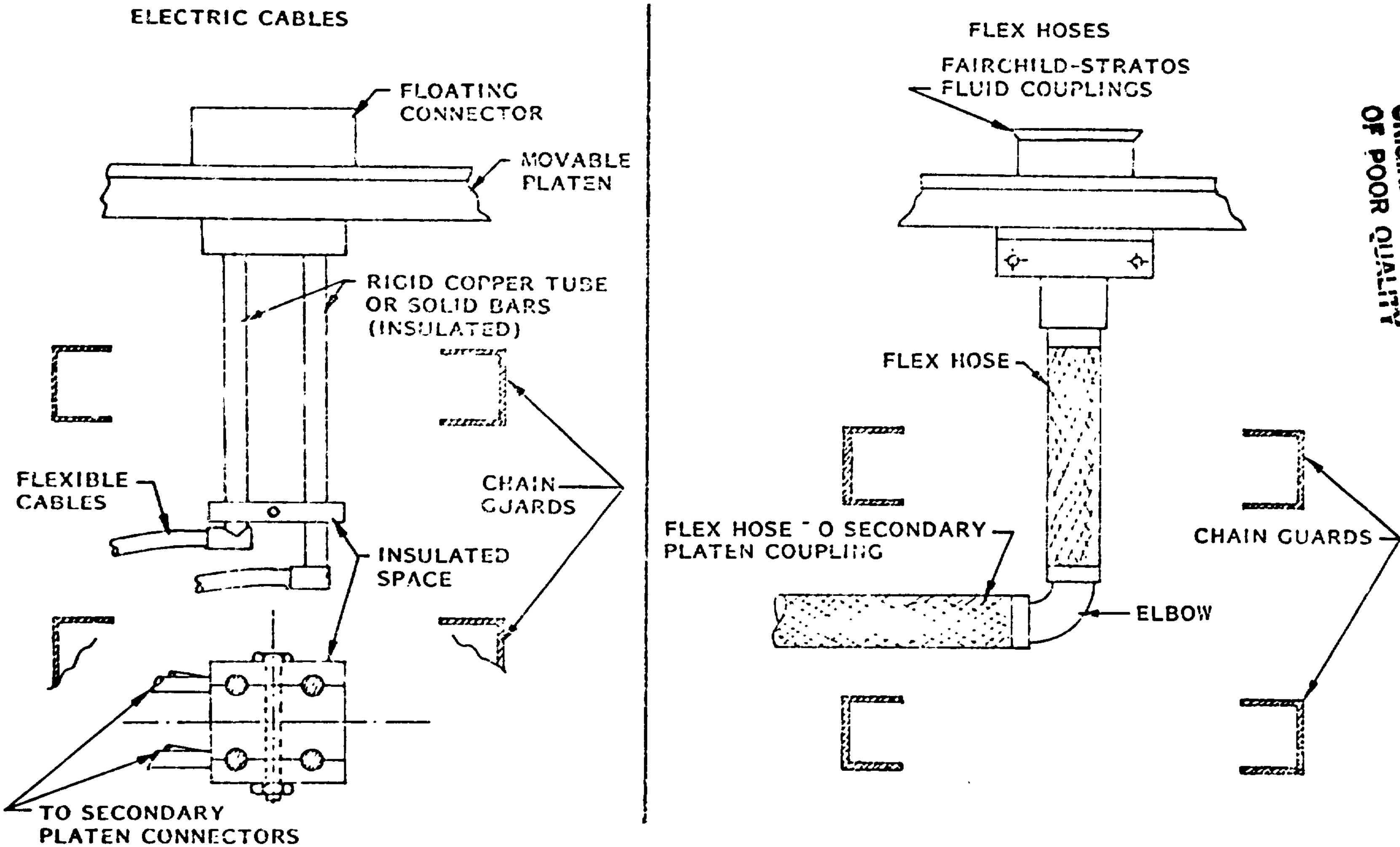

Figure 23. Electric Cables and Flex Hoses

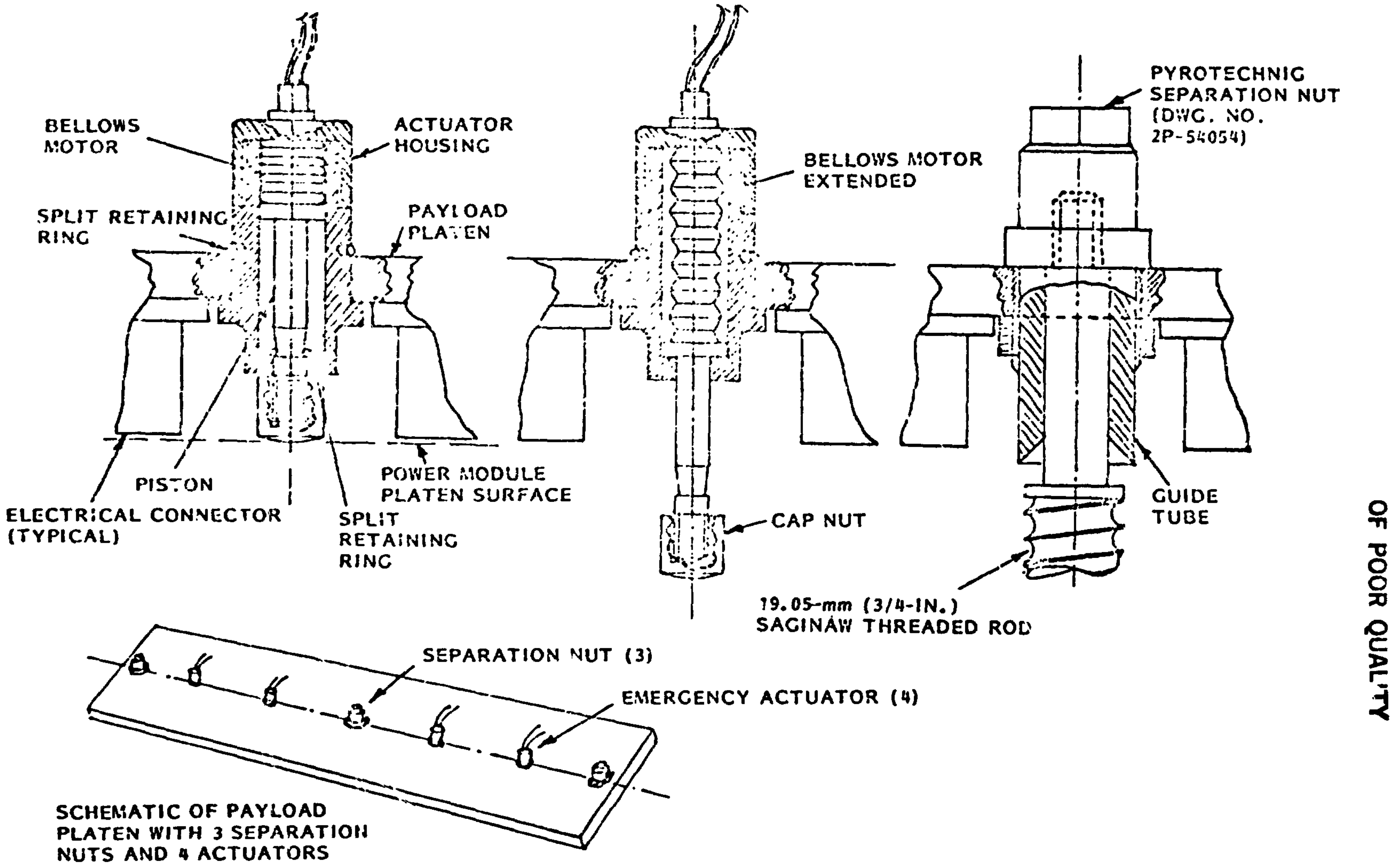

Figure 24. Emergency Relay System

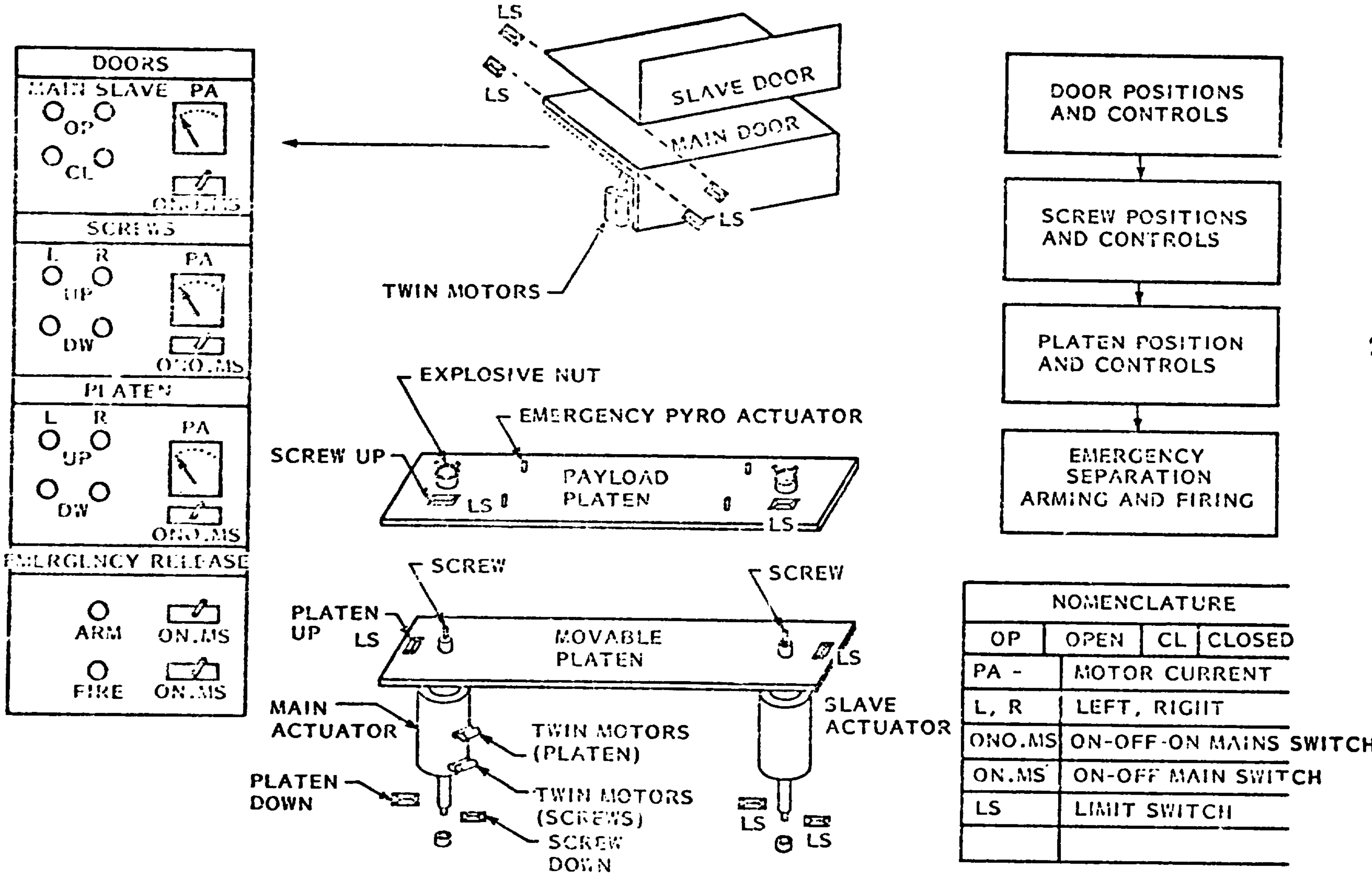

Figure 25. Control System

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TABLE I MECHANISMS

MOVABLE PLATEN ASSEMBLY

- CENTERING SCREW DRIVE
- RAM DRIVE
- FRAME ASSEMBLY ORU ROLLER/TRACKS
- MOVABLE PLATEN GUIDE TRACKS
- DOOR CLOSURE DRIVE SYSTEM
- UMBILICAL ASCENT AND RE-ENTRY SAFETY LOCKS
- DUAL ELECTRIC MOTORS INSTALLATION
- ELECTRIC MOTOR EMERGENCY SHAFT DISCONNECT
- SECONDARY PLATENS MANUAL DRIVE *
- DOOR TRACKS *
- DOOR OPEN TOGGLE LOCK*
- DOOR CLOSED TOGGLE LOCK *
- ELECTRIC CONNECTORS FLOATING MOUNTS*

SLAVE PLATEN ASSEMBLY

- ORU ROLLER/TRACKS
- DOOR CLOSURE SYSTEM
- SCREW-DRIVE EMERGENCY RELEASE NUT
- EMERGENCY SEPARATION PYRO MOTORS
- DOOR TRACKS *
- SECONDARY PLATENS MANUAL DRIVE *
- DOOR OPEN TOGGLE LOCK *
- DOOR CLOSED TOGGLE LOCK *
- ELECTRICAL CONNECTORS FLOATING MOUNTS *

MISCELLANEOUS

- FLUID COUPLING SELF-ALIGNING SYSTEM (ON ALL SIX PLATENS)

NOTE: ASTERISK * ITEMS ARE THE SAME ON BOTH MAIN ASSEMBLIES.

TEST UNIT MANUFACTURE AND TEST

Engagement of the ram screws in the securing nuts was judged as the most critical feature of umbilical mechanism performance. Accordingly, the pre-prototype test unit was manufactured mainly to test this feature. However, the unit was also made to be capable of incorporation of other elements of the mechanism should future mechanism or test elaboration be required.

Some substitutions were found necessary because of extensive lead times, unavailability, or cost for certain items. In addition, because of connector delivery problems, it was decided to test with connectors in only eight of the platen connector seats. This configuration was regarded as suitable as it filled all of the No. 40 shell platen connector locations and resulted in testing heavy connector pin engagement in the most critically misaligned positions for such pins. Nothing was available to simulate the force-deflection characteristics of the Fairchild-Stratos partially pressure-compensated couplings on which umbilical mechanism design was based. This problem was solved by the design and use of elastomeric spring assemblies to simulate the 3000-spi fluid coupling force-deflection characteristics.

The test unit consisted essentially of a slave (fixed platen; a ram-driven movable platen in its frame; the rams, drive chains, and chain guards; electrical connectors, cabling, spring assemblies (to simulate the fluid coupling loads); and incidentals. For the test project, special torque-limited drive motors with manual controls were mounted on the movable platen drive ram and a two-piece test frame made in which the upper piece provided a fixed support for the upper (fixed) platen and the lower piece provided a fixed base for mounting of the movable platen containment/guide frame. This general test arrangement is shown in Fig.26.

Aside from calibration and shake-down tests, the test project consisted of operating the umbilical mechanism through connector engagement/disengagement cycles under various conditions of alignment.

For tests, the upper (slave or "fixed") platen was clamped in a reference position on the upper platen support (see Fig.26), the null reference position being one in which the upper platen and the lower movable platen were essentially parallel in three directions (along across, and perpendicular to the platen face). Misalignments were then produced by translating or rotating the upper "fixed" platen with respect to its null position on the upper platen support.

The umbilical system was exercised through a minimum of fifty complete cycles. Test data were acquired in a number of conditions, namely: on center, 1/8 inch displacement in X direction, 1/8 inch displacement in Y direction, 1 1/2 deg rotation about X axis, and 1 1/2 deg rotation about Y axis. In all cases, screw thread engagement was achieved with no difficulty and minimal forces.

The on center and linear displacement tests produced full engagement of all electrical connectors with very low engagement forces (100 lb total force order of magnitude). The maximum system load varied between 2150 and 2800 lb, simulating the hydraulic load. This load was uniformly distributed between the two screws. The applied motor torque varied between 38 in-lb and 45 in-lb (motor stall torque), and compares with a calculated 29.4 inch-pounds, and a maximum allowable EVA requirement of 60 inch-pounds (for manual operation). (FIG. 28). The angular rotation tests about the X axis (1 1/2 deg rotation about the long edge of the platen) were similar to the previously described linear displacement tests except that the fixed platen and the moving platen did not align satisfactorily. All electrical connectors made contact, but those far distant from the avis of rotation were not fully engaged.

The loading on the screw threads was asymmetric, reflecting the differential compression of the hydraulic load simulators. The maximum load on a single screw was approximately 7200 lb (far in excess of design requirements) with no observable deleterious effects. (FIG 27)

Upon review of the design, it was decided that the difficulties encountered in mating the initially inclined (at 1 1/2 deg) platens could be surmounted in at least two ways: 1) mount the movable platen guide and containment frame in a simple gimbal system, and 2) use a ball screw "pilot" lead which would provide more torque on the movable platen during initial ball screw contact and which hence would remove the initial misalignment prior to full engagement of the Acme thread portion of the ball screw with the securing nut. A gimbal system between the platen and frame may also be required in order to meet the $\pm 3^{o}$ alignment requirement. Neither of these 2 design changes have been incorporated into the hardware.

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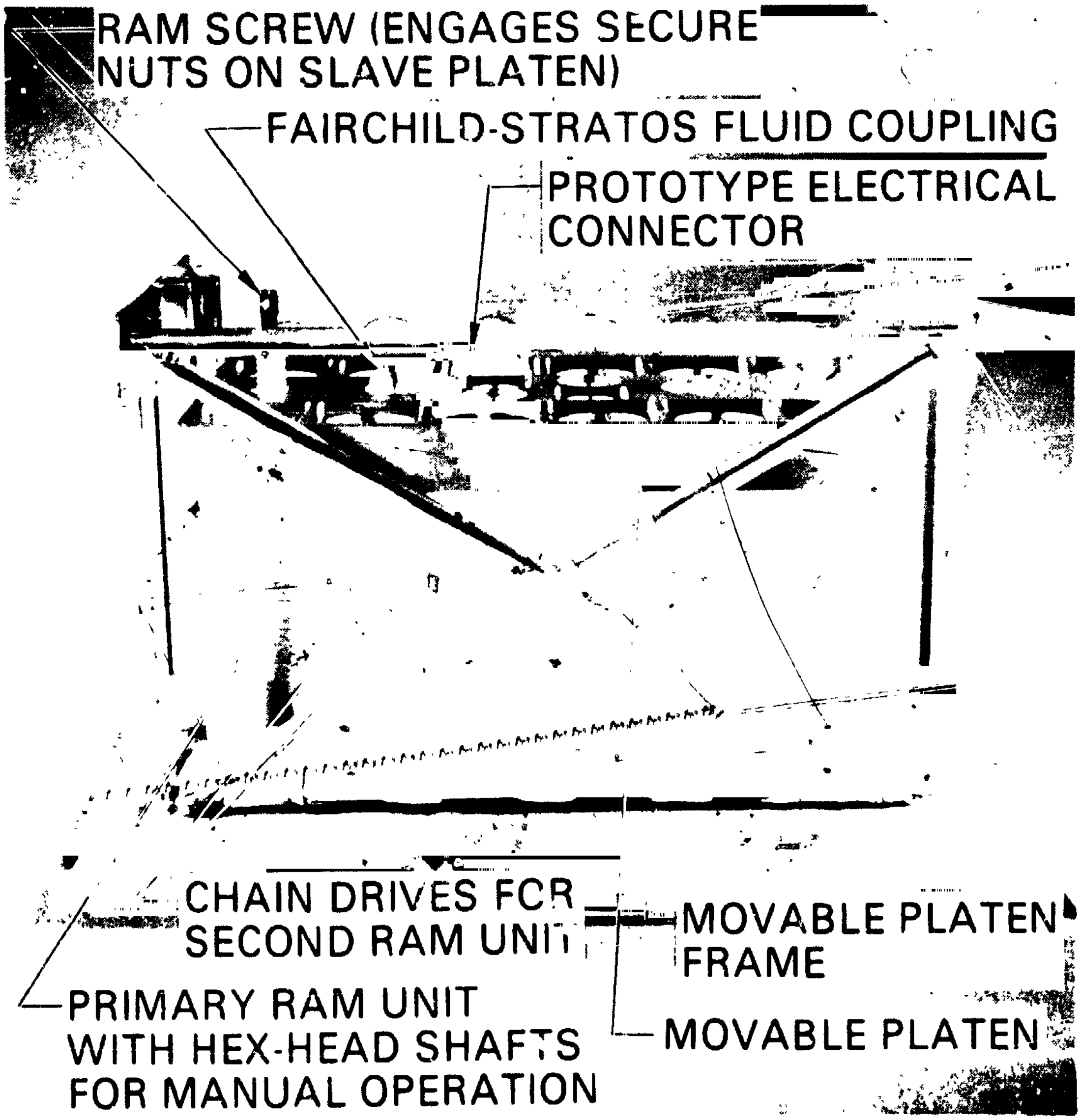
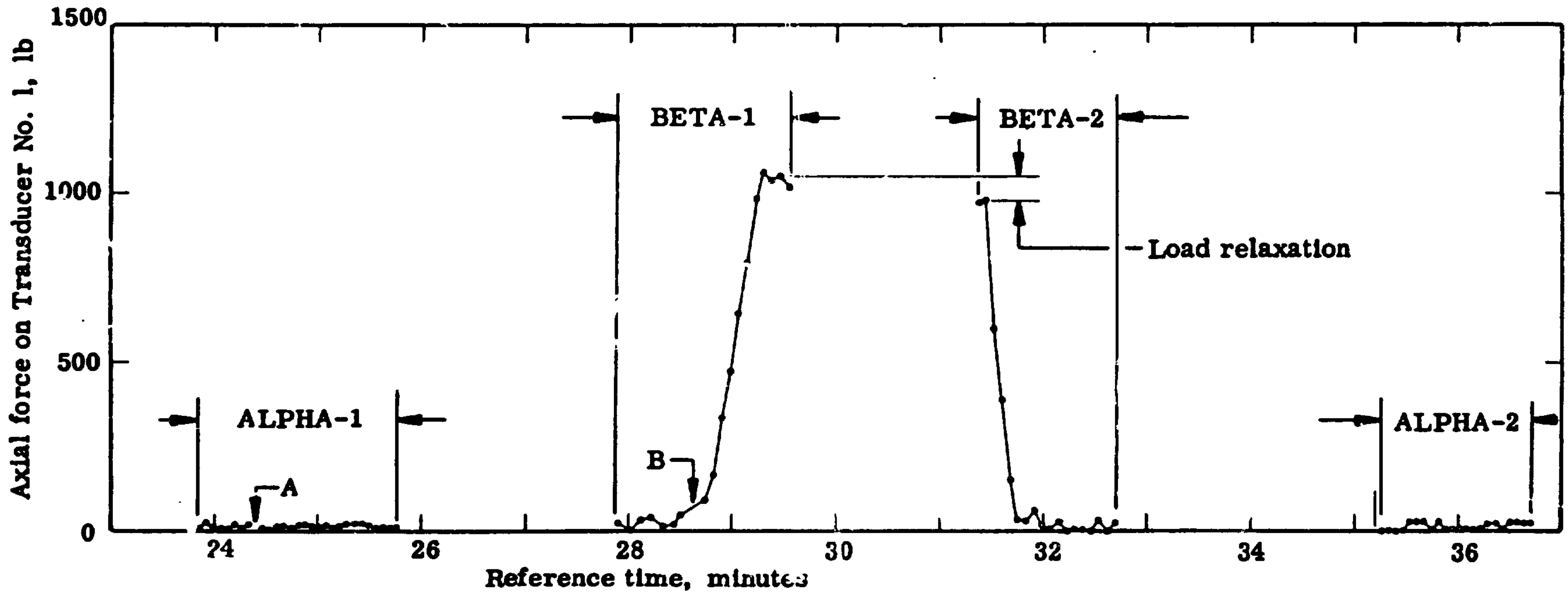


Test Unit as Initially Manufactured

FIG. 26



Note: a. A = time at which screw makes first contact with securing nut on fixed platen.
b. B = time at which electrical connectors make first contact.
c. Test sections as follows:
ALPHA-1: advance ball screws into engagement with securing nuts on fixed platen.
BETA-1: advance movable platen, engage electrical connectors, compress elastomers.
BETA-2: disengage electrical connectors and retract movable platen.
ALPHA-2: disengage ball screws from securing nuts and retract ball screws.
d. Axial force on transducer No. 2 was within 80 lb of force on transducer No. 1 at all times.

TRANSDUCER No. 1 FORCE VS. TIME FOR TEST No. 1 OF TEST SEQUENCE No. 4

FIG. 27

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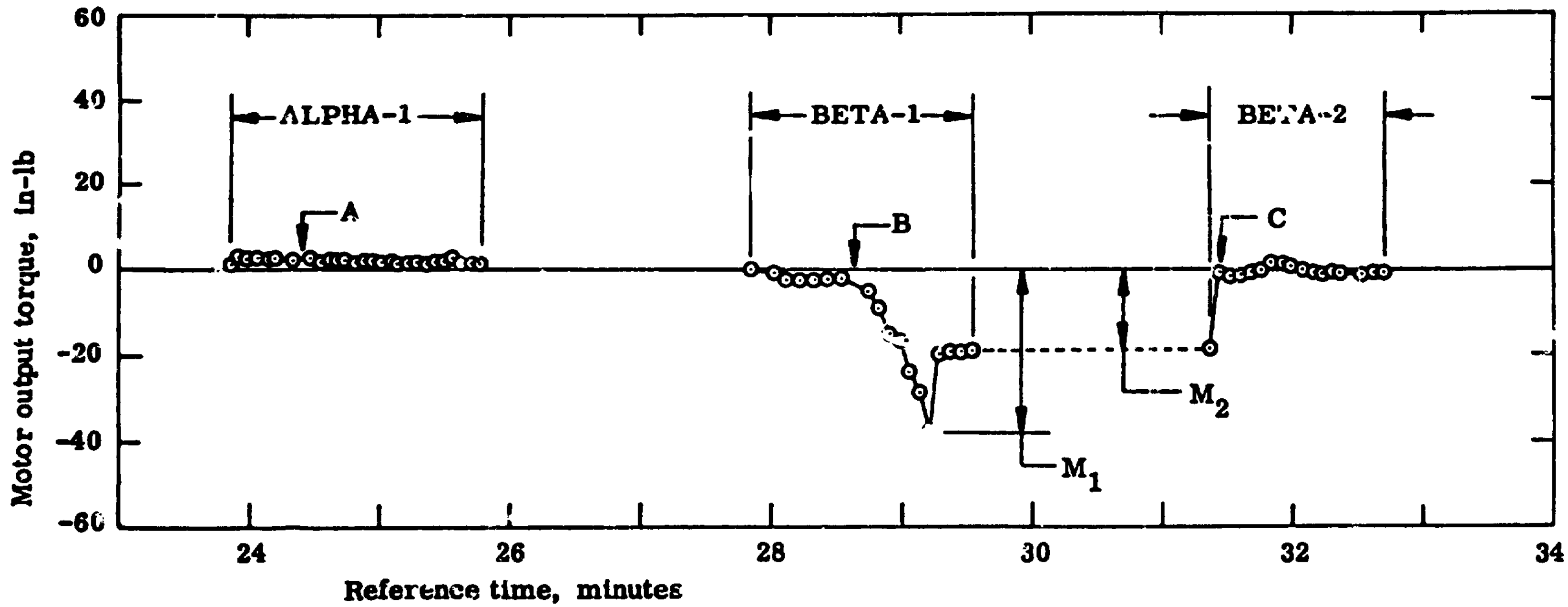

Notes: a. A = time at which ball screw made first contact with securing nut on fixed platen
b. B = time at which all electrical connectors had made initial contact.
c. C = time at which motor turned on (reversed).
d. M_1 = torque required to drive electrical connectors to full engagement simultaneously with full compression of elastomer fluid force simulators.
e. M_2 = relaxation torque maintained by motor brake.
f. Test sections as follows:

ALPHA-1:	Advancement of ball screws and engagement with securing nuts.
BETA-1:	Advancement of movable platen, engagement of electrical connectors.
BETA-2:	Disengagement of electrical connectors, retraction of movable platen.

MOTOR TORQUE VS. TIME FOR TEST No. 1 OF TEST SEQUENCE No. 4

FIG 28

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SPACE OPERATIONS CENTER APPLICATIONS OF SATELLITE SERVICE EQUIPMENT

By

Ronald W. McCaffrey

GRUMMAN AEROSPACE CORPORATION
BETHPAGE, N.Y.

For

SATELLITE SERVICES WORKSHOP

JUNE 22-24, 1982
NASA JOHNSON SPACE CENTER
HOUSTON, TEXAS

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SPACE OPERATIONS CENTER APPLICATIONS OF

SATELLITE SERVICE EQUIPMENT

Ronald W. McCaffrey
Grumman Aerospace Corporation, Bethpage, N. Y.

ABSTRACT

This paper discusses satellite servicing requirements for a continuously manned Space Operations Center (SOC). Applications for Orbiter developed service equipment are described, together with representative satellite servicing operations for use on SOC. This paper is based on recent work performed for NASA/JSC under contract to Boeing, on the SOC System Analysis Extension Study (P.O. K63591), and on related work performed under Grumman IR&D.

INTRODUCTION

Recent studies on satellite servicing near the Orbiter (Ref. 1 and 2) have identified a broad range of services which could be made available to the satellite user community. These services cover the full mission cycle from orbital deployment to on-orbit maintenance/repair and, eventually, removal from orbit. An orbiting base, such as the SOC, will be able to provide many of the same services at less cost than the Space Shuttle transportation system. The SOC will add a new dimension to these services since they can be decoupled from Shuttle launch delays, performance constraints and Orbiter availability limitations.

SOC SATELLITE SERVICE REQUIREMENTS

The SOC concept provides a spaceport for extending satellite services beyond its 400 km altitude, 28.5° inclined orbit. Several modes for using SOC to service satellites are illustrated in Fig. 1. The SOC is used as a transportation mode for the domination of orbits which includes: assembly and deployment of satellites; on-orbit support of attached and retrieval payloads; and as a base for in-situ servicing of remote satellites in LEO and GEO. Since the SOC is decoupled from ground launch constraints, it can provide on-demand service to examine and repair satellite random failure situations. The probability of random failure prior to end of mission or scheduled maintenance for observatory class satellites could be as high as 20%. The SOC can also support the build up of large systems in orbit, such as an IR interferometer in LEO, a cosmic coherent optical system for GEO or, perhaps, a new large interplanetary spacecraft.

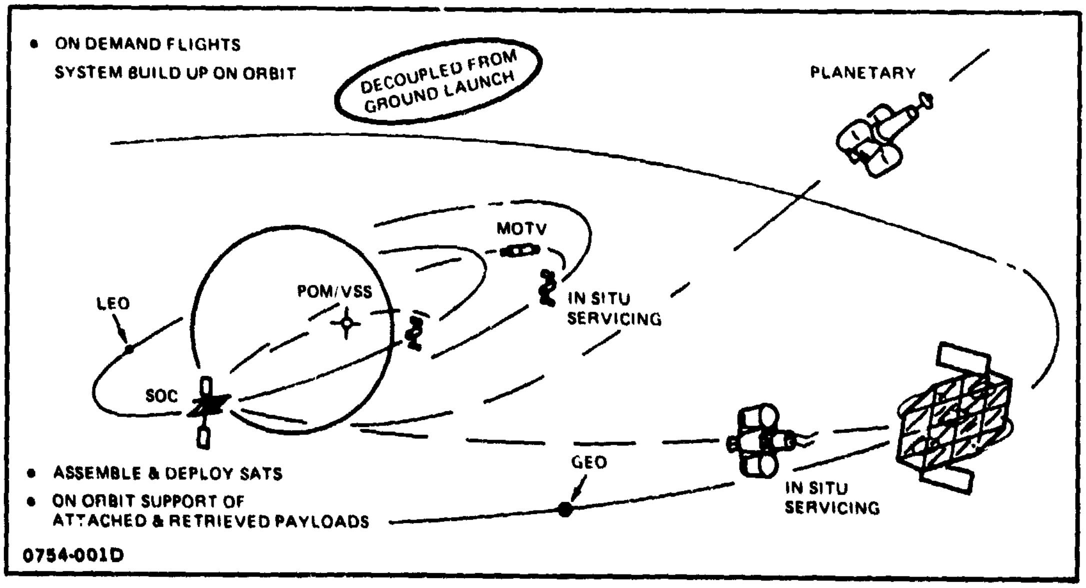

Fig. 1 Satellite Service Modes

Servicing missions for the SOC includes those satellites which are in orbit and require periodic tending for continued operations, as well as those satellites which are ready for initial launch into orbit. Tended satellites encompass attached payloads, co-orbiting, and remote accessible satellites. Co-orbiting satellites include those which station keep with SOC, those that are initially in the same orbital plane and similar altitude (within ≈100 km) and those that are transferred to SOC by a propulsion stage. Remote accessible satellites covers those too distant or impractical to return to SOC, but which are accessible for servicing in-situ from a manned/remote teleoperated service stage.

The types of service operations that can be performed on SOC are shown in Fig. 2 and are keyed to the respective missions. Many of the co-orbiting satellite services are the same as those required for attached payloads. Much of the equipment required to perform these service operations have been previously identified in satellite service studies and some are already under development. While most of these service operations can be performed with the Shuttle Orbiter, the SOC can also offer other services. These services include on-orbit assembly of large systems, mating of large upper stages and the option for on-orbit storage of satellite hardware if pre-deployment test and checkout fails.

	TEND			SAT. LAUNCH	
SERVICE OPERATIONS	ATTACHED PAYLOADS	CO-ORBITING SATELLITES	REMOTE ACCESSIBLE SATELLITES	LOW ENERGY ORBIT	HIGH ENERGY ORBIT
EXAMINATION	•	•	•		
RETRIEVAL		•			
MAINTENANCE/REPAIR	•	•	•		
RESUPPLY	•	•	•		
RE CONFIGURATION	•	•	•		
ON-ORBIT ASSEMBLY				•	•
MATE UPPER STAGES					•
TEST & CHECKOUT	•	•	•	•	•
ON-ORBIT STORAGE		•		•	•
DEPLOY		•		•	•

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Fig. 2 SOC Satellite Service Missions

CO-ORBITING SATELLITE RETRIEVAL METHODS

Three strategies for retrieving co-orbiting satellites, for maintenance/resupply/reconfiguration at SOC, are shown in Fig. 3. The requirements imposed on SOC will vary in accordance with the proximity or relative position of each co-orbiting satellite to SOC and the satellite's orbit adjust capabilities. In the first retrieval scenario, the satellite is shown to be in the same orbit (altitude and inclination) where it keeps station with SOC. In this situation, the satellite could either be a free flyer, which can be controlled by SOC, or any satellite which operates under ground control. When free flying vehicles return to berth or to operate in close proximity to SOC, they will be controlled by the SOC. For on-orbit safety, ground controlled satellites would not be flown in the final docking maneuver to SOC. Nor is it practical to maneuver the SOC toward the satellite for terminal acquisition. Instead, final satellite retrieval is accomplished by a proximity operations module (POM) which can be readily deployed and controlled from the SOC.

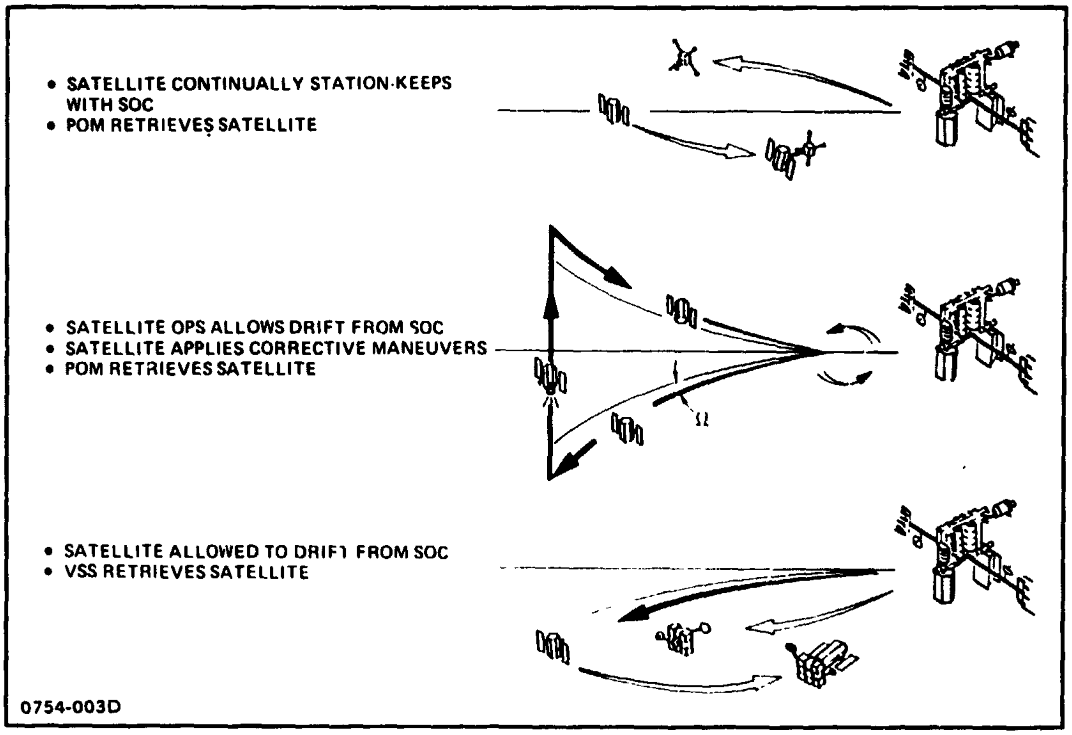

Fig. 3 Strategies for Retrieving Co-Orbiting Satellites

Many satellites will not actively station keep with SOC but will be allowed to decay in altitude and drift out-of-plane. If the satellite has an orbital maneuvering system, as shown in the second scenario, it could be used to adjust its altitude so that it will drift back toward SOC when it is time for maintenance. A SOC controlled POM can then retrieve these satellites as before. On the other hand if the satellite does not have an orbit adjust capability, it will continue to drift out-of-plane from SOC, as shown in the third scenario. The latter satellite must be retrieved by a more capable vehicle, such as a versatile service stage, which can rendezvous, dock and transport the satellite back to SOC. Figures 4 and 5 illustrate alternative vehicles which can be used to retrieve co-orbiting satellites from SOC.

Figure 4 shows two different POM concepts, manned and unmanned. Both concepts use an extendible mast with an RMS end effector to snare the satellite. The manned POM is an adaptation of the Manned Maneuvering Unit/Work Restraint Unit (MMU/WRU) which was developed by Grumman for Orbiter tile repair mission requirements. This is a limited range vehicle, which operates within 300-400 M of SOC. The MMU/WRU-POM would be able to tow 2500 kg satellites (similar to Solar Maximum Mission) to within reach of the SOC manipulator. The unmanned POM has similar satellite retrieval capabilities and would be able to operate about 1 km away from SOC. This unmanned POM vehicle is an adaptation of the Maneuvering Television Vehicle (MTV) being developed at NASA/JSC. The MTV/POM concept shown in Fig. 4, has been modified with an extendible grappling mast and more control authority for its non-contaminating cold gas propulsion system. This vehicle would operate via remote command and control from the SOC crew station.

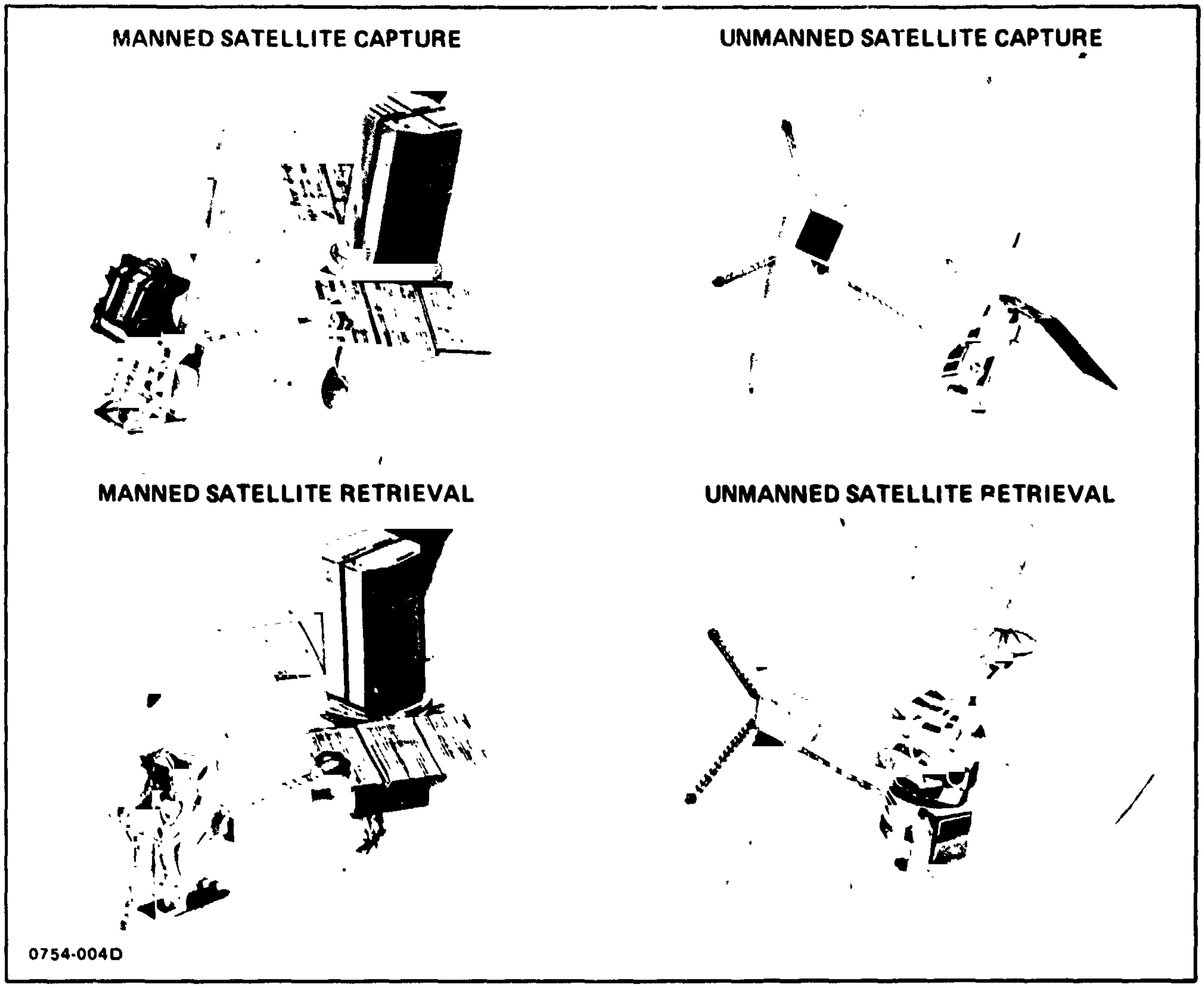

Fig. 4 Alternate Proximity Operations Equipment

Figure 5 shows the versatile service stage (VSS) which is needed for the transfer and return of satellites to and from higher energy low earth orbits. The VSS is designed to operate with several front-end attachments to satisfy a wide assortment of mission needs. Included are, a snare end effector on an extendible mast for grappling satellites rotating at higher rates than that permissible for docking, a docking/berthing system for attaching to compatible spacecraft, and manipulators that provide berthing capability to uncooperative or tumbling satellites and debris. VSS is equipped with a high performance propulsion system for performing large delta-V maneuvers and a clean-firing cold gas propulsion system for satellite and SOC close proximity operations. An on-orbit refueling capability is also provided. The VSS is also equipped with TV systems for satellite examination.

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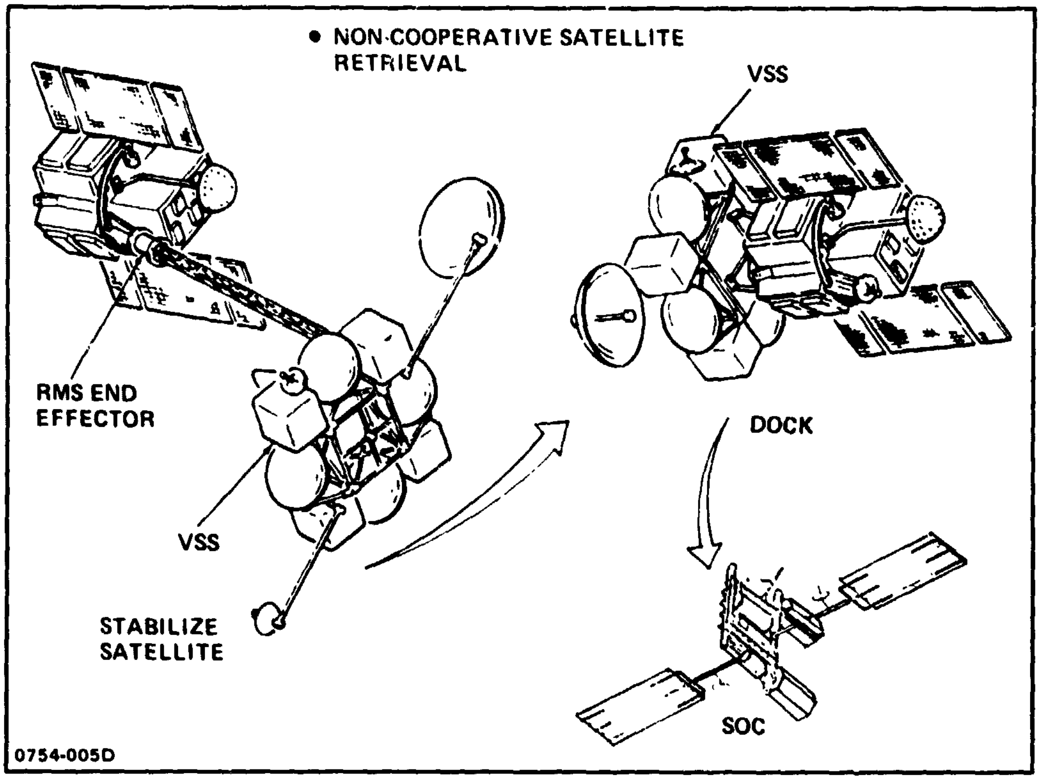

Fig. 5 Versatile Service Stage (VSS)

MANNED MANEUVERING SERVICING VEHICLE

An all propulsive Manned Orbital Transfer Vehicle (MOTV) is illustrated in Fig. 6. This stage and a half design, which is configured for servicing MMS type satellites in geosynchronous orbit (Ref. 3), can also be used to service low earth orbiting satellites which are remote, but yet accessible from SOC. The core stage in this vehicle has a total propellant capacity of 17,500 kg. Liquid hydrogen and liquid oxygen propellants are carried within the forward and aft tanks of this core stage, respectively. Vehicle thrust is provided by two RL10 CAT IIB engines. If needed for a particular mission, additional propellant is carried in external drop tanks attached to the side of the core. The drop tanks are sized for delivery to LEO in the Orbiter payload bay. As many as four drop tanks can be carried and thus provide 109,080 kg of added propellant. These tanks are usually jettisoned from the stage when empty and then de-orbited for direct reentry and burn up in the atmosphere. If desired the tanks could also be designed for LEO return, recovery and reuse. A 2-man crew capsule, equipped with dexterous manipulators and a satellite grappler, is mounted forward of the core. The grappler positions the satellite to enhance operator viewing and to facilitate manipulator servicing operations. These manipulators are bi-lateral force reflecting type and are operated by a master/slave system within the crew capsule.

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Fig. 6 Manned Orbital Transfer Vehicle

SOC LEO SERVICING REGIONS

Whenever practical all co-orbiting satellites in need of maintenance/resupply should be returned to the SOC for that purpose. Out sized platforms of comparable size to SOC should, of course, be serviced in-situ. Cost effective satellite servicing regions in LEO are shown in Fig. 7 for SOC based vehicles. The region identified for service at SOC versus service in-situ is bounded by MOTV core stage capabilities for half range and maximum range payload retrieval performance, when limited to one STS propellant delivery flight. For example, the MOTV half-range retrieval capability defines the maximum plane change maneuver for bringing a satellite back to SOC for servicing and then returning the satellite to its original orbit. Satellites beyond the MOTV half range capability can also be returned to SOC for servicing, if needed. However, it would be more economical if they were serviced in-situ. As shown in the figure, an MOTV can provide in-situ service to an MMS class satellite in a 185-km higher orbit and which is almost 20° out of plane with respect to the SOC. The maximum payload retrieve range of the VSS is also shown for comparison.

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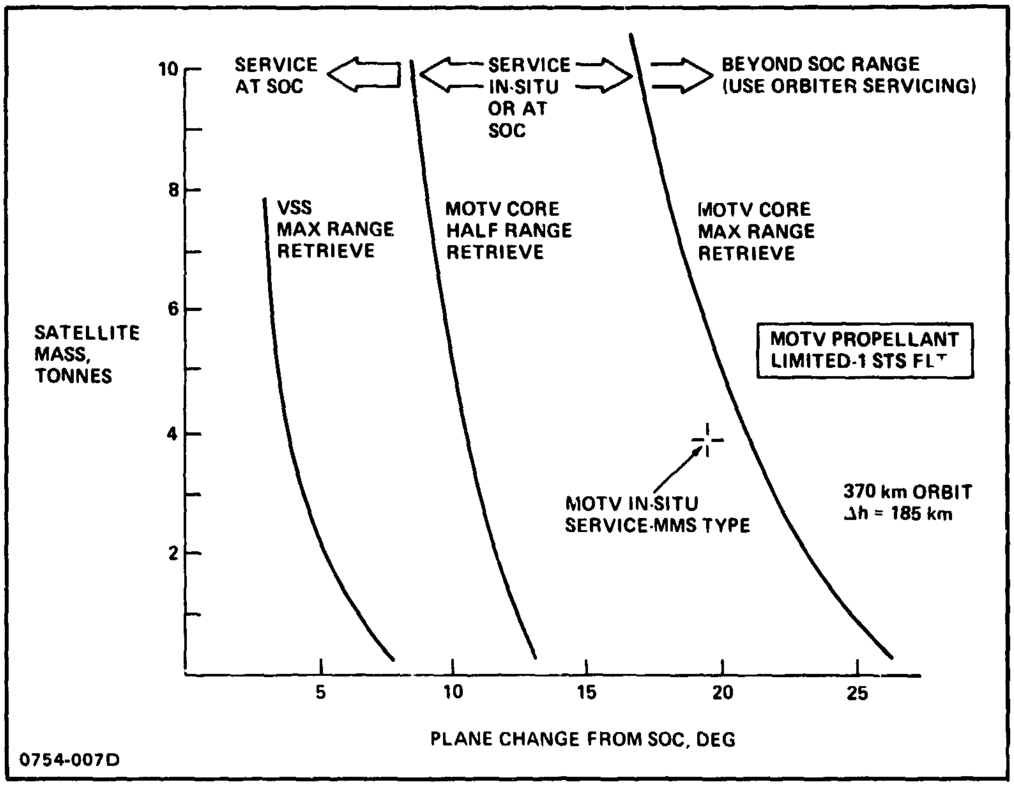

Fig. 7 LEO Satellite Servicing Regions for SOC Based Vehicles

SOC SATELLITE SERVICING FACILITIES

Boeing's operational SOC configuration was used as the baseline for satellite servicing operations. The tracks running around three sides of the two habitat modules are part of the basic configuration, as are the service modules and transfer tunnel/docking module shown in Fig. 8.

For satellite servicing operations, a 7.5 m extension pier is added to one arm of the SOC track system in the direction outboard of the docking module. A handling and positioning aid (HPA) is mounted on a truss structure at the tip of the pier. An end effector, suitable for the particular mission, is attached to the tip of the HPA. An EVA crewman can operate from an open cherry picker (OCP) mounted on a track running along the HPA arm. A mobile platform runs around the existing track system, as well as along the extension pier, to locate a twin manipulator system where it is required for the particular service mission. These manipulators are based on the Shuttle remote manipulator system (RMS) and one of them mounts an OCP at its tip, while the other mounts a standard snare end effector. The EVA crewman on the OCP controls both manipulator arms and the HPA, each in selective sequence. These facilities can also be controlled from a station in the SOC habitation module.

Unless self-propelled, free-flying satellites must be brought to SOC by a propulsion stage. It is necessary to service and refuel these propulsion stages. OTV/MOTV have their own service hangar but smaller propulsion stages, such as VSS and POM, are used for close-in retrieval. These require another facility which is located on the "underside" of the extension pier, as illustrated. A second HPA is mounted on a truss structure to handle VSS and POM. An OCP mounts to a track on the HPA arm and hold an EVA crewman who controls the HPA and thus, the servicing and refueling operations.

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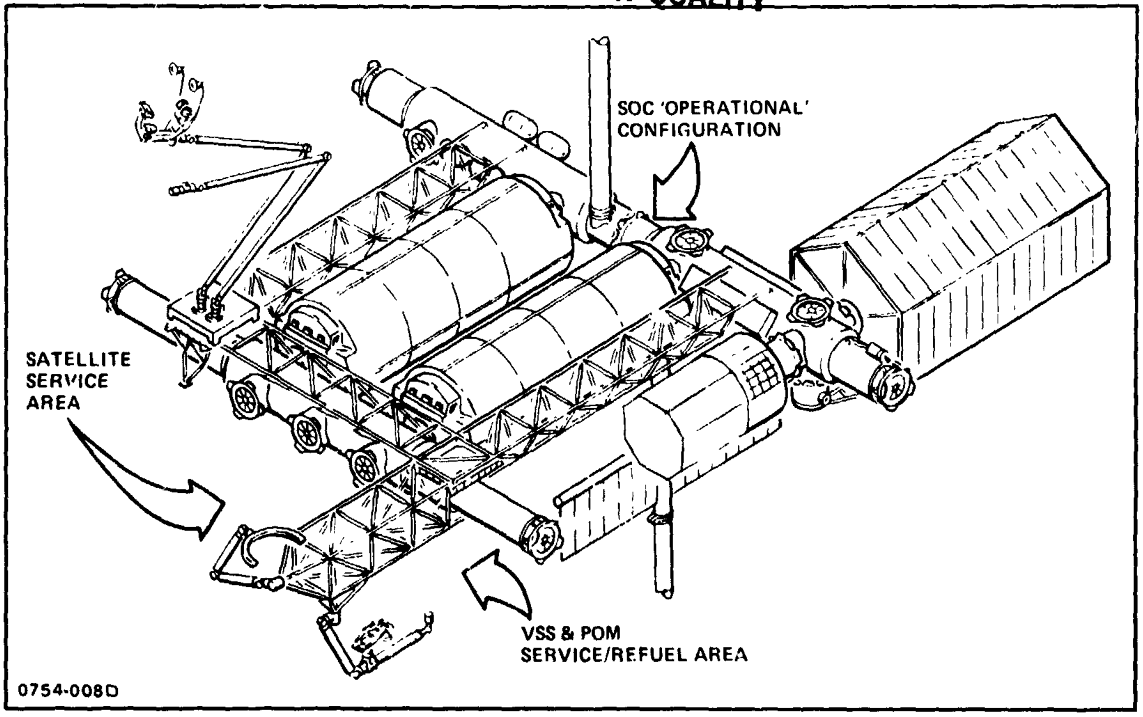

Fig. 8 Operational SOC with Satellite Servicing Facilities

REPRESENTATIVE SERVICE OPERATIONS

Servicing requirements were analyzed for scheduled maintenance of a co-orbiting Advanced X-ray Astrophysics Facility (AXAF) and the launch of a large GEO Communications Platform. Operational techniques were defined for both missions, together with timelines equipment, and required crew support for servicing these satellites from either SOC or the Orbiter.

The AXAF normally operates in a 28.5° orbit at 450 Km altitude or 50 Km above SOC. Since this satellite has no propulsion system of its own, a SOC based tug, similar to the VSS, is used to rendezvous with AXAF and return it to SOC as shown in Fig. 9. After servicing and checkout the VSS returns the AXAF to its operational orbit.

The satellite servicing facility on SOC is shown in the foreground of Fig. 10. In this rendering, a 11.5 m long AXAF satellite is berthed, on its end, to a handling and positioning aid (HPA), which is mounted at the tip of the servicing pier. The HPA can position the AXAF as shown, or it can swing it 90° so that it is parallel with the SOC transfer tunnel/docking module, depending upon accessibility requirements. The AXAF is able to be serviced at two different locations. AXAF subsystems, for example, are undergoing maintenance and repair by an EVA crewman, who is positioned on the OCP mounted to the mobile platform manipulator. Another EVA crewman is shown positioned on the HPA-OCP to reconfigure or repair AXAF instruments. A second HPA is also mounted on the underside of the servicing pier to support the resupply and maintenance of POM and VSS. The VSS is portrayed as being serviced on SOC in the lower foreground while the POM is depicted, in the upper background, towing a satellite to SOC.

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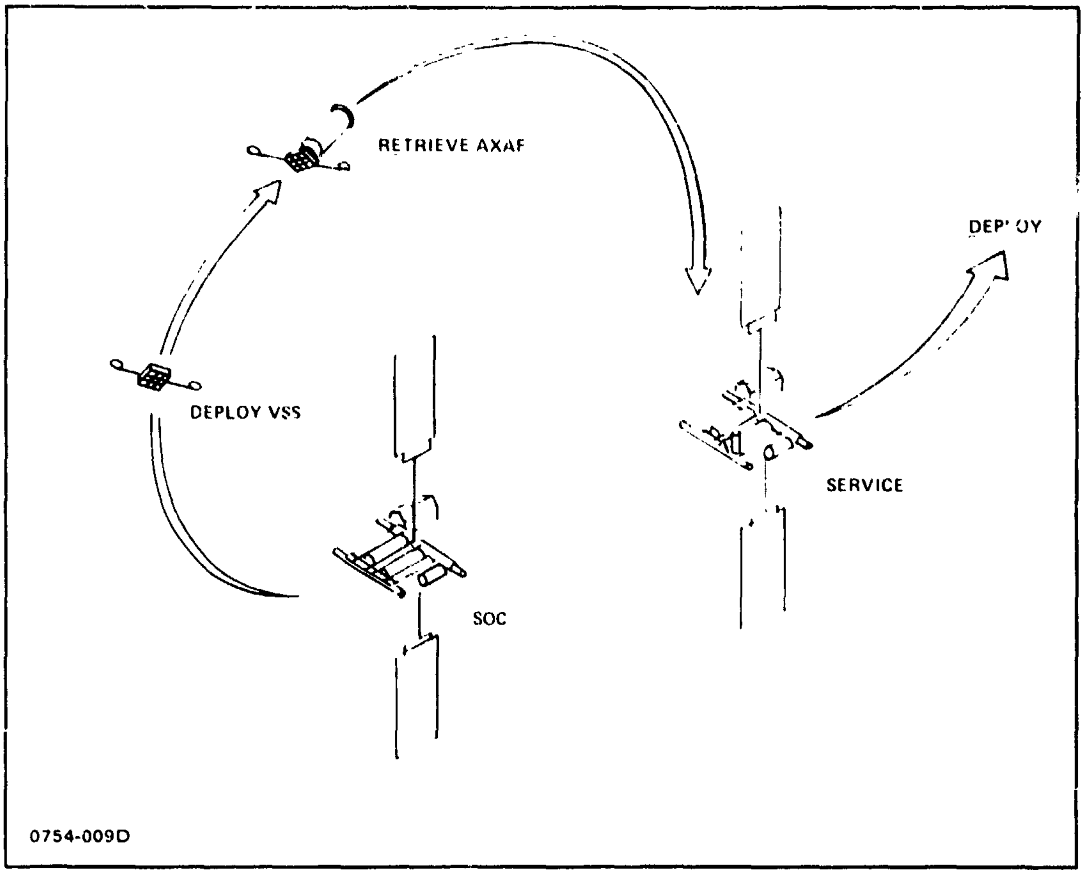


Fig. 9 AXAF Service Mission Scenario

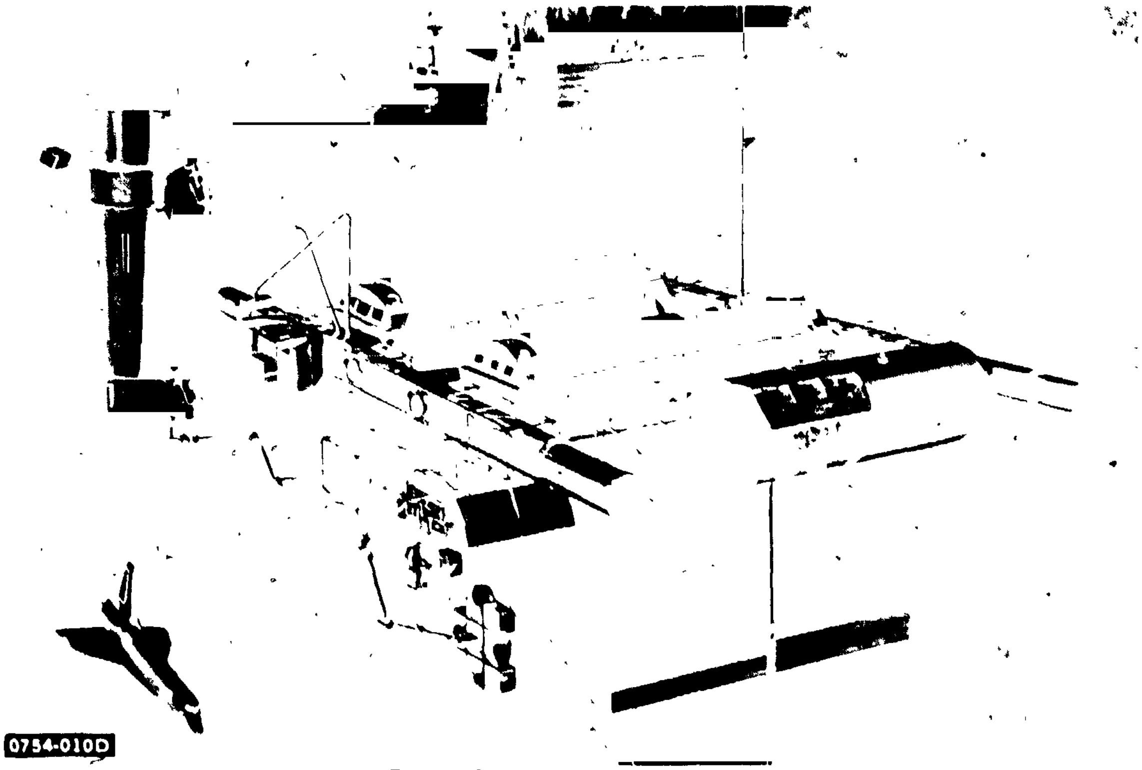


Fig. 10 Servicing Operations at SOC

Satellite maintenance operations will, of course, be tailored to the needs of each satellite. Scheduled servicing of observatory class satellites like AXAF could, for example, entail changing scientific instruments, replacing degraded components and resupplying consumables. As shown in Fig. 11, similar services can be performed on other satellites provided the critical components are serviceable and accessible to the suited astronaut.

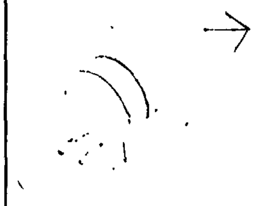
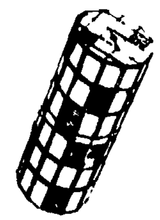
	SCHEDULED SUPPORT	POTENTIAL SUPPORT	
TYPICAL SATELLITES	 AXAF	 LDEF	 LAMAR
ON-ORBIT SERVICE			
• RECONFIG/REPLACE	SCI INSTR	EXPMT TRAY	
• MAINTAIN/REPAIR	S/S ORUS GYRO SENSOR ANTENNAS RATE GYRO STAR TRACKERS SOLAR ARRAY		COMM/DATA MOD. ATT CTL MOD POWER MOD PROPULSION MOD ANTENNAS SOLAR ARRAY
• RESUPPLY	DETECTOR GASES CRYOGENS ?		PROPELLANTS
ON ORBIT REVISIT	2 - 3 YEARS	~ 1/YR	~ 2 YR (EXTEND OPS) OR AS NEEDED
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Fig. 11 On Orbit Satellite Maintenance Operations

Launch of a 60 m diameter GEO Communications Platform begins with delivery of the folded platform to SOC, as shown in Fig. 12. A dedicated Orbiter flight is needed since the folded platform completely fills the payload bay. The platform is transferred to SOC where it is supported by an HPA during unfolding operations. After check out, the GEO Platform is mated to a SOC based orbital transfer vehicle (OTV), and then launched to GEO.

Fig. 13 illustrates the use of SOC service facilities to support the deployment and launch of large complex spacecraft, such as a GEO communications platform. SOC crewmen would monitor the deployment of folded appendages, release hangups if needed, support platform operations ground control during system checkout and calibration, checkout OTV, control upper stage mating, verify system interfaces, control vehicle separation and monitor launch to GEO. While these tasks could also be performed by the Orbiter, the SOC is not constrained by Orbiter mission duration limits. Hence, the SOC offers greater flexibility to deal with satellite deployment situations which require extended calibration operations for the payload operations control center or other system activation contingencies that might arise.

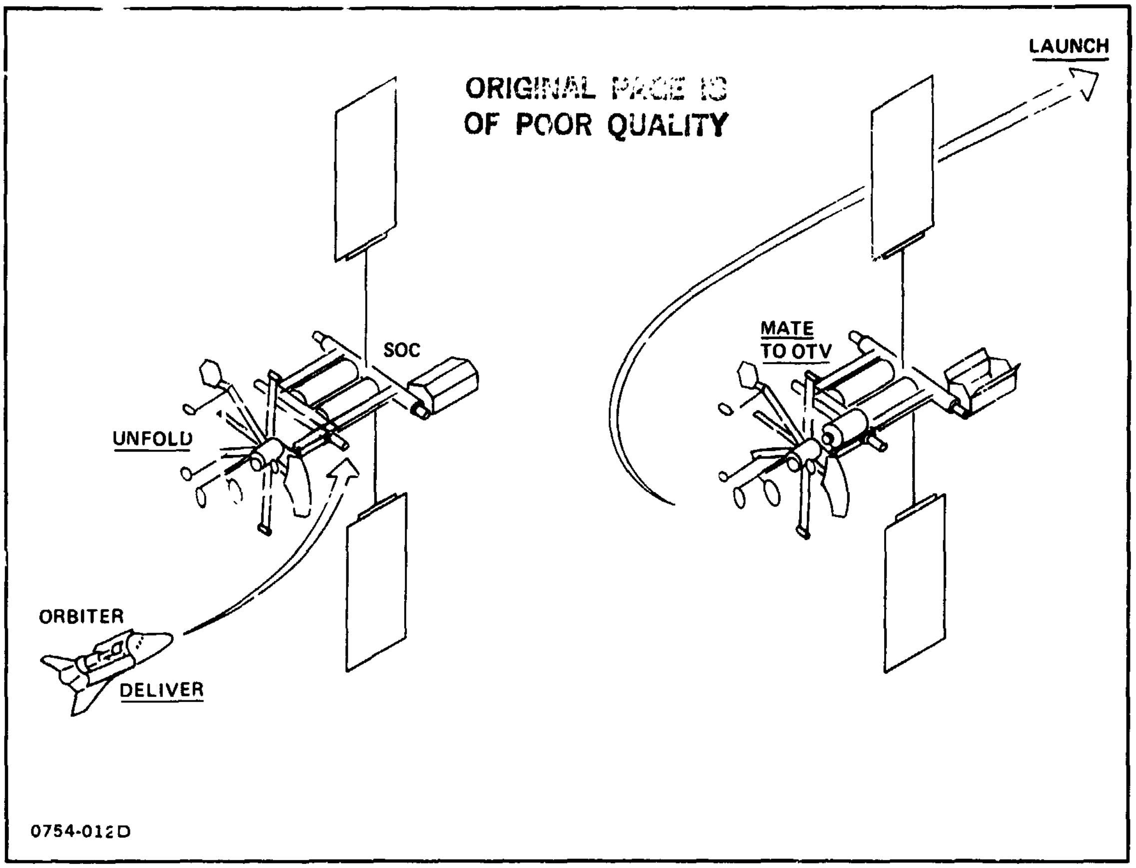

Fig. 12 GEO Communications Platform Launched by SOC

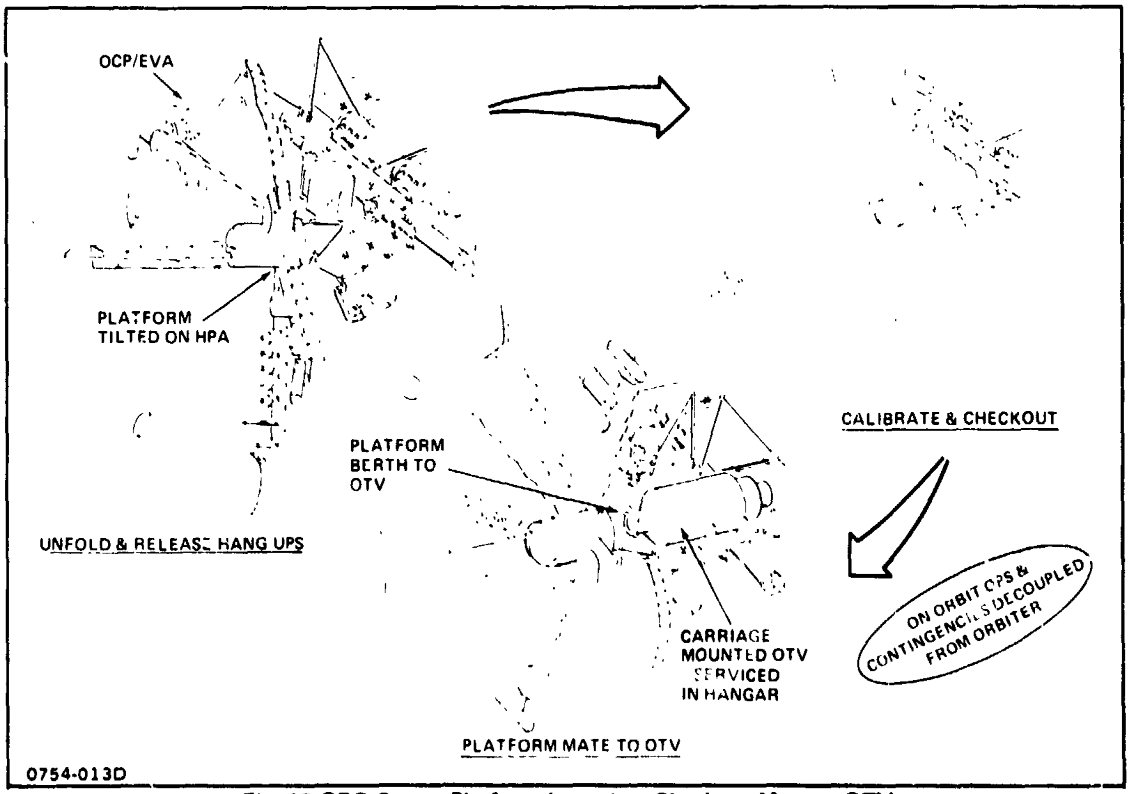

Fig. 13 GEO Comm Platform Launch – Checkout Mate to OTV

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Servicing satellites in orbit is more cost effective for space based systems than ground based systems. Representative mission service costs for the Orbiter and SOC are compared in Fig. 14. These costs include charges for using the satellite service equipment, miscellaneous fees for ground handling support and Orbiter crew operations, and associated costs for transporting service equipment hardware and consumables to orbit. The largest part of the Orbiter costs for the AXAF revisit results from the charge for transporting servicing equipment to and from orbit (i.e. POM, HPA, OCP etc.). If an OMS kit is needed to reach the satellite, the cost will increase further. The major cost driver associated with deployment of the GEO communications platform by the Orbiter results from two STS flights. The first flight delivers the folded platform to orbit and the second flight brings the OTV for on-orbit mating and launch to GEO. In contrast, operations from the SOC assume a space based OTV that is refueled by external tank propellants scavenged during routine Orbiter missions. If propellant scavenging technology was not developed, then another flight would be required to carry the OTV fuel in the Orbiter cargo bay. However, the added transportation costs associated with refueling the OTV would still be less than the cost for carrying the fully loaded stage.

With advanced mission planning and early provisioning of satellite replaceable items and supplies onboard, the SOC should be immune to STS launch delays and vehicle availability problems. By operating in a 28.5° orbit, the SOC will be able to service 50% of the satellites in LEO, a launch all GEO and planetary spacecraft, and support satellite service missions to GEO.

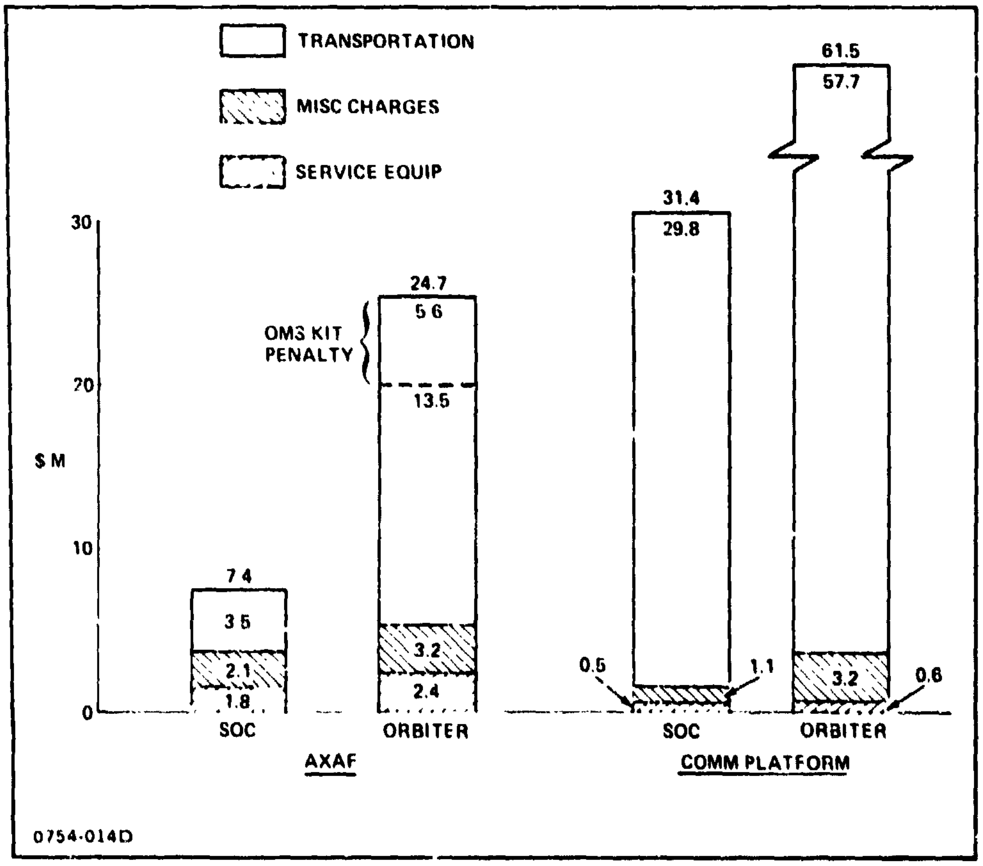

Fig. 14 Representative Mission Service Costs (1981 Constant $)

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REMOTE ORBITAL SERVICING SYSTEM CONCEPT

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NASA Johnson Space Center
June 22-24, 1982

REMOTE ORBITAL SERVICING SYSTEM CONCEPT

Alfred J. Meintel, Jr., Langley Research Center
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SUMMARY

Increased application of automation technology has been identified as necessary for NASA to carry out its missions within the constraints of future funding and available physical resources. In particular, the application of emerging technology in manipulators and machine intelligence will allow the development of robotic devices remotely commanded by human operators to increase man's productivity in space. Under the automation technology research program at Langley Research Center, Martin-Marietta Aerospace has developed a concept for a Remote Orbital Servicing System (ROSS) based on present teleoperator technology. The specific objective was to conceptualize a single servicer design that would be compatible with three specified spacecraft, be capable of performing service to the same extent as manned extravehicular activity, be controlled from a ground control station, and use currently available technology. This paper summarizes the ROSS concept, which has become the focus for the agency research and technology program in teleoperators and robotics.

INTRODUCTION

Currently, space operations such as inspection, servicing, repair, and assembly require close proximity to the Shuttle, using the Remote Manipulator System (RMS) and manned extravehicular activity (EVA). EVA requires significant costs in training and equipment, is strenuous and time consuming, and has inherent risk. In addition, the space operations are restricted to Shuttle orbital capabilities. For satellites outside Shuttle orbital capabilities an orbital transfer vehicle would be required to retrieve the satellite for maintenance or service and then return it to the desired orbit. A remote operations system would significantly enhance the capability to safely and efficiently perform space operations currently and is required for future missions such as space construction and manufacturing and to support an operational space station. Even for operations in or near the Shuttle bay or in close proximity to a permanent space base, the capability to perform operations remotely would increase the operational flexibility and safety.

The development of a totally autonomous robotic system to accomplish the varied tasks envisioned is currently and, for the foreseeable future, beyond the state of the art, particularly for unplanned or unforeseen contingencies. However, by retaining man in the control loop, a remotely controlled (teleoperator) system can be developed with today's technology. With man's flexibility and adaptability, the teleoperator system can significantly increase the capabilities to perform space operations. As robotic technologies mature, they can be integrated to further automate the remote systems, moving man to higher levels of supervisory control.

In the past, a number of NASA centers and contractors have conducted research and conceptual activities addressing remote space operations. Also, extensive research activities are in progress throughout the world, applying advanced technology to develop smart machines for industrial applications. Technological advances in computing systems, coupled with advances in industrial robotics and new programming techniques, are beginning to allow researchers to consider the application of theoretical concepts to the solution of real-time problems. A prime area for the early application of the technology is the development of robotic devices remotely commanded by a human operator to perform space servicing and repair.

For several years the Automation Technology Branch of the Control Theory and Flight Management Division at Langley Research Center has been concerned with the development and application of technology in controls, sensing, man-machine interaction, and computer science (including artificial intelligence), to the problem of remote teleoperator/robotic operation under manned supervision. Expertise has been developed during this period through grants and contracts, meetings and conferences, development and integration of the Intelligent Systems Research Laboratory, and initiation of a real-time teleoperator/robotic systems simulation. Both the laboratory and the simulation will be used extensively in future research. The laboratory is a testbed for computer science research, hardware integration, and enables evaluation of new technology. The simulation will be used to develop and test control algorithms, theoretical models, advanced displays, and to conduct systems integration and analysis. Man will interact with the automated system (physical or simulated) through an intelligent interface control station.

In November 1980, Martin-Marietta Aerospace created a separate organization, the Advanced Automation Technology (AAT) Section, dedicated to the development of the required technology for advanced automation. This activity has been extensively funded through internal research and development funds at Martin and is addressing many areas of advanced automation technology including machine intelligence, robotic systems, and smart sensors. One major activity of this group is the development of an integrated robotic system simulation (ROBSIM) tool for analysis of a broad spectrum of advanced robotic systems.

Currently, Langley is developing the framework for a robotics software simulation. This effort includes both in-house development and a contract with Martin to supply portions of the ROBSIM. In 1981, Langley added a subtask to this contract for the design of a ground control station for a space servicing system. This control station would be the initial man-machine interface to the software simulation and the laboratory. The ground station design task required Martin to examine current technology for remote servicing. The result was a unique, feasible Remote Orbital Servicing System (ROSS) concept. This paper describes the ROSS concept and its implication for future NASA activities.

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REMOTE ORBITAL SERVICING SYSTEM CONCEPT

Objective

The objective of the effort was to design a ground control station to control a remote servicing system. The approach was to review past studies, select one or a mixture of concepts, and further develop the program definition of a Remote Orbital Servicing System (ROSS) supported by the space transportation system (STS). The study guidelines are shown in Table 1.

Table 1

Study Guidelines

- Use State-of-the-Art Technology
- The Servicing System Shall be Capable of Servicing the Space Telescope (ST), Solar Maximum Mission (SMM), and Long Duration Exposure Facility (LDEF).
- Extent of Servicing Shall be the Same as Man in EVA or in the STS Payload Bay
- A GFE Propulsion/Navigation/Power Module Will Provide Transportation and Stabilization of the Servicing System into Close Proximity of the Satellite.
- Control of the Combined Maneuvering System and Attached Servicing Kit Shall be Provided Through Ground Control with Manned Interaction.

The overall emphasis was to develop a feasible remote orbital servicing system controlled from a realistic ground control station using state-of-the-art technology. Three serviceable vehicles representing various maintenance modes of existing spacecraft designs were specified.

The specific objective was to conceptualize a single servicer design that would be compatible with the three specified spacecraft and possibly many more. This required that the selected configuration use current technology and be cost effective in the sense of orbital operations and spacecraft minimum modification impacts. A later modification extended the task to include a cost and schedule analysis of a remote space servicing system concept and a Rough Order-of-Magnitude (ROM) cost.

A major emphasis of this effort was the application of current state-of-the-art technology. No major road blocks were uncovered; however, items requiring additional work have been identified and are discussed later.

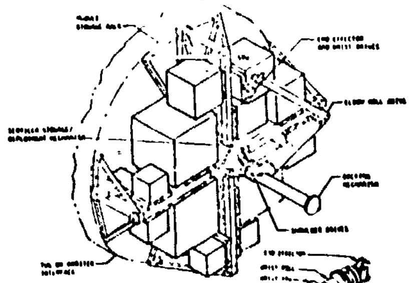
For purposes of this study the government furnished equipment (GFE) propulsion/navigation/power module was represented by the Martin-Marietta version of the Teleoperator Maneuvering System (TMS). However, this servicer kit could be transported and stabilized by any of the various generic carrier, or transport vehicles.

Background

Various conceptual studies on orbital satellite maintenance have been performed over the last decade. Results for three prior efforts that were particularly helpful to this study are illustrated in figure 1.

ROSS Related Efforts

STUDY AREA	LEGACY
STOWAGE RACK	IOSS SERVICER
MANIPULATOR ARMS	PROTO-FLIGHT MANIPULATOR ARM
VIDEO SYSTEM	TELEOPERATOR VISUAL SYSTEM CONCEPTUAL DESIGN STUDY



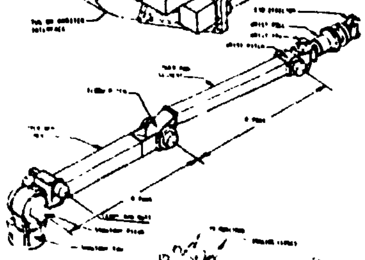


Figure 1

Each of these efforts were performed by Martin under contract to MSFC during the 1970's. Shown are the Integrated Orbital Servicing System (IOSS), the Proto-Flight Manipulator Arm (PFMA) and a stereo visual system. These mockups and early prototype hardware were designed, fabricated and demonstrated to validate subsystems. They increase the confidence for three of the major servicer kit elements required for ROSS.

Spacecraft Mission Models

The three spacecraft chosen as the mission models are shown in figure 2.

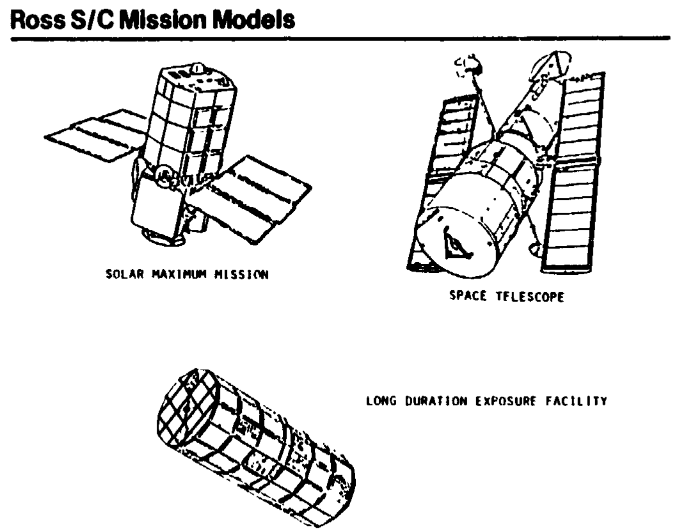


Figure 2

Each of the three is the responsibility of a different NASA center (Solar Maximum Mission (SMM)-GSFC, Long Duration Exposure Facility (LDEF)-LaRC, Space Telescope (ST)-MSFC, and each reflects a different aspect of servicing requirements. Combined, the models thus represent an overall broader range for servicing activities than that afforded by any single spacecraft. The combination of the three models expands the servicing requirements of ROSS. At the same time, the number of modules, the sizes, the weights and the attachment mechanisms impact the ROSS design phase.

The SMM spacecraft is expected to be serviced by the orbiter/EVA several years prior to ROSS flight availability. The spacecraft is composed of an instrument module mated to the Multimission Modular Spacecraft (MMS). An RMS grapple fixture is attached to the MMS portion; however, the Shuttle RMS can only dock to MMS if the solar panels are jettisoned. In the event the solar panels are left intact, NASA has been developing a special grapple tool to attach to the trunnion pins on the transition adapter. Manned servicing plans include the primary objective of replacing the MMS attitude control system module and, secondarily, three repair activities to eliminate anomalies on the instrument module. For ROSS study purposes, the servicing requirements were restricted to the replacement of any of the three standard MMS modules, since the MMS is representative of a generic-type vehicle intended for a variety of missions. The MSS is thus typical of a small spacecraft which has been designed for on-orbit serviceability and would be a prime candidate for future servicing using ROSS.

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The LDEF is representative of a large structure type of spacecraft where the servicing activity encompasses a large number of mechanical, repetitive operations. LDEF is designed for ground rather than on-orbit servicing. Modules are externally accessible and may be mounted anywhere along the periphery and ends of the LDEF structure. Each modular tray contains 60 miscellaneous pieces of noncaptive retention hardware. The precision and dexterity needed to attach one tray would be difficult for a man, and a remote system would be taxed to accomplish the same task on-orbit. Modifications to the retention hardware that would not change the integrity of the system could be utilized to make on-orbit servicing feasible. Such a modification could minimize the number of parts and facilitate servicing with mechanical advantages.

ST is typical of the large observatory type of spacecraft. Inclusion of ST in the mission model allows an assessment of the mechanical attachment and electrical connections for a much wider variety of assemblies than both SMM and LDEF. It also presents a more stringent set of servicing requirements since the assemblies are located internal to the structure and are mounted in both radial and axial orientations. The space telescope is designed to be serviced by an EVA astronaut. Numerous tether rings, footholds and handholds are abundant through the spacecraft. Servicing data for the EVA astronaut is incomplete; therefore, insufficient data were available for a complete analysis for ROSS. If the servicer mechanism can perform as well as man in EVA, then ST servicing should be within the scope of a remote system such as ROSS.

The differences in the three mission models require ROSS to be versatile. The number of replaceable modules and their sizes define the amount of space needed for storage. The sizes range from a compact space telescope RGE box that is handled effortlessly to the bulky ST axial science instrument which is about the size of a telephone booth (36 x 36 x 87 inches). The RGE is the lightest box at 17 pounds and a special LDEF experiment is the heaviest at 1,000 pounds.

Even more diversified are the attachment mechanisms for each of the replaceable modules. The attachment mechanism impacts the removal scenario and the storing configuration on the racks. The different mechanisms dictate assorted end-effector movements such as roll and translation motions.

The spacecraft program schedules span the calendar years of 1982 to 1987. Solar Max was launched in February of 1980 and has been disabled as of November 1980. The present plan is to repair Solar Max by astronaut in the last quarter of 1983. Space Telescope is scheduled for launch in 1985 with a scheduled in-orbit refurbishment in 1987. Flight of LDEF is planned in the second half of 1984, with retrieval by the space Shuttle a year later. Plans to make ROSS functional by 1987 could facilitate the servicing of these three and other spacecraft encompassed by the serviceability range defined. The remote orbital repair capability can be economical to the space program in the 1980's as more expensive spacecraft are launched and are serviced periodically.

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The orbital inclinations and periods of the three spacecraft are almost identical. The altitudes differ by 115 NMI, but are well within the range of ROSS. It is conceivable to service all three spacecraft on one mission due to the similar inclination angles, if the need should arise.

The three mission model spacecraft chosen for servicing represent a balanced group in terms of serviceability. Balanced in terms of one craft being serviceable by the Shuttle RMS, one by an EVA astronaut and one that is not serviceable in its current configuration. With modifications on LDEF, all three spacecraft would be serviceable by ROSS.

ROSS Configuration

The configuration driving rationale are shown in Table 2.

Table 2

Configuration Driving Rationale

- Manipulators
  - Match EVA Capability -- Anthropomorphic Features
  - Number of Arms -- Two, Distributed Actuators, 6-DOF System
  - Length and Joint Motion -- Function of Kinematic Analysis
- Module Stowage Rack
  - Module Rack Depth -- X-Large ST Module
  - Module Rack Diameter -- Orbiter Bay Diameter
- Viewing/Lighting
  - Mount Stereo Cameras to Simulate a Normal Head Position
  - Provide Camera Positioning Devices for Servicing and Docking
- Docking System
  - Baseline System Provided by Carrier Vehicle with Minimum Modifications Due to Servicing Function
- Carrier Vehicle/Servicer Interface
  - Attach to Forward End of Carrier with MMS Adapter Ring
  - Communications from Servicer Kit Through Carrier Vehicle to Ground Control Station
- Ground Control Station
  - Baseline Communication to Ground via TDRSS
  - Configure Station for Man-in-the-Loop Control

The key elements of this rationale are driven by the spacecraft models, the requirements to match the manned EVA capability, the carrier vehicle, and the ground control station.

The servicer kit configuration is shown in figure 3 and shown mounted to the TMS carrier vehicle in figure 4. The servicer kit would be mounted to the carrier vehicle (TMS in this case) via a standard MMS adapter ring and could be cantilevered from the carrier vehicle. The docking probe is an RMS end effector which telescopes from the carrier vehicle through the servicer kit. The basic structure of the kit is a rack design 14 feet in diameter by 36 inches in depth. The actual configuration of the rack is dependent on the parts to be carried. Although shown open, the final design requires covering for thermal control purposes.

Servicer Kit, Isometric

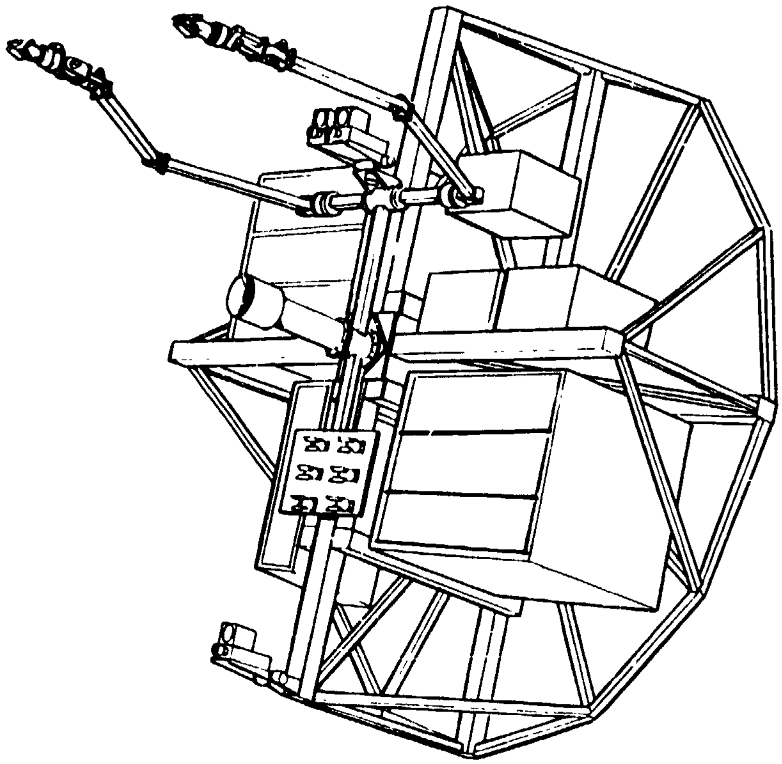

Figure 3

Servicer Kit, Plan View

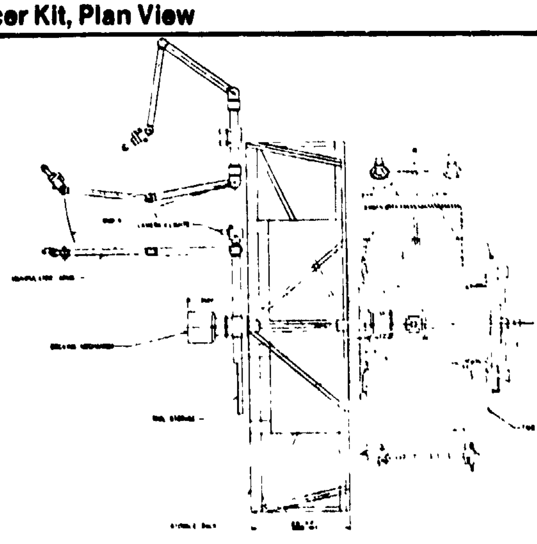

Figure 4

Anthropomorphic considerations were prime in the design of the manipulators and viewing system. An isometric of the dual manipulators is shown in figure 5.

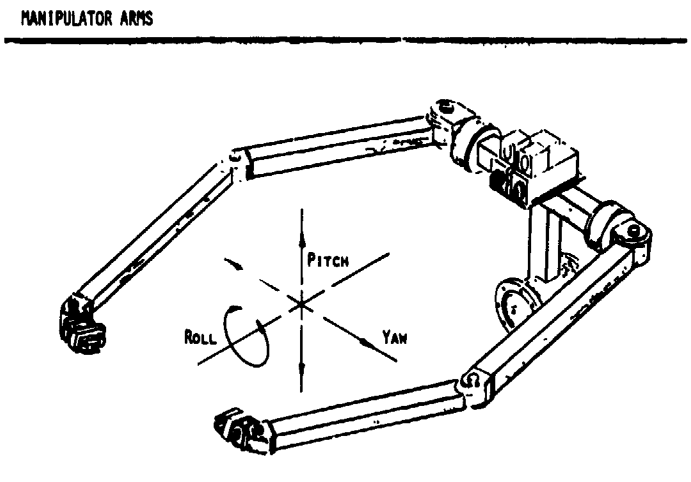

Figure 5

The manipulators, or "arms," are attached to a 360-degree rotating T-section carriage. A stereo camera system, complete with lights and a pan/tilt mechanism, is located at the center of the carriage cross member. The roll, pitch and yaw axes as used herein are defined as shown. The rotating aspect of the carriage significantly increases the kinematic reach of the arms. From the kinematic study, arm segment lengths were established as 4 feet from the shoulder axis to the elbow axis, and also 4 feet from the elbow axis to the first drive axis of the wrist assembly as shown in figure 6.

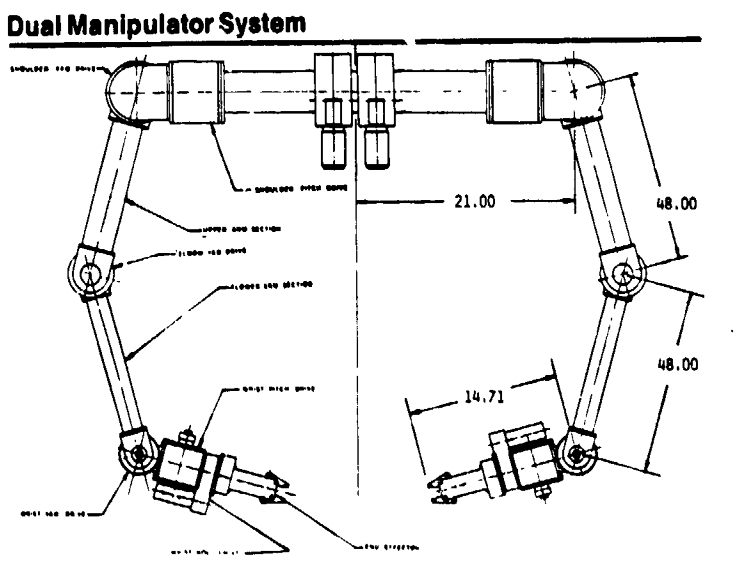

Figure 6

The manipulator arm segments are square aluminum tubing. Both the drives and arm segments are modular, allowing easy changes in lengths, drive sizes, and torques.

The kinematic study dictated the following joint ordering: shoulder pitch and yaw; elbow yaw; and wrist roll, pitch, and yaw. Each joint has a fail-safe brake which is on when power is off and vice versa. Thus, a load or torque may be held with power off. The wrist assembly and the wrist roll interface mechanism are shown in figure 7 and are from a servo manipulator designed for the Oak Ridge National Laboratory (ORNL). The ORNL design is similar in many respects to the current ROSS conceptual design. These drawings are intended to provide an overview of the wrist and details concerning the roll mechanism, exchangeable tool interface, and power take-off capability.

Wrist Roll Interface Mechanism

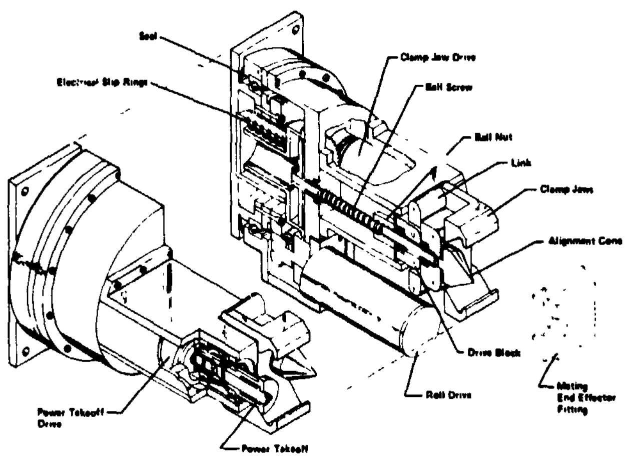

Figure 7

A servicer weight statement is shown in Table 3. The total weight for the dual arms, T-bar carriage, and the stereo TV, with lights and pan/tilt mechanism, is estimated at 309 pounds for the 8-foot arm configuration. The total weight of the servicer kit, excluding repair equipment to be carried, is estimated to be 847 pounds.

Table 3

Servicer Weight Statement

ITEM	WEIGHT (POUNDS)	NOTES
STOWAGE RACK (OPEN)	[illegible]	FROM [illegible] ADJUSTED)
THERMAL CLOSURE	[illegible]	
PERIPHERAL TV, PAN/TILT	[illegible]	
ELECTRONIC ASSEMBLIES	80	ASSUMES VIDEO CONTROL ON SERVICER
CABLING	30	
DOCKING SYSTEM		CHARGEABLE TO CARRIER VEHICLE
MANIPULATOR SYSTEM (R')	309	
TOTAL	[illegible]	EXCLUDING REPLACEMENT MODULES
FLIGHT SPECIFIC OPTIONS		
BATTERIES	180	MISSION DURATION & THERMAL DESIGN DEPENDENT
MANIPULATOR TV (2)	*	ASSUMES ADDITIONAL SENSORS & LENSES ONLY USE COMMON ELECTRONICS

Number and Locations of Cameras

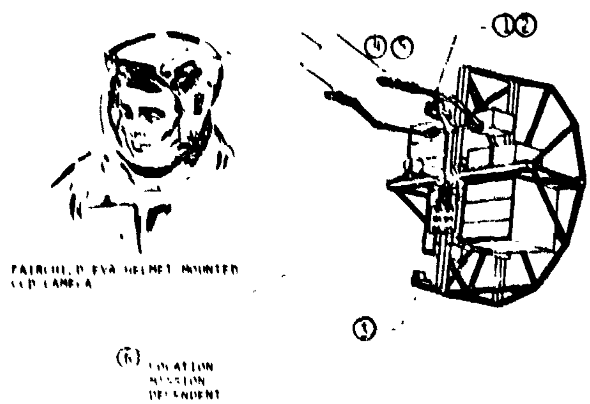

Figure 8

An overall system capability of six separate cameras is envisioned, as shown in figure 8. Three (Cameras 1-3) of these constitute the basic system and three are mission-dependent. Time sharing of the downlink video channel is required even if only two cameras are flown. All cameras except numbers 4 and 5 will be mounted on pan/tilt mechanisms for maximum versatility. Cameras 1 and 2 form the basic stereo system used primarily for the detailed servicing tasks and docking. Camera 3 is mounted peripherally to provide another view angle to assist in servicing, docking and inspection.

Cameras 4 and 5 are small, monocular cameras mounted near the end effectors for viewing operations within a confined or partially enclosed volume. The small size of the Fairchild EVA helmet-mounted CCD camera (488 x 380 CCD array) is shown for size perspective for this application. The camera is currently in the Shuttle inventory, although not yet reflected in all documentation.

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Camera 6 is mission-dependent and can be mounted anywhere on the servicer for missions with viewing requirements exceeding the capabilities of the basic system.

The servicer/TMS interfaces have been kept as simple as possible since a specific carrier vehicle has not been specified. Thus, the servicer should be compatible with any carrier vehicle, or at least very nearly so, at this early conceptual stage. The servicer requires +28V power only. Any other power required is the responsibility of the servicer. If mission requirements dictate, additional batteries can be carried on the servicer. The data bus concept for commands and telemetry is a simple, standard interface. Data are routed to the carrier vehicle for formatting and multiplexing with the carrier vehicle telemetry data. Commands are routed to the servicer, which in turn routes them to the appropriate servicer function. Thus, the servicer appears to the carrier vehicle as simply another of its subsystems. The video data is routed to the carrier vehicle (which requires a video capability independent of the servicer for non-servicing missions) for transmission. Data compression, if needed, would also be performed by the carrier vehicle.

The communications are depicted in figure 9. The servicer system is considered to be under ground control after delivery to orbit by STS. Control is from a ground control station through the Telemetry and Data Relay Satellite System (TDRSS). Time delays for the control loop will fall between 1 to 2 seconds. This delay presents complications in the control design and operation, but previous studies have indicated successful control with these levels of delay.

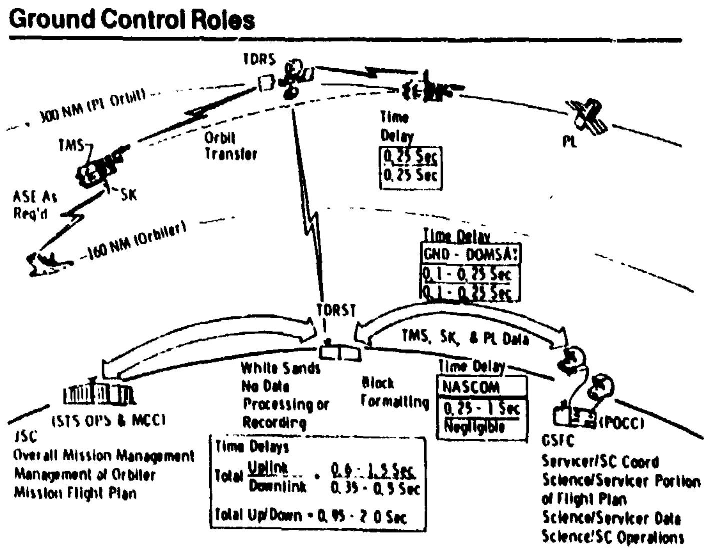

Figure 9

Figure 10 is a functional flow control overview showing the ground control station and servicer kit. The ground control station configuration is shown in figure 11. The system includes a control station for TMS on the right. This station would control the carrier vehicle (TMS) through

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capture of the spacecraft to be serviced. The center section includes two six-degree-of-freedom controllers and a stereo vision system and is the main control for the servicer kit. The left console controller would supply system level activities and support the primary controller as required.

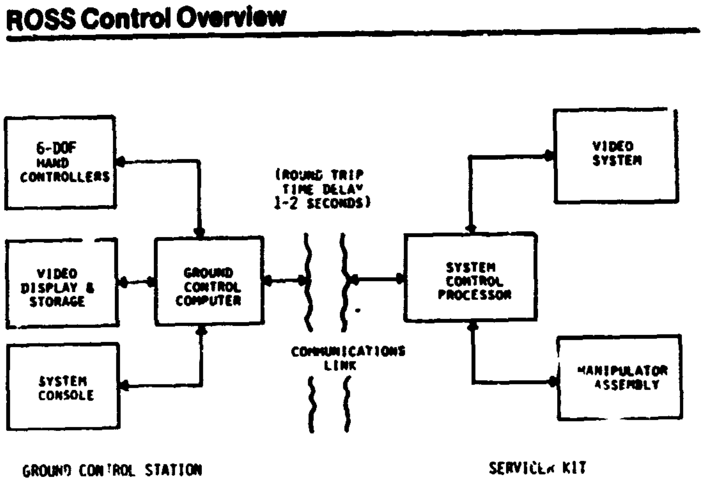

Figure 10

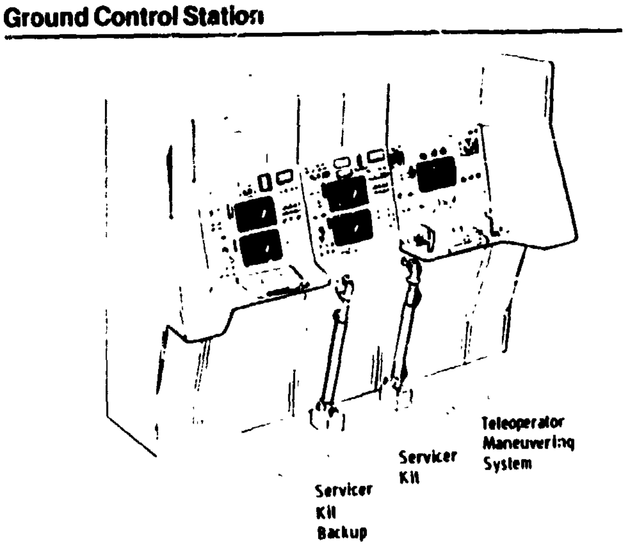

Figure 11

Figure 12 is an artist's concept of the servicer kit mounted to the Martin TMS carrier equipped with solar panels. This configuration could remain in orbit and be on call for servicing, requiring only delivery of repair parts and consumables by STS.

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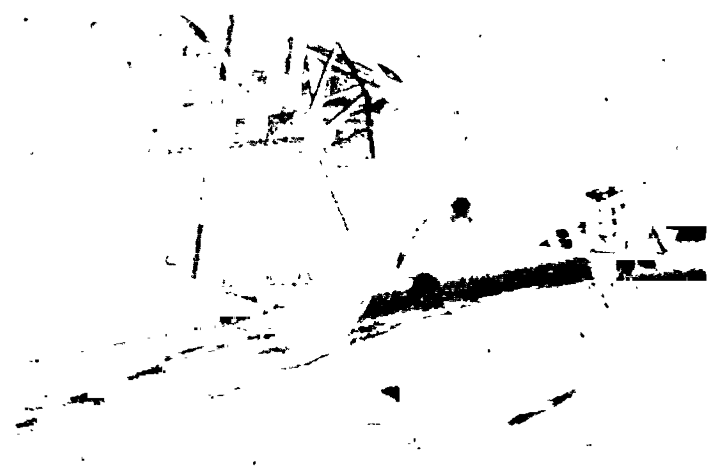

Figure 12

RESULTS

Remote orbital servicing is a feasible concept within the state of the art. The configuration subsystems and components of ROSS are based on considerable previous studies and development programs. The required technologies have been and are currently being addressed by numerous contractors and at many NASA facilities. The ROSS configuration is flexible, allowing serviceability for future spacecraft with minimal impact on spacecraft design. The initial configuration has inherent design flexibility and performance margins which will accommodate modifications to incorporate advanced technologies or additional requirements.

Development of the ROSS servicer kit is estimated to require approximately 3 years with an ROM cost in 1982 dollars of $40 million for design, development, testing and engineering.

The ROSS concept study was an 8-month effort and even though it was based on a significant amount of previous laboratory and development experience, a number of items which require additional review were identified. For the three spacecraft in the mission model three areas require further evalation. LDEF was designed for ground servicing and would require modification to allow in-space servicing by either ROSS or EVA. The noncaptive retention hardware for tray attachment could be changed, and, in addition, a method of grasping the trays for removal and installation would be required. ROSS is defined as having the capability to accomplish the same level of tasks as is possible by a man in a space suit; therefore, it should be possible to service ST. However, the level of details necessary to define the ST servicing requirements could not be identified during this study, and further analysis is needed. The third area requiring further analysis is the docking or attachment mechanisms between the spacecraft and ROSS. The basic method chosen for ROSS was the use of the RMS grapple attachment; other methods of attachment to spacecraft hardpoints need to be investigated, and analysis of the relative motion between spacecraft and servicer is required.

The other major area of concern is the communication between ROSS and the remote control station. Control of ROSS through TDRSS to a ground station requires further evaluation of video bandwidth, data compression techniques, the influence of time delays, and interruption of communication. Control of ROSS from STS or a space operations center, when within line of sight, would eliminate or reduce the communication concerns but would also decrease the operational flexibility of the remote servicing concept.

CONCLUSIONS

The original aim of the ROSS concept study was to identify the current state of the art for remote servicing and conceptualize the initial control station for interfacing with a remote system. The study accomplished both of those aims; and, in addition, identified a focus for the agency research program in automation.

The development and construction of the manipulators, sensor and video systems, control laws, and man-machine interface could be tested in the STS bay or attached to the RMS with control from the aft deck of STS. The flight-qualified system could then be integrated with a TMS-type carrier for further tests in a free-flyer mode with control from STS and/or a ground station.

The ROSS concept has had a significant influence on Research and Technology (R & T) planning for the OAST programs on teleoperators and robotics. The Spacecraft Systems Office (RSS) planning includes system level analysis of the communications requirements, development of a systems simulation capability, supporting hardware simulations, and planning for possible integration with the RMS, STS, and TMS for flight experiments. Within the Electronics and Human Factors Office (RTE), planning for both the Automation and Space Human Factors programs has been identified and sorted into two classes. Those areas in which work is either required or could directly enhance ROSS were classified as direct support. Those areas in which advanced R & T could substantially advance the teleoperator capabilities in the direction of autonomy leading to robotics were classified as opportunity areas.

The ROSS is a feasible concept. The focus of agency R & T automation activities toward developing and advancing the concept allows researchers to apply innovative and theoretical concepts to solving real problems which can have early benefit to the agency.

The development of a ROSS-type space vehicle designed with sufficient flexibility and performance margins would supply a needed flight testbed for the early integration and application of emerging technologies in robotics and automation.

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OPEN CHERRY PICKER SIMULATION RESULTS

By

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For

SATELLITE SERVICES WORKSHOP

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NASA JOHNSON SPACE CENTER
HOUSTON, TEXAS

OPEN CHERRY PICKER SIMULATION RESULTS

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ABSTRACT

The introduction of the Space Transportation System provides opportunities for improved space operations reliability and cost effectiveness. Many applications are now envisioned that require maintenance and repair of satellites, and assembly of large structures. This paper addresses the simulation program associated with a key piece of support equipment to be used to service satellites directly from the Shuttle. This equipment, the Open Cherry Picker (OCP) is a manned platform mounted at the end of the Remote Manipulator System (RMS) and is used to enhance Extra Vehicular Activities (EVA). The results of simulations performed on the Grumman Large Amplitude Space Simulator (LASS) and at the JSC Water Immersion Facility are summarized.

INTRODUCTION

Concepts for an Open Cherry Picker (OCP) were developed for NASA Johnson Space Center (JSC) under contract NAS9-15507 and a development test article designed, fabricated and tested under contract NAS9-15881. The objectives of the initial contract was to evaluate the OCP flight article roles and associated design concepts for fundamental requirements, and to embody key technology developments into a ground simulation program. Detail manufacturing drawings and plans for an Open Cherry Picker simulator Development Test Article (DTA) were prepared for use in the JSC Manipulator Development Facility and Water Immersion Facility. The follow-on contract, NAS 9-15581, supported the fabrication of the test article and the subsequent simulations that evaluated the features of this servicing tool.

The potential applications of the OCP in support of Shuttle operations are listed in Figure 1. A platform mounted at the end of the Remote Manipulator System (RMS) is a convenient means for transporting an EVA astronaut, tools and mission hardware around the Shuttle Cargo bay. Similar in application to terrestrial cherry pickers, the OCP enhances productivity during six-hour EVA periods. Potential OCP service roles are enhancement of Space Lab and sortie operations, maintenance of automated payloads such as the Multimission Modular Spacecraft, assistance in space construction research, and inspection and repair of the Shuttle itself.

The Development Test Article, Figure 2, was designed and fabricated to include all the features needed to perform the functions identified in the mission studies. These features included a rotating foot platform and a set of equipment for RMS control, RMS stabilization, payload handling, lighting and tool storage. The design objective was to maximize crew productivity by maximizing their work zone and conveniently locating all work tools and handling equipment. This was achieved by mechanizing all support equipment to rotate in and out of the crew work zone and placing infrequently used equipment such as lights and tool storage brackets out of the area.

FUNCTIONS IN SUPPORT OF SHUTTLE OPERATIONS

- SPACELAB
  - DEPLOY & RETRACT EXPERIMENTS
  - DATA RETRIEVAL & DATA STORAGE SYSTEM RESUPPLY
  - UNSCHEDULED MAINTENANCE
- AUTOMATED PAYLOADS
  - MMS SUBSYSTEM INTERCHANGE
  - LDEF SAMPLE TRAY INTERCHANGE
  - LST SERVICING
- CONSTRUCTION R&D
  - DEPLOY FIXTURES
  - RESUPPLY FAB MACHINES
  - JOINING & ALIGN OPERATIONS
  - DISASSEMBLE STRUCTURE
- SHUTTLE INSPECTION/REPAIR
  - FORWARD RCS
  - RADIATORS
  - EXPERIMENT TIE DOWN

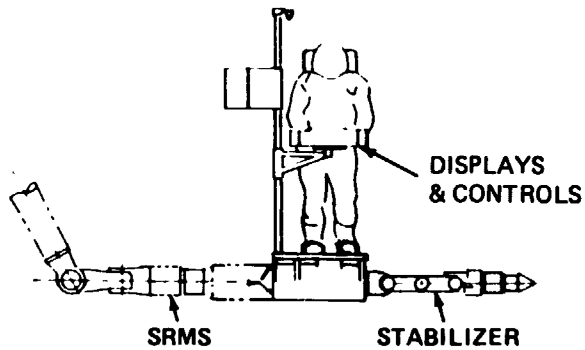

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Fig. 1 Open Cherry Picker Missions

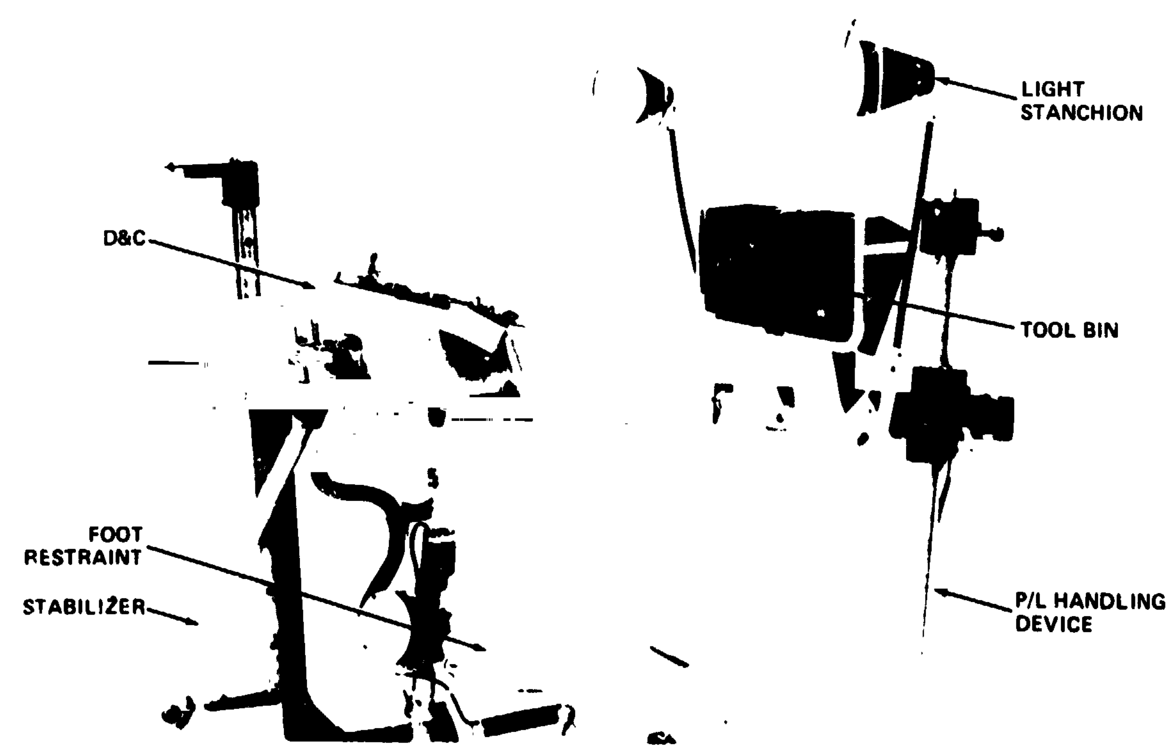

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Fig. 2 Open Cherry Picker Devolopement Test Article

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Techniques for storing, activating and deactivating the OCP were also evaluated during the development program. The OCP Development Test Article was designed to be folded for storage in the side bulkhead of the payload bay. The OCP was folded into a 62x45x36-inch package, as shown in Figure 3, and normally mounted to the cargo bay at STA 622 which is reserved for EVA peculiar equipment. The OCP support structure, shown in Figure 4, was designed with the versatility to interface the OCP with the Shuttle at any station along the length of the Cargo Bay, and to operate the latches remotely from the Aft Flight Deck or manually by the EVA crew.

This test hardware was used in a series of simulations in the JSC Water Immersion Facility and the Manipulator Development Facility, and the Grumman Large Amplitude Space Simulator with the goal to determine the minimum set of design features needed in an OCP system. The results, to be discussed in this paper, significantly reduced the complexity of the system and formulated a growth program that starts with a basic Manipulator Tool Restraint (MFR) and is enhanced with payload handling equipment, tool storage, stabilizers and lighting as the mission need arises.

SIMULATION OVERVIEW

Three simulation facilities were used in the development program. The first series of tests utilized the JSC Water Immersion Facility to test OCP form, fit and function. The second series of tests were performed at the Grumman Large Amplitude Space Simulator and emphasized evaluation of handling qualities and the effects of RMS dynamics on the Cherry Picker's crew productivity. The third test series was performed at the JSC Manipulator Development Facility where the accessibility of the RMS to the OCP support structure was evaluated.

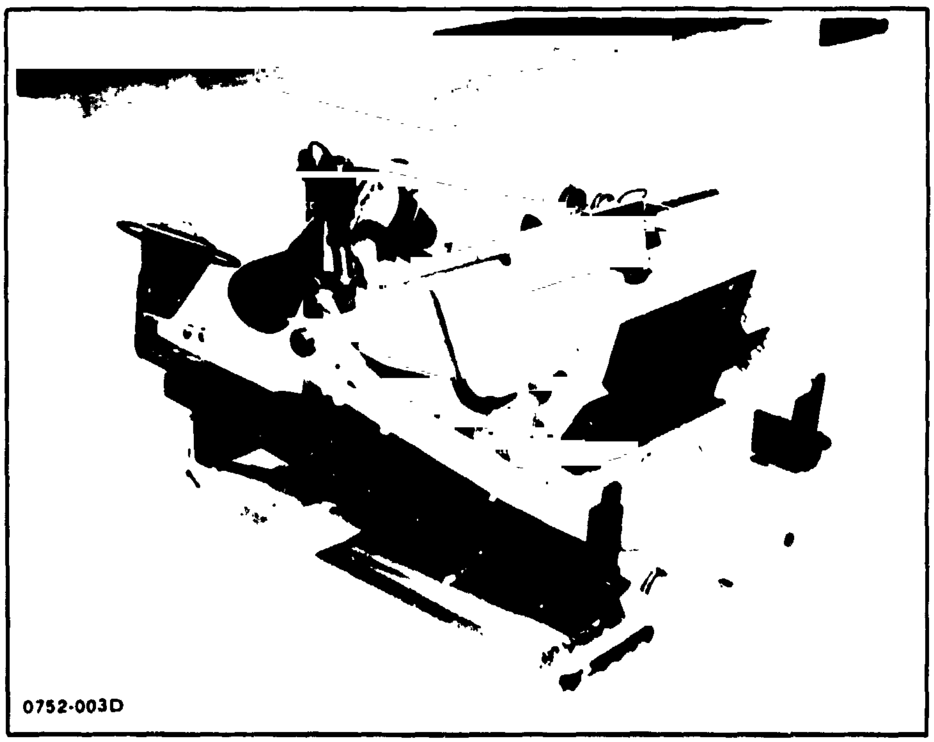

Fig. 3 OCP-DTA Folded Configuration

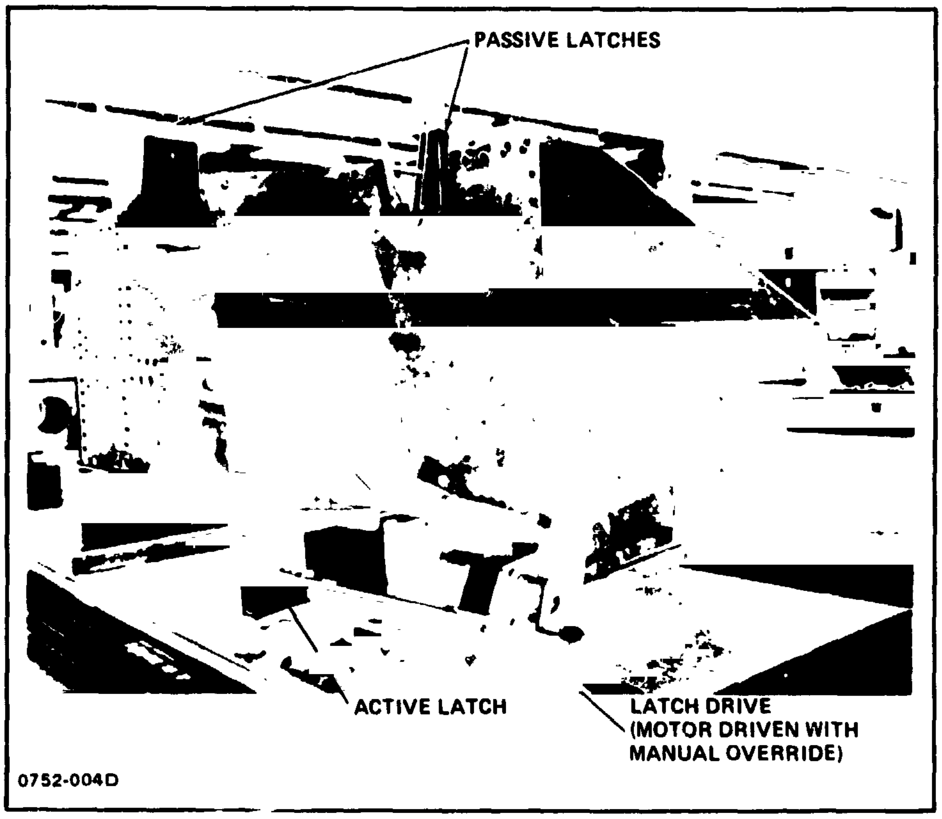

Fig. 4 OCP Support Structure

The first tests performed in the development program were at the Water Immersion Facility (WIF) at NASA-JSC. These tests evaluated crew interfaces and demonstrated the feasibility of maintaining satellites using the OCP. The first test (Test A on Figure 5) satisfactorily demonstrated reach and function of each OCP element. The second test (Test B) verified the ability to unfold and fold the OCP and further demonstrated the feasibility of an EVA role in satellite servicing.

Some configuration issues identified in Test A are listed on the Figure 5. The initial light stanchion arrangement utilized a knurled-knob lock mechanism that required two hands for operation. This design was replaced with a friction lock device to achieve one hand operation. The D&C panel height was found not to have sufficient adjustment to cover the range of subjects participating in the test. A 6-inch sleeve was added to the DTA for further evaluation. The original DTA utilized pippins for lock and release of the aft tool bin. These devices were found difficult to operate with a gloved hand and were replaced with a latch release handle that can be operated with one hand. Test A also demonstrated the ease with which a suited astronaut could replace the Multimission Modular Spacecraft (MMS) subsystems and an LDEF experiment tray. The latter task was found more difficult because of the fastener system used on the tray.

Test B demonstrated the ability to fold and unfold the OCP and evaluated the operation of the stabilizer. The removal and installation of an MMS module was repeated in this test with a higher fidelity mockup; the results indicated that the task could be performed in an EVA mode using the OCP. Reach data to evaluate the extension a crewman can make with the new Shuttle suit was collected for comparisons with one-g tests using a restraint system. These tests indicated a rebound effect in the suit when the subject extended himself by rotations about the ankle. Additional testing is needed to determine if this effect is due to balance of the subject in the tank or due to spring forces in the suit.

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TEST A – FORM & FIT

- DEMONSTRATED SATISFACTORY REACH TO ALL OCP COMPONENTS
- RESULTED IN REDESIGN OF LIGHT STANCHION
- RESULTED IN MORE HEIGHT ADJUSTMENT IN D&C STANCHION
- RESULTED IN REQUIREMENT FOR ONE-HANDED LATCH RELEASE FOR TOOL BIN
- DEMONSTRATED EASE OF MMS MODULE REPLACEMENT
- DEMONSTRATED LDEF TRAY REPLACEMENT AND NEED FOR SIMPLER LDEF TRAY FASTENERS

TEST B – DEPLOYMENT & REACH TESTS

- DEMONSTRATED EASE OF OCP DEPLOYMENT – NEED REDESIGN OF D&C LATCH
- DEMONSTRATED EASE OF MMS REPLACEMENT WITH HIGHER FIDELITY MOCK-UP THAN IN TEST A
- DEMONSTRATED EASE OF OPERATING STABILIZER
- COLLECTED REACH DATA – IDENTIFIED POTENTIAL SUIT REBOUND PROBLEM

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Fig. 5 WIF Test Results

The reach data obtained in Test B was used to develop a crew off-load system to be used in one-g testing. The zero-g restraint system was evaluated using suited subjects at JSC's Manipulator Development Facility (MDF). The results of this evaluation, presented in Figure 6, indicated that the crew's mobility was not restricted by the off-load system; his reach capibility was found to be the same as the WIF reach capability when the arms were above the shoulder, but somewhat less for arm position below the shoulders. It was also found that a dead weight of 16 lb could be handled when the subject was in a fully extended position. The offload system developed jointly by NASA and Grumman was installed on the Grumman LASS motion base for OCP/RMS interaction studies.

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- CAN DON HARD UPPER TORSO WHILE ATTACHED TO RESTRAINT SYSTEM
- VACUUM PRESSURE MAY HAVE TO BE VARIED DEPENDING ON SUBJECT POSITION
- SUBJECT COULD HANDLE 16 LB
- SUBJECT MOBILITY SIMILAR TO THAT IN WIF TESTS
- OPERATION OF COMPONENTS & MODULE PRESENTED NO PROBLEMS
- REACH TEST DATA COMPILED FOR COMPARISON WITH WIF DATA

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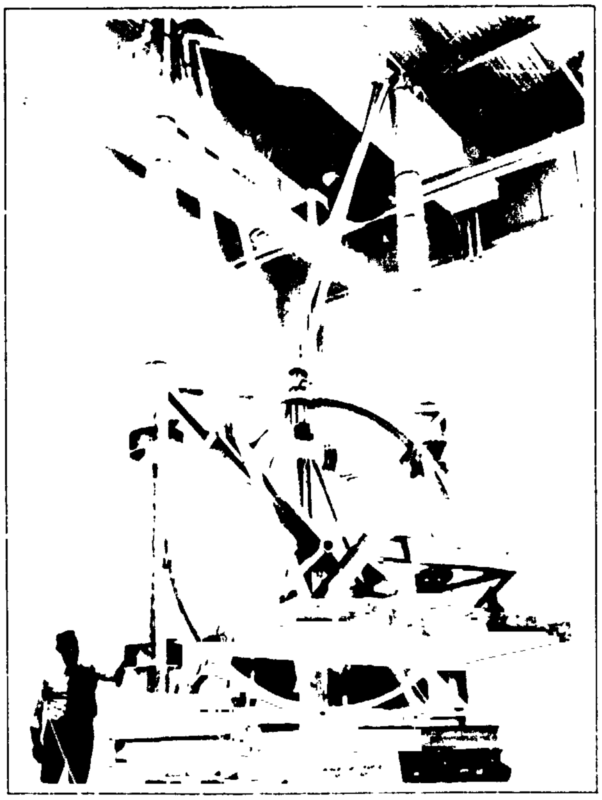
Fig. 6 Zero-G Restraint System Evaluation Results

The objectives of the four tests performed on LASS are summarized in Figure 7. Test 1 emphasized evaluation of OCP handling qualities, control modes and control devices (hand controllers or switches). The second test series evaluated the effect of RMS dynamics on work productivity and the need for an OCP stabilizer. Test series 3A and 3B evaluated operations for replacing the Multimission Modular Spacecraft subsystems using a GSFC-supplied high fidelity mockup, and replacing a LDEF experiment tray. Test Series 3A was performed with shirt-sleeve subjects with emphasis placed on developing procedures. Test Series 3B was a repeat of Test 3A using suited subjects and emphasized the study of work-induced loads on RMS dynamics and a comparison of crew productivity using two cherry picker configurations. The results of these tests are covered later in more detail.



OBJECTIVES

- TEST 1
  - EVALUATE OCP HANDLING QUALITIES
  - COORDINATE REF SYSTEMS CONTROL MODES
  - CONTROL DEVICES
- TEST 2
  - EVALUATE EFFECTS OF RMS STRUCTURAL DYNAMICS ON OPERATOR'S ABILITY TO PERFORM STAELLITE SERVICE TASKS
- TEST 3A
  - EVALUATE REPLACING MULTIMISSION MODULAR SPACECRAFT S/S MODULES
  - EVALUATE REPLACING LDEF EXPERIMENT TRAYS
- TEST 3B
  - SUITED REPLACEMENT OF MMF & LDEF
  - EVALUATE MFR
  - EVALUATE FAILURE MODES
  - EVALUATE FORCE FEEDBACK
  - EVALUATE DIRECT MODE

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Fig. 7 LASS Test Series Objectives

The tests performed at the MDF determined the location of the stored OCP in the Cargo Bay and evaluated the capability of the Aft Flight deck to control the cherry picker using voice commands from the EVA crew The tests (see Figure 8) indicated that the Bay 1 location for OCP storage poses difficulties to the RMS operator. In this location the operator has poor visibility, poor TV coverage, and is continuously encountering RMS singularities. Other analytical studies performed by JSC support the findings that Bay 1 is not an optimum location for the OSS. "Plaid" studies indicated that the front bulkhead starboard mounted TV camera intrudes into the space needed for RMS grapple operations. Control of the maneuver from Bay 1 to a Solar Max Mockup was performed in the Orbiter Unloaded mode and presented no problems to the AFD operator given voice commands from an OCP operator to help in accurately positioning the OCP for a servicing task. As a result of these MDF tests, the baseline location for the OCP support structure switched to Bay 2, and more emphasis was placed in the LASS tests on evaluating aft flight deck control of the cherry picker.

The remainder of this paper will elaborate on the results of tests performed at Grumman's Large Amplitude Space Simulator.

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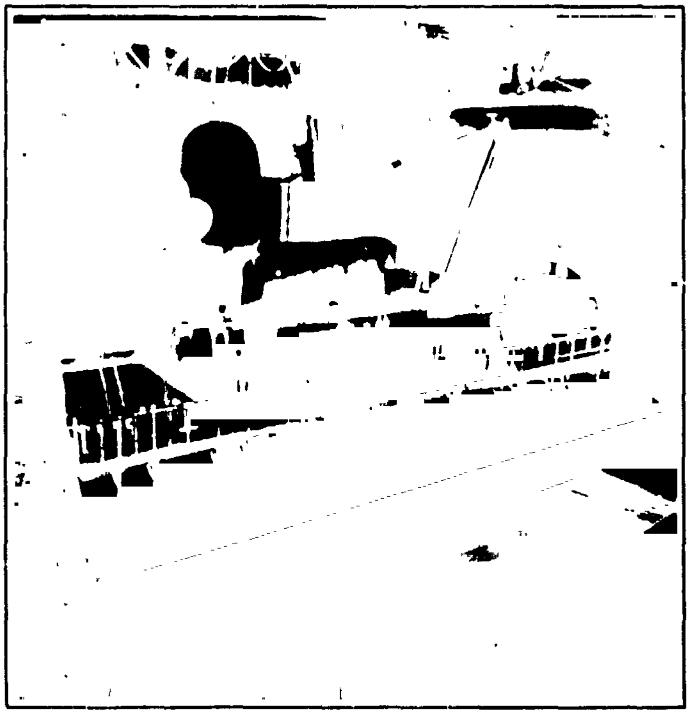

- MOUNTING POSITION OF STOWED OCP IN BAY 1 MAY BE DIFFICULT TO GRAPPLE
  - SINGULARITIES
  - LITTLE VISUAL CONTACT WITH OPERATION
  - POOR TV COVERAGE

  (RECOMMEND BAY 2 LOCATION)
- AFT CREW STATION CONTROL OF OCP TRANSLATION TO SOLAR MAX MAINTENANCE STATION FEASIBLE
- OCP OPERATOR VOICE COMMANDS WHILE IN CLOSE IMPROVES POSITIONING ACCURACY

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Fig. 8 MDF Test Results

FACILITY CHARACTERISTICS

The LASS is a versatile six degree-of-freedom motion base that is certified for manned simulation using suited or unsuited subjects. A hybrid computer is used to control the moving base with a model of RMS (including the first few bending modes). The motion base is hydraulically and electrically actuated and has a frequency response of about 1 Hz. Other characteristics of the facility are presented on Figure 9. The simulator can be controlled or "flown" from either a cherry picker mounted on the motion base or from an aft flight deck mockup. The OCP test subject is counterbalanced to simulate a weightless condition and a force sensing system is installed in the OCP foot platform to measure crew induced work loads.

LASS SIMULATION RESULTS

Four tests were performed at the Grumman Large Amplitude Space Simulator. The objectives of the tests were to evaluate cherry picker functions and design features and determine the minimum requirements in a phased program of manned work station capabilities.

The overall objective of Test Series 1 was to develop an OCP/RMS control system for transport to and from work sites located within the Orbiter Cargo Bay. The test evaluated system performance and handling qualities as a function of coordinate reference system (End Effector, Payload, and Orbiter Unloaded) and control device (toggle switches or hand controllers).

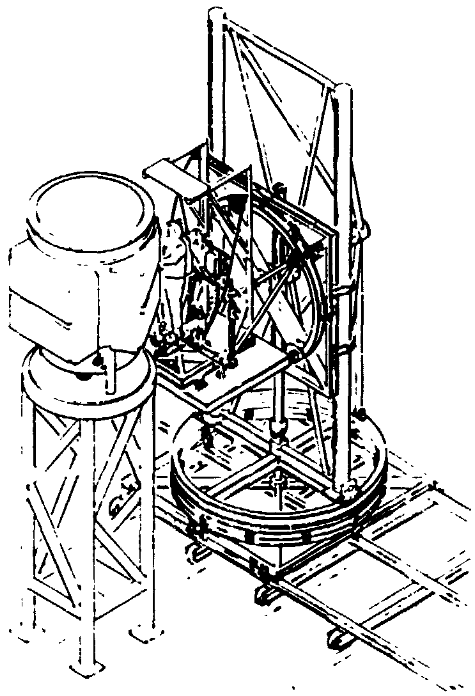

Fig. 9 Large Amplitude Space Simulator

The first part of the tests evaluated handling qualities and utilized three subjects in an experimental design that entailed flying the OCP to three targets using various control modes. The results, summarized in Figure 10, including test subject evaluative comments, indicate that the Payload mode was superior to the other two con trol modes investigated. The End Effector mode was found to be more difficult to use because it required more training to compensate for unwanted motions in translation due to a rotation input. Use of the Orbiter Unloaded mode required more in selecting the right control input when the vehicle has an attitude offset from the Cargo Bay centerline. This mode was more prone to inadvertent selection of commands.

The second part of the test evaluated toggle switches and integrated hand controllers. The integrated hand controllers were clearly superior because of their multi-axis control capability. Use of hand controllers reduced the time to perform a task by 50%. Use of toggle switches mounted to two console flyleafs were found to be acceptable but could cause problems in an emergency condition. Hand controllers with proportional control commands were favored over discrete hand controllers.

The objective of Test Series 2 was: 1) to evaluate the OCP operator's ability to perform satellite servicing tasks with and without a stabilizer; 2) to evaluate the difference in operator performance using different C&D console positions; 3) to evaluate alternative grapple fixture designs; and 4) to determine the reach envelope of the stabilizer.

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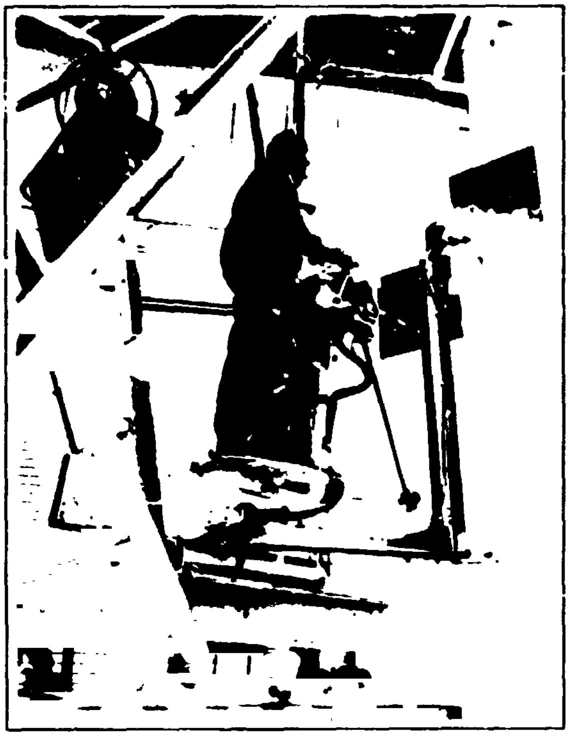

- PAYLOAD MODE PREFERRED – REQUIRES LESS TRAINING & OPERATOR THOUGHT
- INTEGRATED HAND CONTROLLERS IMPROVES PERFORMANCE 50% RELATIVE TO SIMPLE TOGGLE SWITCH FLYLEAFS
- PROPORTIONAL HAND CONTROLLERS PREFERRED OVER DISCRETES
- FLYLEAF CONTROLLERS ACCEPTABLE FOR SIMPLE TASKS
- POTENTIAL EXISTS FOR REDUCED PANEL COMPLEXITY
- RMS CONTROL LAW ANOMALIES IDENTIFIED

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Fig. 10 Test Series 1 Results

The salient findings of the test are summarized on Figure 11. Servicing the MMS took 12 minutes longer using the stabilizer than without the stabilizer. Initial evaluations of RMS back-drive effects on MMS replacement work loads (simulated as a 150 lb-sec impulse in the X direction) indicated that the operator could readily perform the task. With a 300 lb-sec impulse, the task became significantly more difficult and could present safety problems. It was believed, however, that a human operator could react to the motion and stabilize the RMS manually.

The stabilizer was found to gimbal-lock when the OCP was close to the work piece. Two additional "roll-pitch" joints and 12 in. of length were added. These changes simplified stabilizer operation to the point where the operator did not have to visually inspect the stabilizer while maneuvering.

Several configuration anomalies were identified including the operation of the D&C latch and the Payload Handling Device release mechanism. The "push-pull" feature of the D&C stand latch was found to degrade operator performance and intermittently jammed. The interlock between offline and rotate functions also reduced operator performance (time to perform a task).

The ability to use the hand controllers when the D&C console is rotated in the 90° position helps overall task performance. This feature eliminates the need to rotate the D&C to the forward position each time that a vernier position adjustment has to be made.

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- SERVICING MMS USING STABILIZER TAKES 12 MIN LONGER THAN WITHOUT STABILIZER (6 MIN VS 18 MIN)
- INITIAL STUDY OF RMS BACKDRIVE DYNAMICS INDICATE THAT MMS REMOVAL/REPLACEMENT TASK CAN BE PERFORMED W/O USING STABILIZER
- TWO JOINTS AND AN ADDITIONAL 12 IN. OF LENGTH HAD TO BE ADDED TO STABILIZER
- D&C STANCHION LATCH SYSTEM NEEDS REDESIGN
  - "PUSH PULL" MOTION CAUSED PROBLEMS
  - "ROTATE-OFF LINE" INTERLOCK INCREASED TIME
  - MECHANISM JAMS
- FLYING OCP WITH D&C IN 90° POSITION HELPFUL IN TASK PERFORMANCE
- PHD RELEASE HANDLE SHOULD BE RELOCATED (COULD BE A RESTRAINT SYSTEM PROBLEM)

0752-011D

Fig. 11 Test Series 2 Results

The suited test subject was required to perform a MMS module replacement task using the stabilizer and without it. For the non-stabilizer case, a simulated force feedback of 50 lb acting for three seconds in the X-direction was inputted to the software and the resulting RMS motions commanded to the LASS drive servos. The force feedback was triggered by a set of microswitches which were activated when the MMS module was either inserted or removed from the support structure. The timelines for performing the module replacement task are shown on Figure 12. It took only 6 min to perform the task without a stabilizer compared to 18 min with a stabilizer. The suited test subject had no difficulty in handling the module with the force feedback oscillations of the OCP. Force feedbacks of 100 and 150 lb were imposed on the non-suited test subjects and they were still able to cope with the resulting RMS oscillations.

These test results strongly recommended that further efforts be placed on determining the need for a stabilizer. The time to manage the stabilizer was found to be excessive, and the ability to perform the task with the RMS oscillating as a result of representative induced workloads was found to be feasible. As a result of Test 2, more fidelity in representing the effects of RMS work-induced oscillations on productivity was planned for subsequent tests.

The objectives of Test Series 3A were to demonstrate the feasibility of operating the OCP from the Aft Flight Deck (AFD), demonstrate further the feasibility of operating without the stabilizer and refine the procedures for replacing the MMS module and LDEF experiment tray. To meet these objectives, a working engineering mockup of the RMS operator station was added to the facility and a two-axis force sensing system added to the foot platform of the OCP DTA. A high fidelity mockup of the MMS including flight latch systems was provided by NASA GSFC. In addition, a mockup of the Universal Service Tool for interface with the MMS was provided by SPAR, Ltd.

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TASK	TIME, MIN
• START	0
• MANEUVER TO GRAPPLE FITTING	2
• ACTUATE SAFING SWITCH, OFFLINE D&C CONSOLE TO 90°, GRAPPLE STABILIZER	1
• CANCEL SAFING, MANEUVER TO MODULE, D&C IN 0	3
• OFFLINE C&D TO 90°, MOVE RH PHD TO 0°, REMOVE MODULE AND CLAMP IN PHD, MOVE RH PHD TO OFFLINE	2
• MOVE LH PHD TO 0 , REPLACE MODULE	4
• MOVE LH PHD TO OFFLINE, D&C @ 90°, MANEUVER TO GRAPPLE FITTING	3
• UNGRAPPLE & STOW STABILIZER, D&C @ 90°, MANEUVER BACK TO START	3
TOTAL	18 MIN

TASK	TIME, MIN
• START	0
• MANEUVER TO MODULE, ACTUATE SAFING, OFFLINE D&C TO 90°	1
• REMOVE MODULE AND CLAMP IN RH PHD AND MOVE RH PHD OFFLINE	1
• MOVE LH PHD TO 0°, REPLACE MODULE, MOVE LH PHD TO OFFLINE	2
• D&C @ 90°, CANCEL SAFING, MANEUVER BACK TO START	2
TOTAL	6 MIN

- PAYLOAD MODE
- COARSE RATE (2 0 FPS MAX TRANS, 4 b /SEC MAX ROTATIONAL)
- VERNIER RATE (0 2 FPS MAX TRANS 0 48° SEC MAX ROTATIONAL)
- PROPORTIONAL RHC AND THC
- FORCE FEEDBACK 50 LB FOR 3 SEC IN X DIRECTION FOR NON STAB
- HAND HOLD GRAPPLE FITTING

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Fig. 12 Test Series 2 Timelines Stabilizer vs No Stabilizer

The tests demonstrated that the MMS module replacement task and the LDEF experiment tray replacement could be performed with control from the AFD using voice commands from the OCP operator (see Figure 13). It was also demonstrated that the MMS and LDEF tasks could potentially be performed without a stabilizer. Although the force feedback system was only in the horizontal plane, it was felt most of the induced loads would be in the forward direction and that the results did have validity.

Studies involving the preferred location of the D&C console indicated that if the console were located to the side of the operator, he could control the vehicle position while leaving the forward work zone open.

A timeline for replacing three modules of the MMS was established using the results of Test Series 3A. Figure 14 presents the functions and their associated times. Egress and MFR deployment times were based on WIF tests and the time to translate to the satellite based on LASS test results. The overall time for the replacement task was based on Test 3A results. The total time to replace an entire MMS subsystem complement is approximately 1 hour.

These tests supported the viewpoint that a stabilizer is not needed; also it became clear control at the OCP is superfluous for the types of tasks evaluated. Plans were then made for a Test Series 3B, in which a suited subject would be used and a third force sensing unit added to assure that a stabilizer was not needed.

- SUCCESSFULLY DEMONSTRATED FEASIBILITY OF REMOTELY CONTROLLING THE OCP FROM RMS OPERATOR STATION USING ORAL COMMANDS FROM OCP OPERATOR
- LDEF TRAY AND MMS MODULE REPLACEMENT CAN BE ACCOMPLISHED WITHOUT USE OF A STABILIZER
- DEMONSTRATED THE CONCEPTUAL DESIGN, OPERATION, AND STOWAGE OF SPECIAL TOOL FOR EXCHANGING OF LDEF TRAY AND MMS MODULE
- OCP OPERATOR IS CAPABLE OF POSITIONING OCP AT WORKSITE WITH D&C CONSOLE AT 90 DEGREE OFFLINE POSITION
- TIME TO PERFORM LDEF TRAY EXCHANGE 25 MINUTES
- TIME TO PERFORM MMS MODULE CHANGEOUT 10 MINUTES

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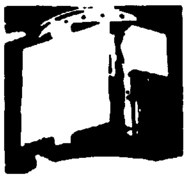
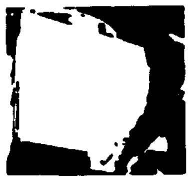
Fig. 13 Test Series 3A Results

- EGRESS THRU AIRLOCK TO BAY 2 – 15 MIN.
- DEPLOY & INGRESS MFR – 1 MIN



- TRANSLATE TO FSS – 1.5 MIN.
- REMOVE MMS REPLACEMENT MODULE FROM FSS & TRANSLATE TO MMS MODULE LOWER FITTING – 5 MIN.



- WITHDRAW TOOL FROM PHD, INSERT INTO LOWER FITTING & UNLOCK – 1.5 MIN.

- TRANSLATE TO UPPER FITTING, INSERT TOOL & UNLOCK – 1.5 MIN

- REMOVE MMS MODULE & PLACE ON PHD – 1 MIN.

TOTAL TIME = 53.5 MIN.

- INSERT REPLACEMENT MODULE – 4 MIN.
- TRANSLATE TO FSS & SECURE SPENT MODULE–5MIN.
- TRANSLATE TO BAY 2 – 1.5 MIN
- EGRESS & STOW MFR – 1.5 MIN.
- RETURN TO ORBITER – 15 MIN

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Fig. 14 Test Series 3A Results

The objectives of Test 3B were the same as Test 3A, except for introduction of suited operation and an experienced astronaut test subject. A third force sensor was added to the feedback system; Don Lind was assigned to the program as a test subject. In addition, the OCP DTA was reconfigured as a Manipulator Foot Restraint, in which the OCP Displays and Control panel was replaced with a tool carrier and the Payload Handling Device and lights removed as a means of comparing overall system performance with the basic OCP.

The simulation results, summarized on Figure 15, indicated that control of the RMS from the Aft Flight Deck is acceptable. A test of control modes concluded that augmented modes be used as primary and that joint-by-joint direct control be used only if a failure occurs.

Selected tests of failure modes were also done. A technique to "fend-off" obstructions in the event of a runaway arm were made and concluded that the technique was feasible but not recommended until further evaluations were made. Tests to detect and stop a runaway arm were also made. Preliminary conclusions recommend that if control from the AFD is baselined, that as a minimum, a "kill button" be made available on the cherry picker. This feature is potentially needed when the OCP is in tight places, in places where the RMS operator does not have visual contact with the OCP, and in places where there may be communications blockage between the RMS operator and OCP. Further evaluations, however, are being made to determine if this emergency feature is required based on suit strength and the reliability of the RMS consistency check software.

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- A CREWMAN SUPPORTED ON THE END OF RMS IS FEASIBLE
- AFT FLIGHT DECK CONTROL OF RMS IS ACCEPTABLE
- AUGMENTED MODE FOR RMS OPERATION IS ACCEPTABLE & RECOMMENDED
- DIRECT MODE (ONE JOINT AT A TIME IS ACCEPTABLE FOR "BACK-OUT" IF A FAILURE OCCURS). IT IS NOT RECOMMENDED FOR NOMINAL OPERATIONS
- TO "FEND OFF" (BACK-DRIVING RMS MOTORS IN A RUNAWAY) IS FEASIBLE BUT NOT RECOMMENDED FOR NOMINAL PLANNING
- A "KILL BUTTON" (BRAKES ON) WAS RECOMMENDED TO ENHANCE SAFETY WHEN:
  - IN TIGHT PLACES
  - IN PLACES WHERE RMS OPERATOR DOES NOT HAVE VISIBLE CONTACT WITH OCP
  - IN PLACES WHERE THERE IS RESTRICTED RF COMMUNICATIONS

Fig. 15 Test Series 3B Results

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The measurement of work induced forces are presented on Figure 16. The forces for replacement of the MMS varied between 7 and 25 lb. The difference in forces using suited and unsuited subjects was found not to be significant. This result infers that a high percentage of engineering development simulation can be performed in a shirt sleeve mode and the results verified with a minimum of suited simulation hours.

WORK TASK	FORCE APPLIED, lbf	
	UNSUITED	SUITED
MMS TOOL INSERTION	15-28	15
MMS TOOL CRANKING	15	10-20
MMS MODULE REMOVAL	15	7
MMS MODULE REPLACEMENT	15	21

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Fig. 16 Work-Induced Forces Applied During MMS Servicing

CONCLUSIONS

The objective of the simulation program was to take a full capability OCP, with its complement of design features, and reduce its complexity to those features that are mandatory for initial Shuttle applications. Figure 17 represents the final configuration and growth path for the OCP. The initial version is a Manipulator Foot Restraint that is made up of a torque box, a rotating foot restraint and a rotating stanchion that houses handholds, and a tool storage section with an interface for payload modules. The torque box is designed large enough for the addition of kits at a later date. These add-on kits include the stabilizer, payload handling device and rear mounted lights and tool storage area.

Simulations indicated that the crew can readily control the RMS/MFR from the Aft Flight Deck using voice commands from the MFR crewman. This eliminated the need for a D&C panel on the cherry picker. There was no apparent need found for a stabilizer. The dynamics of the RMS due to crew-induced workloads were found to be minor enough as not to impair task productivity. The use of the stanchion used for the D&C console as a payload handling device was found to be acceptable for such tasks as replacing a Multimission Modular Spacecraft subsystem. If the size or complexity of the payload increases, the payload handling devices could be added at a later date. The need for additional tool storage or lighting was not apparent in the missions evaluated. Like the need for dedicated payload handling devices, mission peculiar requirements for tool storage and lighting could be added as the missions demand them.

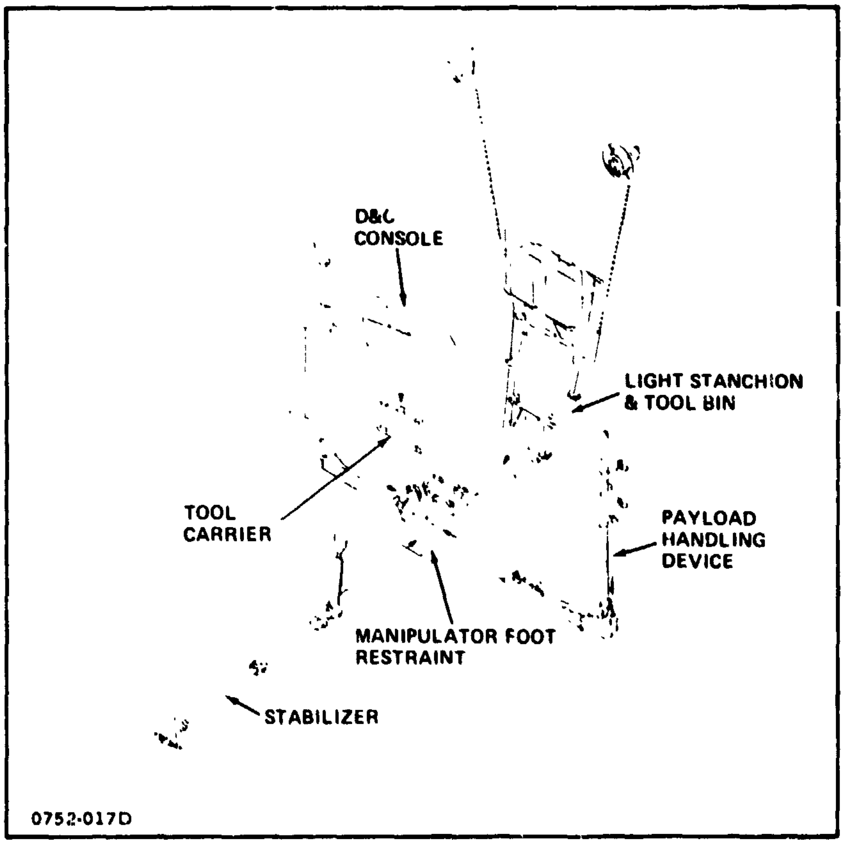

Fig. 17 Salient Simulation Result – Manipulator Foot Restraint with Add - On Kits

SERVICING COSTS FOR REPRESENTATIVE SERVICE MISSIONS

By

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For

SATELLITE SERVICES WORKSHOP

JUNE 22-24 1982
NASA JOHNSON SPACE CENTER
HOUSTON, TEXAS

SERVICING COSTS FOR REPRESENTATIVE SERVICE MISSIONS

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Satellite services user costs are the sum of the equipment use charge, miscellaneous service charges, and the STS transportation charges that apply to the service equipment utilized for a particular STS mission manifest. User costs were assessed for three representative service missions:

- Advanced X-Ray Astrophysics Facility (AXAF) revisit
- Upper Atmosphere Research Satellite (UARS) revisit
- Solar Maximum Mission (SMM) earth return.

SERVICING SCENARIOS

Servicing scenarios were developed for these missions to identify service equipment needs and on-orbit usage. Within these scenarios are considerations of the following:

- Backups for hangup of mechanical devices
- Retrieval/servicing modes for contamination-sensitive satellites
- Plume impingement/satellite control implications during close proximity operations.

Additional assumptions made in developing the on-orbit servicing scenarios are:

- Status monitoring, checkout, and activation/deactivation of satellites is user controlled (satellite communications via Orbiter S-band or satellite's communications system, as appropriate)
- Minimize Orbiter status/checkout involvement
  - Power (as required)
  - Overall health (extent tbd, but standardized for all satellites)
  - Go/No-Go for deployment and servicing verification/effectiveness is a satellite user decision
- Satellite deployment is via Orbiter command

- EVA is an acceptable service mode
- Maneuverable Television (MTV) usage
  - Record LEO/GEO upper stage firings
  - Examine all satellites prior to Orbiter capture/berthing
- Satellite separation ΔV during deployment imparted by RMS or HPA
- Consider various close proximity operations
  - Orbiter closure
  - "Clean" vehicle closure from 1000 ft separation
  - Versatile Service Stage Closure
    - All unmanned vehicle closures are controlled by the Orbiter crew
- Orbiter safety considerations

- Satellite RCS firings	>200 ft separation
- Liquid rocket engine firings	>2700 ft separation
- Solid rocket engine firings	OMS separation burn required to assure Orbiter exit of hazard envelope.

The AXAF revisit represents a service mission involving a contamination-sensitive satellite. The Orbiter would rendezvous with the spacecraft to within 1000 ft, and an unmanned Proximity Operations Module (POM) is then dispatched to retrieve and bring the payload within Remote Manipulator System (RMS) reach distance where it is captured by the RMS and placed on a Handling and Positioning Aid (HPA) for on-orbit servicing. Following servicing and checkout, the spacecraft is redeployed from the orbiter.

Figure 1 shows the Level-1 revisit on-orbit sequence of events for this service mission. Service equipment needs are highlighted as they appear in the scenario and for backup situations. The initial events call for:

- POM which can be adaptation of the MTV
- RMS and associated AFD Controls/Displays
- HPA
- AFD Controls/Displays for checkout of the POM
- AFD Controls/Displays for close proximity flight control of the POM.

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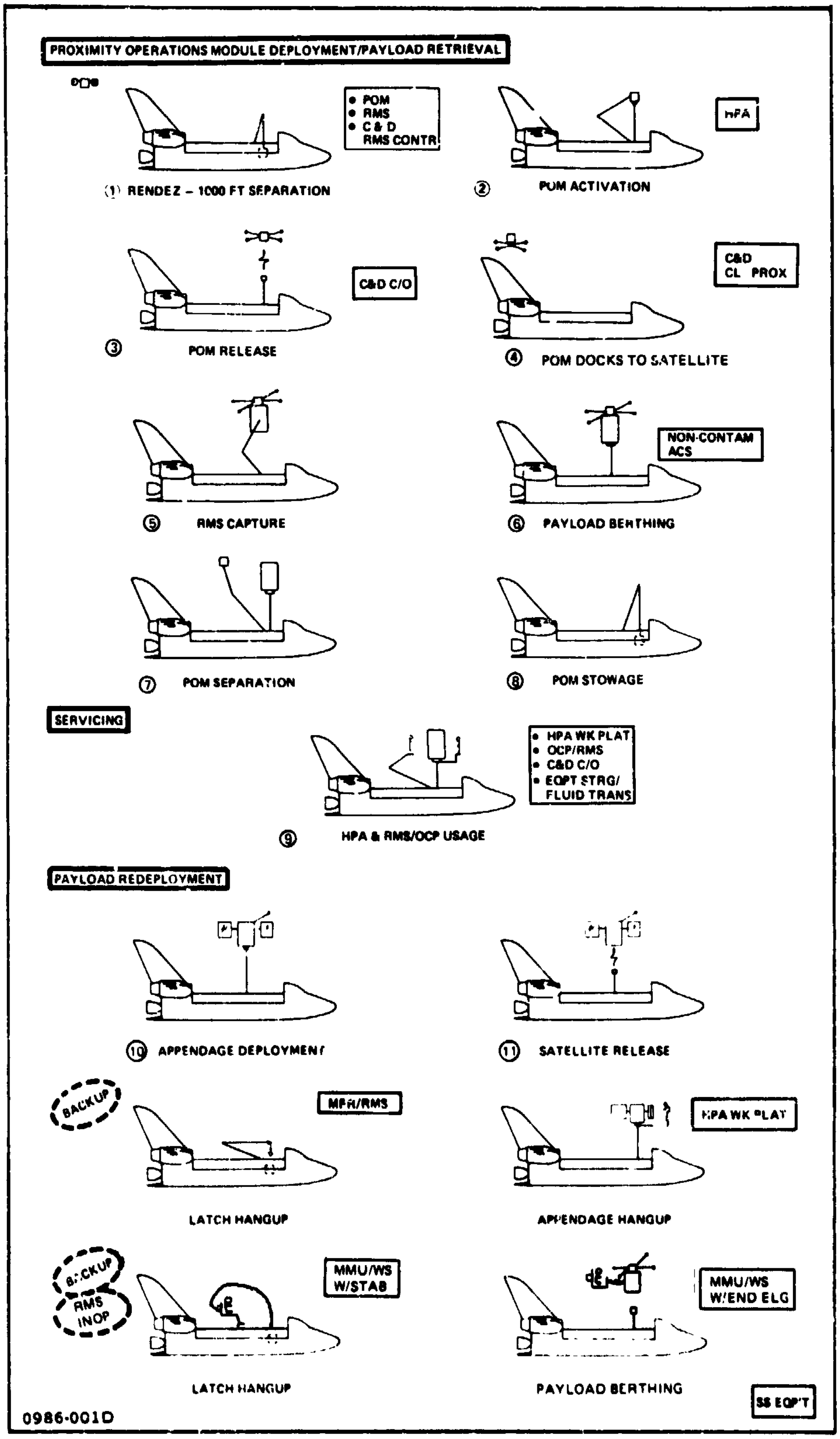

Fig. 1 AXAF Revisit Scenario

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Subsequent operations identify:

- Noncontaminating Orbiter ACS to enable servicing of the contamination-sensitive satellite
- Work Platform for the HPA
- Open Cherry Picker (OCP) and RMS
- AFD Controls/Displays for satellite checkout/servicing support
- Equipment storage/fluid transfer for servicing support.

Backup situations identify the following equipment needs:

- Manipulator Foot Restraint (MFR) to cover latch hangups
- HPA work platform (also identified above) to assist a potential satellite appendage hangup
- Manned Maneuvering Unit/Work Station (MMU/WS) adaptations with stabilizer and end effector to cover RMS inoperative situations.

Figure 2 illustrates a representative servicing concept for the AXAF revisit mission.

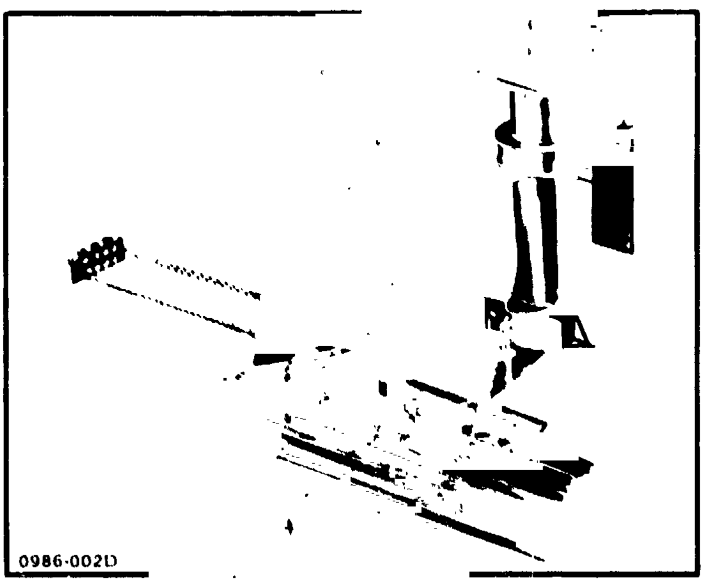

Fig. 2 AXAF Servicing Concept

Similar service scenarios were developed for the UARS revisit and SMM earth return missions. The UARS revisit represents a class of service mission in which the satellite's LEO operational altitude is above the nominal delivery altitude of the Orbiter. A Versatile Service Stage (VSS) is used to retrieve, and subsequently return, the satellite to its operational orbit after servicing at the Orbiter.

The SMM earth return mission illustrates a complement of service equipment for an earth return mission. The Orbiter would rendezvous with the satellite to within 1000 ft and accomplish retrieval with a manned POM that is an adaptation of an MMU/Work Station.

A summary of the service equipment complements for these three service missions is shown in Fig. 3.

SERVICE MISSION	SERVICE EQUIPMENT COMPLEMENT
AXAF REVISIT	• HANDLING AND POSITIONING AID • FLUID TRANSFER/EQUIPMENT STOWAGE • NON-CONTAMINATING ACS • PROXIMITY OPERATIONS MODULE - MTV ADAPTATION • OPEN CHERRY PICKER • MANNED MANEUVERING UNIT • WORK STATION - END EFFECTOR AND STABILIZER ADAPTATIONS • REMOTE MANIPULATOR SYSTEM
UARS REVISIT	• VERSATILE SERVICE STAGE • FLUID TRANSFER/EQUIPMENT STOWAGE • HANDLING AND POSITIONING AID • MANEUVERABLE TELEVISION • OPEN CHERRY PICKER • MANNED MANEUVERING UNIT • WORK STATION - END EFFECTOR AND STABILIZER ADAPTATIONS • REMOTE MANIPULATOR SYSTEM
SMM EARTH RETURN 0986-003D	• FLIGHT SUPPORT SYSTEM • MANEUVERABLE TELEVISION • OPEN CHERRY PICKER • MANNED MANEUVERING UNIT • PROXIMITY OPERATIONS MODULE - MANNED WORK STATION ADAPTATION • REMOTE MANIPULATOR SYSTEM

Fig. 3 Representative Service Missions and Equipment Complements

ORBITER/SERVICE EQUIPMENT PACKAGING

Packaging layouts were developed for each service mission to determine whether the packaged service equipment length or weight was the driver for STS transportation charges. Figures 4, 5, and 6 illustrate the service equipment packaging arrangements. The payload bay length and weight associated with the service missions are:

Mission	Service Equipment	
	Length (m)	Weight (k)
AXAF Revisit	6.4	4260
UARS Revisit	4.9	5369
SMM Earth Return	3.7	4561

For each of these missions, the STS user charges were driven by the length of the payload bay occupied by the service equipment.

SERVICE COSTS

Service mission costs represent a summation of costs that include:

- Equipment usage charges
- STS Transportation charges, and
- Miscellaneous charges.

To determine equipment use charges the following assumptions applied:

- User charges over the equipment's lifetime (est. 10 years) recover production and operations costs
- Operations costs are prorated against equipment utilization
- Equipment utilization was based on Grumman's Satellite and Services User Model (Ref. 1).

Similiarly, to determine STS Transportation charges the following assumptions applied:

- Costs are based on the STS Reimbursement Guides groundrules
- All service missions are shared payload flights. The revisit and earth return missions are missions of convenience, planned as events following initial launch (deployment) of payloads from the Orbiter.

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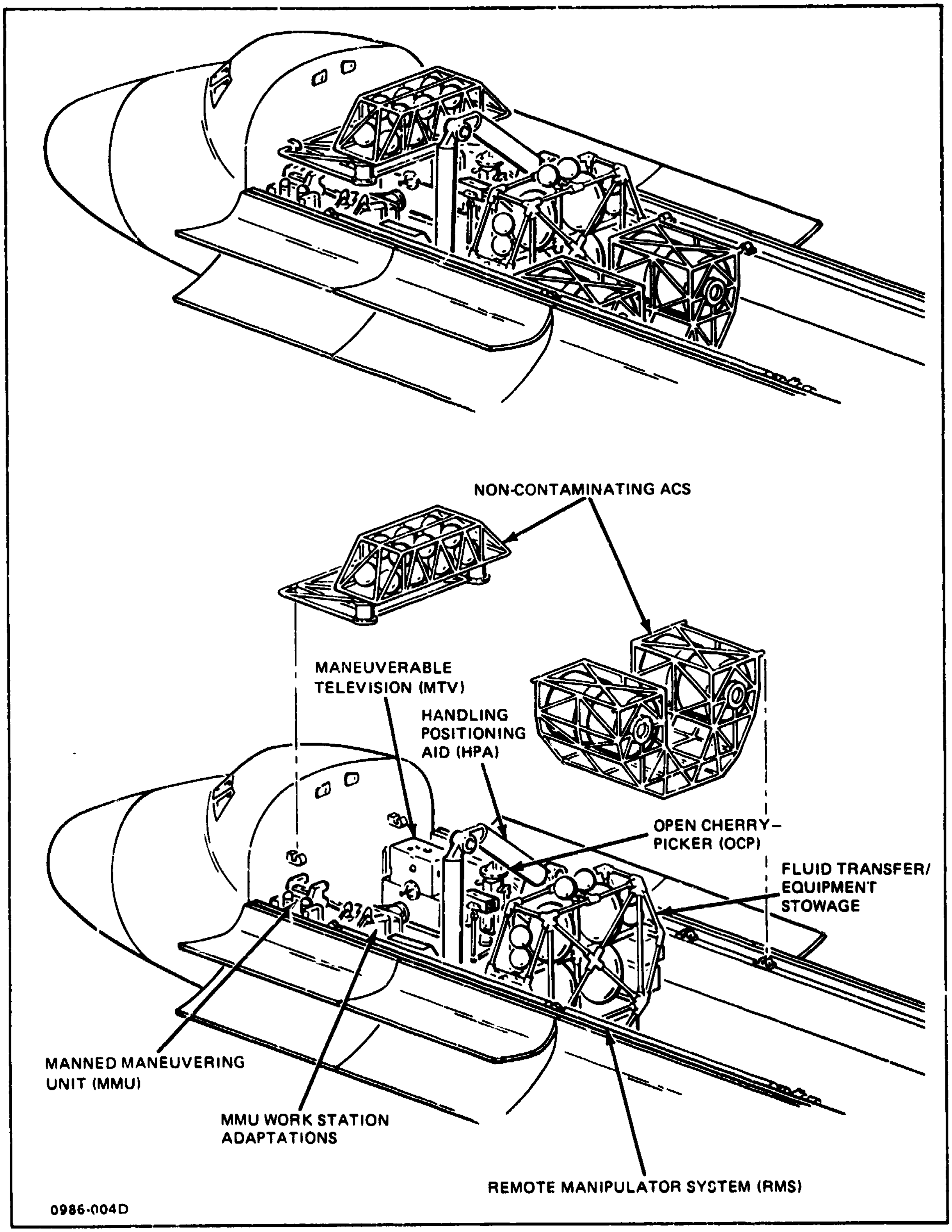

Fig. 4 AXAF Revisit - Service Equipment Packaging Arrangement

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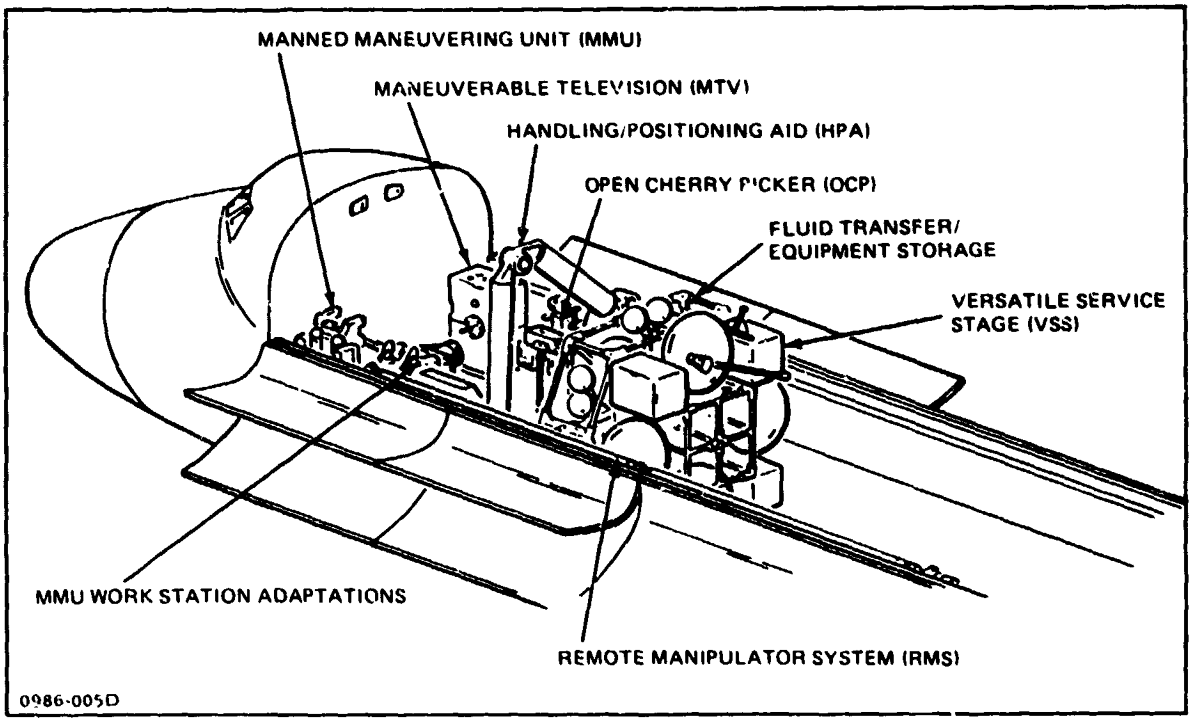

Fig. 5 UARS Revisit-Service Equipment Packaging Arrangement

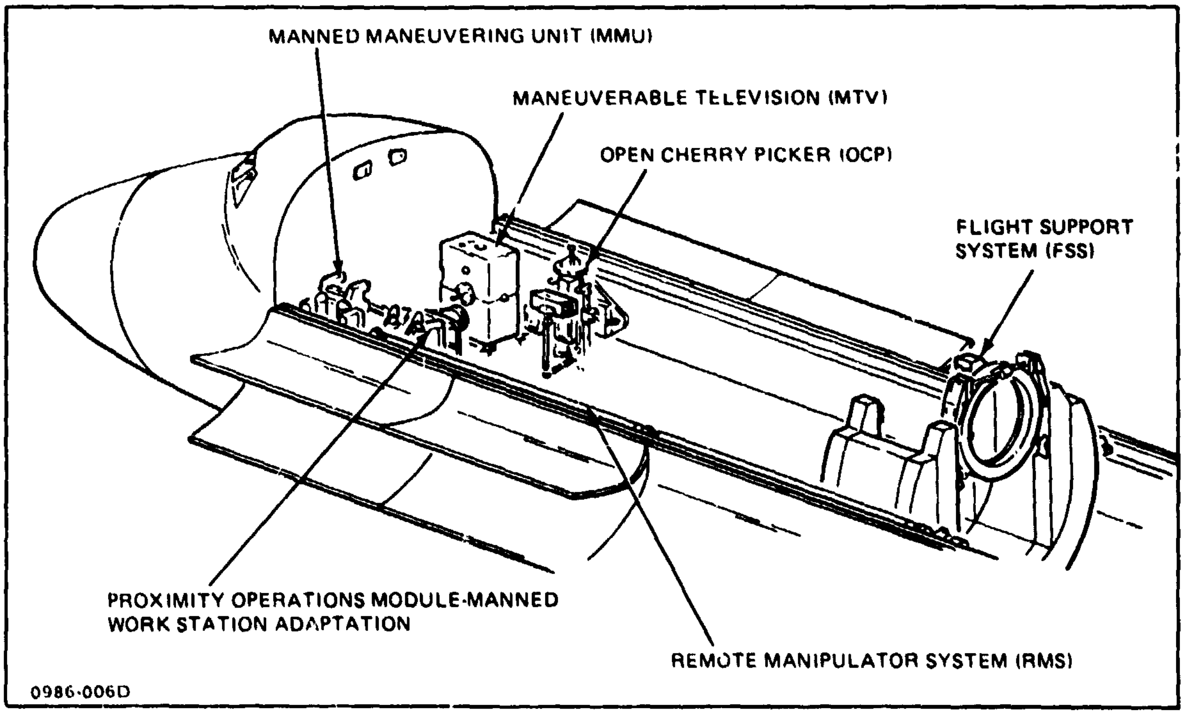

Fig. 6 SMM Earth Return-Service Equipment Packaging Arrangement

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All costs were assumed in 1980 dollars and, in addition, each mission was analyzed for miscellaneous charges in terms of impact on KSC flow, requirements for EVA, days on Orbit, and payload specialists need/usage.

Based on the equipment packaging, costing assumptions, and operations scenarios applying to these representative missions, the estimated total user costs are:

- AXAF Revisit - $16.0 million
- UARS Revisit - $13.2 million
- SMM Earth Return - $ 9.1 million

Figure 7 clearly shows that the dominant user charge for satellite services is that associated with the STS transportation charges for the service

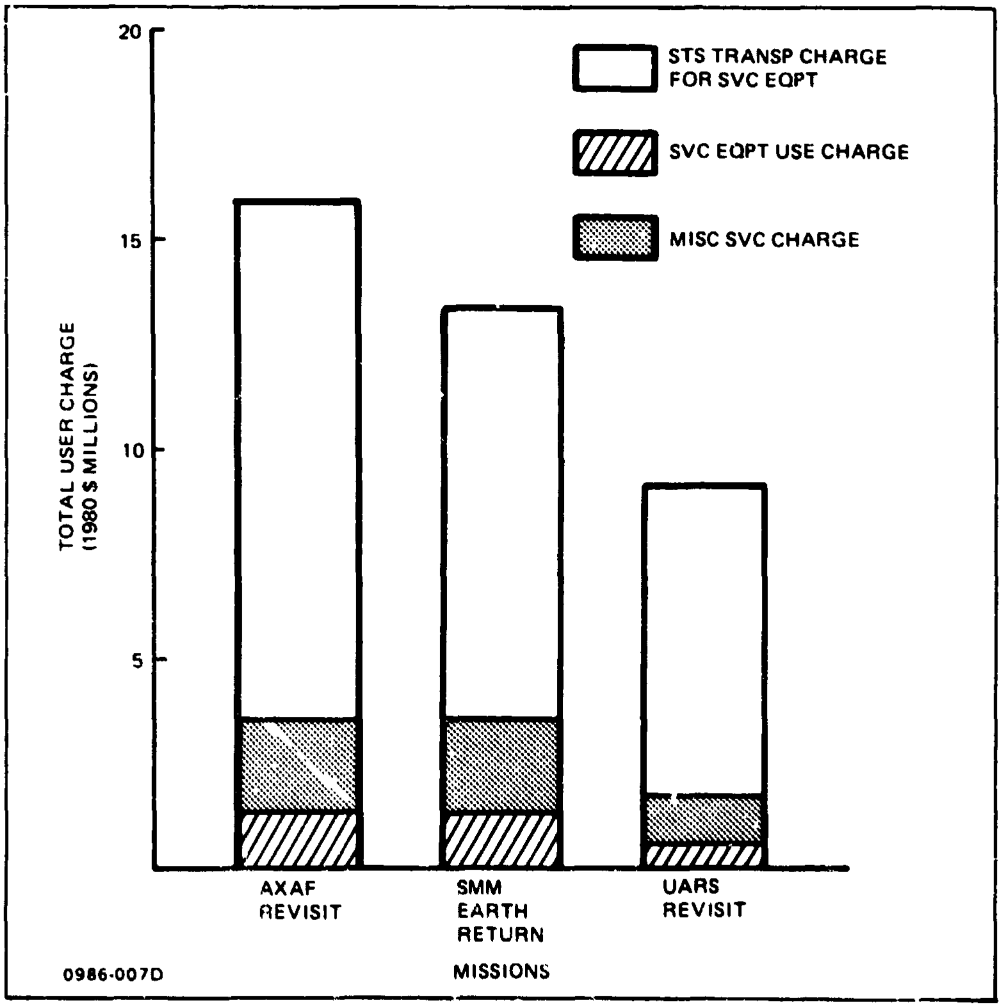

Fig. 7 Representatives Missions – Service Costs

equipment. Miscellaneous service charges are moderate, and equipment usage charges (amortized as a function of total estimated usage) are minimal. The overall service costs appear "about equal" for revisit or earth return missions.

Of key significance, however, is the fact that total user charges for revisit missions appear to be about 5 - 10% (or less) of the cost to build and relaunch a replacement satellite. This clearly indicates that satellite servicing from the Orbiter is cost effective.

REFERENCES:

1 - Satellite Services System Analysis Study, NASA Contr. NAS 9-16120, Grumman Aerospace Corporation, August 1981

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Presentation On The
Low-Earth Orbit Satellite Servicing Economics
To The Satellite Services Workshop
On
June 23, 1982

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3 - Goddard Missions 1970 Decade

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R. E. Davis
F. J. Cepollina
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LOW-EARTH ORBIT SATELLITE SERVICING ECONOMICS

SATELLITE SERVICES WORKSHOP

JUNE 23, 1982

R. F. DAVIS
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(GODDARD SPACE FLIGHT CENTER)

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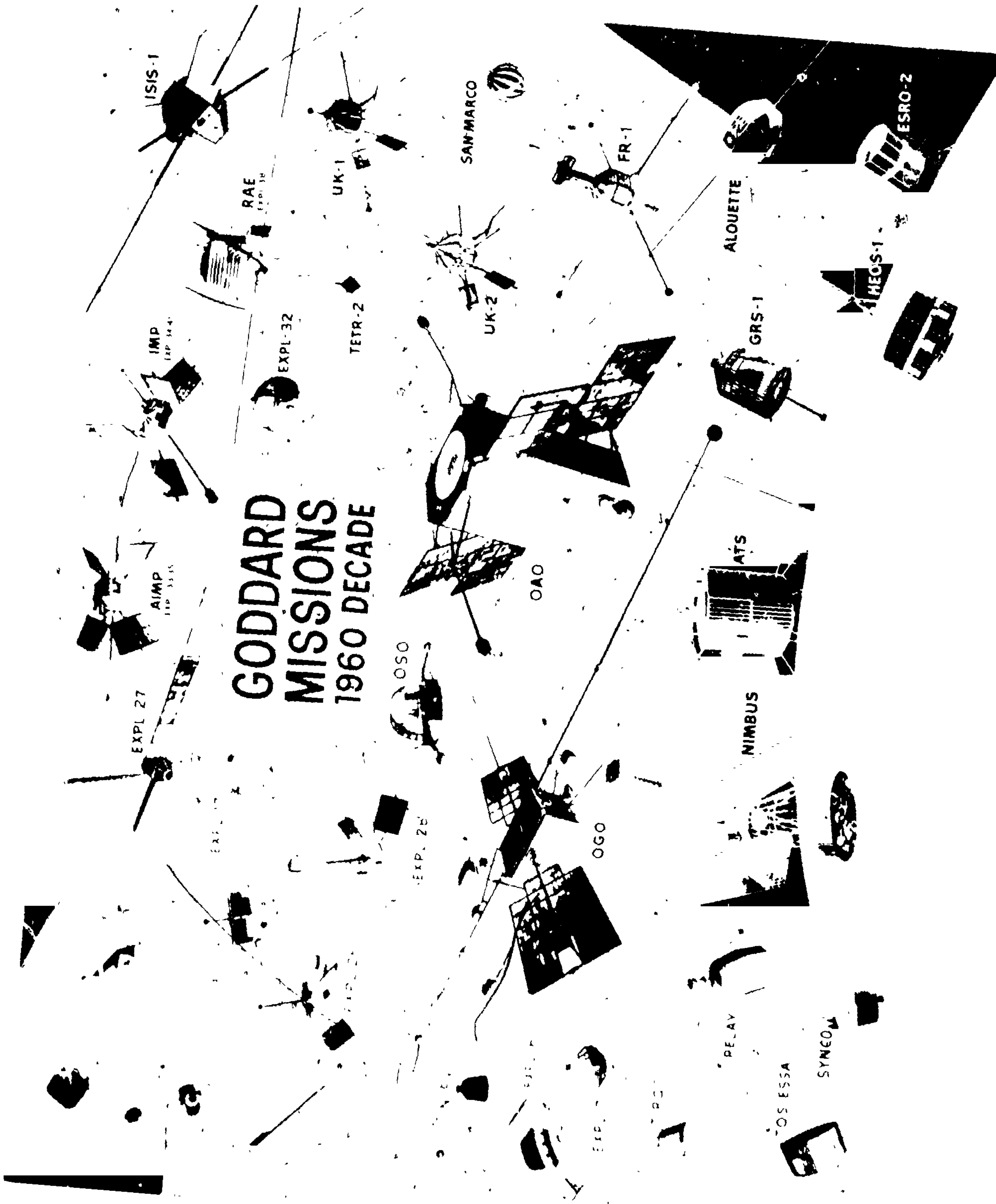

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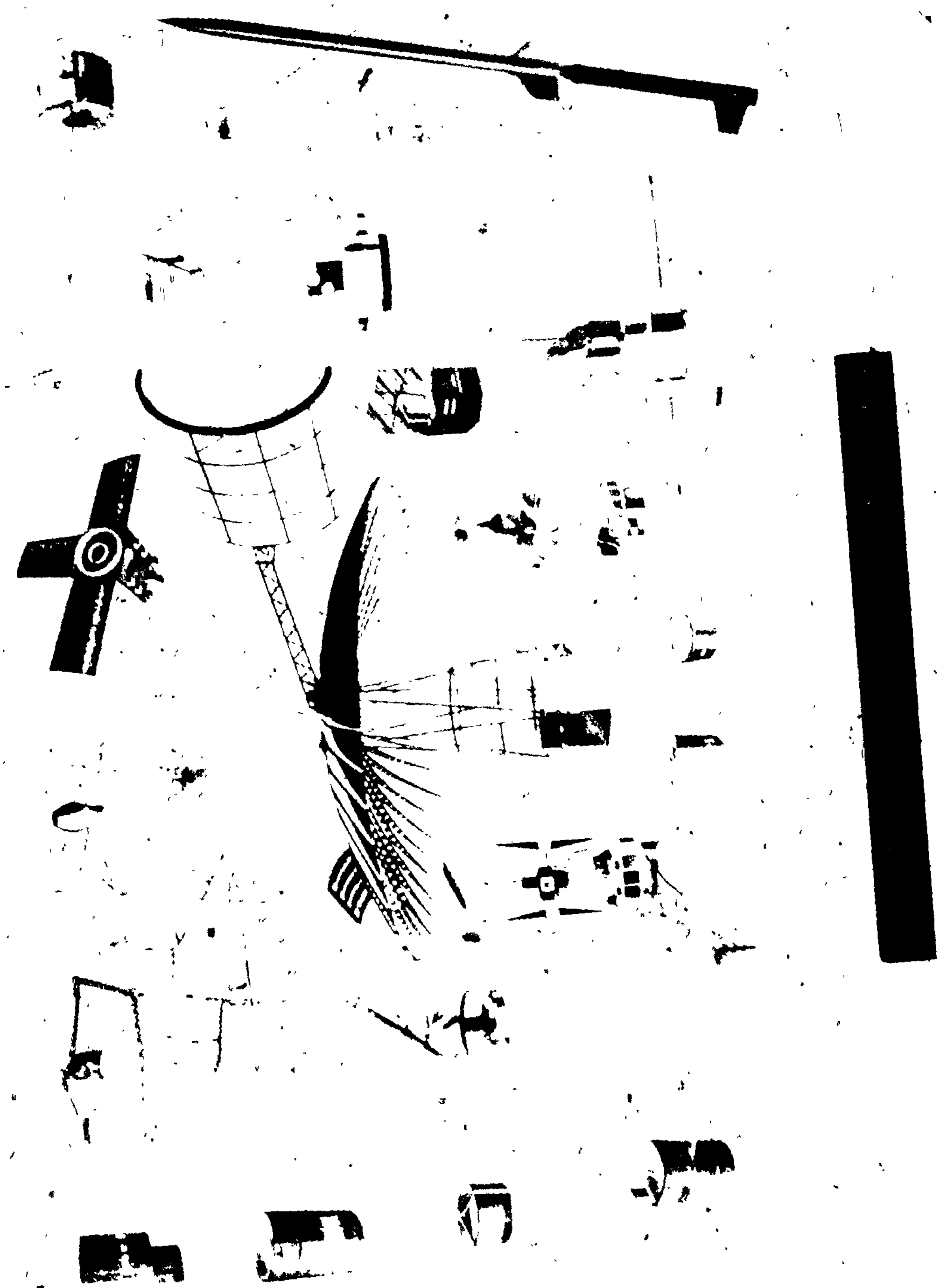

WHAT IF WE HAD USED THE SHUTTLE?


OAO-1
EARLY S/C FAILURE


OAO-2
50 MONTHS OF OPERATION

OAO-B
NOSE CONE FAILURE


OAO-3
6 + YEARS OF OPERATION

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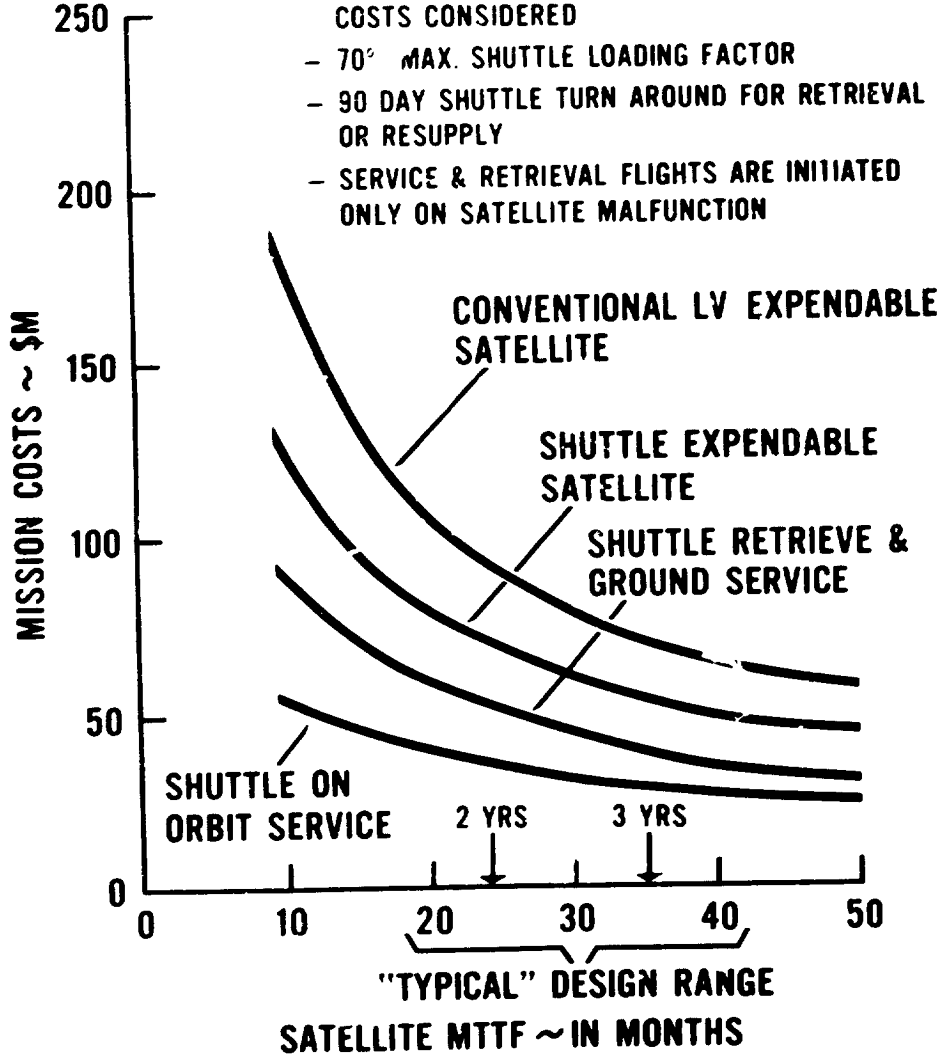
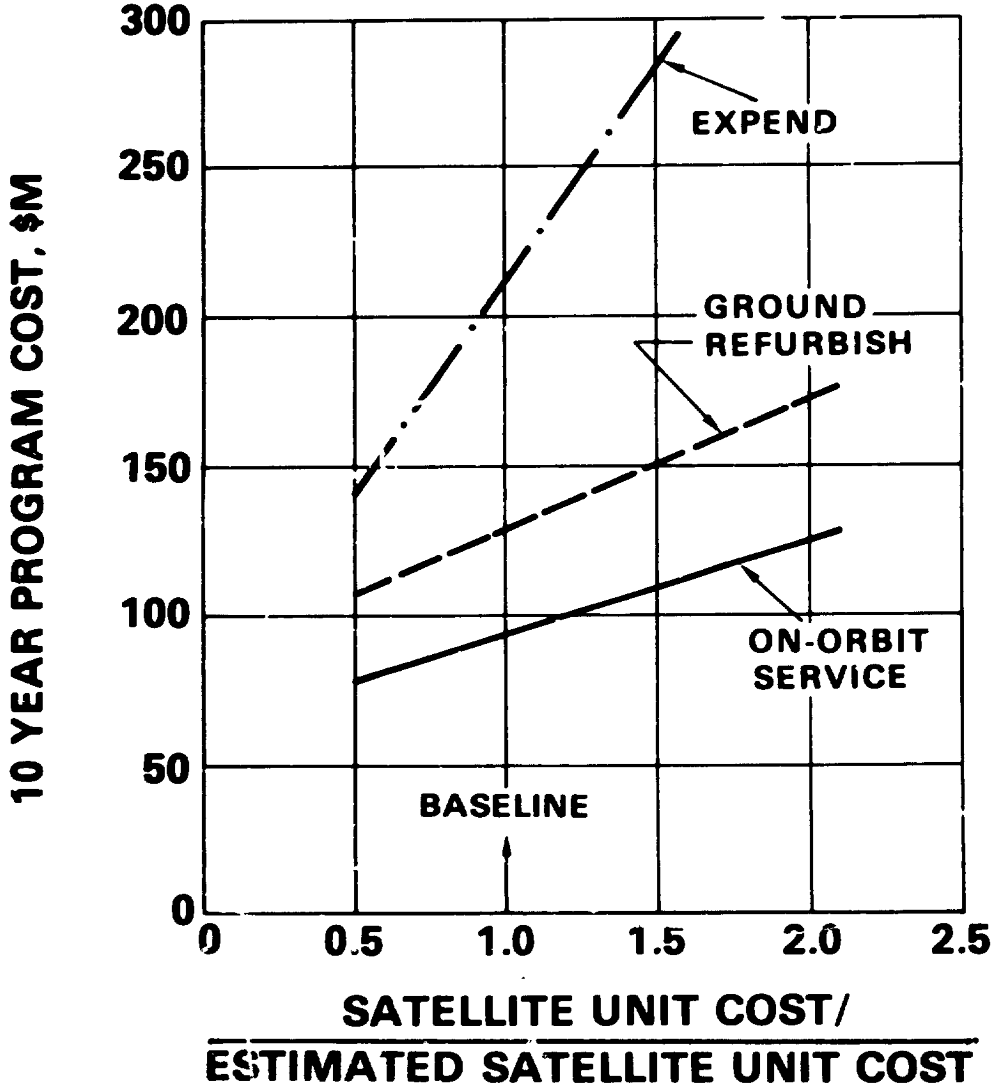
METER LARGE SPACE TELESCOPE
STRIICTURAL CONCEPT

MISSION COSTS VS SATELLITE MTTF FOR 10 YR CONTINUOUS OBSERVATIONS

ASSUMPTIONS:
- ONLY MISSION RECURRING & OPERATIONAL COSTS CONSIDERED
- 70% MAX. SHUTTLE LOADING FACTOR
- 90 DAY SHUTTLE TURN AROUND FOR RETRIEVAL OR RESUPPLY
- SERVICE & RETRIEVAL FLIGHTS ARE INITIATED ONLY ON SATELLITE MALFUNCTION



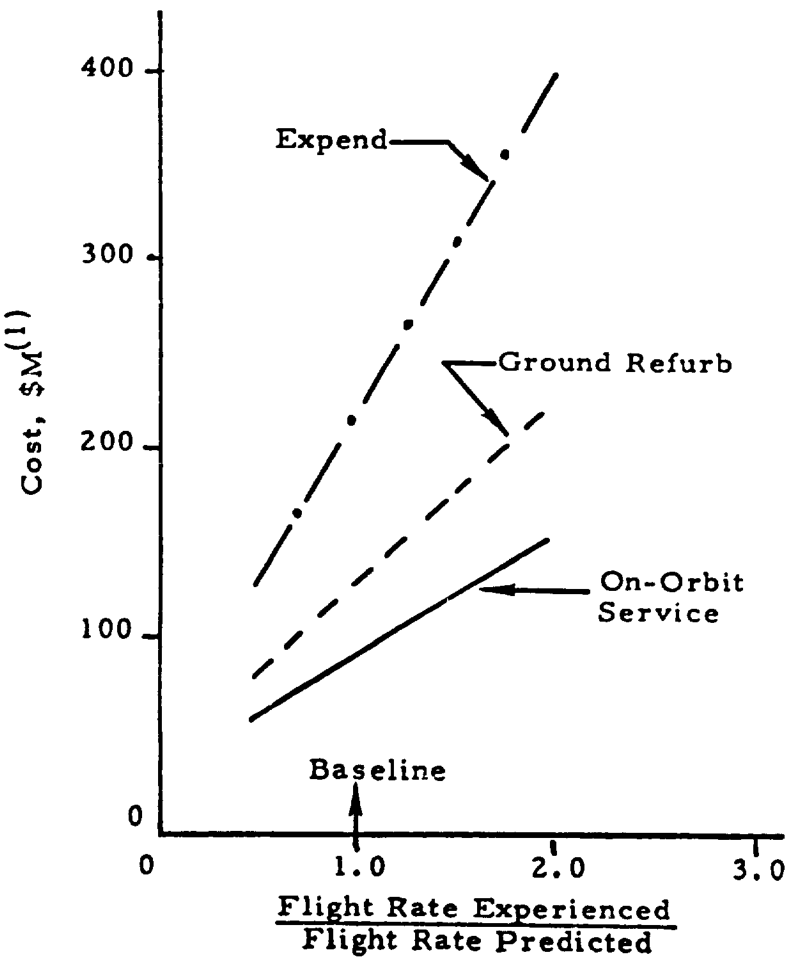
EFFECT OF SATELLITE UNIT INVESTMENT COST



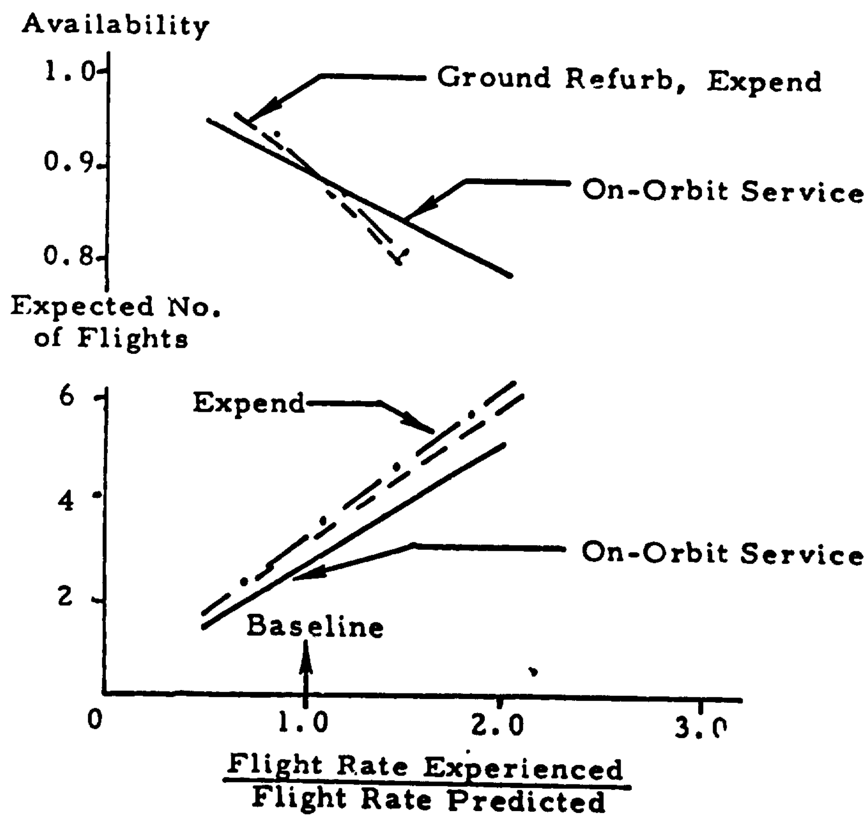
- NO ORBITAL SPARE
- BASELINE AVAILABILITY ≥ 0.90
- SATELLITE MMD = 3 YEARS
- SATELLITE ΔV = 1,000 fps
- 3 MODULE INSTRUMENT BAY FOR ON-ORBIT SERVICE SATELLITE
- SHUTTLE DELAY TIME = 5 MONTHS
- EXPECTED NO. OF FLIGHTS = 2.6 (SERVICE); 3.0 (EXPEND OR GROUND REFURBISH)

EFFECT OF SATELLITE FLIGHT RATE CHANGE DUE TO CHANGES IN SATELLITE FAILURE RATE

- No Orbital Spare
- Baseline Availability ≥ 0.90
- Satellite MMD = 3 years
- Satellite ΔV = 1,000 fps
- 3 Module Instrument Bay for On-Orbit Service Satellite
- Shuttle Delay Time = 5 Months
- Dedicated Flight User Charges



(1)Recurring Cost + DDT&E Cost



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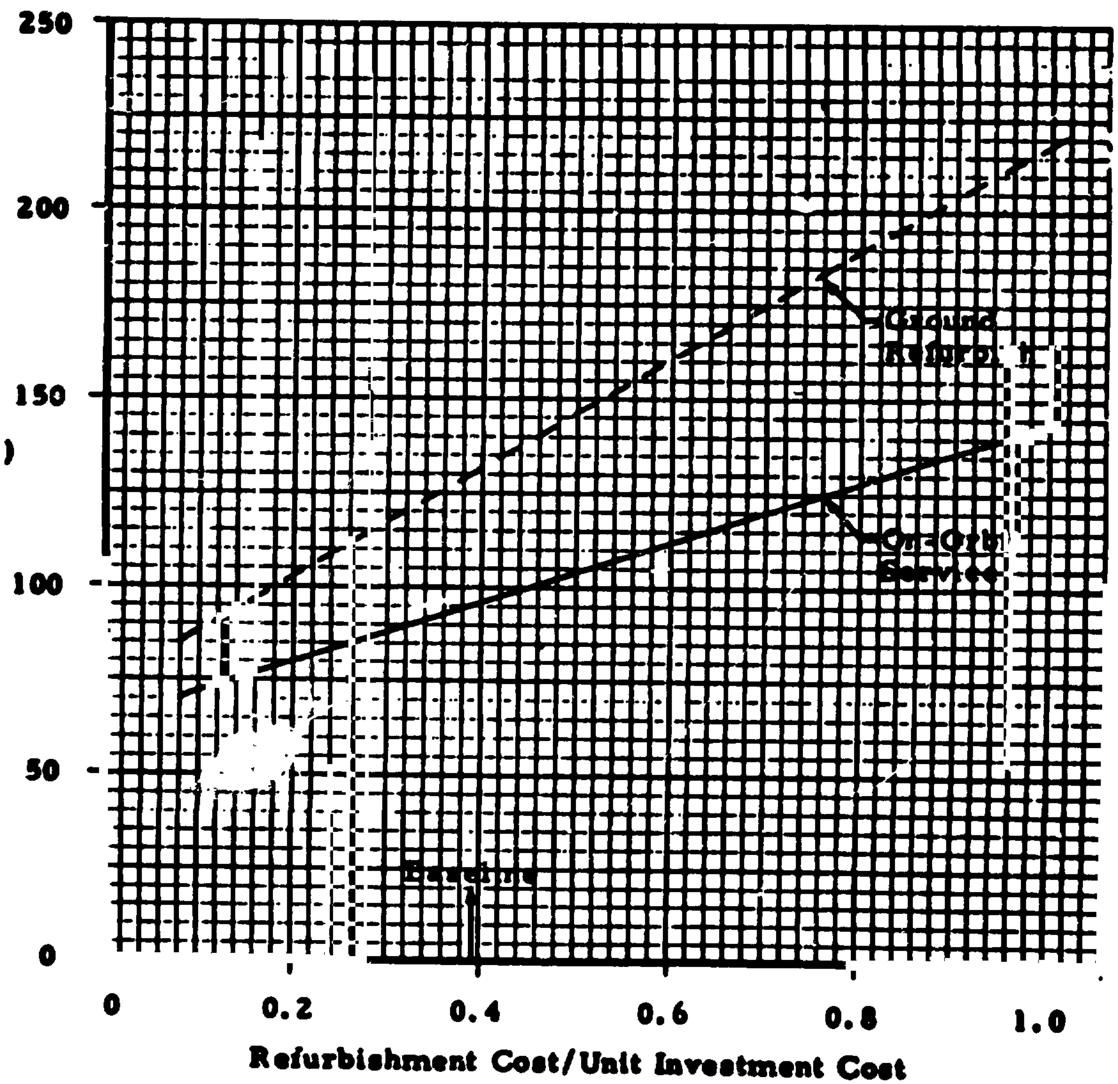
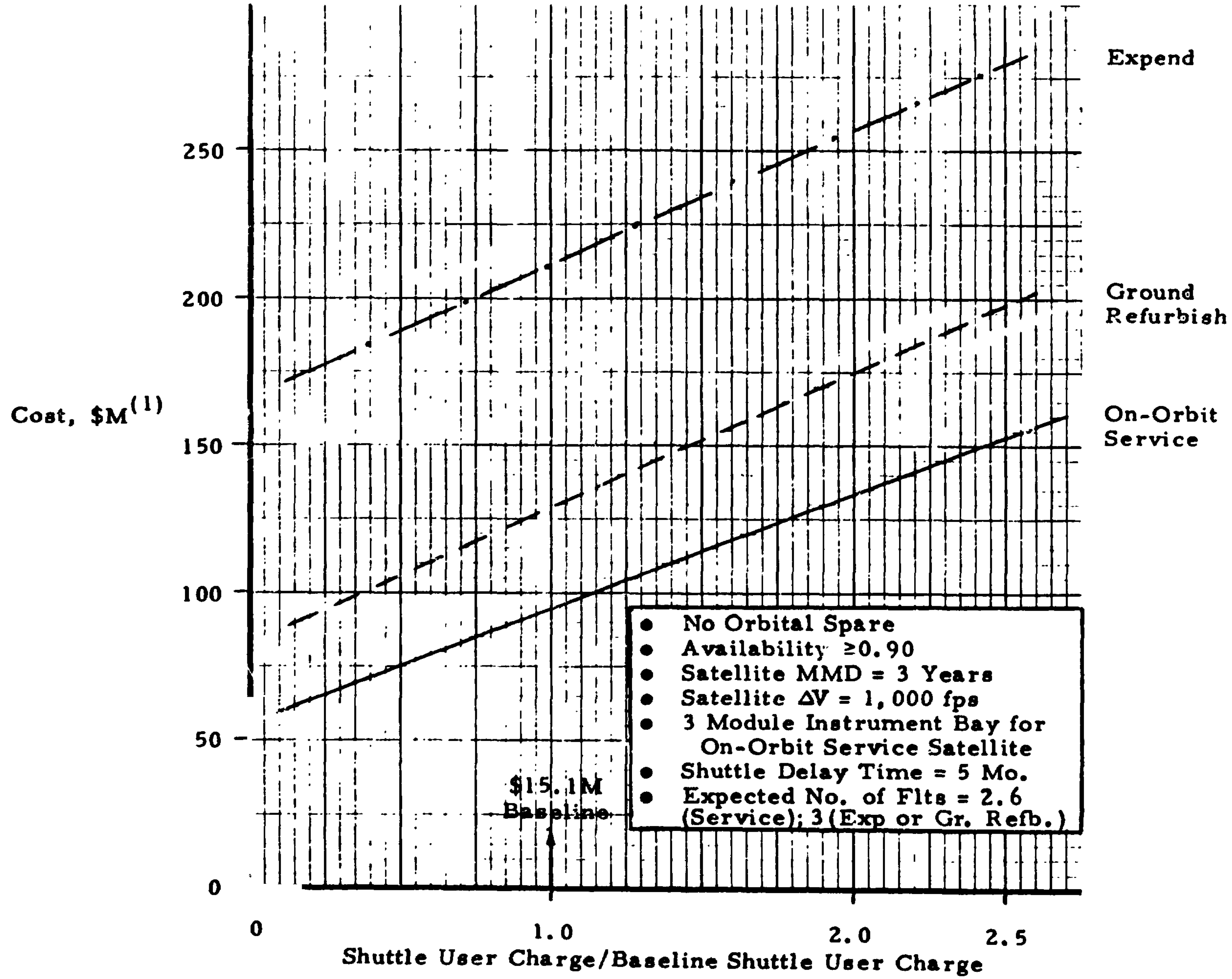
TEN YEAR PERIOD OF SHUTTLE-SUPPORTED OPERATION
EFFECT OF REFURBISHMENT COST ON OPERATIONAL MODE COMPARISON



(1) Recurring Cost + DDT&E Increment



(1) Recurring Cost + DDT&E Increment

Figure 7-3. Effect of Shuttle User Charge on Operational Mode Comparison Ten-Year Period of Shuttle-Supported Operation

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Table 3-2. Study Results - Effect of Satellite Logistics Mode, Shuttle Parking Orbits for a Ten-Year Period of Shuttle-Supported Operation with a Three-Year MMD Satellite, Availability ≥0.9, Dedicated Shuttle Flight User Charges

Mode of Operation				Number of Satellite Flights		Costs ($M)							Total Relative Cost Decrease (%)	
							Non-Recurring		Recurring					
Satellite Logistics (Satellite Design Used)	Shuttle Parking Orbit Altitude (km)	Availability	Shuttle Delay (Mo)	Retrieve & Refurbish To Establish Shuttle Supported System	Logistic Operations	Total Relative Costs	DDT&E Increment	Cost To Establish Shuttle Supported System(1)	Investment	Transportation	Satellite Refurb	Operations	Over Expend Mode	Over Ground Refurb. Mode
On-Orbit Service (3 Module Instrument Bay)	296	0.96	2	1.0	2.71	101	13	21	---	41	18	8	55	26
	705 x 185	0.96	2	1.0	2.71	97	11	19	---	41	18	8	56	27
	705	0.96	2	1.0	2.71	89	6	17	---	41	17	8	59	28
Ground Refurbish (Instrument Bay Not Modularized)	296	0.95	2	1.0	3.21	137	11	21	---	48	47	10	39	--
	705 x 185	0.95	2	1.0	3.21	133	10	19	---	48	46	10	40	--
	705	0.95	2	1.0	3.21	124	4	17	--	48	45	10	42	--
Expend (Instrument Bay Not Modularized)	296	0.96	2	---	3.23	223	8	37	119	49	--	10	--	--
	705 x 185	0.96	2	---	3.23	220	7	36	118	49	--	10	--	--
	705	0.96	2	---	3.23	215	4	36	116	49	--	10	--	--

(1) Assuming reuse of Delta-launched satellites

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ESTIMATE OF SATELLITE COST REDUCTIONS WITH SHUTTLE OPERATION
TYPICAL NASA PROJECTS FROM THE 1960s & 1970s

- POSTULATE: AUTOMATED SATELLITE PROJECTS FROM THE 1960s, 1970s
  TEN YEARS OF OPERATION FOR EACH PROJECT
  SHUTTLE OPERATIONS AT ETR & WTR
  MULTIMISSION MODULAR SPACECRAFT (MMS OR SMMS)
  ON-ORBIT SERVICE OPERATIONS
- SATELLITE PROJECTS

NAME	MMS	SMMS	ON-ORBIT SERVICE
ERTS/LANDSAT	✓		✓
HEAO	✓		✓
TIROS/TOS/ESSA/ITOS/NOAA	✓		✓
OGO*	✓		✓
OSO	✓		✓
OAO	✓		✓
NIMBUS	✓		✓
SEASAT	✓		✓
ATS*	✓		✓
EXPLORERS		✓	

*HALF OF LAUNCHES SERVICEABLE.

COST CATEGORY	COST	REDUCTION	COST REDUCTION DRIVER
DDT&E	1.9	0.8	SATELLITE STANDARDIZATION
UNIT PROCUREMENT, REFURB., & OPS.	2.8	1.0	FEWER SATELLITE MODULES FLOWN, MANY MODULES REUSED
TOTAL	4.7	1.8	--

- SATELLITE COST ESTIMATES (BILLIONS OF 1976 DOLLARS)

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CHRONOLOGY OF SERVICING ECONOMIC STUDIES

YEAR	COMPANIES	TITLE DESCRIPTION	STUDY CONDUCTED FOR	BASIC FINDINGS: COST BENEFICIAL	BASIC FINDINGS: NOT COST BENEFICIAL
1970	GRUMMAN AEROSPACE CORP.	OAO LST SHUTTLE ECONOMIC STUDY	GSFC, OSS	X	
1971	LOCKHEED CORP	PAYLOAD WEIGHT COST EFFECTS	HDQTRS OSF	X	
1972	PLANNING RESEARCH CORP	USE OF STS TO AVOID S C FAILURES	HDQTRS OSF	X	
1973	ROCKWELL CORP.	SPACE SERVICING AN OPPORTUNITY TO REDUCE SPACE PROGRAM COSTS	INTERNAL STUDY	X	
1973/ 1974	TRW	EOS SYSTEM DEFINITION STS UTILIZATION TRADE-OFFS	GSFC, OA	X	
	GENERAL ELECTRIC CORP.	EOS SYSTEM DEFINITION STS UTILIZATION TRADE-OFFS	GSFC, OA	X	
	GRUMMAN AEROSPACE CORP.	EOS SYSTEM DEFINITION STS UTILIZATION TRADE-OFFS	GSFC, OA	X	
	AEROSPACE CORP.	EOS IN ORBIT SERVICING ECONOMIC STUDY	HDQTRS O	X	
1975/ 1976	MARTIN MARIETTA CORP.	INTEGRATED ORBITAL SERVICING STUDY	MSFC, O	X	
1976	IDA	DCS III, DSP CONCEPT STUDY	DOD	X (FOR LOW EARTH ORBIT)	
1977	AEROSPACE CORP.	BRAVO ECONOMIC STUDY OF LANDSAT FOLLOW-ON	GSFC, OA	X	

DESIGN OBJECTIVES OF THE MULTIMISSION MODULAR SPACECRAFT

- O RETRIEVAL
- O MULTIMISSION CAPABILITY
- O STANDARD FLIGHT SUPPORT SYSTEM
- O STANDARD HARDWARE
- O REPAIR AND REFURBISHMENT ON-ORBIT
- O INSTRUMENT REPLACEMENT
- O STANDARD GROUND SUPPORT SYSTEM
- O STANDARD SOFTWARE

STS/SPACECRAFT VERIFICATION TESTS FOR IN ORBIT REPAIR & MAINTENANCE

Tests	Use of MMS/FSS as Baseline	MMS Subsys/FSS as Baseline	Use FSS Only, Unique S/C	Totally Unique System
Spacecraft Compatibility Tests	25%	75%	90%	100%
POCC/MCC End-to-End Test	75	90	100	100
CITE Validation	75	100	100	100
Crew Equipment Interface Test	25	50	100	100
Water Tank Training	25	75	90	100
Six Degree of Freedom (SOS) Simulations	50	100	100	100
Manipulator Develop Facility and Air Bearing Test	25	80	80	100
Lighting Test	50	100	100	100
Shuttle Mission Simulations	50	50	80	100

LOGISTICAL HARDWARE/SOFTWARE ELEMENTS REQUIRED FOR S/C

ON-ORBIT REPAIR AND MAINTENANCE

	Use of MMS/FSS as Baseline	MMS Subsys/FSS as Baseline	Use FSS Only, Unique S/C	Totally Unique System
STS Training and Development Mockups				
• Instrument Module	100%	100%	100%	100
• Spacecraft Bus	0	50	100	100
• Airborne Support Equip. (FSS)	0	0	0	100
• Subsystem Module	0	0	100	100
• Module Servicing Tool	0	0	100	100
• Experiment Repair Kits	100	100	100	100
• 1-G Simulator	25	70	80	100
Spare Spacecraft Hardware	0	40	100	100
Airborne Support Equipment (FSS Cradle A')	0	0	25	100
Spacecraft Servicing Tools				
• Module Service Tools	0	0	100	100
• Standard Tool Rack	0	0	50	100
• Module Storage Rack	0	9	100	100
• Portable Foot Restraints	10	50	50	100
Ground Support Equipment	0	40	80	100
Spacecraft Hanger Queen for Final Verification				
• Structure Interfaces	0	100	100	100
• Electrical Interfaces	0	100	100	100
• Spacecraft Electrical Simulator	0	50	100	100
Analyses ([illegible] Therm. EMI/EMC)	40	70	80	100

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STS ANALYTICAL INTEGRATION AS A FUNCTION OF CARRIER HARDWARE CONFIGURATION

Analytical Elements	Mission Hardware Configuration: Use of MMS/FSS as Baseline	MMS Subsys/FSS as Baseline	Use FSS Only, Unique S/C	Total Unique System
Safety Reviews/Hazzard Analysis	30%	50%	70%	100%
Interface Control Documents MMS/FSS Baseline Mission Unique Appendix	30	50	70	100
Payload Integration Plan	70	80	90	100
Launch				
Retrieval				
Servicing (Repair)				
Payload Integration Annexes				
Payload Data Package	30	50	70	100
Flight Planning	80	90	90	100
Flight Operations Support	80	90	90	100
Orbiter Command & Data	50	75	100	100
Payload Operations Control Center	50	75	100	100
Orbiter Crew Compartment	25	25	25	100
Training	30	70	90	100
Launch Site Support Plan	70	90	90	100
Payload Interface Verfication Summary	25	50	75	100
EVA	20	70	90	100
Hardware/Software Integration Reviews				
Cargo Integration Review	100	100	100	100
Integration Hardware Software	50	75	90	100
Flight Operations	100	100	100	100
Ground Operations	100	100	100	100
Flight Readiness	100	100	100	100

ECONOMIC IMPORTANCE OF THE REPAIR MISSION

- The United States has maintained continuous observations of the sun from space since 1962

- The Space Shuttle will restore and prolong the life of this country's only solar observatory

- The demonstration of this capability is both technical and economic:

  - SMM cost in FY83 dollars is $216M *
  - The cost to perform this repair is 40M *
  - The scientific return on SMM will be enhanced by 2 to 3 years
  - Therefore the savings will be 176M

- If sucessfully restored, the SMM observatory will provide an additional two to three years of solar observation

* Excludes launch costs

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PRESENTATION ON THE
MULTIMISSION MODULAR SPACECRAFT (MMS) DESIGN
TO THE SATELLITE SERVICES WORKSHOP
ON
JUNE 23, 1982

R. E. Davis
Goddard Space Flight Center

PRESENTATION ON THE

SERVICING ECONOMICS FOR LANDSAT

TO THE SATELLITE SERVICE WORKSHOP

ON

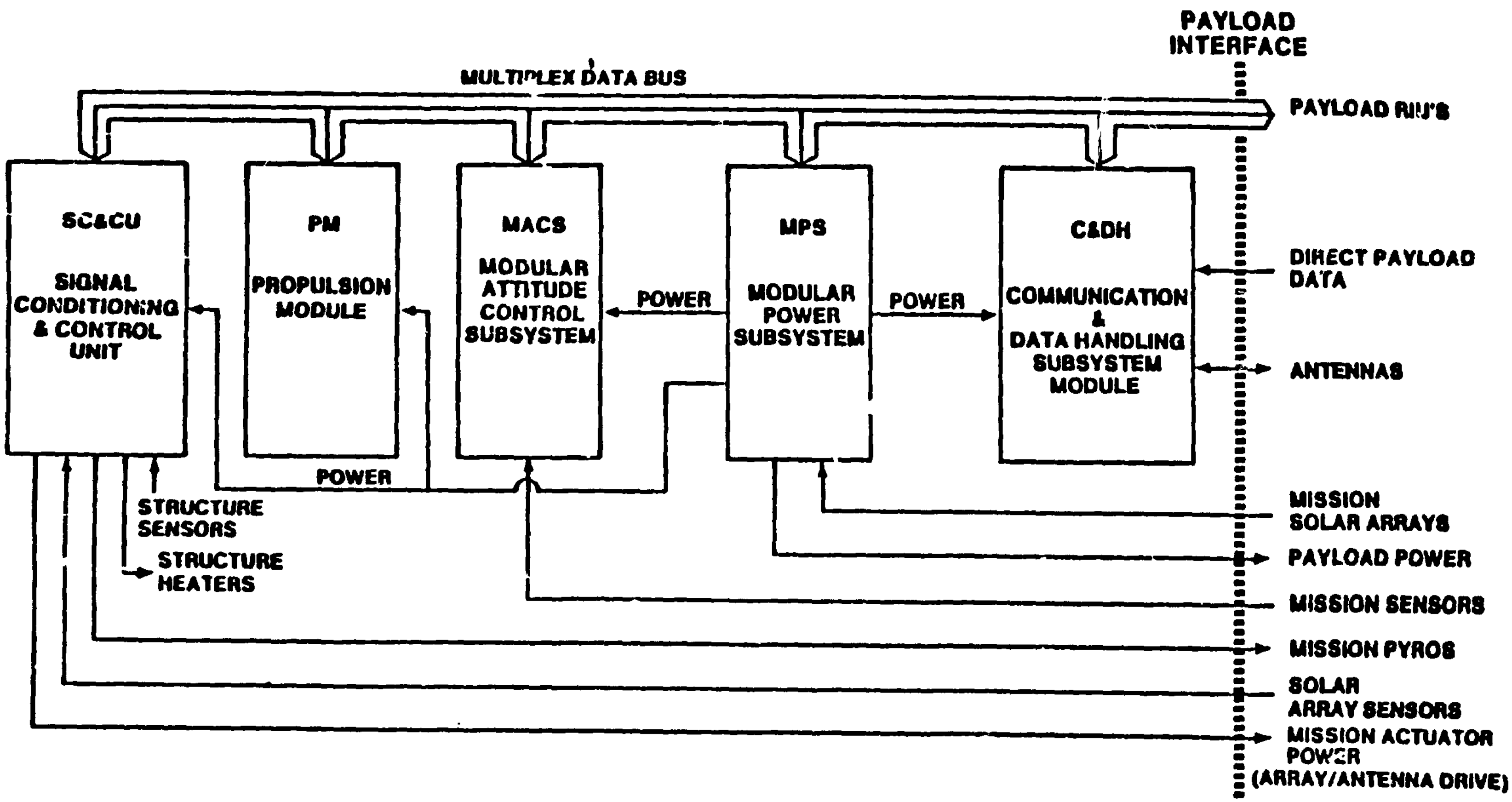
JUNE 23, 1982

R. E. Davis
Goddard Space Flight Center

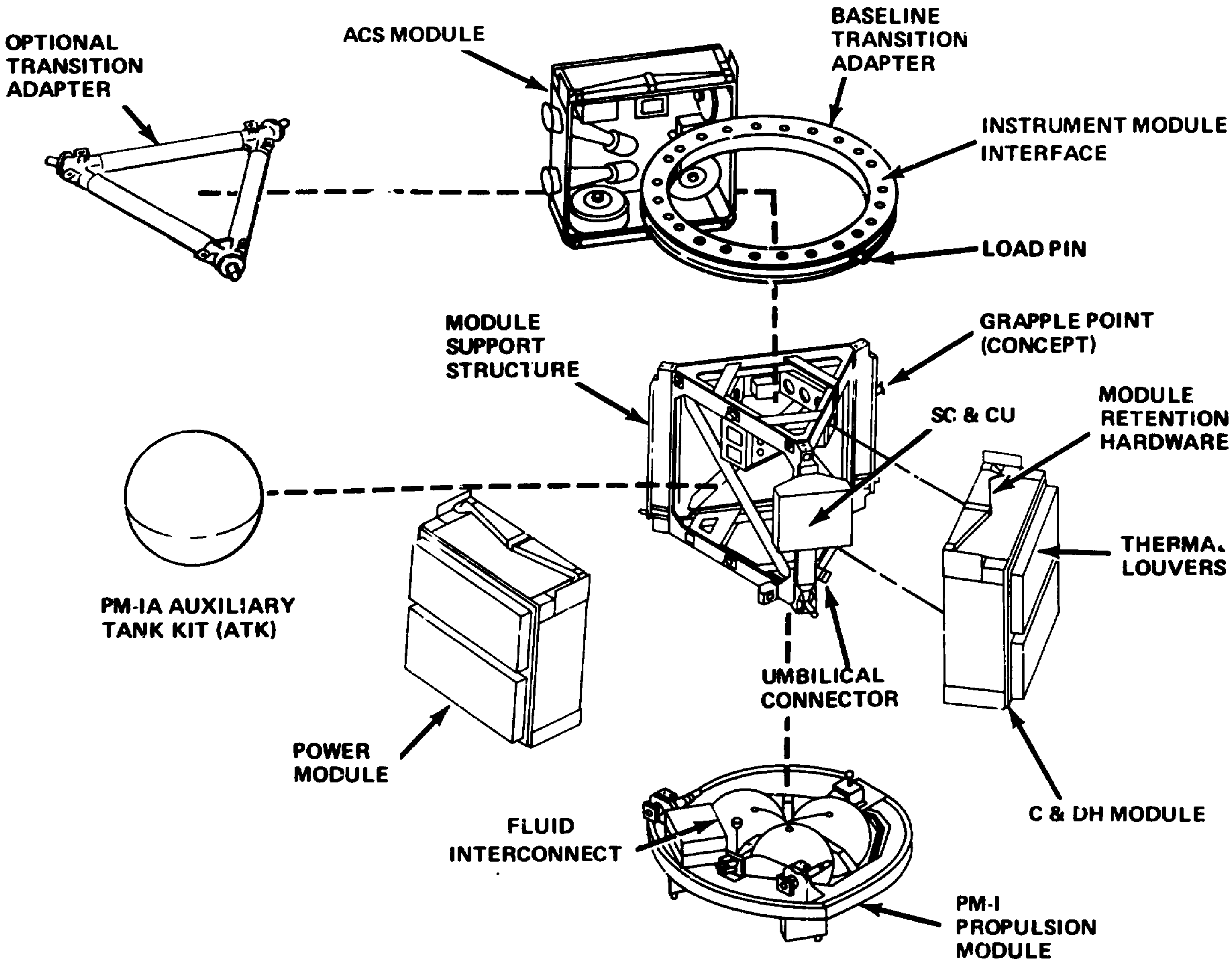
DESIGN OBJECTIVES OF THE MULTIMISSION MODULAR SPACECRAFT

- O RETRIEVAL
- O MULTIMISSION CAPABILITY
- O STANDARD FLIGHT SUPPORT SYSTEM
- O STANDARD HARDWARE
- O REPAIR AND REFURBISHMENT ON-ORBIT
- O INSTRUMENT REPLACEMENT
- O STANDARD GROUND SUPPORT SYSTEM
- O STANDARD SOFTWARE

MMS FUNCTIONAL BLOCK DIAGRAM.



MODULAR SUBSYSTEM COMPONENTS OF THE MMS



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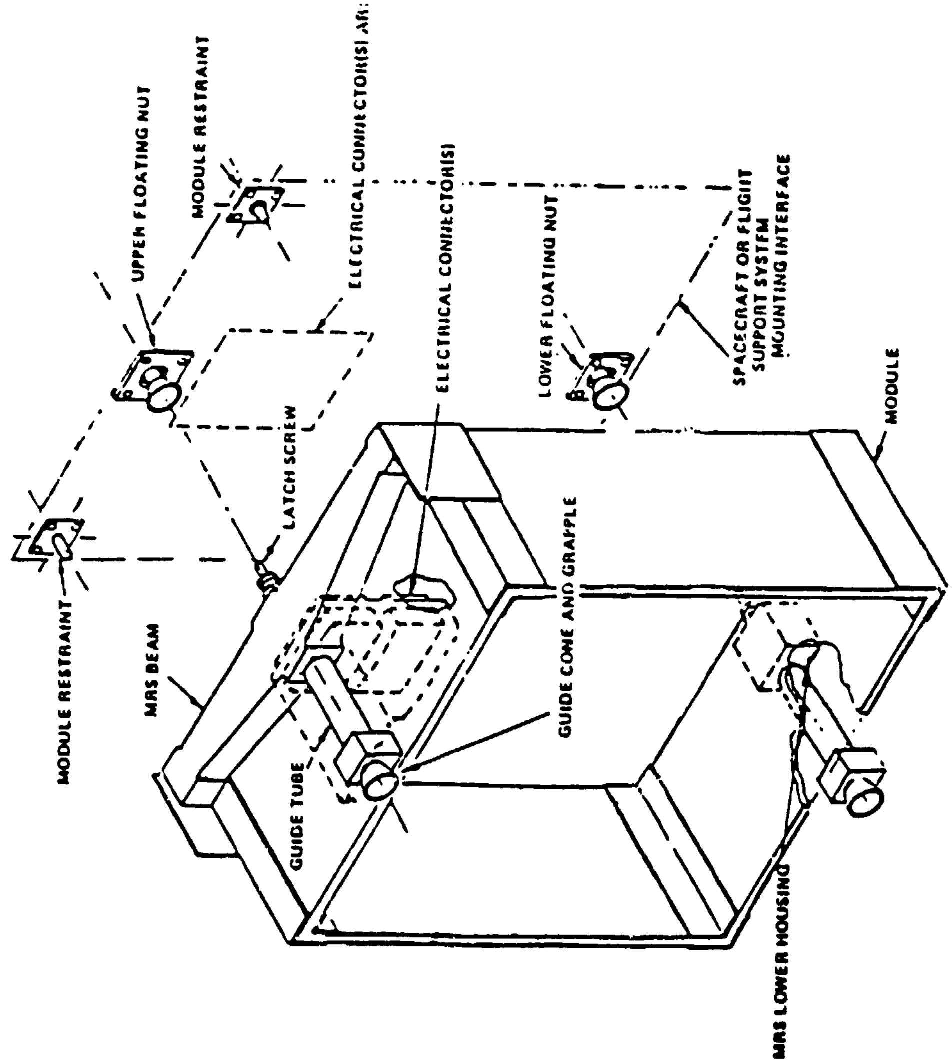

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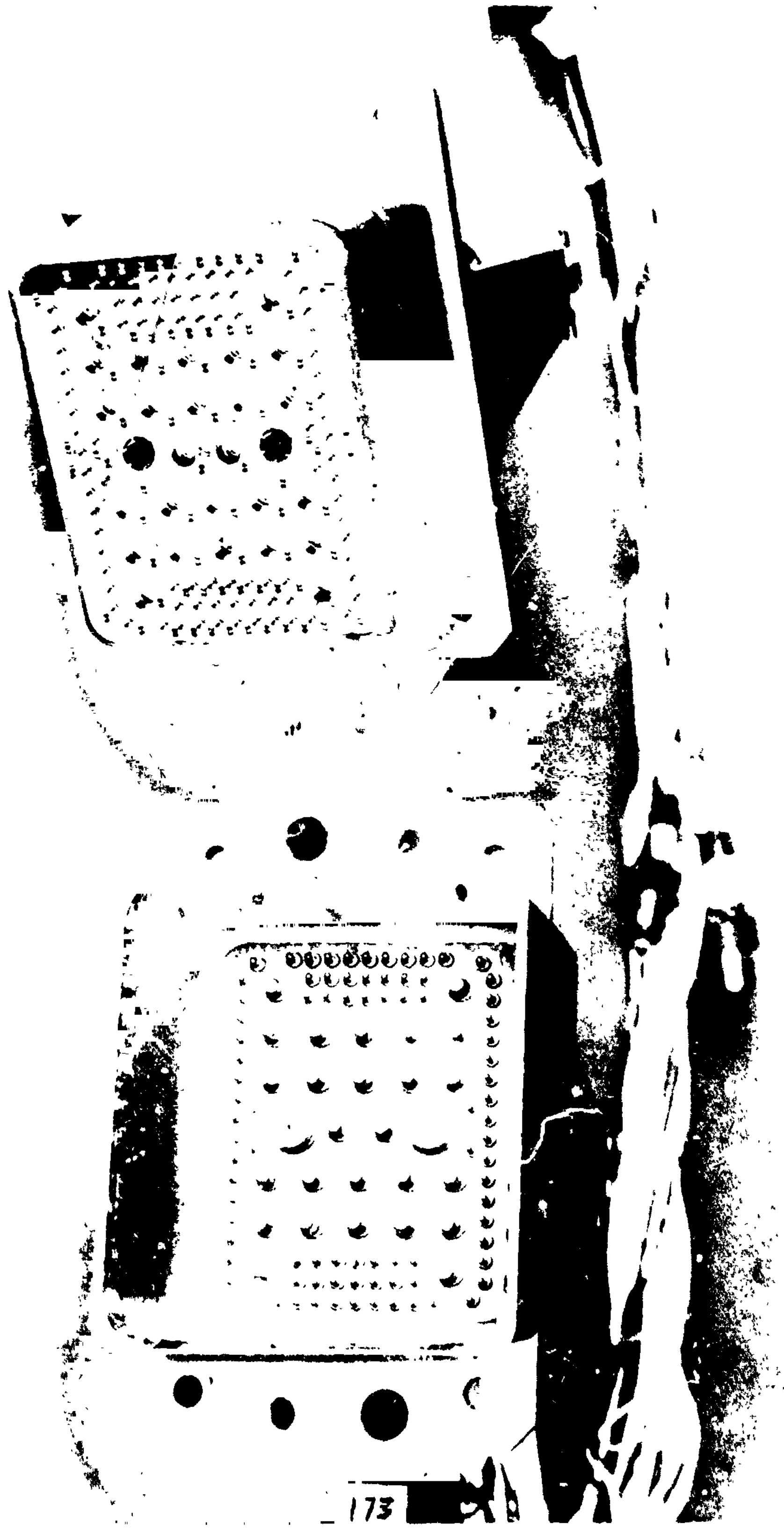

NASA

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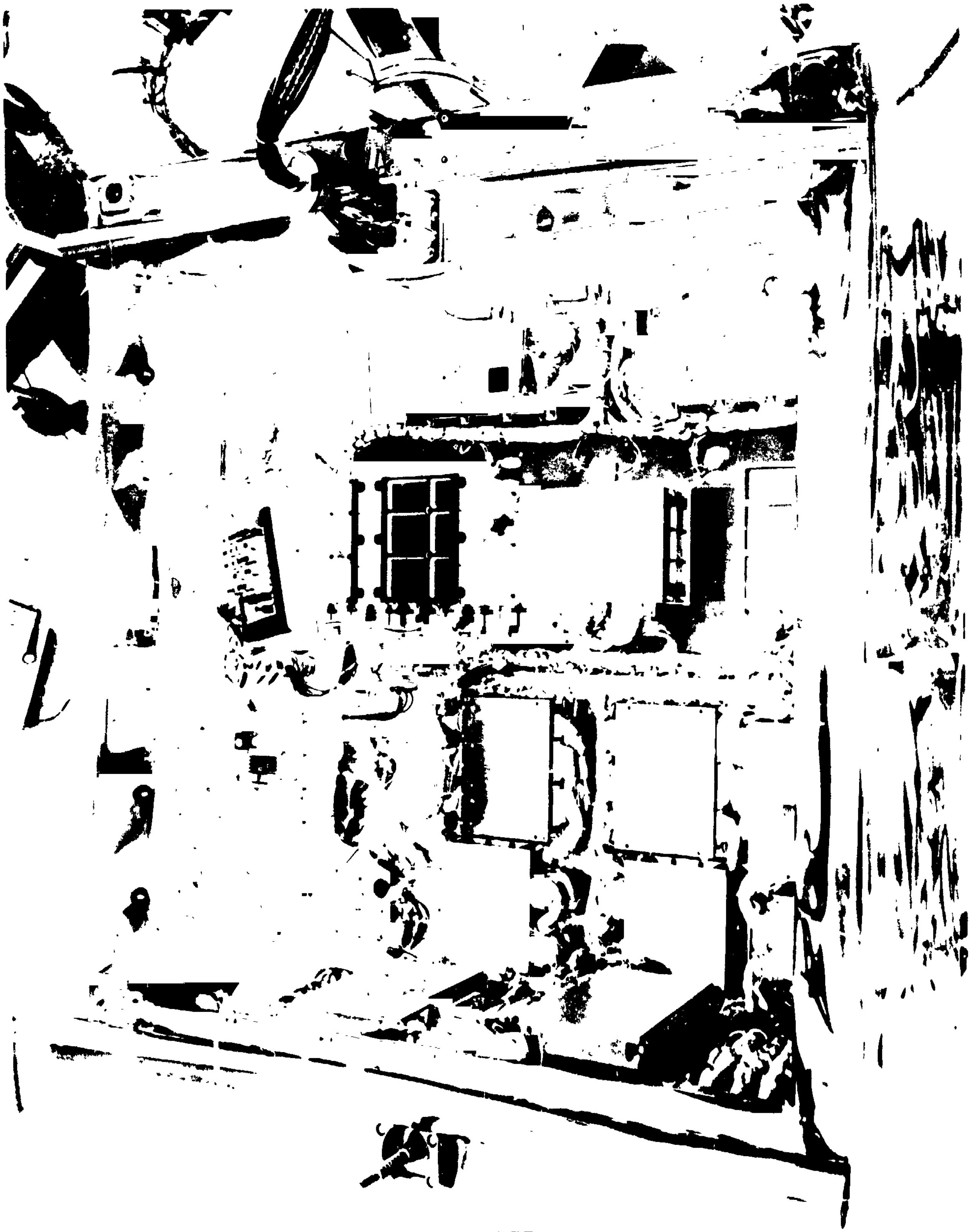

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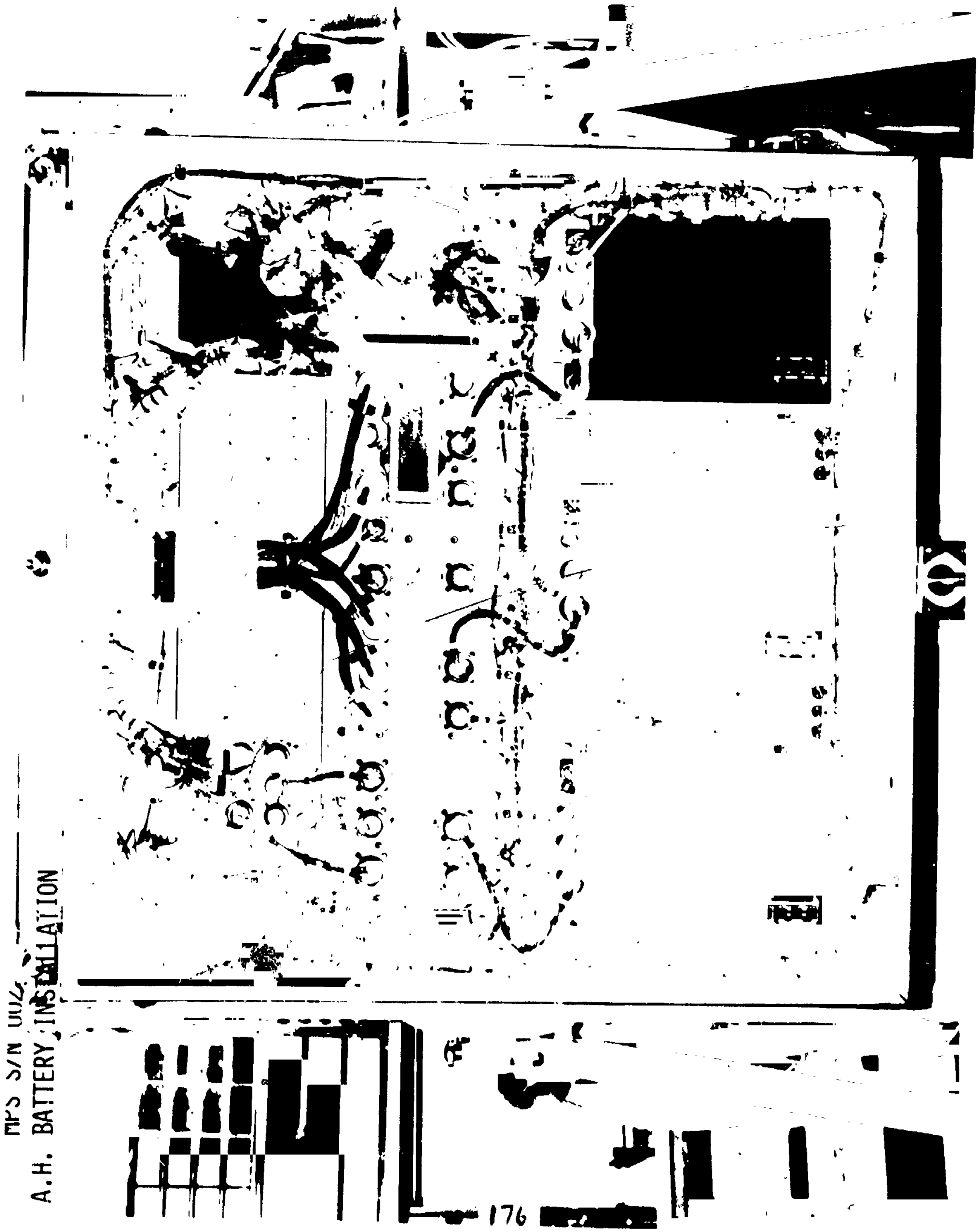

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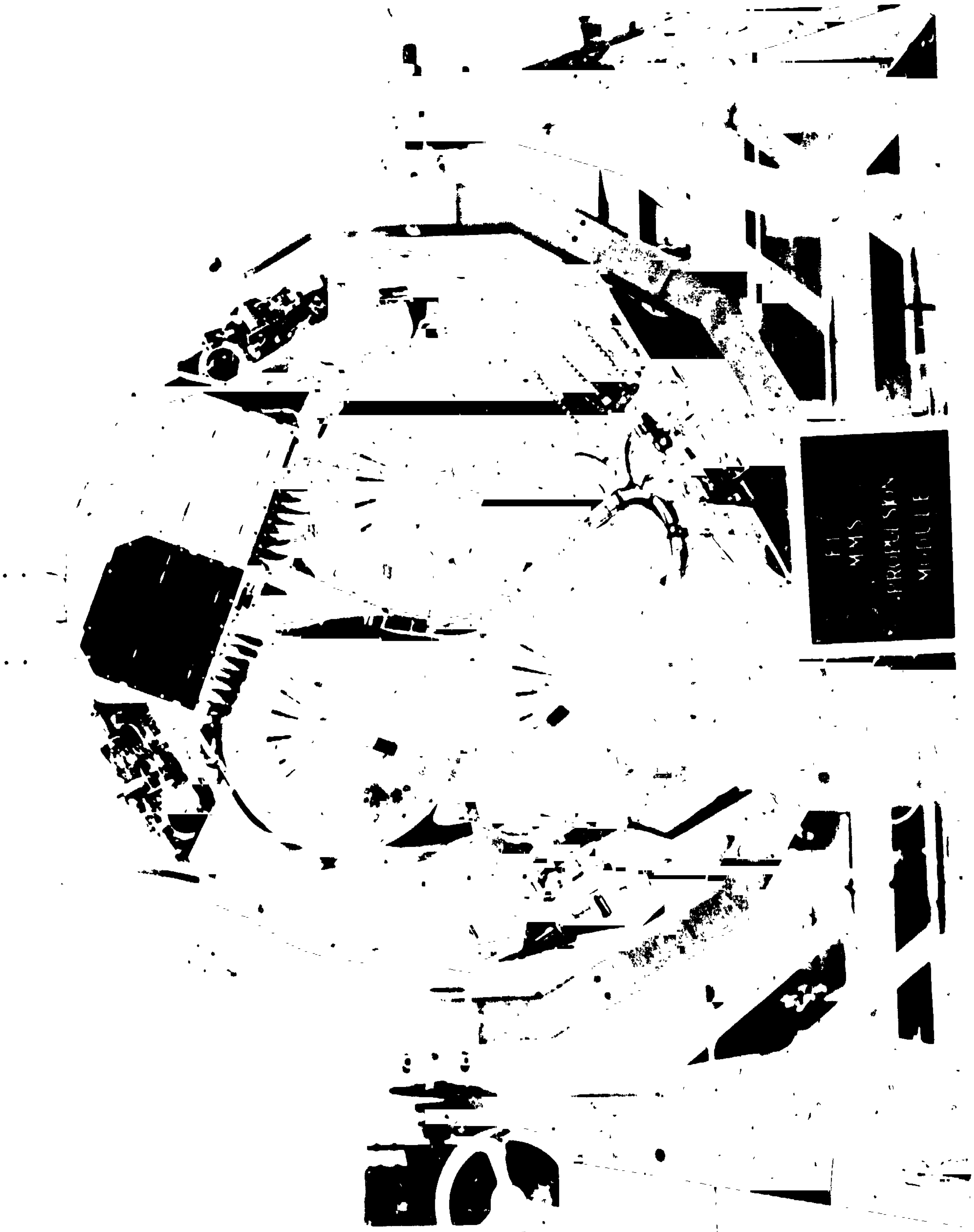

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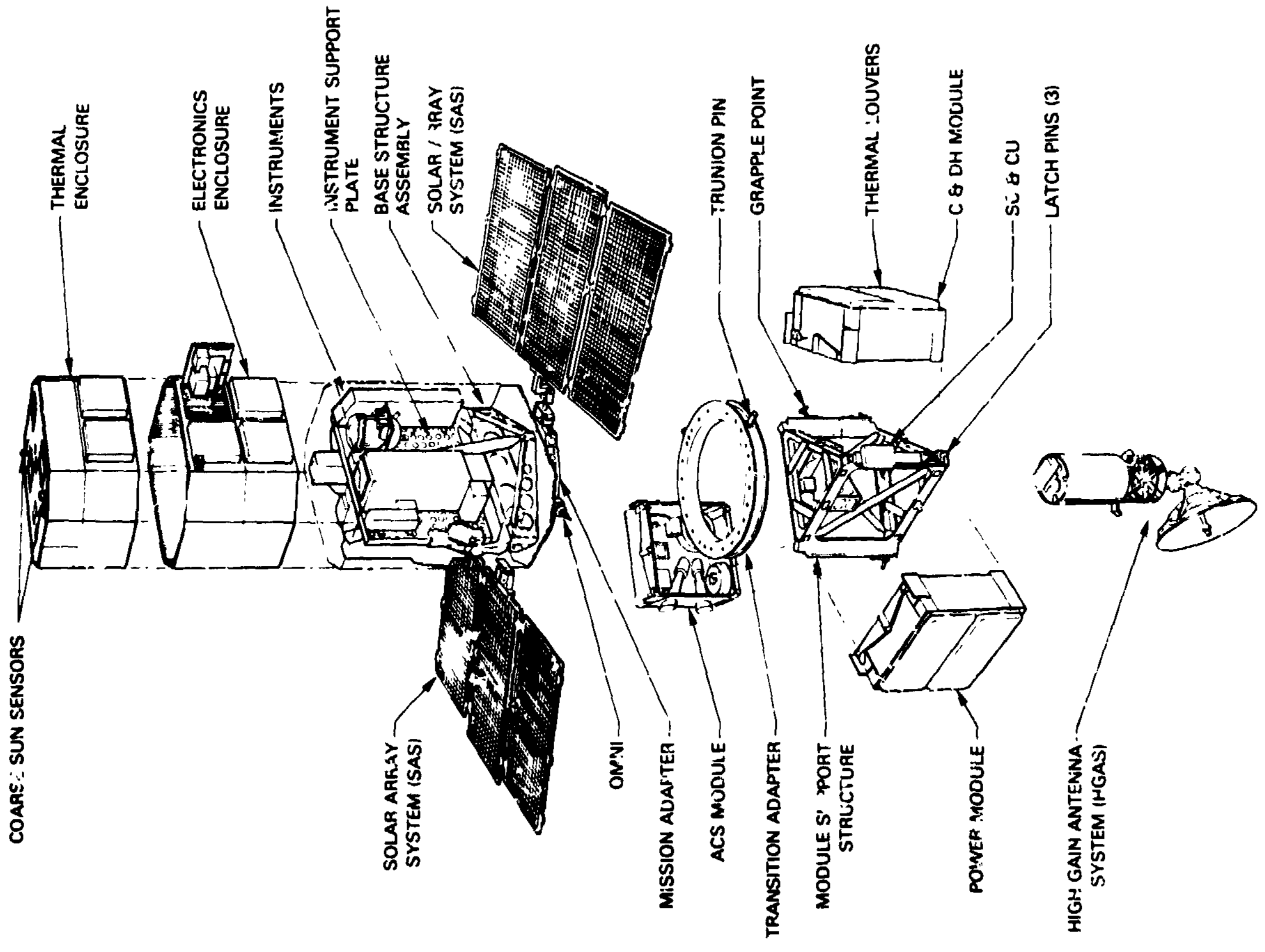
SMM OBSERVATION EXPLODED VIEW



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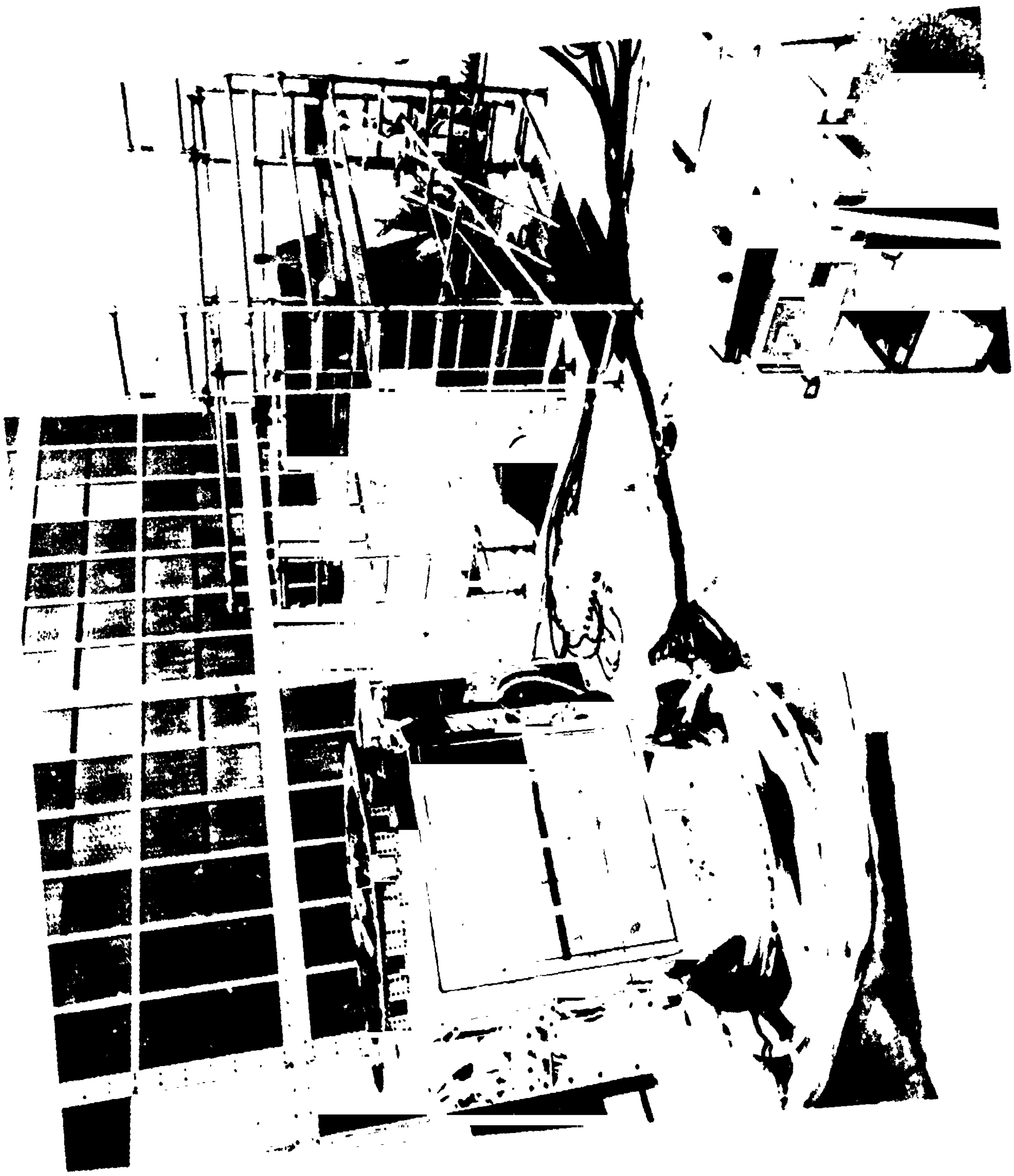

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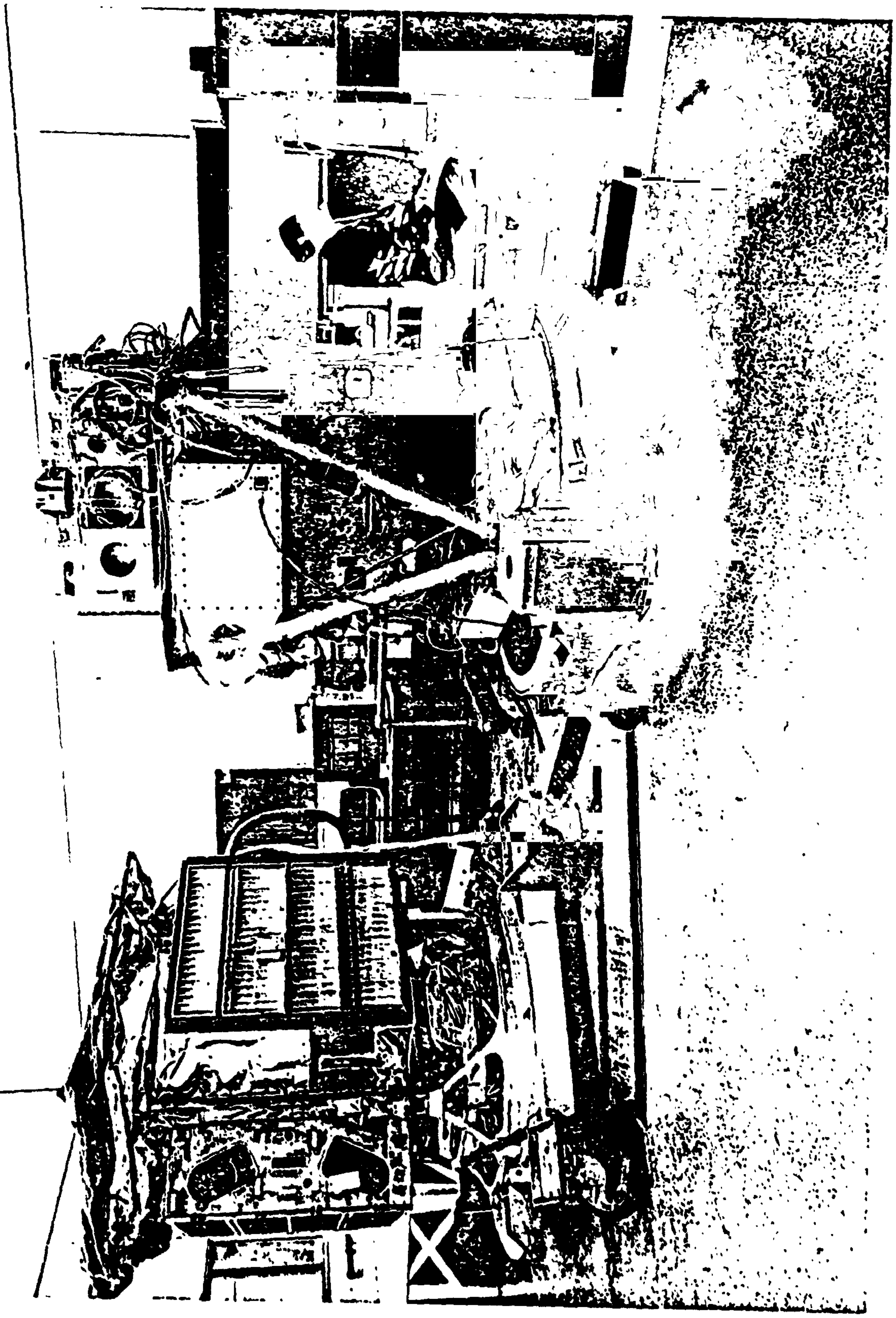

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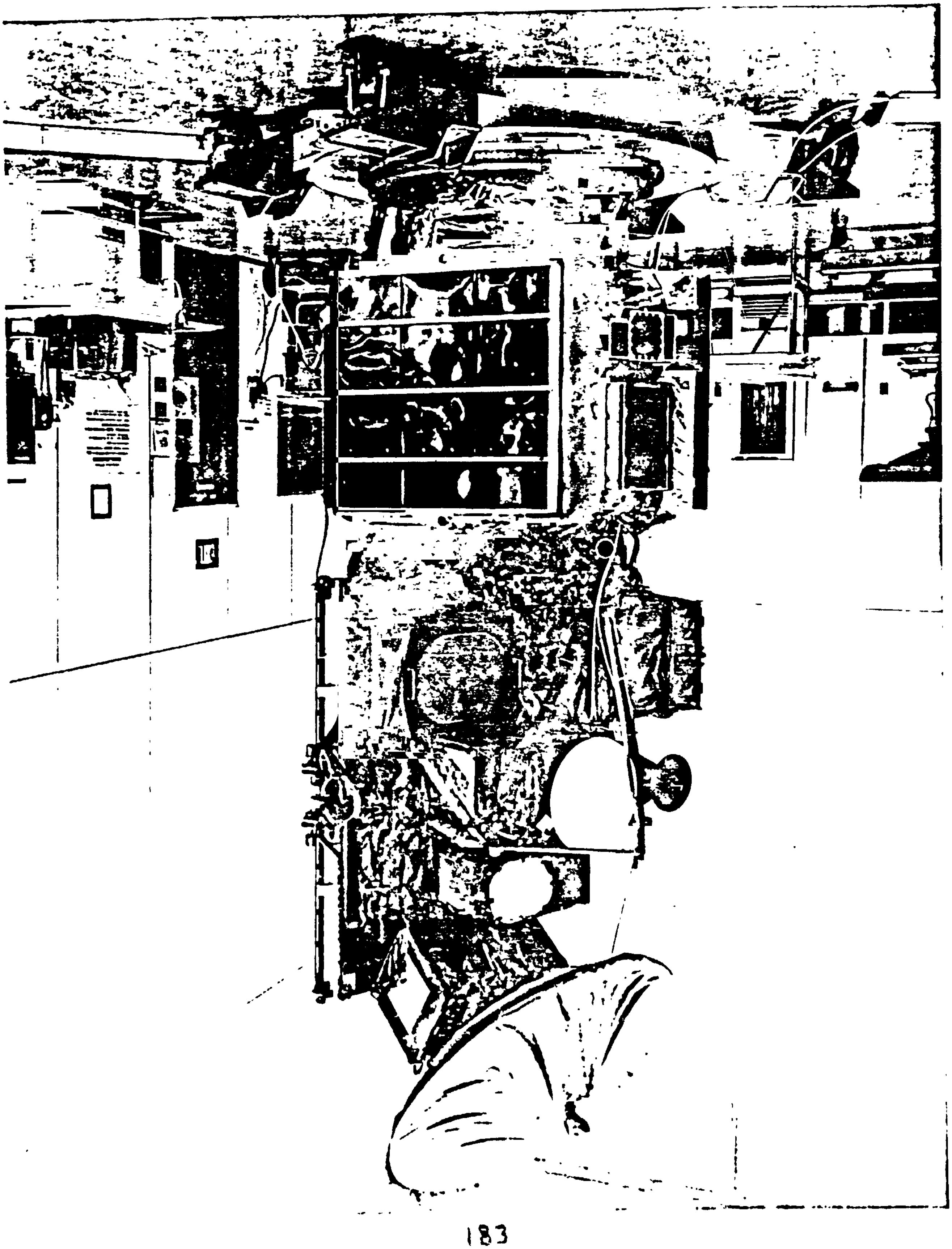

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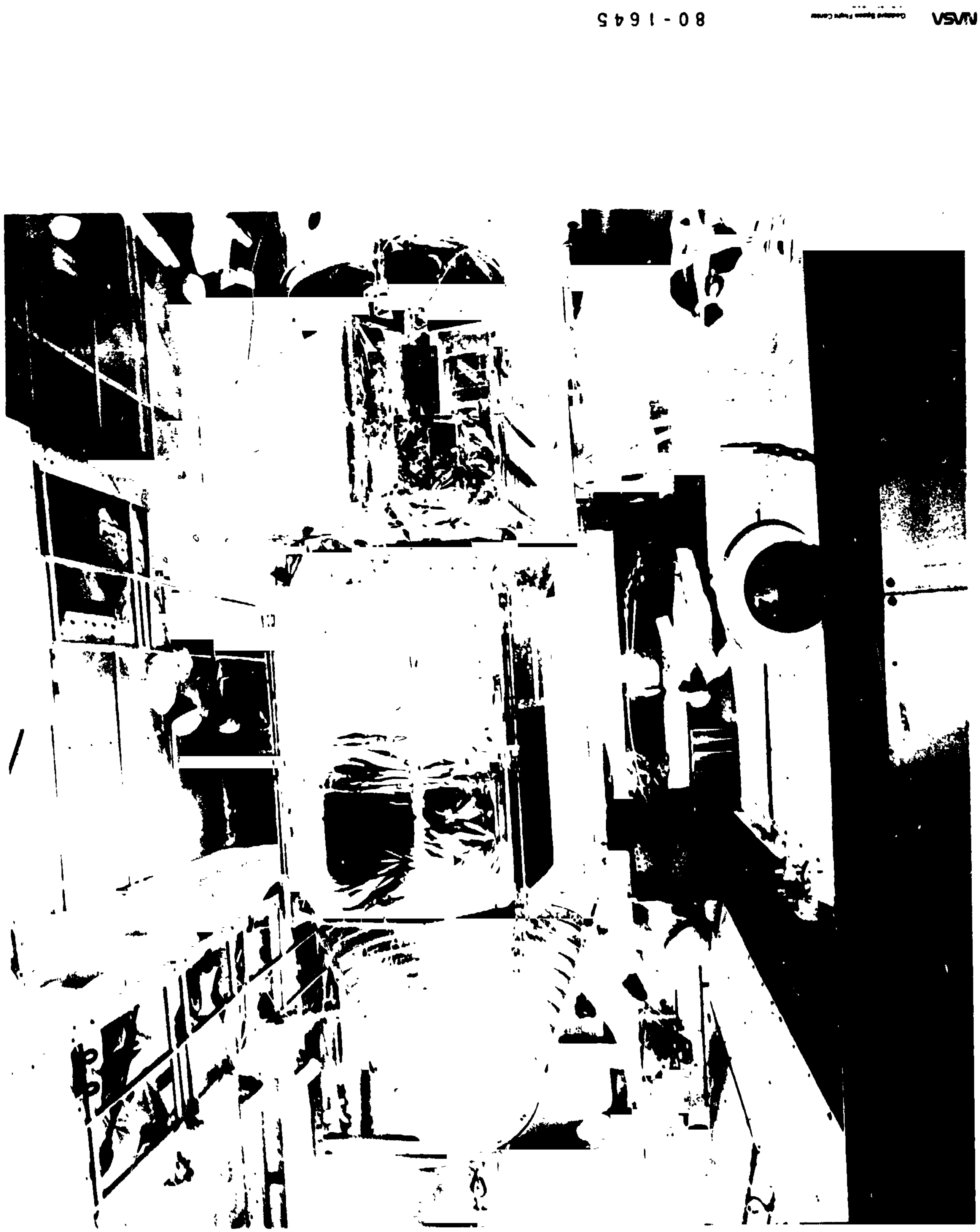

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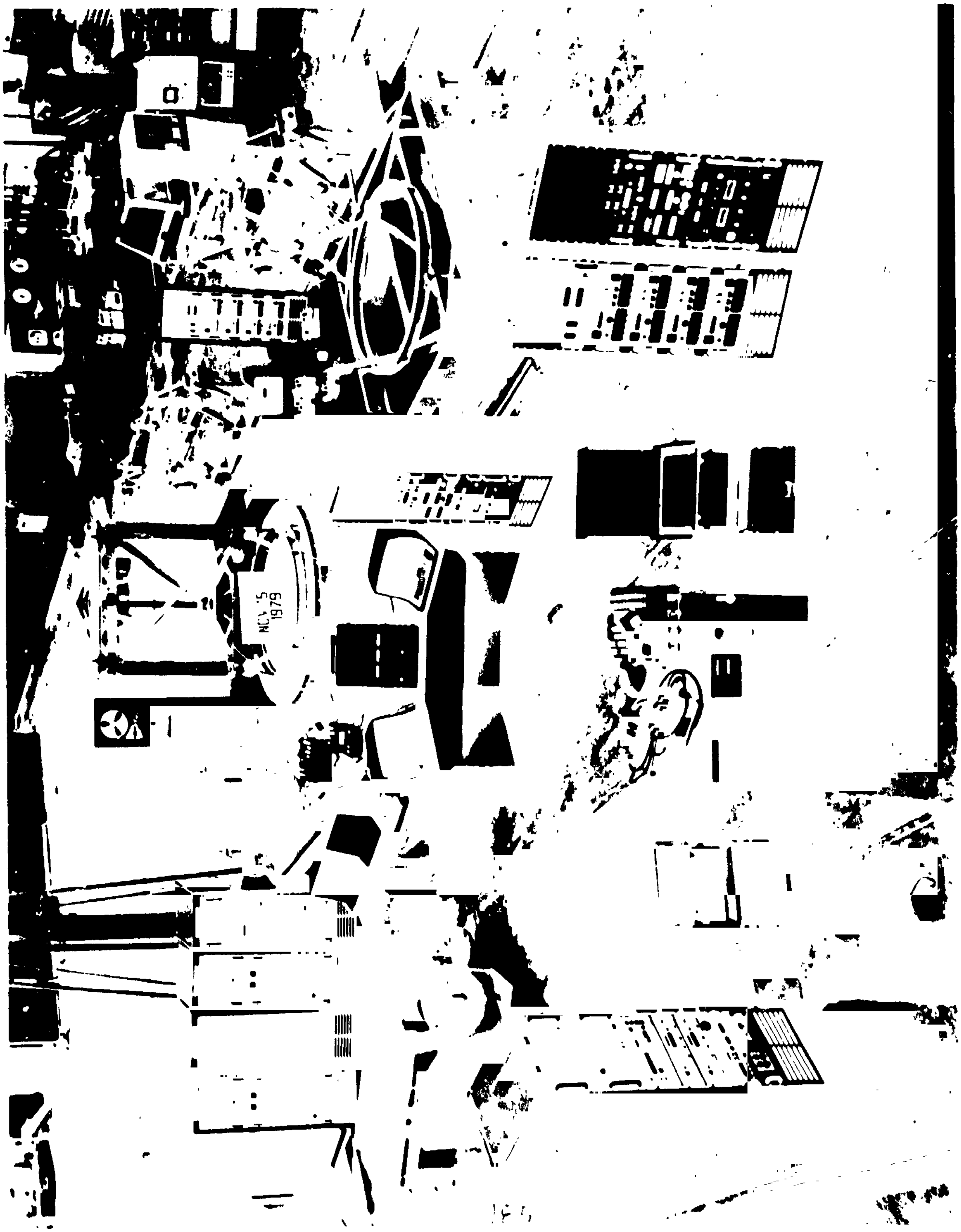


MMS SOFTWARE

CURRENT MISSIONS

SMM - FINE SOLAR POINTING

LANDSAT-D/D' - FINE EARTH POINTING

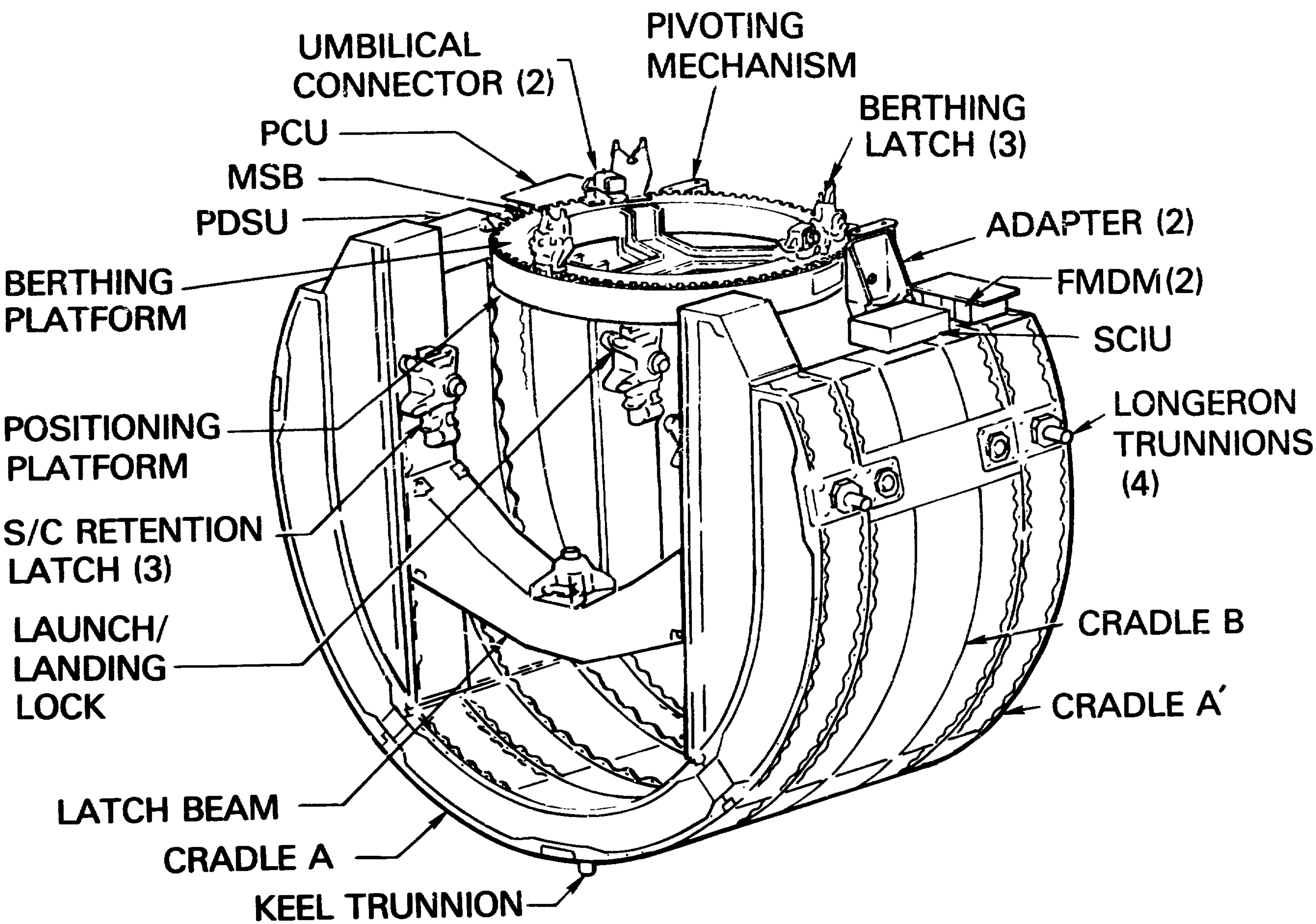
PROJECTED MISSIONS

GRO - FINE CELESTIAL POINTING

UARS - FINE EARTH POINTING AND FINE SOLAR PLATFORM POINTING

MPS - LOW-G/COARSE SOLAR POINTING

(ALL MISSIONS WILL HAVE COMPUTER OVER-RIDE
AND ANALOG SAFE-HOLD MODES)


UMBILICAL CONNECTOR (2)
PIVOTING MECHANISM
BERTHING LATCH (3)
PCU
MSB
PDSU
ADAPTER (2)
FMDM (2)
BERTHING PLATFORM
SCIU
POSITIONING PLATFORM
LONGERON TRUNNIONS (4)
S/C RETENTION LATCH (3)
LAUNCH/ LANDING LOCK
CRADLE B
CRADLE A´
LATCH BEAM
CRADLE A
KEEL TRUNNION

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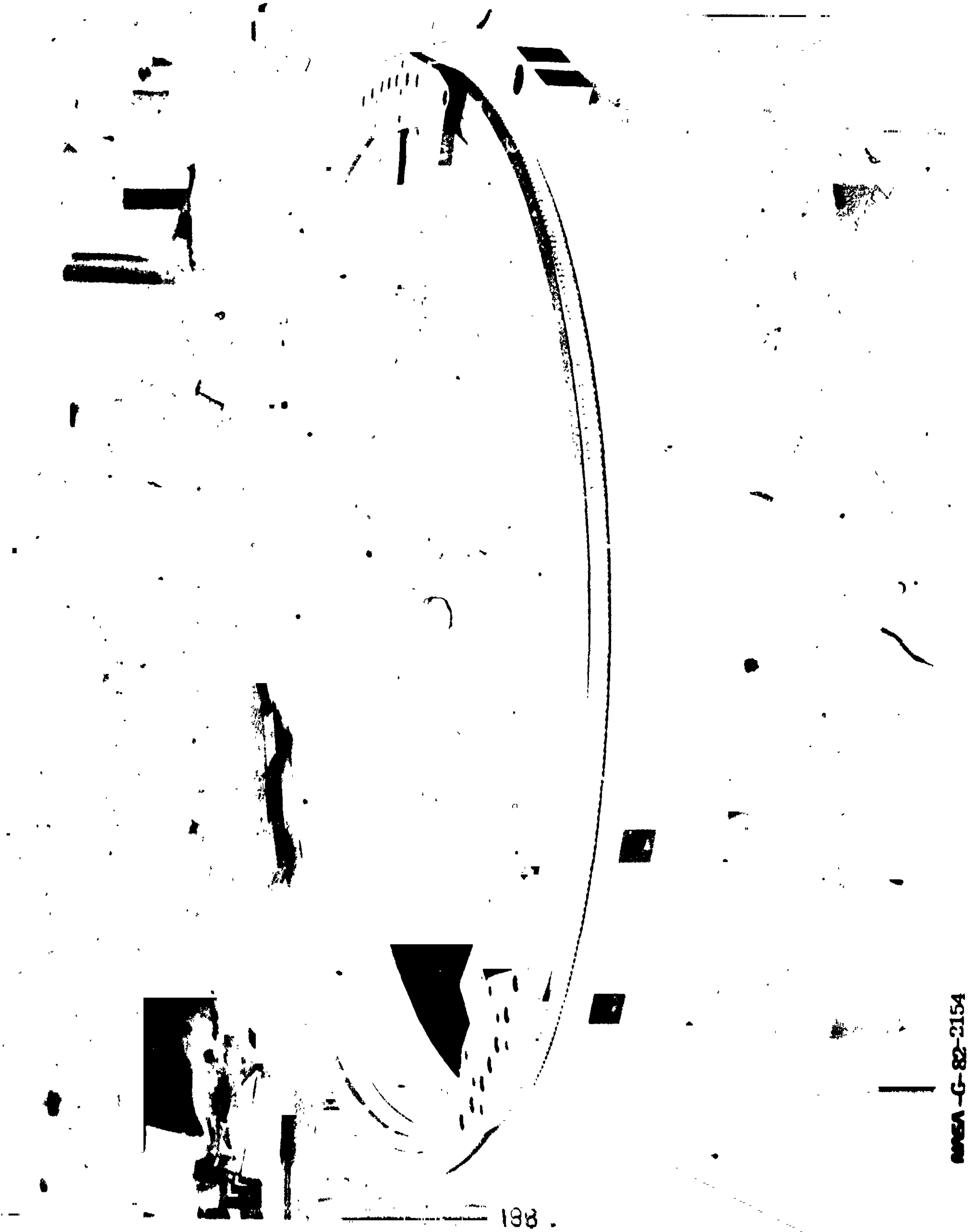

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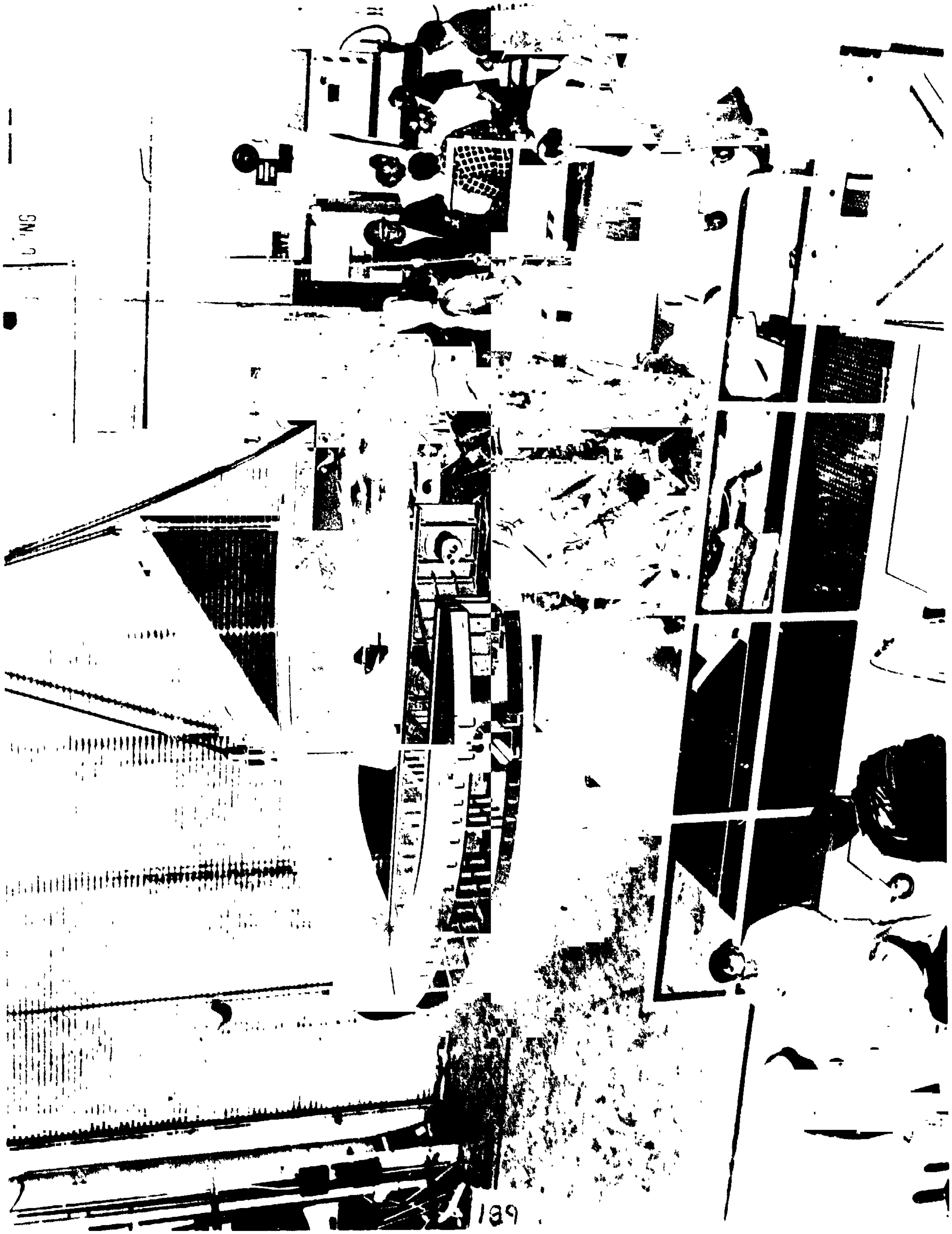

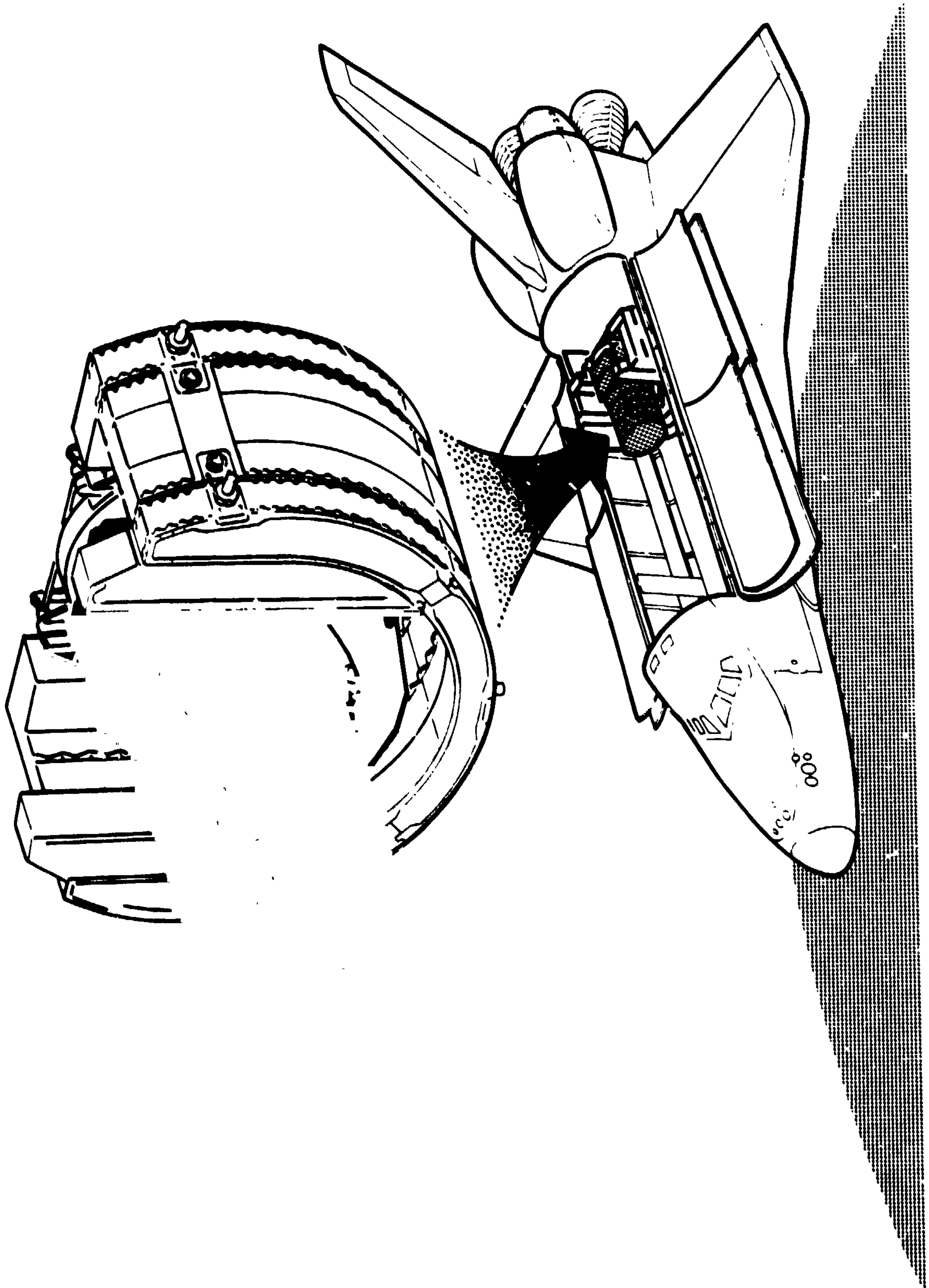

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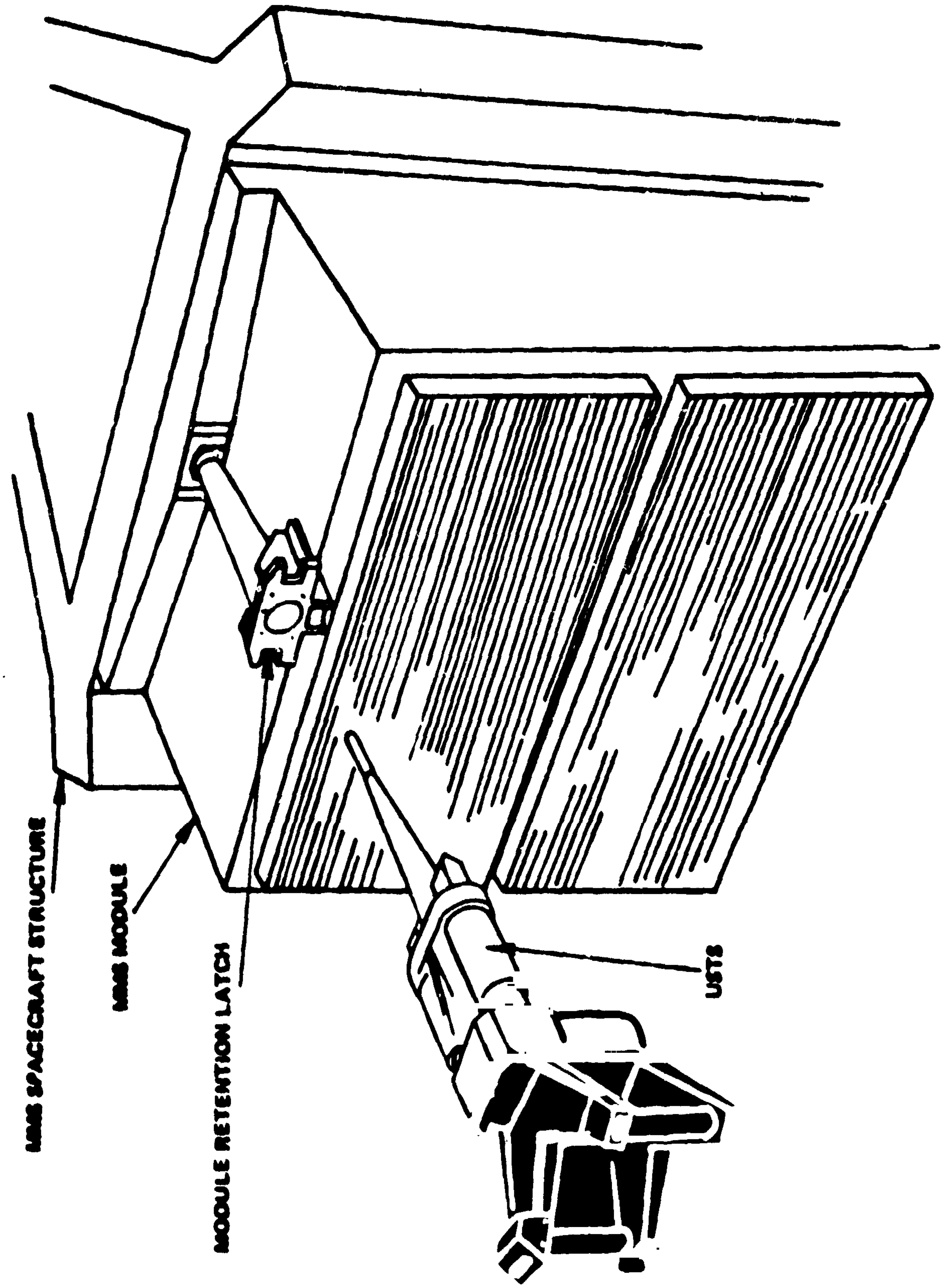

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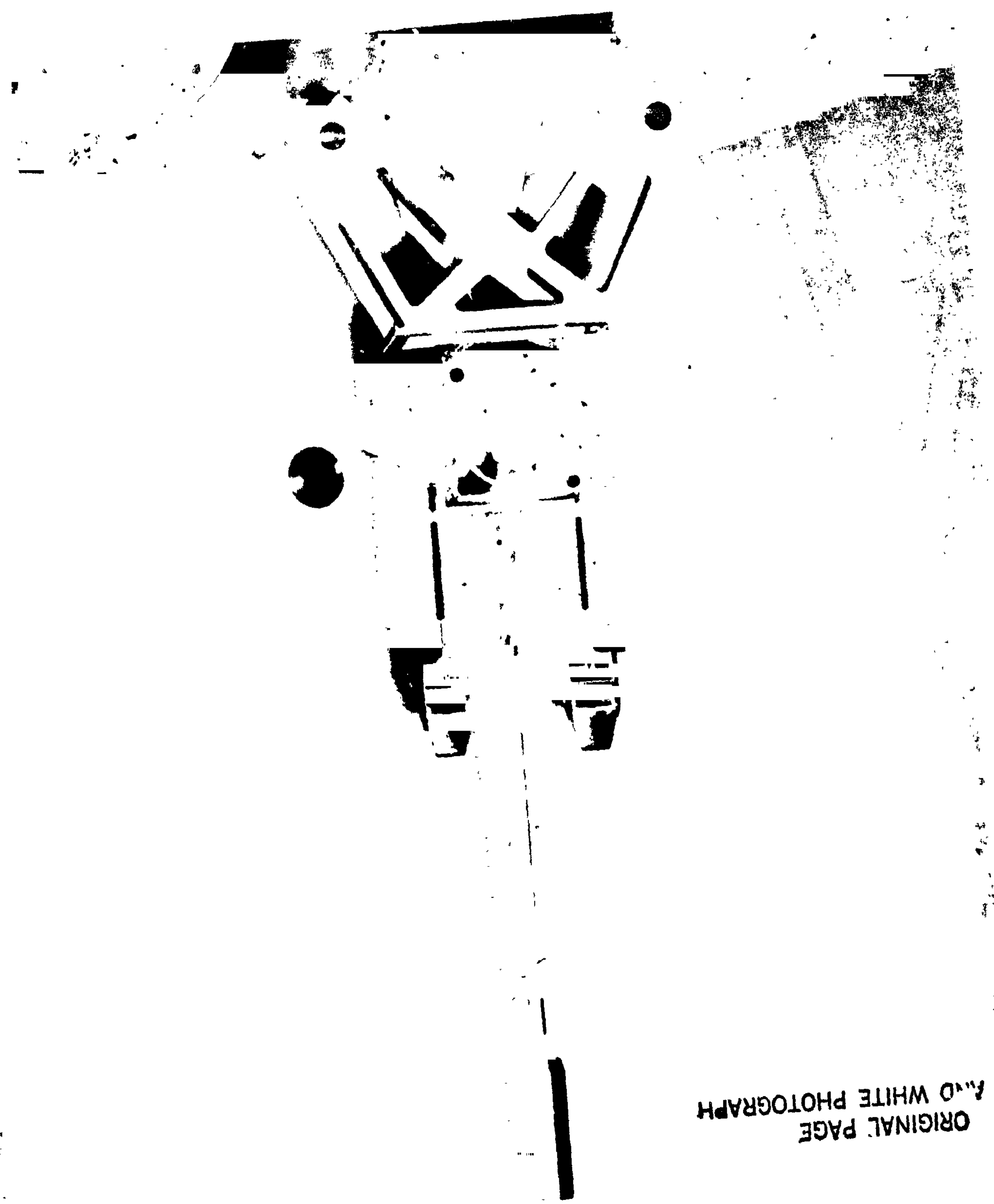

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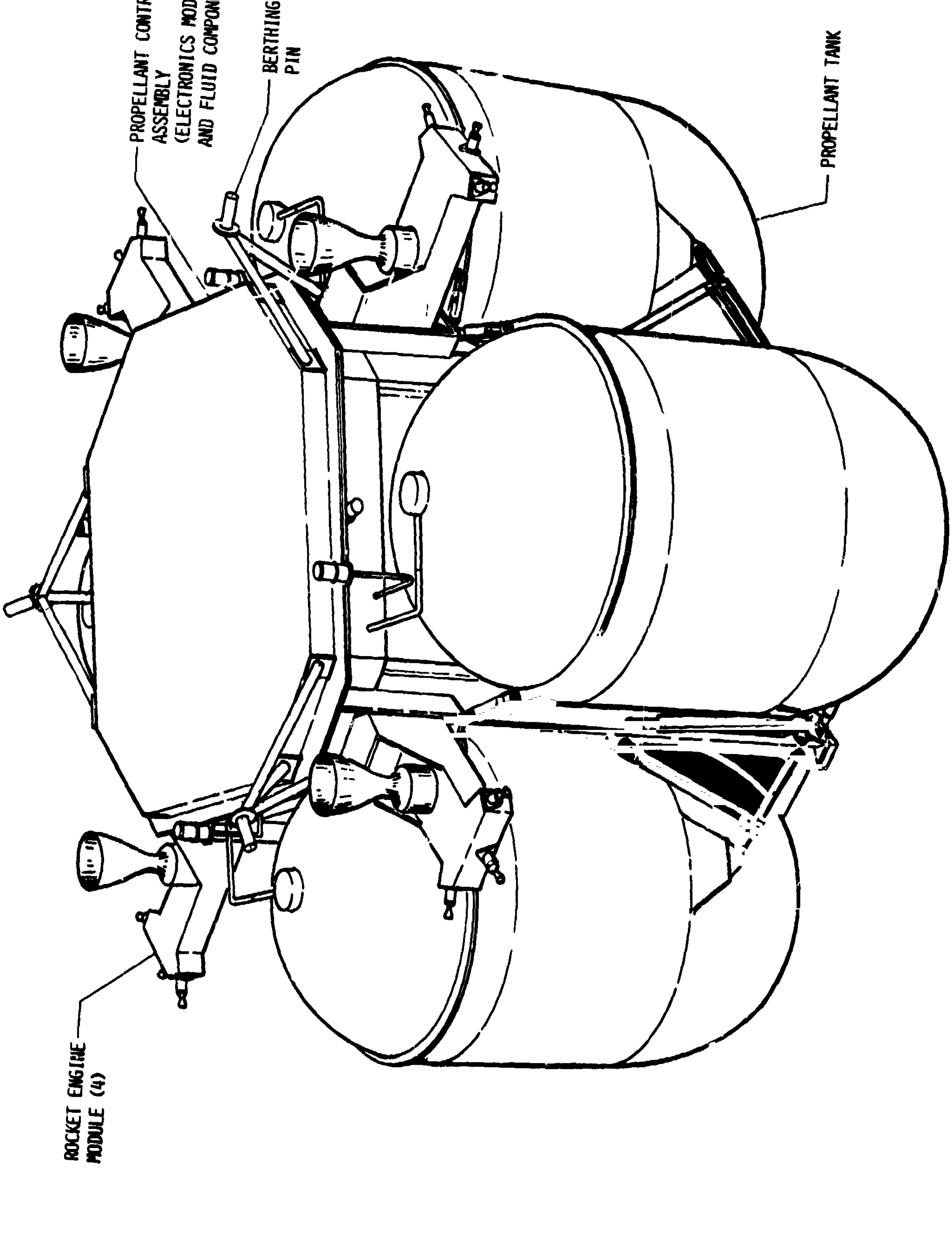
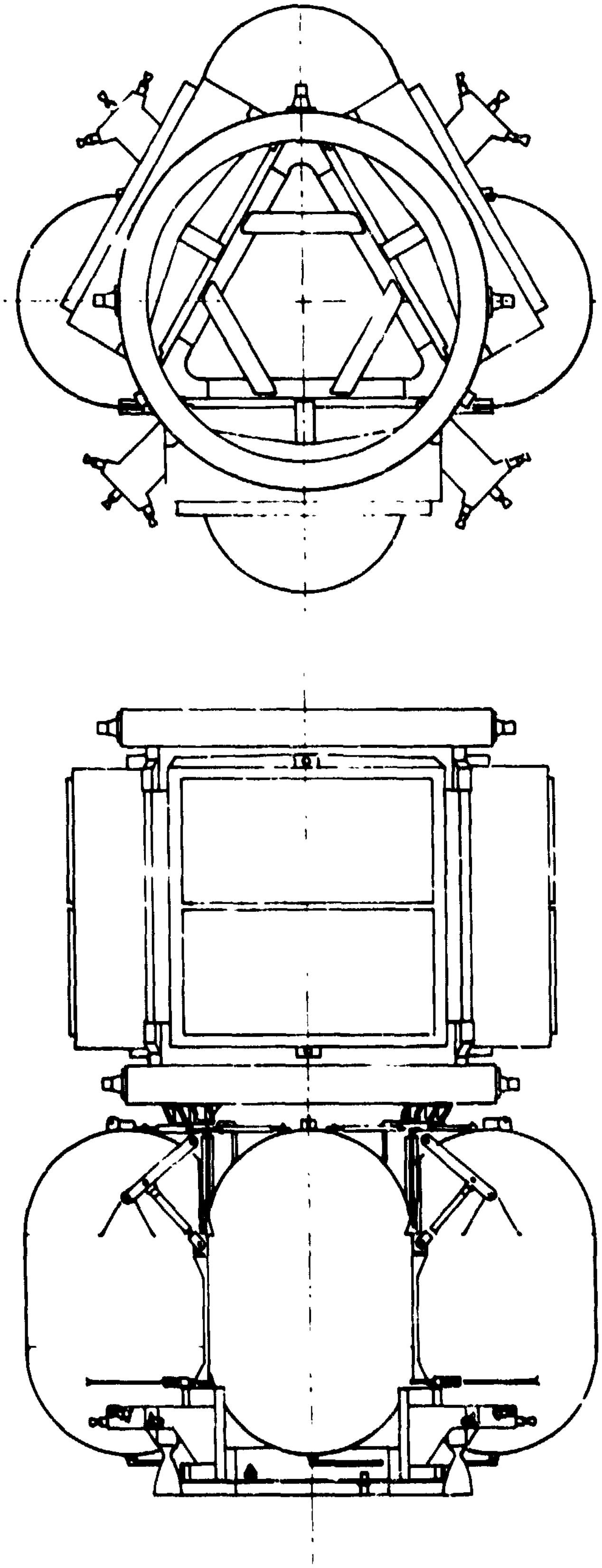

PROPELLANT CONTROL
ASSEMBLY
(ELECTRONICS MODULES
AND FLUID COMPONENTS)
BERTHING LATCH
PIN
PROPELLANT TANK
ROCKET ENGINE
MODULE (4)



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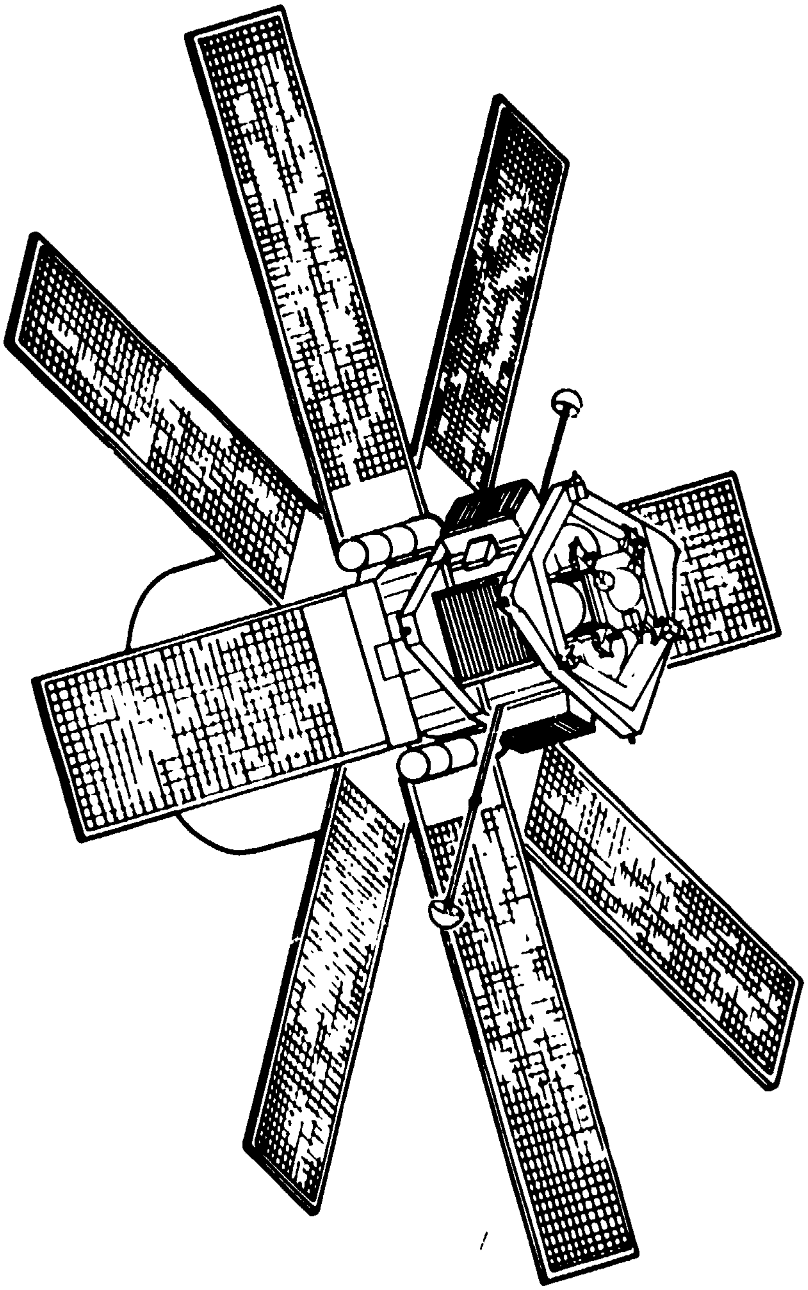
USERS OF MMS HARDWARE

PROJECTS / MMS SUPPORT	LSD AND D' (1978/1982)	SMM REPAIR (1982/1985)	DOD (1980/1990)	UARS (1984/1988)	SPACE TELESCOPE (1980/1987)	GRO (1983/1986)	OSS-3 (1985/?)	ERBE (1982/1984)
C&DH	√	√	√	√	√	√		
POWER SYSTEMS	√	√	√	√		√		√*
ACS	√	√	√	√			√*	
FSS	√	√	√	√	√	(√)		
PROPULSION	√	√	√	√				
STS INTERFACE	√	√		√	√			
SPACECRAFT SERVICING	√	√	√	√	√	(√)		
MECHANICAL SYSTEMS	√	√	√	√	√	(√)		
ELECTRICAL SYSTEM SC/CU	√	√	√	√				
THERMAL SYSTEMS	√	√	√	√				
MISSION INTEGRATION GSE	√	√	√	√		(√)		

√ SUPPORT * COMPONENTS (√) POSSIBLE SUPPORT



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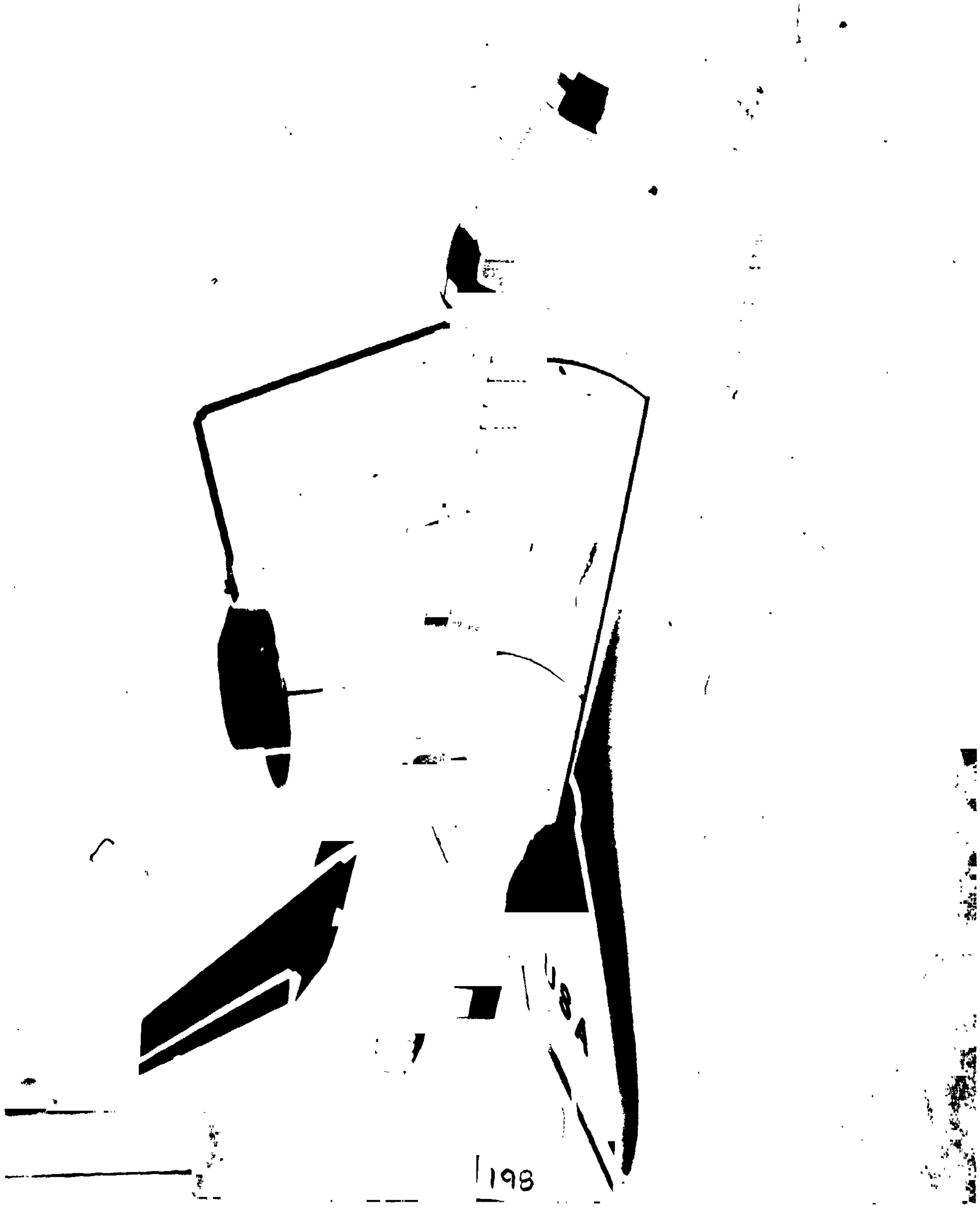

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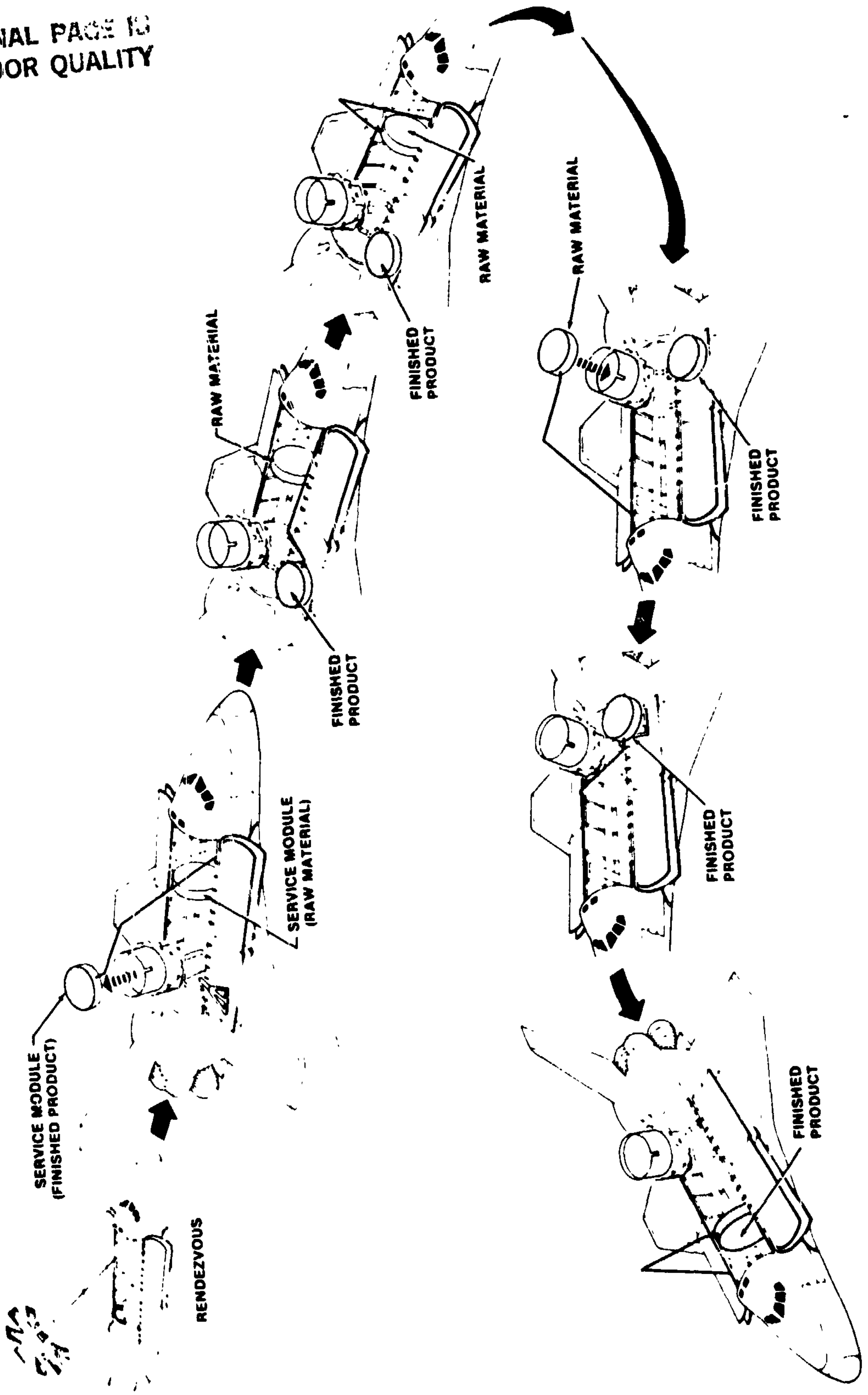
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CHARACTERISTICS OF THE HANDLING AND POSITIONING AID

By

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For

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CHARACTERISTICS OF THE HANDLING AND POSITIONING AID

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INTRODUCTION

It has long been recognized that in order to exploit the full Shuttle on-orbit capability - a large cargo bay, massive payload, the presence of astronauts - the need exists for several pieces of Orbiter-based equipment. These include the space suit, the Manned Maneuvering Unit, (MMU) the Remote Manipulator (RMS), and the Handling and Positioning Aid (HPA). The last item, HPA, is the subject of this paper.

The HPA that has emerged from the NASA/JSC contracted Phase 1 study provides a wide range of adjustable work stations both inboard and outboard of the cargo bay. It can assist with berthing and docking. It is robust, stiff, has a simple control system, and is modular.

Figure 1 shows an articulated arm version of HPA employed in a typical servicing mission. Mounted on a base frame that spans the Orbiter cargo bay, the 6 m arm is long enough to hold the satellite being serviced and keep its solar array clear of the Orbiter radiators. A suited astronaut anchors his feet to an auxiliary support platform as he works. By adjusting the length and angle of the support platform mast, and rotating the tip of the HPA arm, almost every item on the satellite can be reached.

Spares and change-out units can be brought to and from the work site by the RMS, which is controlled from the aft flight deck. The fore and aft position of the base frame can be changed between Orbiter flights and this, together with the 5 degrees of freedom (DOF) of the long arm, allows work sites to be chosen (several on one sortie if desired) that meet the clearance, reach and vision requirements of many missions.

The overall program, which includes the Phase 1 study, is illustrated in Fig. 2. Flight article activities are shown above the dashed horizontal line and Development Test Article (DTA) work below. Phase 1 focused first on developing flight article requirements and concepts; then it concentrated on the design of the DTA. Phase 2 will continue the program with the manufacture of the DTA. Once the DTA is available, simulation will commence and, as the lessons and results accumulate, attention will return to the flight article.

HPA CHARACTERISTICS

The final form of the HPA flight article is not yet frozen - but Fig. 3 which illustrates the DTA is a useful guide.

The DTA will be installed in the cargo bay mockup of the Manned Development Facility (MDF) at NASA-JSC. The side longerons of this bay were

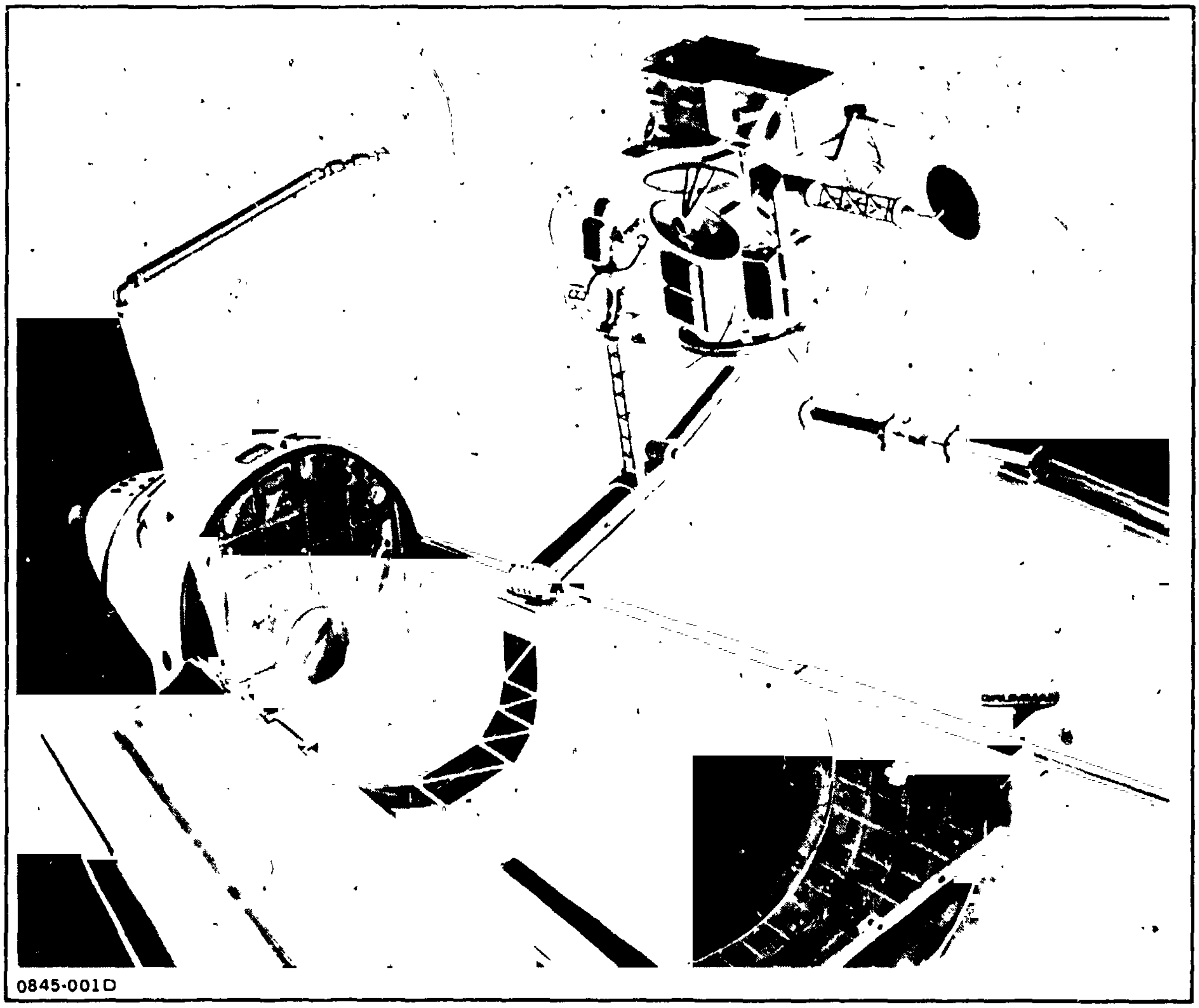

Fig. 1 Handling & Positioning Aid — Articulated Arm Version

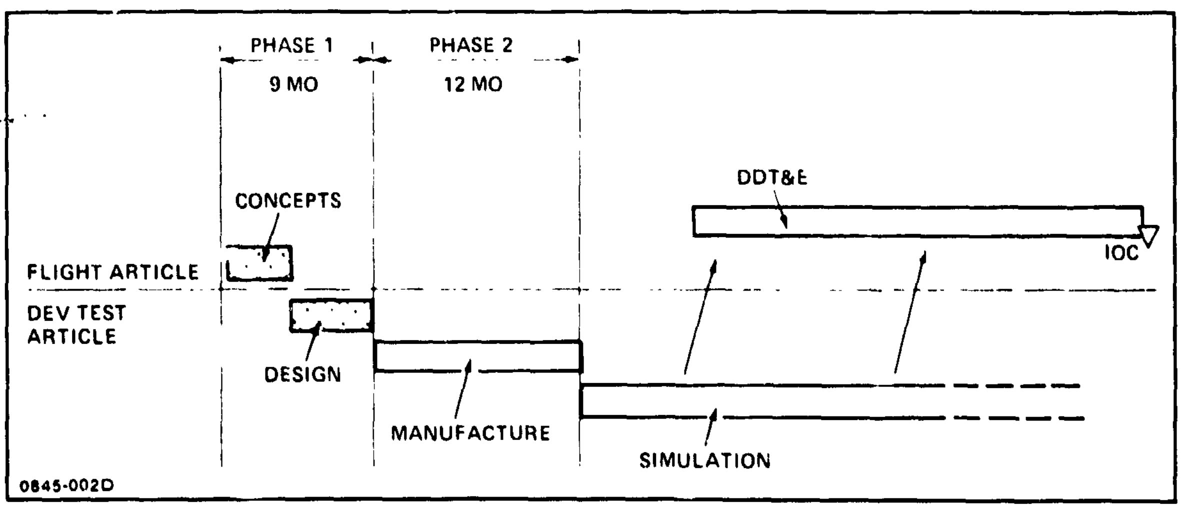

Fig. 2 Overall HPA Program

SHOULDER PITCH	ELBOW PITCH	WRIST PITCH	WRIST ROLL
• MAIN GEAR (CW) 2534:1	• MAIN GEAR (NASA) 162:1	• MAIN GEAR (NASA) 768:1	• MAIN GEAR (CW) 102:1
• SECOND GEAR 5.1	• SECOND GEAR 50:1	• SECOND GEAR 15:1	• SECOND GEAR 10:1
• MOTOR 1.9 HP	• MOTOR 1.9 HP	• MOTOR 0.2 HP	• THIRD GEAR 10:1
• BRAKES (2)	• BRAKE (1)	• BRAKES (2)	• MOTOR 0.2 HP
			• BRAKE (1)

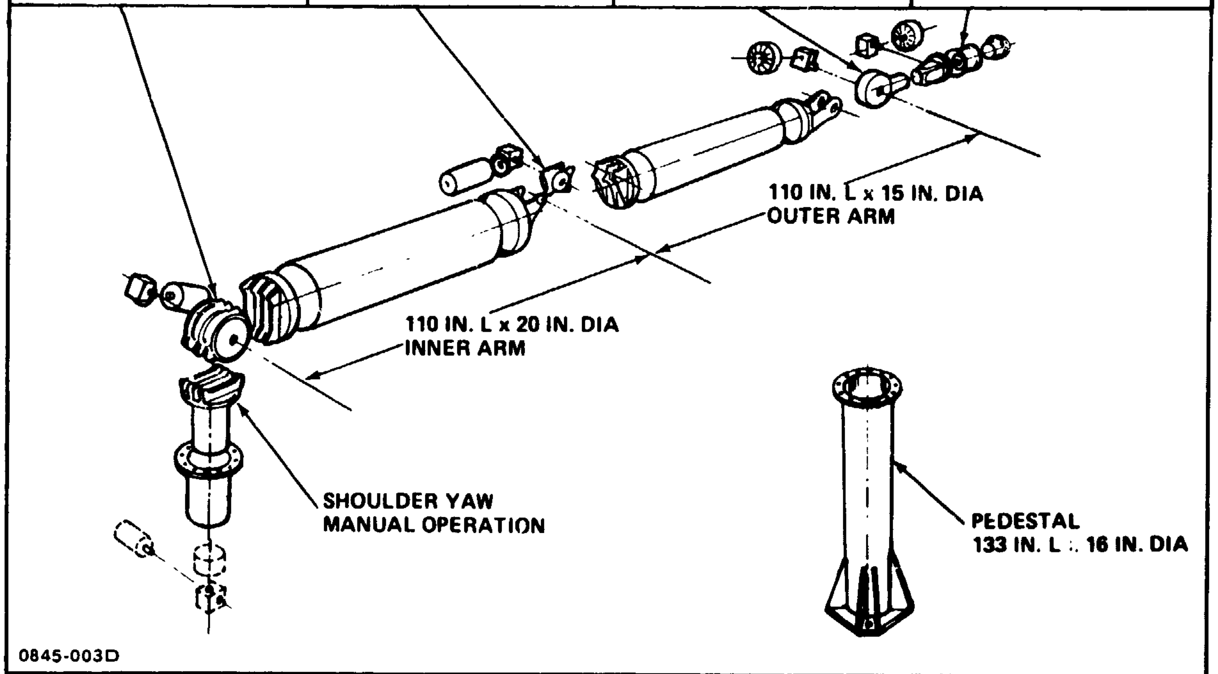

Fig. 3 HPA DTA Arm & Pedestal

not considered to be strong enough or stiff enough to support the DTA arm in the flight article manner (i.e., with a base frame that spanned the bay from longeron to longeron and depended on them for support). As a low cost alternative, we designed a pedestal that bolts directly to grill-work on the MDF floor and rises about 3 m to support the shoulder joint of the main arm. This welded steel pedestal is shown in Fig. 3 together with the structural and mechanical assemblies that make up the arm.

To minimize costs in Phase 2, only four of the five joints are powered; the shoulder is yawed manually. Space is available for powering the shoulder joint later. The four powered joints utilized off-the-shelf hardware for gears, motors, and brakes. Furthermore, the main gear boxes for each of these joints will be loaned to this program - two from NASA and two from Curtiss-Wright, a significant and welcome cost saving. The elbow pitch and wrist roll joints have irreversible gear boxes in the drive trains and, therefore, need only one brake each. The other joints have two brakes each for safety.

The inner and outer sections of the DTA arm are tubular, 51 cm and 38 cm diameter respectively. Each tube is rolled from an aluminum plate with a longitudinal seam weld. Both tubes have open handholes and are bolted to end fittings which mate with the appropriate main gear box.

As shown in Fig. 4, HPA mission dedicated requirements fall naturally into two groups:

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- Requirements for missions satisfied by a tilt table
- Requirements for missions needing the greater mobility of an articulated arm.

The articulated arm version is used to provide reach, more DOF, and greater angular ranges. Based upon mission analysis, the tilt table version of the HPA must be located at various stations in the back half of the cargo bay. The articulated arm version is located at various stations in the front half. The weight of payloads supported by the HPA during Orbiter ascent and descent show a wider range for the articulated arm group. Power and data interfaces should generally be connected to a spacecraft soon after it is berthed, and disconnected only a short while before it is released. This indicates, for both types of HPA, the need for automated connection and disconnection. Fluid interfaces with the spacecraft are needed only for the missions using articulated arms. The making and breaking of the fluid connections are not time-critical, and manual operation was judged to be lighter, cheaper, and simpler. Further mission analysis is needed to confirm the differences, in HPA location and payload weight supported, between the two groups.

To determine the proper level of stiffness for the tilt table and articulated arm HPAs (particularly when supporting heavy spacecraft), a parametric study of fundamental frequency of the Orbiter/HPA/Spacecraft combination was performed. Variables included the spacecraft weight, its radius of gyration, the distance from the HPA interface to the spacecraft cg, and the arm stiffness. The stiffness values shown in Fig. 4 allow the spacecraft to weigh up to

	MOBILITY LEVEL	
	TILT TABLE	ARTICULATED ARM
• REFERENCE MISSION	ST UARS INTELSAT OAO GEO PLATFORM	SOC ORB SERVICE PLATFORM 25 kW PWR MOD LSSD
• REACH DOF ANGULAR RANGE	NA 1 OR 2 90 TO ±180°	4.5 – 5.5 m 3 TO 5 130 TO ±180°
• LOCATION VARIES	MID TO AFT BAY	FORWARD TO MID BAY
• CARGO SUPPORT	2000/4000 kg	0 TO 10000 kg
• SPACECRAFT INTERFACES POWER DATA	AUTOMATIC CONNECT & DISCONNECT REQUIRED.	
FLUID	NA	MANUAL CONNECT & DISCONNECT FEASIBLE
• ARM STIFFNESS	1.8×10^6 N.m/RAD	@1.15×10^7 N.m^2 EFFECTIVE EI
• BERTHING DEVICE	SINGLE STANDARD?	SOME STANDARDIZATION BUT NOT COMPLETE
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Fig. 4 Two Categories of Mission Requirement

100,000 kg with a radius of gyration, and an offset between HPA tip and spacecraft cg, of about 5 m each. Any larger, heavier spacecraft would infringe the low frequency limit, but any such platform would in all probability have a cmg or inertia wheel control system of its own that could control the combination without the impulse problems inherent in the Orbiter RCS.

In addition to these mission-related requirements, there are, of course, generic HPA requirements in the areas of shuttle compatibility, man rating, reliability and safety.

The wrap-around base frame concept, chosen for its inherent stiffness, fits comfortably into the modular system shown in Fig. 5. The main hardware elements are:

- The base frame
- One of the various mobility mechanisms (tilt tables, articulated arm)
- A berthing or docking end effector.

Now shown are the control systems and such auxiliary items as the EVA crew foot restraint, lights, etc. Usually a combination of the three main elements listed above will be used, but the base frame can be employed on its own as a payload support.

Our current weight estimates of the HPA articulated arm and HPA tilt table versions are summarized in Fig. 6. This tilt table is stiff enough to provide cantilever support for a modest sized payload during ascent and descent. If a pitch support latch, with its mechanical complexity, were added, the structural weight of the tilt table could be reduced.

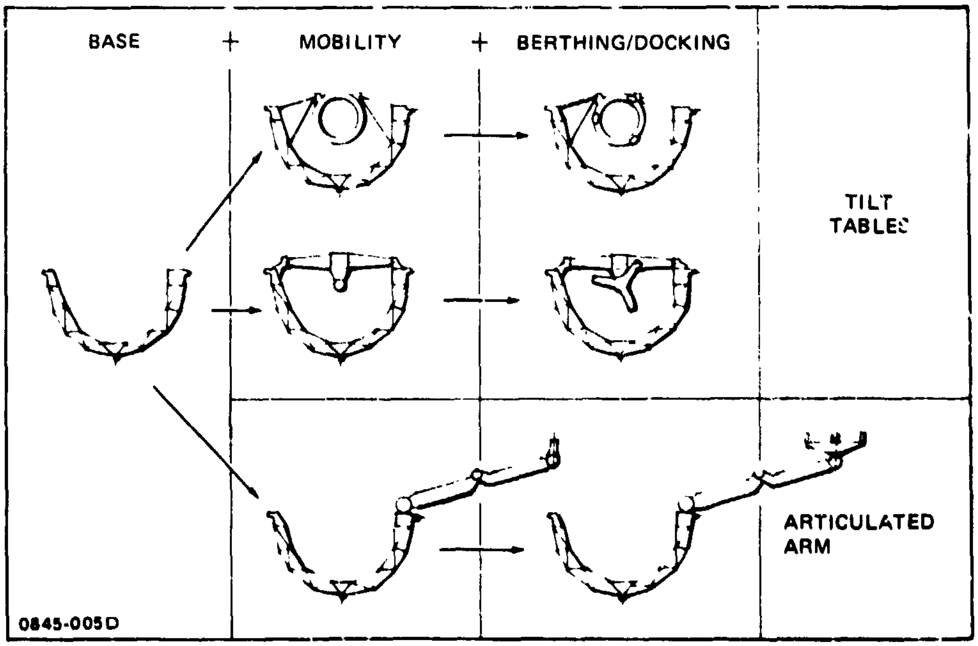
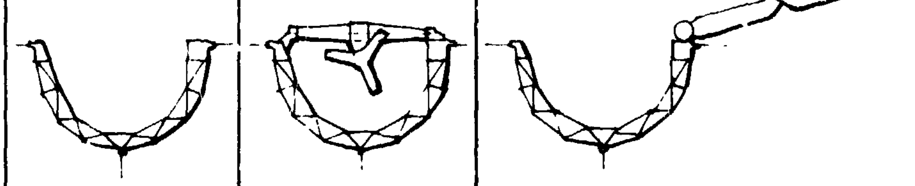


Fig. 5 HPA Modularity



	BASE	TILT TABLE	ARTICULATED ARM
FRAME STRUCTURE	383	383	383
CROSS BEAM		206	
MOUNTING		68	
MOTORS		51	
SPIDER		74	74
ARM			275
GEARS & MOTORS			475
CONTROLS, POWER DISTRIB		109	109
	383 (845lb)	891 (1965lb)	1316 (2902lb)
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Fig. 6 Flight Article HPA Modular Weights, kg

The angular ranges required at the HPA arm and tilt table joints were derived from a step-by-step analysis of representative missions in Phase 1. The maximum values were rounded off upwards and are presented in Fig. 7. The only angles which might have to be enlarged when more missions are studied are the pitch joints - shoulder, elbow, wrist. But the values shown here cover a slightly larger range than the corresponding joints in the RMS; and we believe that they will prove adequate.

The control modes and speeds of the flight article arm are to be explored by simulation after Phase 2. Until this is done, the modes and control panel of the DTA can be treated as representative of the flight article.

For routine operations, such as arm deployment and storage, payload movement and rotation, the control mode is one-joint-at-a-time. These operations do not take up a large proportion of the HPA use time; during most of the mission it is stationary, providing location and support. As a result, the speeds chosen for these operations (1 degree per second high speed, 0.3 degrees per second low speed) are modest, and the one-joint-at-a-time procedure keeps the controls simple without impacting mission duration significantly. The "lunge mode," used when the HPA assumes an active role in berthing, involves the operation of the three pitch plane joints in unison (Fig. 8). However, this control mode is not complex since the motor speeds are coordinated by straightforward hard-wired logic.

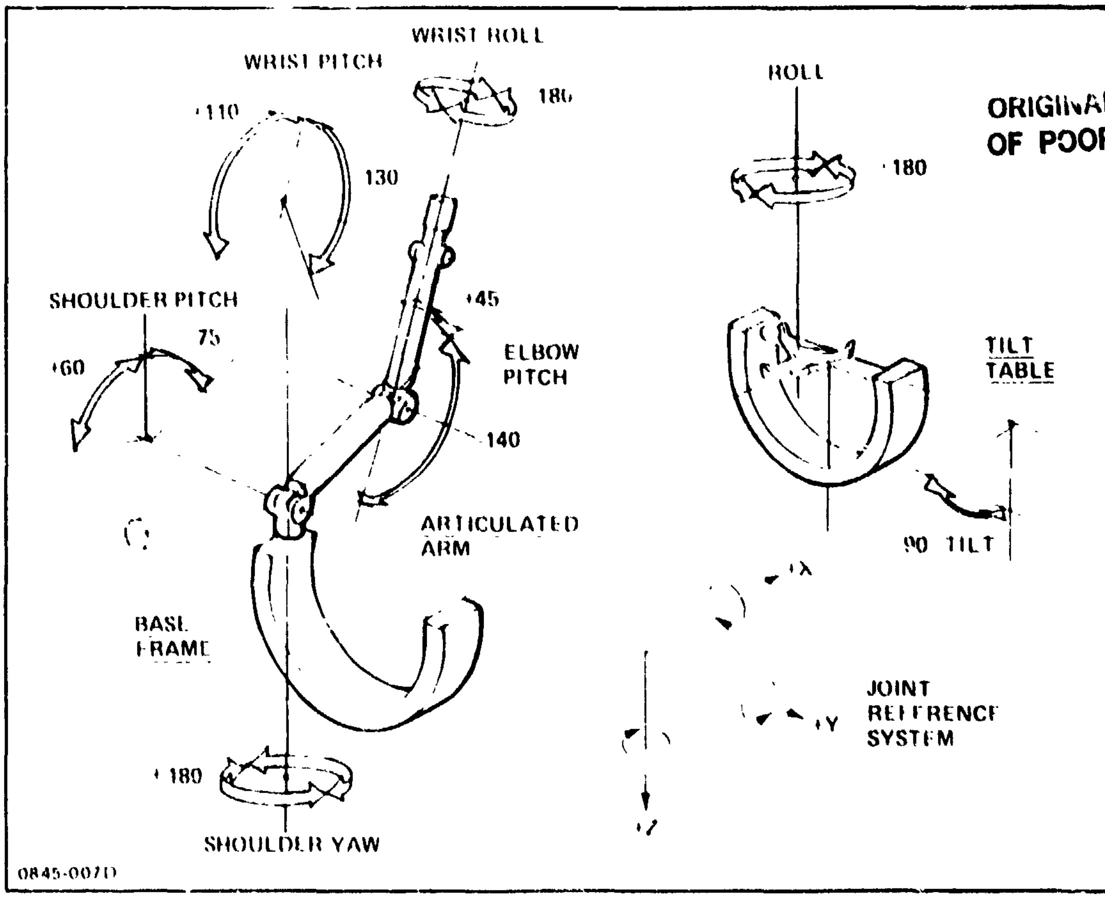

Fig. 7 HPA Flight Article Joint Angles

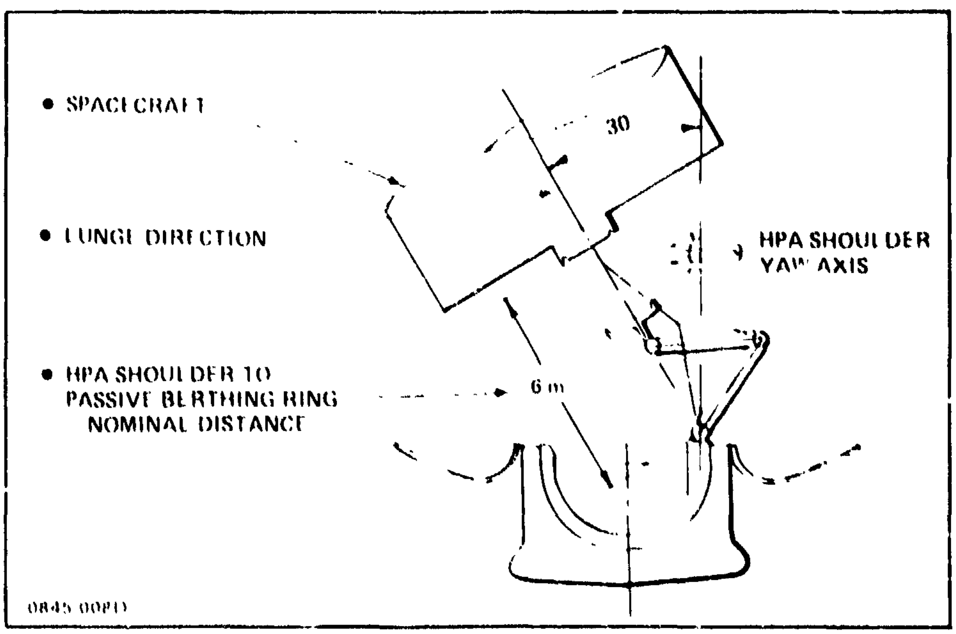

Fig. 8 Inclined Lunge Berthing Geometry

The control panel illustrated in Fig. 9 will normally be located on the MDF aft flight deck. The sequence of panel operations is:

- Turn on master power; activate logic circuitry and servo amplifiers
- Select desired channel and adjust speed control
- Actuate the run switch which releases the brakes, initiates a delay circuit allowing time for brake release, powers the servo amplifier and applies the drive signal to the servo motor. The motor accelerates to its selected speed in approximately 200 milliseconds and continues to drive the joint until the run switch is released
- Release the run switch which returns the drive command to zero and initiates a delay circuit to allow time for motor deceleration, again taking 200 milliseconds. The servo motor is now disabled and the brakes are reapplied to hold the joint in position.

If the joint drives to its "end of travel" limit prior to release of the run switch, limit microswitches on the arm will disable the servo amplifier, stop the drive motor and illuminate the limit lamp for that channel. To drive the joint out of the limit stop, the run switch must be operated in the reverse direction. An emergency stop switch is provided which, when activated, will bypass all control circuitry, remove the servo amplifier power, and apply all brakes. A digital display for each channel reads displacement (±180.0 degrees) from normal joint position.

TYPICAL MISSION APPLICATIONS

In this section, four missions (three using an articulated arm version of the HPA, the fourth using a tilt table version) are described. These are followed by a short discussion of the results of our recent simulation of the HPA in a berthing role.

a) Large Space Structure Demonstration (LSSD) Mission - The mission involved the assembly of Automatic Beam Builder (ABB) Fabricated beams to form a spacecraft structure. The assembly sequence is shown in Figs. 10 and 11.

The 1 m beams used for the LSSD structure are fabricated by the AAB mounted in the cargo bay. An HPA is used as the base to hold the beams during assembly.

Two RMSs are required. The first RMS is used to stabilize the structure during assembly. The second is used by the astronaut on an MFR/RMS to transport and join beams and to transport LDEF and radiometer panels to the structure for assembly. After the LDEF panels are installed, the LSSD is removed from the HPA by the second RMS. Next the MFR/RMS removes the radiometer from the cargo bay and installs it on the HPA. Then the LSSD is positioned on the radiometer, structurally attached, and electrical connections are completed.

b) Power Module Servicing - The Power Module (PM) generally goes on Spacelab missions to provide power, etc., for experiments. With Spacelab mod-

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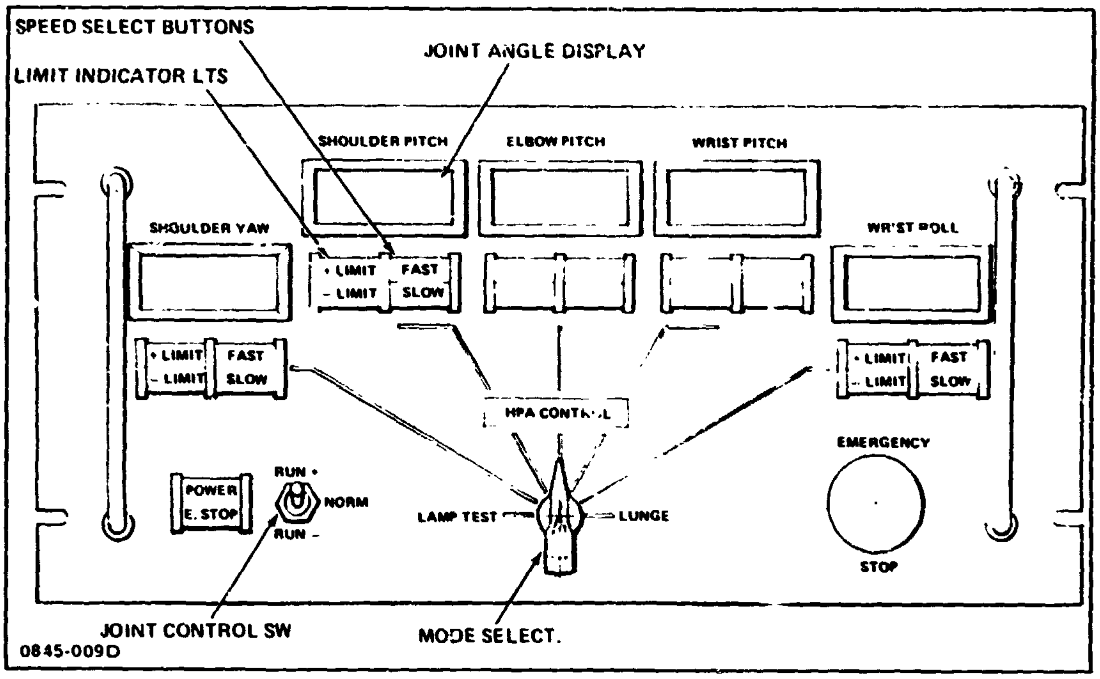

Fig. 9 DTA Control Panel

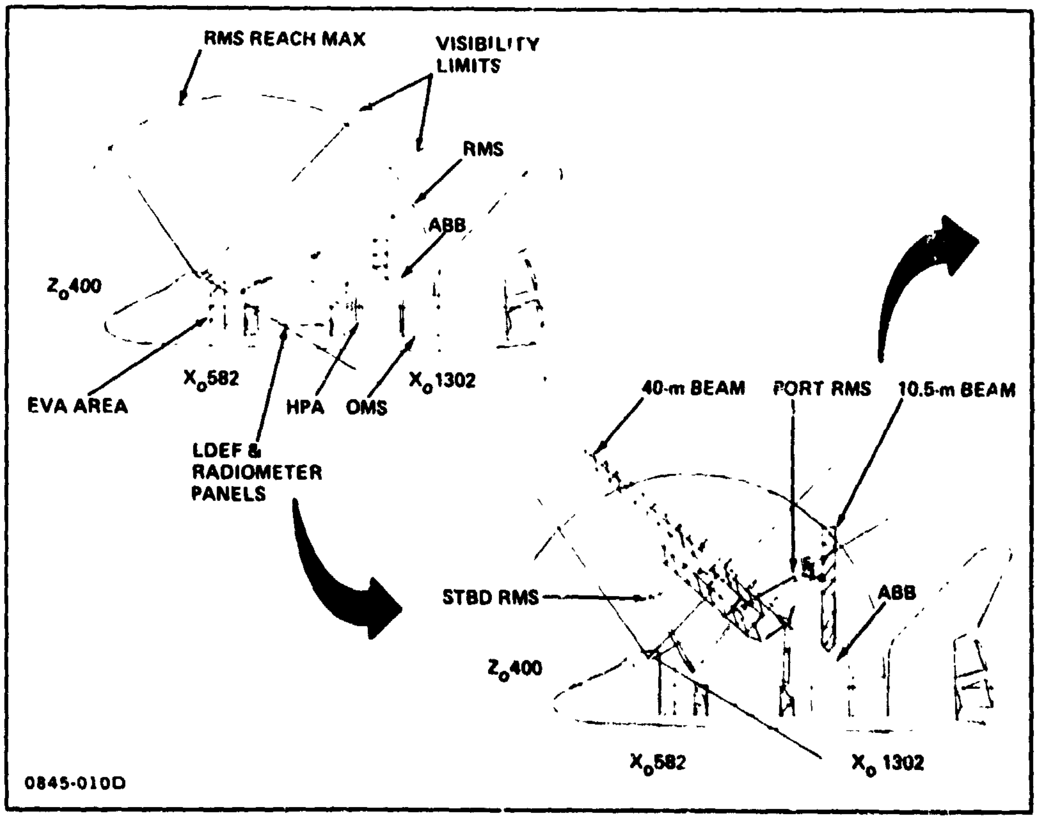

Fig. 10 LSSD Construction Sequence

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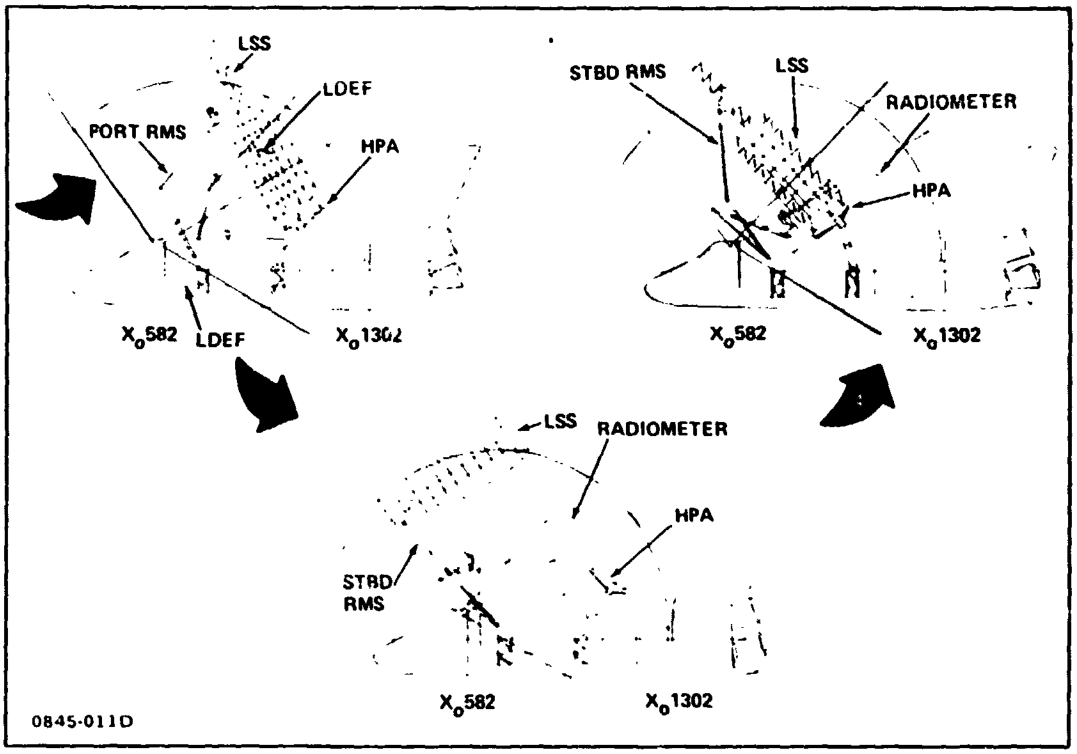

Fig. 11 LSSD Construction Sequence (Contd)

ules and pallets in the cargo bay, the only area consistently available for an HPA is at the forward end. To a great extent, the airlock and the tunnels connecting Spacelab to the cabin govern the shape of the HPA base frame and the lengths of the articulating arms for an outboard HPA.

Figure 12 shows two locations for the arm tip fixture. The PM will be berthed to the arm tip fixture after capture by the RMS. The PM is conventionally mated to the Orbiter above the cabin at Orbiter centerline. This is shown in Fig. 12. For convenience, the HPA could be extended outboard for viewing of the berthing operation, then moved in board to position the PM at the conventional position.

c) Orbiter to SOC Docking - As presently proposed, direct docking of the Orbiter to the SOC has an interface close to the Orbiter cabin. There is some concern that a mis-docking maneuver could result in the piercing of the pressure cabin by a SOC member. To obviate this, it is proposed that the docking ring now be mounted on the tip of the Orbiter HPA arm. The hard docking will then take place between this ring and the SOC active docking ring.

As shown in Fig. 13, this effectively extends the reach of the RMS, inasmuch as an Orbiter which is remotely docked to the standard docking location on SOC, can use its RMS to transfer cargo to a SOC location which it could not normally reach. In this instance, the tunnel joining two habitation modules is being transferred and installed.

After docking and, if appropriate, after cargo transfer, the HPA articulates to berth the Orbiter to the SOC by mating the crew transfer tunnel to a sealing face on the docking ring. Crew can now transfer.

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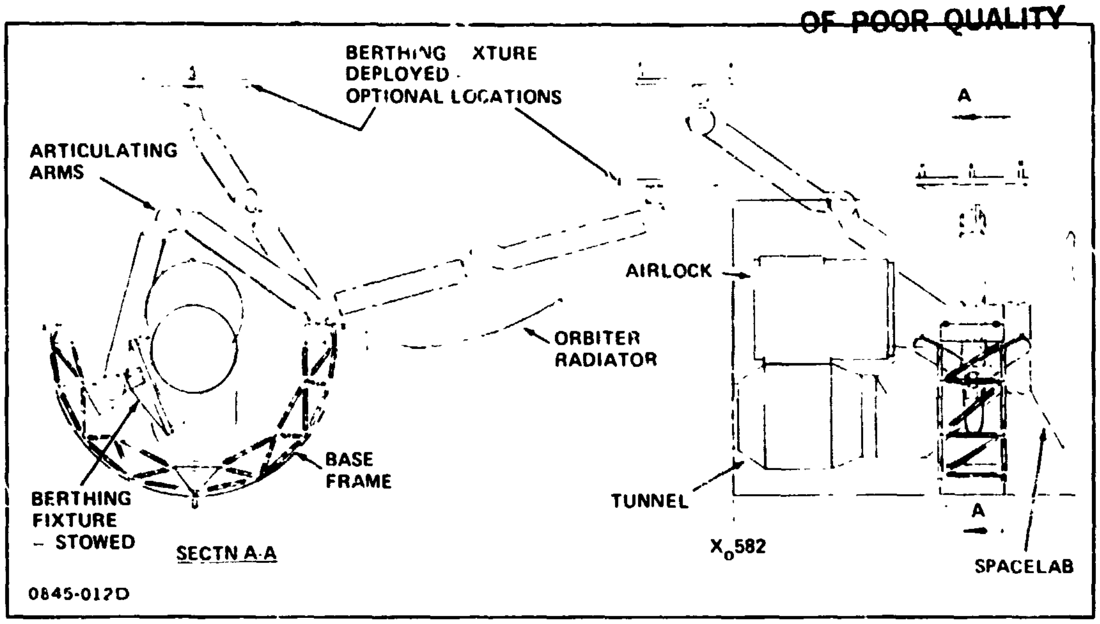

Fig. 12 Bay Packing of HPA for 25 kW Power Module Support

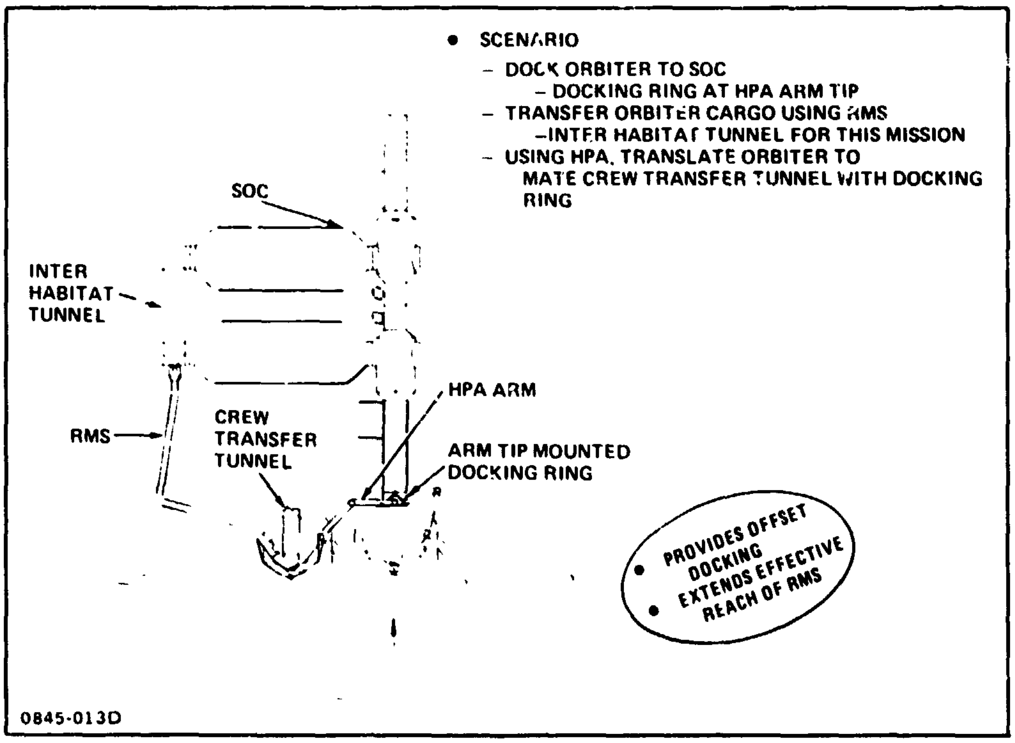

Fig. 13 HPA Provides Orbiter to SOC Docking

d) Upper Atmosphere Research Satellite (UARS) Servicing - Figure 14 shows a tilt table version of the HPA supporting the first servicing mission of the UARS. Here, the initial propulsion module, PM1, is exchanged for a PM2 propulsion module and MMS modules are replaced. For this mission, the HPA base frame carries two tilt tables. Location of the HPA in the cargo bay is mainly dependent upon other payloads flown on the mission and on the Orbiter cg envelope. The only specific requirement is that it be within reach of the RMS and within the operator's view.

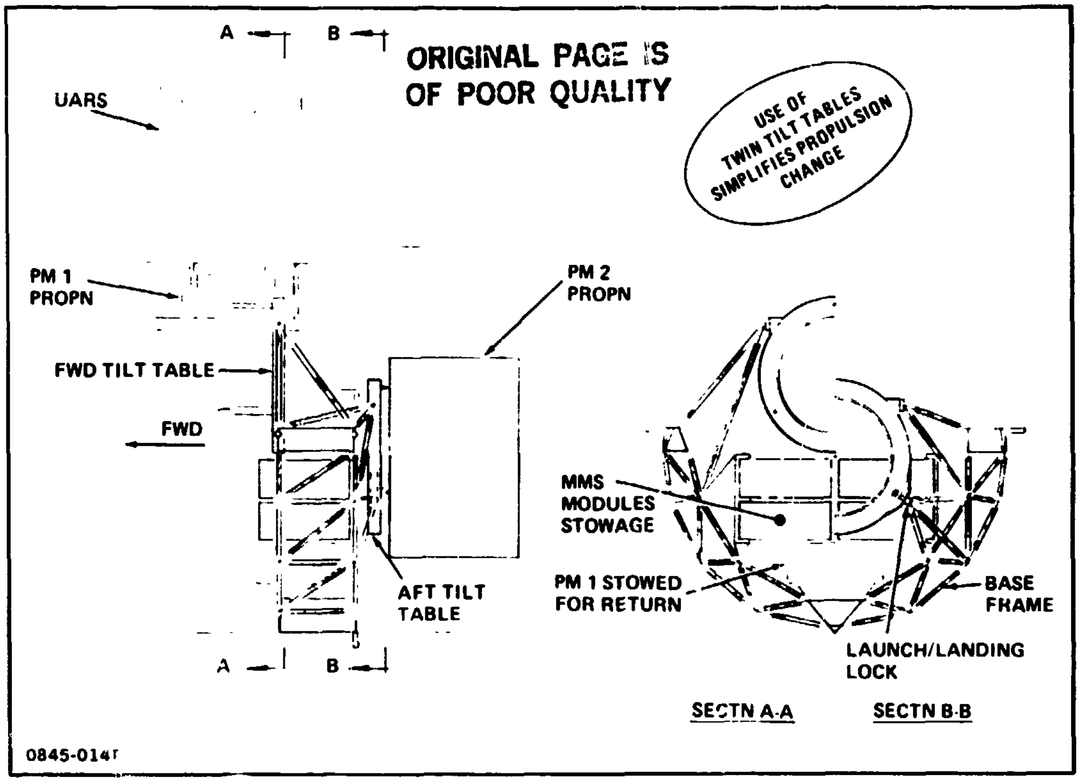

Fig. 14 Upper Atmosphere Research Satellite Service Mission

The sequence of events for UARS servicing is as follows: First the forward tilt table is deployed and the UARS berthed to it, using the RMS. This table has berthing extensions which directly support the UARS body, leaving PM1 attached only to UARS. The PM1 is then released from the UARS and moved sideways by the RMS, through the gap between UARS and tilt table. It is then stowed on the base frame. The aft tilt table, which mounted PM2 for launch, is now deployed to place PM2 above the HPA. The UARS is now transferred by RMS to mate with the PM2 in its correct operational location, The forward tilt table is stowed back into its launch position. Using the RMS/OCP combination, MMS modules are replaced on the UARS, with the old modules occupying those mounts on the HPA base frame which supported the new MMS modules during launch. After checkout and separation of the UARS, the aft tilt table is stowed.

e) The Results of Simulating an HPA in a Berthing Role - Figure 15 shows a typical sequence of spacecraft capture followed by berthing. In this case, the HPA is included in the berthing sequence because the spacecraft is assumed to be relatively massive. Orbiter, RMS, and HPA functions follow one after another. The Orbiter achieves a near match of orbits. The RMS captures the spacecraft, moves it into the HPA envelope, and ensures that the passive berthing fixture on the spacecraft is turned towards the Orbiter. The HPA, which carries the active berthing fixture, aligns it with the spacecraft and berths. At the top of the chart is an indication of the period when the Orbiter's RCS will be inhibited because the spacecraft is massive. The period of interest in the simulation covered the second and third HPA activities, "aligns berthing fixture" and "berth."

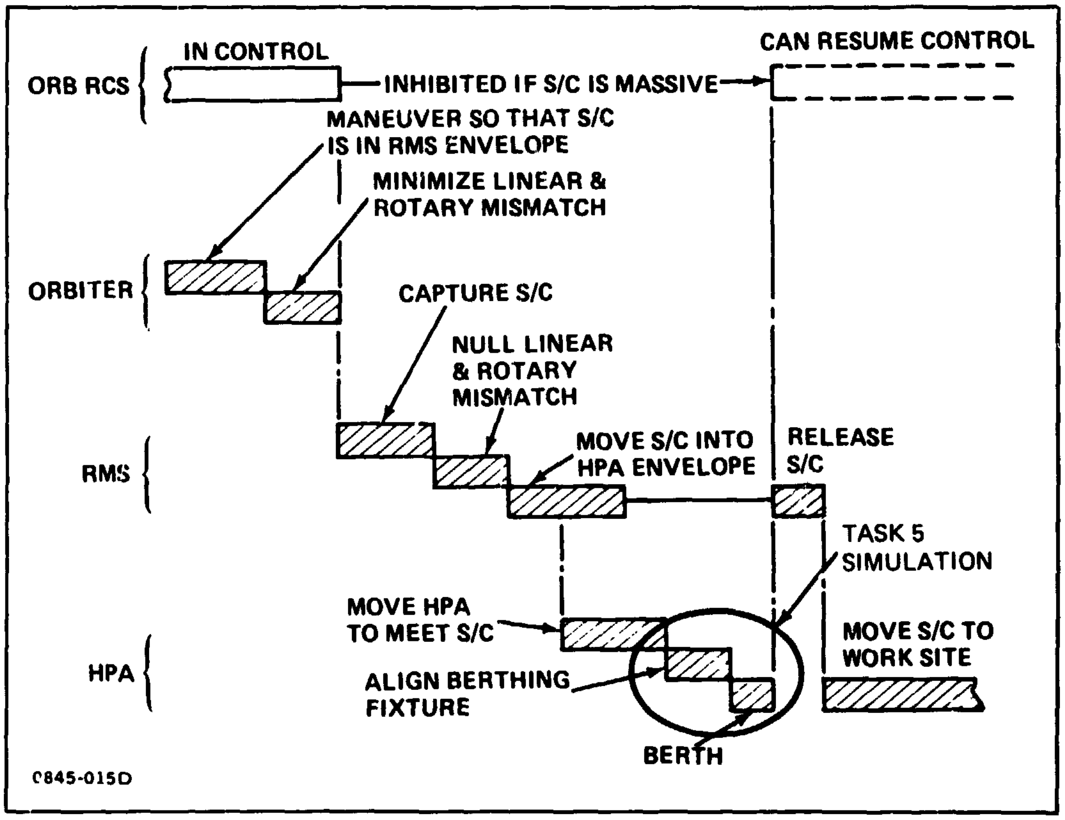

Fig. 15 Typical Spacecraft Capture & Berthing Sequence

The simulation, which was carried out at Grummans Large Amplitude Space Simulator, explores the degree of control complexity needed for berthing.

It was determined that coping with misalignment and drift normal to the arm pitch plane were the main influences in driving up the control complexity. Therefore, we re-examined the initial straight up (Y axis) lunge approach and hit upon the inclined lunge concept. This requires only 5 DOF and is illustrated in Fig. 16; the key feature is that the nominal lunge direction is inclined at (say) 30° to the vertical shoulder yaw axis. Quite large (20 cm) out of arm plane misalignments can be corrected with small (5°) shoulder yaw rotation. Lateral adjustments are swift, essential linear and introduce only small angular perturbations.

The use of a 5 DOF arm with a combination of with one-joint-at-a-time control and the lunge mode is judged to be satisfactory.

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ARM LENGTH	20 FT
REMOVABLE MASS	2770 LB
TIP FORCE	1000 LB
TIP STIFFNESS	1000 LB/INCH

CONTROL MODES:

- ONE-JOINT-AT-A-TIME MAX SPEED — 1°/SEC
- LUNGE MODE (SEE BELOW) MAX TIP SPEED — 3 IN/SEC

JOINT MOTIONS:

SHOULDER YAW	+180°	-180°
SHOULDER PITCH	+ 60°	- 90°
ELBOW PITCH	+140°	- 45°
WRIST PITCH	+130°	-130°
WRIST ROLL	+180°	-180°

BASE FRAME IS GROUND ADJUSTABLE FOR POSITION IN CARGO BAY.

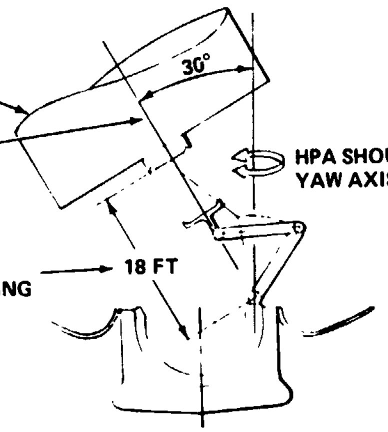

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Fig. 16 HPA Flight Article Characteristics

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THE SPACE SHUTTLE ORBITER PAYLOAD RETENTION SYSTEMS

J. H. Hardee
Rockwell International

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Space Transportation &
Systems Group

Rockwell
International

THE SPACE SHUTTLE ORBITER

PAYLOAD RETENTION SYSTEMS

By J.H. Hardee

INTRODUCTION

The Space Shuttle orbiter provides structural support attachment points for carrier/payloads along the length of the cargo bay. Nondeployable payloads are retained by passive retention devices, and deployable payloads are secured by motor-driven, active retention devices. Payloads are secured in the orbiter payload bay by means of the payload retention system or are equipped with their own unique retention systems.

The orbiter payload retention mechanisms provide structural attachments for each payload by using four or five attachment points to secure the payload within the orbiter payload bay during all phases of the orbiter mission (see Figures 1 and 2). The payload retention system (PRS) is an electromechanical system that

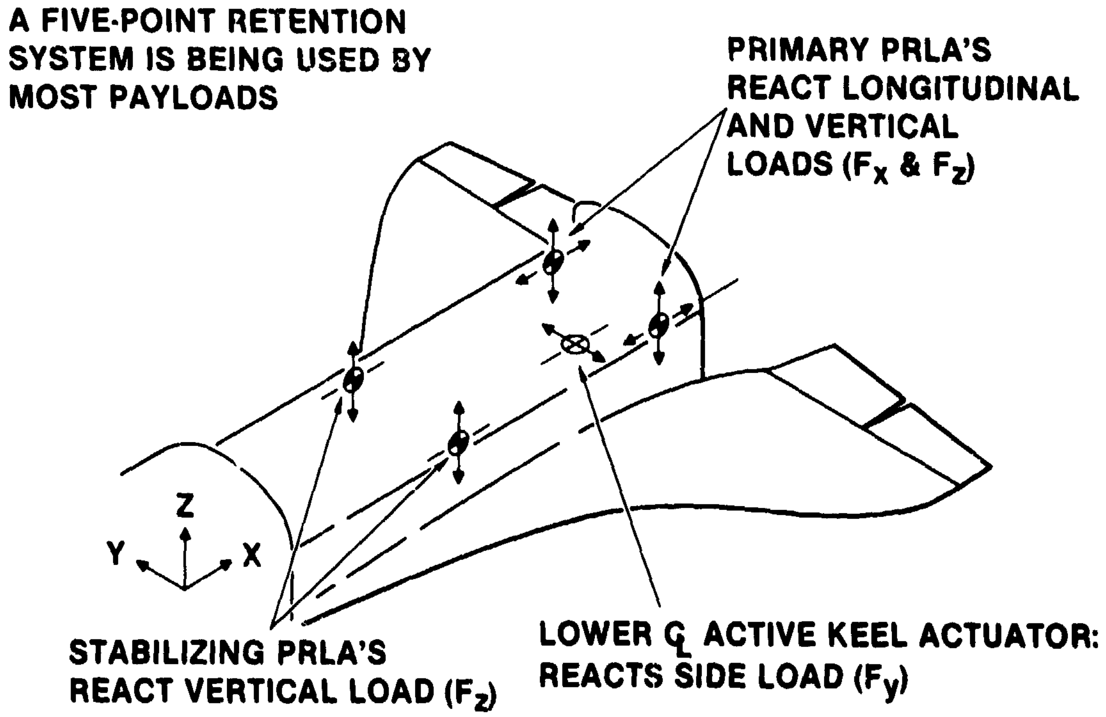

Figure 1. Payload Five-Point Retention

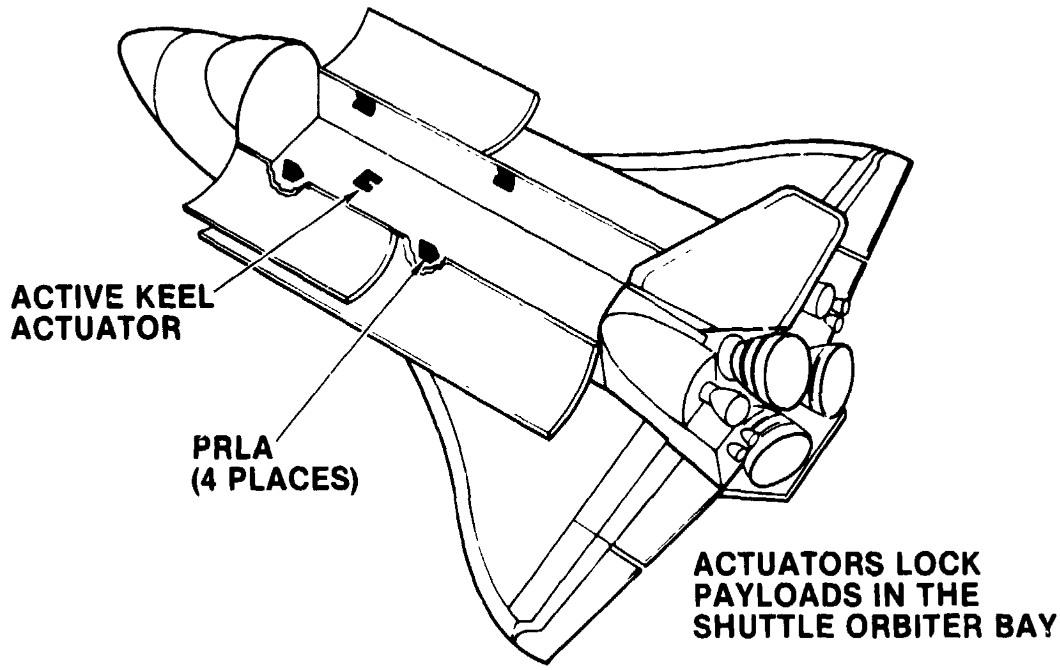

Figure 2. Payload Retention Actuators for the Space Shuttle Program

provides standardized payload carrier attachment fittings to accommodate up to five payloads for each orbiter flight. The mechanisms are able to function under either 1g or zero-g conditions. Payload berthing or deberthing on-orbit is accomplished by utilizing the remote manipulator system (RMS). The retention mechanisms provide the capability for either vertical or horizontal payload installation or removal.

The payload support points are selected to minimize point torsional, bending, and radial loads imparted to the payloads. Trunnion friction loads are minimized; however, at the present time there is a $C_F = 0.10$ to 0.25 depending upon environment and load. In addition to the remotely controlled latching system, the passive system used for nondeployable payloads performs the same function as the RMS except it provides fixed attachments to the orbiter.

PAYLOAD/ORBITER INTERFACE

Retention lugs (trunnions) on the payload extend beyond the 15-foot diameter payload envelope, where they are grasped and secured by the payload retention mechanisms. The location of the attachment points may vary along the length of the payload bay, depending on a flight's assigned payload. These locations are established (and changed as needed) during orbiter ground turnaround. The remotely operated latches and release mechanisms for deployable payloads provide guides to facilitate proper orientation of payloads during remote operation (Figure 3).

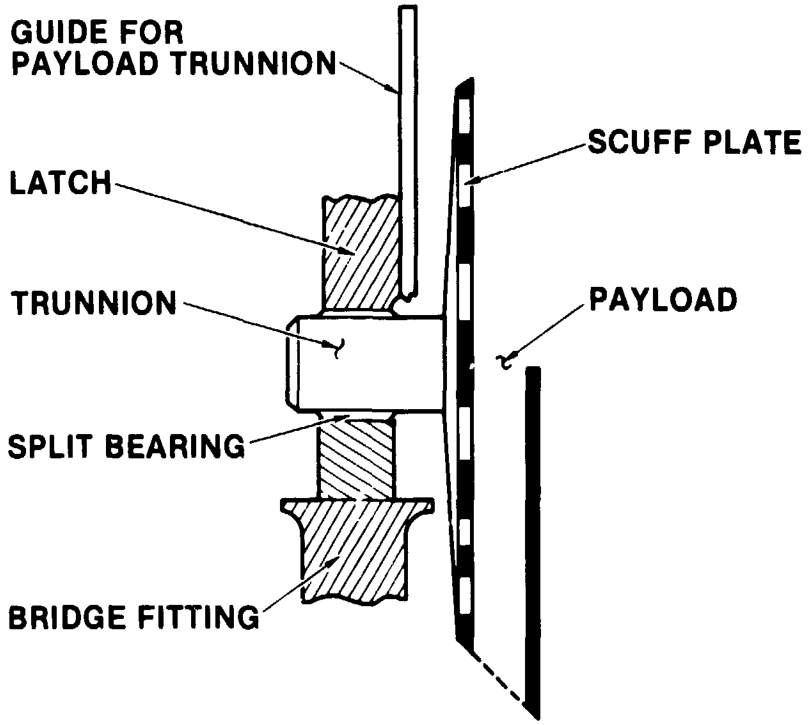

Figure 3. Payload/Orbiter Interface

FUNCTIONAL DESCRIPTION

The PRS generally consists of a set of five retention devices, four of which are located on fittings attached to the upper longeron and contain the payload guides. Two of these devices react against loads in both the vertical and horizontal planes, and two react against loads in the vertical plane only. The fifth retention device is located at the lower centerline of the payload bay and reacts against side loads only (see Figures 4 through 8).

Load and temperature usage limits for standard longeron latches and lightweight longeron latches are presented in Tables 1 and 2.

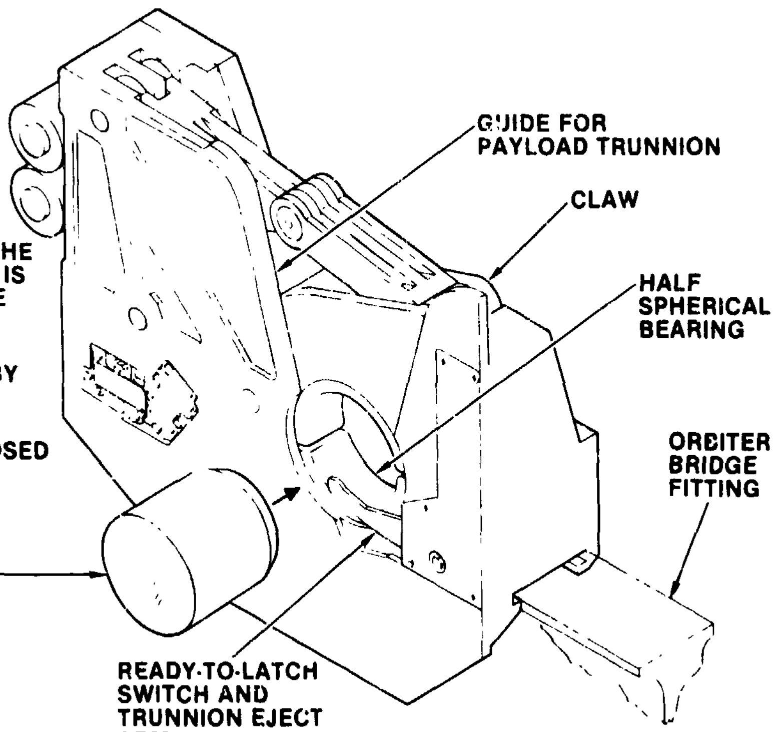

Figure 4. PRLA Payload Installation Sequence

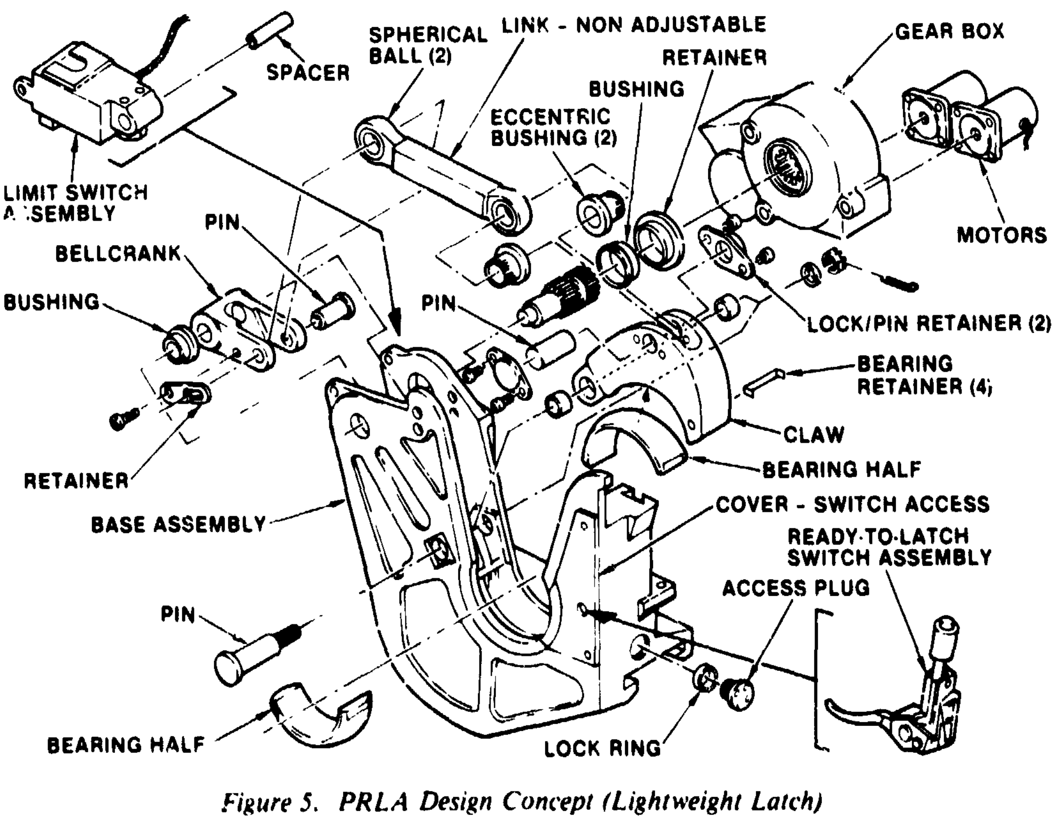

Figure 5. PRLA Design Concept (Lightweight Latch)

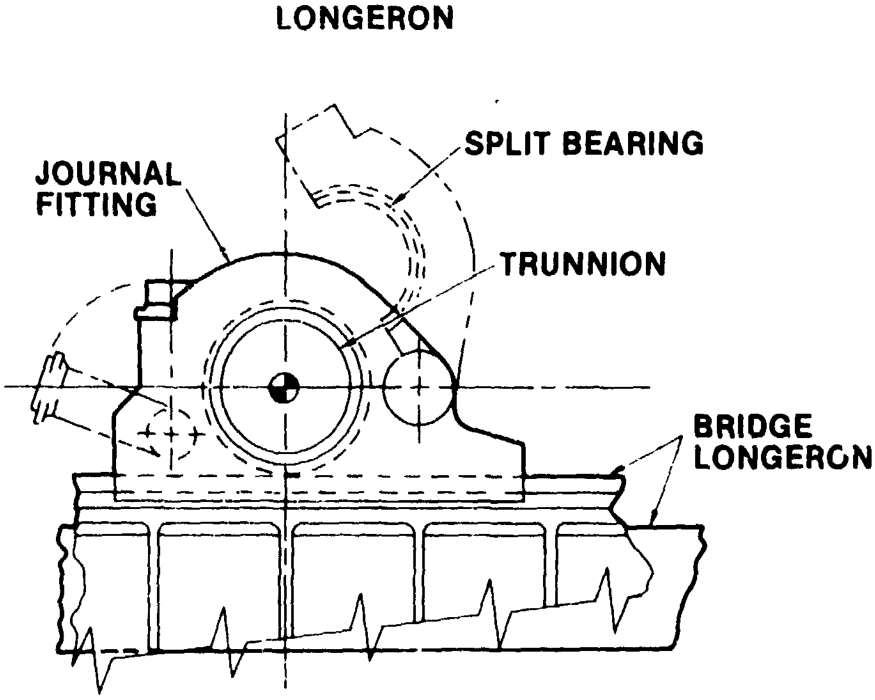

Figure 6. Passive Longeron Fittings

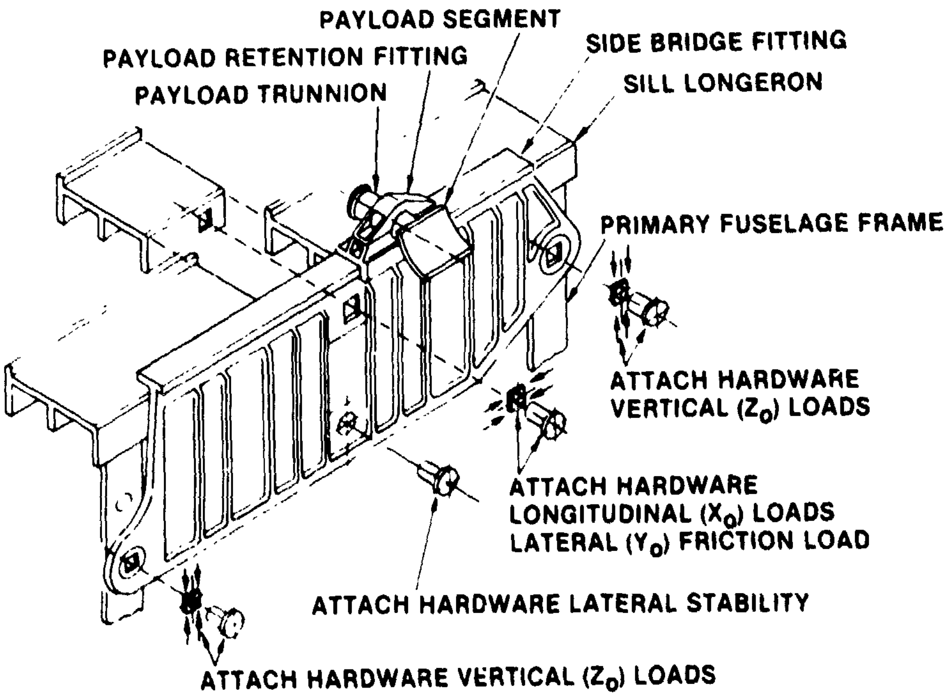

Figure 7. Typical Longeron Bridge

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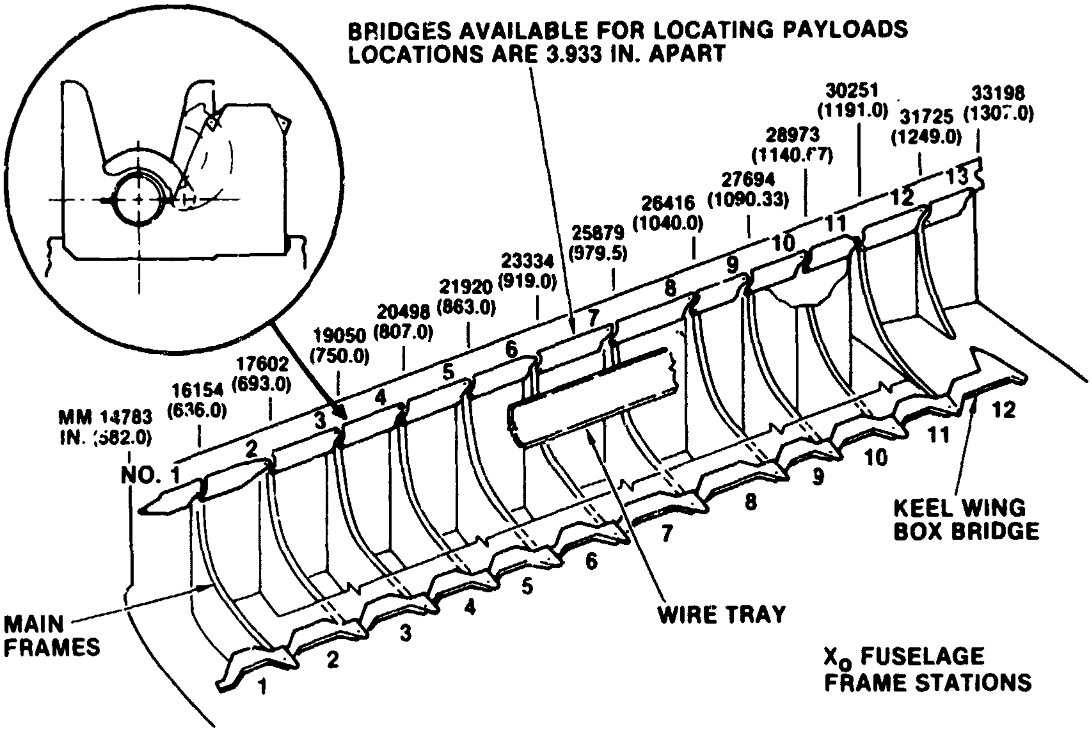

Figure 8. Payload Retention System General Mechanism Locations

Table 1. Load and Temperature Usage Limits—Flight Conditions; Standard Latches (Longeron)

Loads and Conditions \ Latches	Longeron Latch
	Standard Weight
Ultimate load at X-Z plane	169,400 lb (radius)
Temperature range	+290°F to −200°F (−100°F operational)

Table 2. Load and Temperature Usage Limits—Flight Conditions; Lightweight Latches (Longeron)

Load and Conditions \ Latches	Longeron Latch — Lightweight
Ultimate load at X-Z plane	67,600 lb (radius)
Temperature range	+275°F to −200°F (−100°F operational)

DESIGN-PAYLOAD RETENTION LATCH ACTUATORS (PRLA)

The PRLA is comprised of the following major assemblies:

1. Electromechanical actuator - gear box, and redundant motor/brake assembly
2. Drive linkage - mechanical drive from gear box to movable retention device (claw)
3. Base assembly - structural mount to bridge fitting
4. Switch modules - READY-TO-LATCH and LATCHED indicators
5. Guide assembly - centering device for payload berthing
6. Eject spring mechanism - to overcome any stiction of payload trunnion to bearing

DESIGN-ACTIVE KEEL ACTUATOR

The active keel actuator is comprised of the following major assemblies:

1. Electromechanical actuator - gear box and redundant motor/brake assembly systems
2. Drive linkage - ball screw mechanical drive from gear box to movable latch
3. Base assembly - structural mount to bridge fitting
4. Switch modules - READY-TO-LATCH and LATCHED indicators

See Figures 9 through 13.

Load and temperature usage limits for standard and lightweight keel latches are presented in Tables 3 and 4.

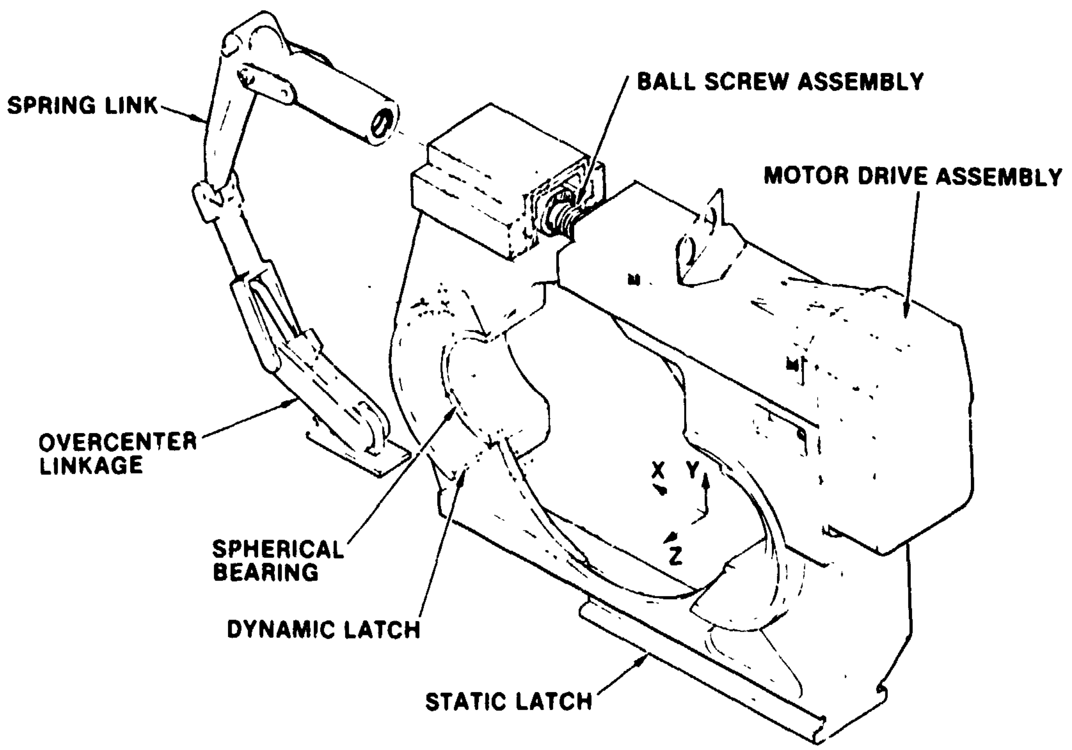

Figure 9. Active Keel Actuator

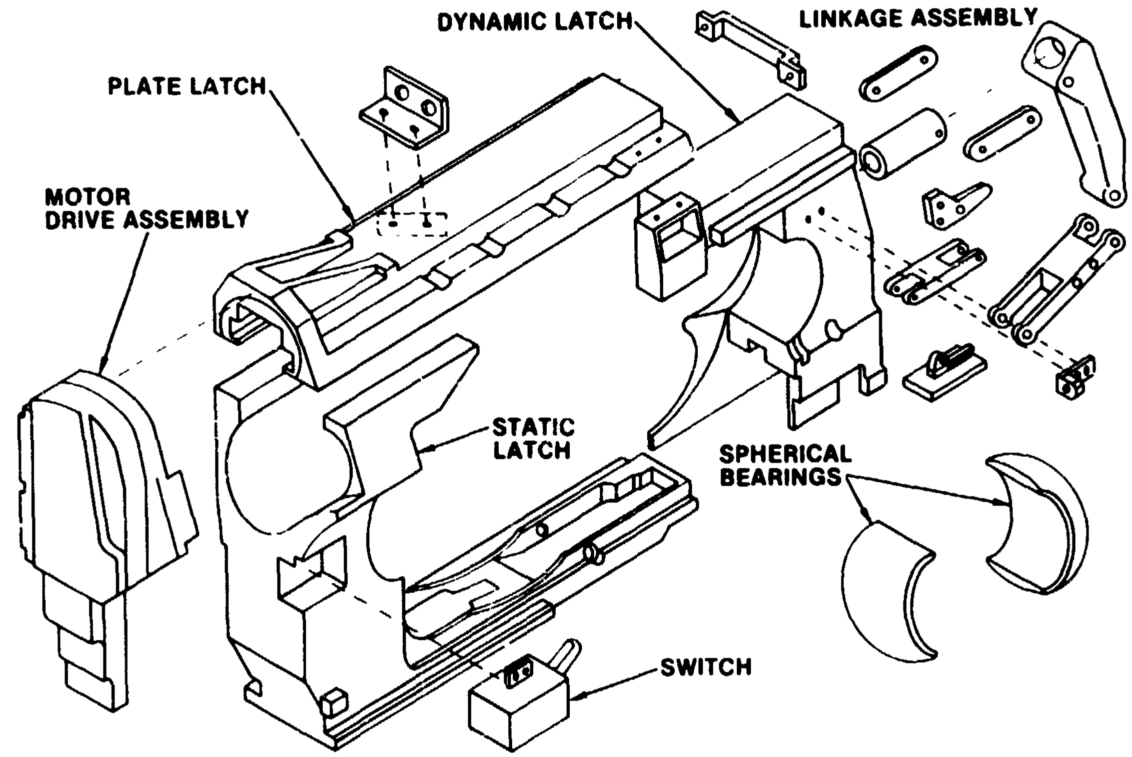

Figure 10. Active Keel Actuator Design Concept (Lightweight Concept)

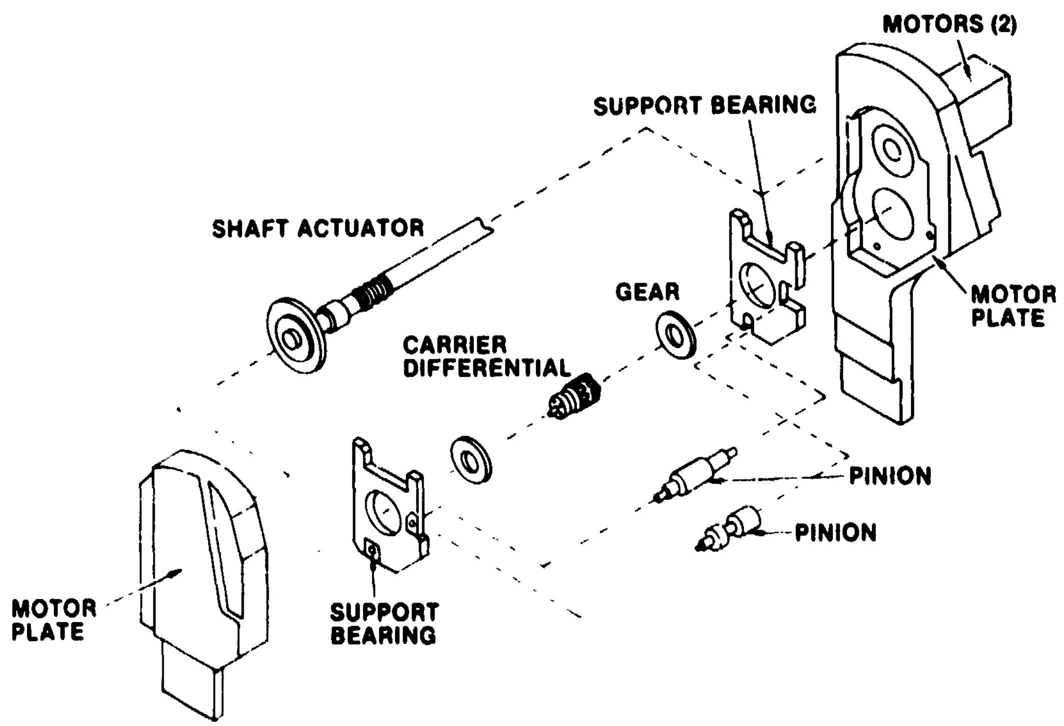

Figure 11. Mot r Drive Assembly (Active Keel Actuator Lightweight Latch)

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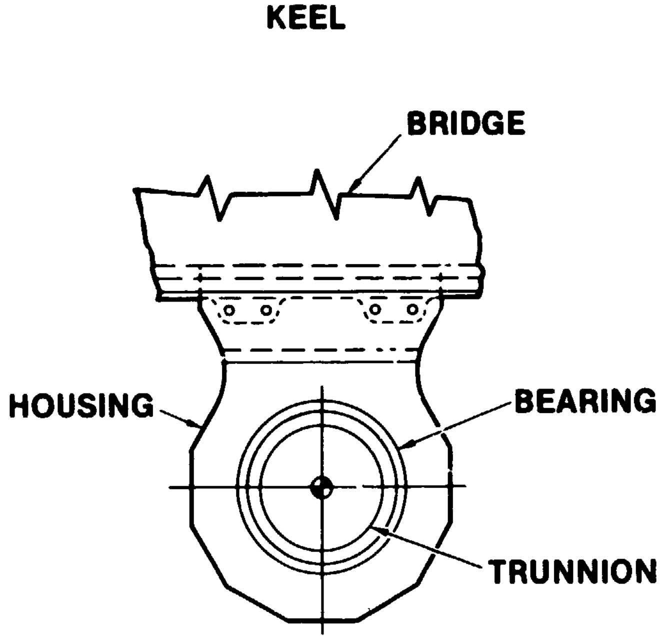

Figure 12. Passive Keel Fitting

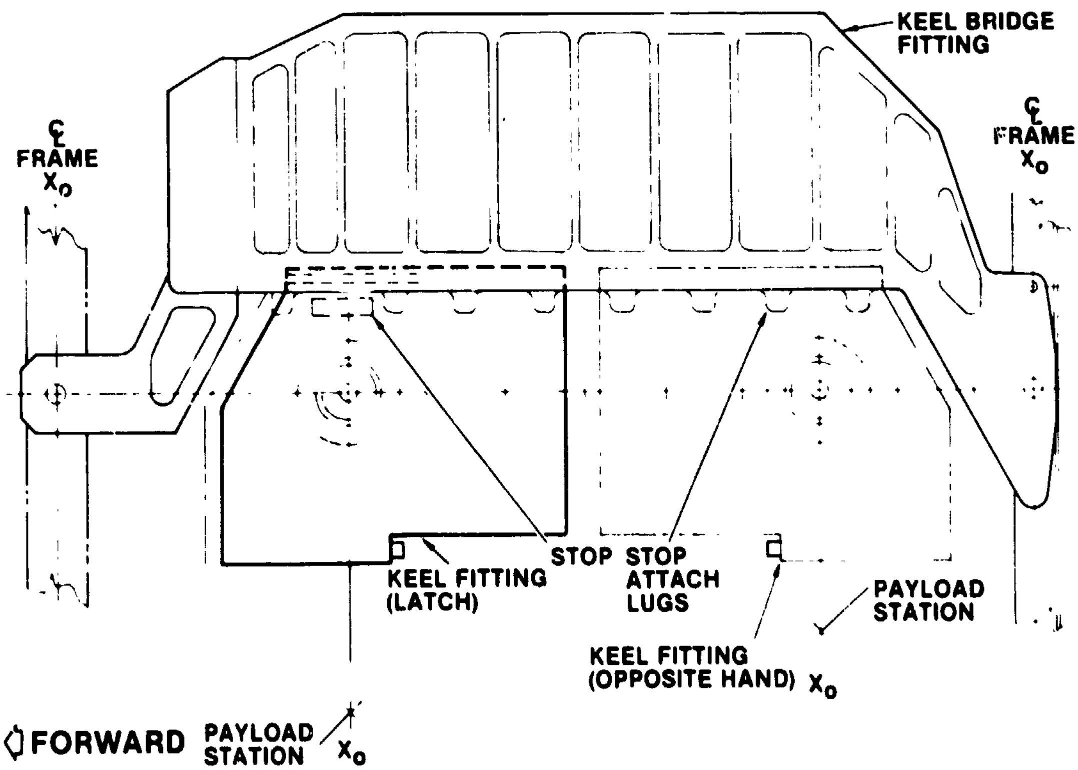

Figure 13. Payload Retention System Active Keel Fitting (Closed Position)

Table 3. Load and Temperature Usage Limits—Flight Conditions; Standard Latches (Keel)

Loads and Conditions \ Latches	Keel Latch
	Standard Weight
Ultimate load at X-Z plane	P_Y = ± 144,200 lb
Temperature range	+330°F to −200°F (−100°F operational)

Table 4. Load and Temperature Usage Limits—Flight Conditions; Lightweight Latches (Keel)

Load and Conditions \ Latches	Keel Latch
	Lightweight
Ultimate load at X-Z plane	P_Y = ± 57,600
Temperature range	+275°F to −200°F (−100°F operational)

FUTURE APPLICATIONS

At the present time payload retention mechanisms have been designed, developed, and certified that are capable of supporting a payload of up to 65,000 pounds through all phases of flight. For weight saving purposes and handling enhancement, a lightweight latch system is now being designed to accommodate the lighter payloads. Intermediate capability latching mechanisms are in the proposal stage. Unique latching systems can be developed at minimum cost and with minimum schedule impact by using already developed and certified system components.